AIRPLANE MAINTENANCE MANUAL

PA-34-220T

Seneca IV

SENECA

(SN'S 3447001 THRU 3447029)

(SN'S 3449001 AND UP)

PIPER AIRCRAFT, INC.

PART NUMBER 761-888

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Member
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Manufacturers Association

REVISION STATUS

1. Definitions

The follow definitions apply to the columns in paragraph 2.

A. Revision

The data in the revision column is comprised of two elements:

(1) A Type of Revision Code: ORG = Original, CR = Complete Revision, and PR = Partial Revision.

NOTE: Partial Revisions (PR) are listed only until the next Complete Revision (CR) is published. Then they are removed.

(2) The Revision Date in YYYYMMDD format.

NOTE: The Revision Date is the date placed on each revised page. It exists to separate one version of a page from another. Revision Date does not indicate the calendar date when the revision was actually published and available to the public. However, ***this is the date that appears in the Current Revision Checklist*** in the Customer Service Information File.

B. Publication Date

Publication Date usage has varied over the years. 1937–1996 is unknown. 1997–2013 it generally was synchronized with the revision date regardless of when the revision was published. In mid-2013 and later, the Publication Date is the calendar date when the revision was actually published and available to the public.

NOTE: ***This date does not appear in the Current Revision Checklist*** in the Customer Service Information File.

2. Revisions

Revisions to this maintenance manual (P/N 761-888) originally issued July 12, 1995, are as follows:

Revision	**Publication Date**
ORG19950712	November 7, 1996
CR20050819	August 19, 2005
CR20070530	May 30, 2007
CR20180930	December 11, 2018
PR20190831	September 13, 2019
PR20231010	June 27, 2024

3. Availability

This maintenance manual, related inspection reports and manuals, service publications (SB, SL, etc.) and other Piper publications are available as described in the Owner Publications Catalog (part of the Customer Service Information File, see below).

Consult the “Customer Service Information File” (a free download from the Piper Aircraft, Inc., website at https://www.piper.com/technical-publications/) to verify that you have the latest revision.

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INTRODUCTION

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	2	Sep 30/18			
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INTRODUCTION

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INTRODUCTION

1. Instructions for Continued Airworthiness

WARNING: INSTRUCTIONS FOR CONTINUED AIRWORTHINESS (ICA) FOR ALL NON-PIPER APPROVED STC INSTALLATIONS ARE NOT INCLUDED IN THIS MANUAL. WHEN A NON-PIPER APPROVED STC INSTALLATION IS INCORPORATED ON THE AIRPLANE, THOSE PORTIONS OF THE AIRPLANE AFFECTED BY THE INSTALLATION MUST BE INSPECTED IN ACCORDANCE WITH THE ICA PUBLISHED BY THE OWNER OF THE STC. SINCE NON-PIPER APPROVED STC INSTALLATIONS MAY CHANGE SYSTEMS INTERFACE, OPERATING CHARACTERISTICS AND COMPONENT LOADS OR STRESSES ON ADJACENT STRUCTURES, THE PIPER PROVIDED ICA MAY NOT BE VALID FOR AIRPLANES SO MODIFIED.

The Piper PA-34-220T Seneca IV / V Maintenance Manual constitutes the Instructions for Continued Airworthiness (ICA) in accordance with Federal Aviation Regulations (FAR) Part 23, Appendix A. Chapter 4 contains the Airworthiness Limitations section (4-00-00) and the Inspection Program is in Chapter 5 (5-20-00).

2. General

This publication is prepared in accordance with the General Aviation Manufacturers Association (GAMA) Specification No. 2, with respect to the arrangement and content of the System/Chapters within the designated Chapter/Section-numbering system.

WARNING: USE ONLY GENUINE PIPER AIRCRAFT PARTS OR PIPER AIRCRAFT APPROVED PARTS OBTAINED FROM PIPER APPROVED SOURCES, IN CONNECTION WITH THE MAINTENANCE AND REPAIR OF PIPER AIRPLANES.

This manual generally does not contain hardware callouts for installation. Hardware callouts are only indicated where a special application is required. Even then, confirm the correct hardware per the PA-34-220T Parts Catalog, P/N 761-887, and FAR 43 for proper utilization.

Genuine PIPER parts are produced and inspected under rigorous procedures to insure airworthiness and suitability for use in PIPER airplane applications. Parts purchased from sources other than PIPER, even though identical in appearance, may not have had the required tests and inspections performed, may be different in fabrication techniques and materials, and may be dangerous when installed in an airplane.

Additionally, reworked or salvaged parts or those parts obtained from non-PIPER approved sources, may have service histories which are unknown or cannot be authenticated, may have been subjected to unacceptable stresses or temperatures or may have other hidden damage not discernible through routine visual or nondestructive testing. This may render the part, component or structural assembly, even though originally manufactured by PIPER, unsuitable and unsafe for airplane use.

PIPER AIRCRAFT, INC. expressly disclaims any responsibility for malfunctions, failures, damage or injury caused by use of non-PIPER approved parts.

NOTE: PIPER AIRCRAFT, INC. expressly reserves the right to supersede, cancel and/or declare obsolete any part, part number (P/N), kit, or publication that may be referenced in this manual without prior notice.

Also, Piper Aircraft, Inc. may possess manufacturer's data which defines minimum type design beyond what may be assumed by an authorized repair entity. When a repair is proposed, it is the responsibility of the repairer per AC 43.13-1 to determine that the proposed repair is not contrary to manufacturer's data. The repairer or aircraft owner or his agent should contact Piper directly to determine that a proposed repair is not in conflict with minimum type design capability.

If you have any question concerning the care of your airplane, be sure to include the airplane serial number in any correspondence to Piper.

3. Effectivity

This maintenance manual is effective for PA-34-220T Seneca IV airplanes serial numbers 3447001 thru 3447029, and Seneca V airplanes serial numbers 3449001 and up. This encompasses the model years shown in Chart 1.

NOTE: Chart 1 is provided as a general reference only.

NOTE: The maintenance manual for Seneca IV airplanes serial numbers 3448001–3448079 is Piper P/N 761-751.

4. Serial Number Explanation

Example:

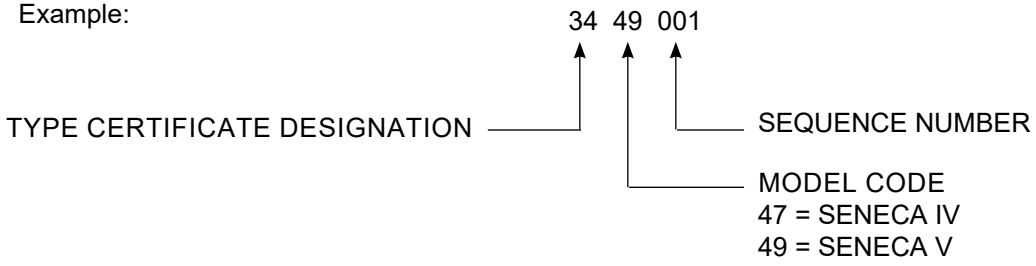


CHART 1
MODEL YEARS

Seneca IV	
Model Year	**Serial Numbers**
1995	3447001 thru 3447012
1996	3447013 thru 3447029

Seneca V	
Model Year	**Serial Numbers**
Prototype	3449001
1997	3449002 thru 3449041
1998	3449042 thru 3449096
1999	3449097 thru 3449151
2000	3449152 thru 3449195
2001	3449196 thru 3449231
2002	3449232, 3449237 thru 3449239, and 3449241 thru 3449268
2003	3449269, 3449273, and 3449275 thru 3449300
2004	3449301 thru 3449310
2005	3449311 thru 3449322
2006	3449323 thru 3449346
2007	3449347 thru 3449366
2008	3449367 thru 3449394
2009	3449395 thru 3449403
2010	3449404 thru 3449426
2011	3449427 thru 3449447
2012	3449448 thru 3449463
2013	3449464 thru 3449486
2014	3449487 thru 3449494
2015	3449495 thru 3449506
2016	3449507
2017	3449508
2018	3449509 thru 3449510
2019	3449511 thru 3449513
2020	3449514

5. Assignment of Subject Material

This publication is divided into industry standard, three element, numeric subject groupings as follows:

A. System/Chapter - The various groups are broken down into major systems such as Environmental Systems, Electrical Power, Landing Gear, etc. They are assigned a number, which becomes the first element of the standardized numbering system. Thus, the element “28” of the number 28-40-01 refers to the chapter “Fuel”. Everything concerning the fuel system will be covered in this chapter.

B. Sub-System/Section - The major systems/chapters of an airplane are broken down into subsystems. These sub-systems are identified by the second element of the standard numbering system. The element “40” of the number 28-40-01 concerns itself with the indicating section of the fuel system.

C. Unit/Subject - The individual units within a sub-system/section may be identified by the third element of the standard numbering system. The element “01” of the number 28-40-01 is a subject designator. This element is assigned at the option of the manufacturer and is normally zeroed out by PIPER.

Refer to paragraph 14, Chapter/Section Index Guide, for a complete breakdown and list. The material is arranged in ascending numerical sequence.

6. Pagination

The Chapter - Section (i.e., 28-40-00) numbering system (explained above) forms the primary page numbering system for this manual. Within each Section, pages are numbered consecutively beginning with Page 1 (i.e., 28-40-00, Page 1).

7. Identifying Revised Material

A vertical line (i. e. - change bar) along the left-hand margin of the page (or text column) is used to identify revised text or illustrations.

Example.

A change bar in the left-hand margin opposite the footer (i.e. - chapter/section/subject, page number and date), indicates that the text was unchanged but the material was relocated.

Change bars in the individual chapter Tables of Contents do not indicate a change to that page, but rather that the information in the actual paragraph has changed.

NOTE: Change bars are not used in the title pages, list of effective pages, index, or Wiring Diagram Cross-Reference.

A revision to a page is defined as a change to the text or illustrations that existed previously. Revisions, additions and deletions are identified by a vertical line (i. e., change bar) along the left-hand margin of the page opposite only the text or illustration that was changed. Reformatted, but otherwise unchanged, text is not identified by a change bar.

8. Indexing

An alphabetically arranged subject Index follows this introduction to assist the user in locating desired information. In addition, each System/Chapter begins with an individual Table of Contents.

9. List of Effective Pages

Each System/Chapter has a List of Effective Pages preceding the Table of Contents to identify the effective revision date for each page in that chapter.

10. Warnings, Cautions and Notes

These adjuncts to the text are used to highlight or emphasize important points when necessary. Warnings call attention to use of materials, processes, methods, procedures or limits which must be followed precisely to avoid injury or death to persons. Cautions call attention to methods and procedures which must be followed to avoid damage to equipment. Notes call attention to methods which make the job easier. Warnings and Cautions shall be located directly above and Notes directly beneath the text and be in line with the paragraphs to which they apply.

11. Accident/Incident Reporting

To improve our Service and Reliability system and aid in Piper's compliance with FAR 21.3, knowledge of all incidents and/or accidents must be reported to Piper immediately. To expedite and assist in reporting all incidents and accidents, Piper Form 420-01 has been created. See Service Letter 1041, latest revision, for the current form. This procedure is to be used by all Dealers, Service Centers and Repair Facilities.

12. Supplementary Publications

The following publications/sources provide servicing, overhaul and parts information for the PA-34-220T Seneca IV–V airplanes and their various components. Use them to supplement this manual.

A. Piper Publications: (by Part Number)

	Seneca IV	Seneca V
(1) Parts Catalog:	761-887	761-887
(2) Periodic Inspection Report:	767-012	767-012
(3) Progressive Inspection Manual (50 Hour):	767-006	767-006

B. Vendor Publications:

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY.

(1) AIR CONDITIONING COMPRESSOR:

Vendor: Sanden International (USA), Inc.
601 South Sanden Blvd.
Wylie, TX 75098-4999
http://www.sanden.com
PH: (972) 442-8400

(2) AUTOFLIGHT:

Vendor(s): Honeywell
One Technology Center
23500 W. 105th St., M/D #45
Olathe, Kansas 66061-1950
http://www.bendixking.com/

(or)

S-TEC Corporation
One S-TEC Way
Mineral Wells, TX 76067
PH: (940) 325-9406
http://www.s-tec.com

Flight Control: Bendix/King

System Flight Line Installation Manual: KFC 150
P/N 006-0287-00

(3) BATTERY:

Vendor:	Concorde Battery Corporation 2009 San Bernardino Road West Corvina, California 91790 http://www.concordebattery.com	PH:	(626) 813-1234
RG® Series Emergency Aircraft Battery CMM:	Document No. 5-0167		
RG® Series Main Aircraft Battery CMM:	Document No. 5-0171		

or

Vendor:	GILL Batteries A Division of Continental http://www.gillbatteries.com	PH:	(800) 456-0070

(4) BRAKES AND WHEELS:

Vendor:	Parker Hannifin Corp. Aircraft Wheel and Brake Division 1160 Center Road Avon, Ohio 44011 http://www.parker.com/portal/site/PARKER/menuitem. (Select Divisions and Aircraft Wheel and Brake Division.)	PH:	(800) 272-5464

(5) ELECTRONIC FLIGHT INSTRUMENT SYSTEM (EFIS):

Vendor:	Meggitt Avionics, Inc. 10 Ammon Drive Manchester, NH 03103-7406 http://www.meggitt.com/	PH: FAX:	(603) 669-0940 (603) 669-0931

(6) EMERGENCY LOCATOR TRANSMITTER:

Vendor:	Artex Aircraft Supplies 14405 Keil Road NE Aurora, Oregon 97002 http://www.artex.net/	PH:	(800) 547-8901

(7) ENGINE:

Vendor:	Continental Aerospace Technologies P.O. Box 90 Mobile, AL 36601 http://www.continentalmotors.aero/	PH: (800) 718-3411

SENECA IV

Maintenance Manual =	M-8
Operator's Manual =	OI-8
Overhaul Manual =	M-8
Parts Catalog =	See Continental website.

SENECA V

Maintenance Manual =	M-8
Operator's Manual =	OI-8
Overhaul Manual =	M-8
Parts Catalog =	See Continental website.

(8) FIRE EXTINGUISHER (PORTABLE):

Vendor:	H3R Inc. 43 Magnolia Ave # 4 San Francisco, California 94123-2911 http://www.h3r.com/index.htm	PH: (800) 249-4289

(9) FUEL CELLS:

Vendor:	Meggitt (Rockmart), Inc 669 Goodyear Street Rockmart, Georgia 30153-0548 https://www.meggitt.com/meggitt-polymers-composites/	PH: (770) 684-7855 FAX: (770) 684-7438

(10) FUEL PUMP:

Vendor:	Weldon Pumps P.O. Box 46579 640 Golden Oak Parkway Oakwood Village, Ohio 44146 https://www.weldonpumps.com	PH: (440) 232-2282 FAX: (440) 232-0606

(11) GEAR LOCKING ACTUATORS, HYDRAULIC PUMP, & ALL HYDRAULIC COMPONENTS:

Vendor:	Parker Hannifin Corporation 6035 Parkland Boulevard Cleveland, OH 44124-4141 USA http://www.parker.com/portal/site/PARKER/menuitem. (Select Divisions and Hydraulic Systems Division.) email: c-parker@parker.com	PH: (800) C-PARKER (800) 272-7537 FAX: (440) 266-7400

(12) HEATER:

Vendor:	Hartzell Engine Technologies 2900 Selma Highway Montgomery, AL 36108 http://www.hartzellenginetech.com/	PH: FAX:	(877) 359-5355 (334) 386-5410

Model B-3040, P/N 91E88-1 Heater:

Maintenance and Overhaul Manual:	Aircraft Heater Overhaul and Maintenance Manual - 24E25-1

Model B-3500, P/N 91E88-1(EL) Heaters:

Maintenance and Overhaul Manual:	Aircraft Heater Overhaul and Maintenance Manual - 94E47

Model I-3500, P/N 91E88-1(IS) Heaters:

Owners Manual:	99M90
Overhaul Manual:	99M91

(13) INTEGRATED AVIONICS SYSTEM (IAS):

Vendor:	Garmin International 1200 East 151ST Street Olathe, KS 66062 http://www.garmin.com	PH:	(913) 397-8200
Garmin G1000	PA-34-220T Pilots Guide:	Part No. 190-01404-00	
	G1000 Line Maintenance and Configuration Manual:	Part No. 190-00303-04	
Garmin G1000 NXi	PA-34-220T Pilots Guide:	Part No. 190-02477-00	

(14) INTEGRATED FLIGHT DISPLAY SYSTEM (IFDS):

Vendor:	Avidyne Corporation 55 Old Bedford Road Lincoln, MA 01773 http://www.avidyne.com/support/support.asp	PH:	(800) 284-3963

Instructions for Continued Airworthiness

Primary Flight Display and Magnetometer/OAT:	Document No. AVPFD-174
Multifunction Display:	Document No. AVMFD-167
Data Acquisition Unit:	Document No. AVSIU-011

(15) KEVLAR:

Vendor:	KEVLAR Special Products E.I. DuPont De Nemours & Co. Inc. Textile Fibers Department Centre Road Building Wilmington, Delaware 19898
Guide:	A Guide to Cutting and Machining Kevlar Aramid

(16) LIGHTS - NAVIGATION, STROBE, AND MAP LIGHTS:

Vendor:	Whelen Engineering Co. Inc. Route 145, Winthrop Rd. Chester, Connecticut 06412 http://www.whelen.com/aviation/index	PH: FAX:	(860) 526-9504 (860) 526-2009

(17) MAGNETOS:

Vendor:	TCM Aircraft Products P. O. Box 90 Mobile, AL 36601 http://www.tcmlink.com/	PH: FAX:	(800) 718-3411 (251) 432-7352
Installation, Operation and Maintenance Instructions:	Continental Service Support Manual, P/N X42002-3, S-20 / S-200 Series High Tension Magnetos		
or,			
Vendor:	Slick Aircraft Products Champion Aerospace P.O. Box 686 1230 Old Norris Road Liberty, SC 29657 http://www.championaerospace.com/	PH:	(904) 772-1909
Installation, Operation And Maintenance Instructions:	F1100 Master Service Manual, 4300/6300 Series Magneto Maintenance and Overhaul Manual - L-1363		

(18) NAVIGATION, COMMUNICATIONS, AND GPS (NAV/COM/GPS):

Vendor:	Garmin International 1200 East 151ST Street Olathe, KS 66062 http://www.garmin.com	PH:	(913) 397-8200

(19) OXYGEN SYSTEM:

Vendor:	Avox Systems Inc (Zodiac Oxygen Systems) 2225 Erie Street Lancaster, New York 14086 http://www.zodiacaerospace.com/en/zodiac-oxygen-systems-us	PH: FAX:	(716) 683-5100 (716) 681-1089

(20) PROPELLER:

Vendor:	Hartzell Propeller Inc. One Propeller Place Paqua, Ohio 45356-2634 http://hartzellprop.com/	PH: FAX:	(937) 778-4379 (937) 778-4321
Overhaul Instructions:	Manual No. 117		
Owner's Manual:	Manual No. 115N		
or,			
Vendor:	McCauley Accessory Division 3535 McCauley Drive P.O. Drawer 5053 Vandalia, Ohio 45377-5053		
Service Manual:	McCAULEY C500 SERIES SERVICE MANUAL - P/N 810915		

(21) STANDBY ATTITUDE INDICATOR:

Vendor:	Mid-Continent Instruments Co., Inc. 9400 E. 34 TH Street N. Wichita, KS 67226 http://www.mcico.com/index.html	PH: FAX:	(316) 630-0101 (316) 630-0723
Installation Manual and Operating Instructions:	Manual No. 9015762		

(22) STANDBY ELECTRONIC FLIGHT DISPLAY:

Vendor:	Aspen Avionics, Inc. 5100 Indian School Road NE Albuquerque, NM 87110 http://www.aspenavionics.com	PH:	(505) 856-5034
Tech Note for Air Data Calibration:	Tech Note 2009-09, Document No. 991-00023-001		

(23) STARTER:

Vendor:	Continental Motors P.O. Box 90 Mobile, AL 36601 http://www.continentalmotors.aero/	PH:	(800) 718-3411

(24) VACUUM PUMP:

Vendor:	Aero Accessories, Inc. 1240 Springwood Avenue Gibsonville, NC 27249 http://www.aeroaccessories.com/index.html	PH:	(800) 822-3200

(25) VACUUM REGULATOR:

Vendor:	Parker Hannifin Corp. Pneumatic Division North America 8676 EAST M89 Richland, MI 49083 USA http://www.parker.com/portal/site/PARKER/menuitem. (Select Divisions and Pneumatic Division.)	PH: Fax:	(269)-629-5000 (269)-629-5385

(26) VOLTAGE REGULATOR:

Vendor:	Electro Systems, Inc. Airport Complex P. O. Box 273 Fort Deposit, Alabama 36032 http://www.kellyaerospace.com	PH:	(888) 461-6077

13. Chapter/Section Index Guide

NOTE: The following GAMA Specification No. 2 standard chapters are not included in this Maintenance Manual: 36, 38, 49, 54, 60, 72, 75, 78, and 83. These chapters are omitted because the subject system is either: not installed in these airplanes; adequately covered in vendor or other manuals; or, for ease of use, has been combined with another chapter.

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	10	Jacking
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	20	Leveling
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CHAPTER

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AIRWORTHINESS LIMITATIONS

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CHAPTER 4

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AIRWORTHINESS LIMITATIONS

NOTE: The Airworthiness Limitations section is FAA-approved and specifies maintenance required under §§ 43.16 and 91.403 of the Federal Aviation Regulations unless an alternative program has been FAA-approved.

1. Approved Mandatory Replacement Times for Type Certification

(PIR-TCDS A7SO, Rev. 23.)

The following limitations related to the fatigue life of the airplane and its components have been established for the PA-34-220T Seneca IV–V airplane:

See Chart 1 for safe life limits of components.

2. Approved Mandatory Structural Inspection Intervals

None.

3. Inspection Procedures for those Approved Mandatory Structural Inspection Items

None.

CHART 1
COMPONENT LIFE LIMITS

Component	Life Limit (Flight Hours)
Upper Drag Link Attachment to Nose Gear Trunnion • P/N 400-274 (AN7-35) bolt or P/N 693-215 (NAS6207-50D) bolt • P/N 407-591 (AN960-716L) washer, as applicable • P/N 407-568 (AN 960-716) washer, as applicable • P/N 404-396 (AN 320-7) nut • P/N 424-085 (MS24665-302) cotter pin	500

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CHAPTER

5

TIME LIMITS / MAINTENANCE CHECKS

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CHAPTER 5

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CHAPTER 5 - TIME LIMITS / MAINTENANCE CHECKS

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Each 1000 Hours		2
Each 1800 Hours		3
Each 2000 Hours		3
Each 2200 Hours		3
Each 2400 Hours		3
Per Heater Operation Hour		4
Each 100 Hours		4
Each 200 Hours		4
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CHAPTER 5 - TIME LIMITS / MAINTENANCE CHECKS

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Each Four (4) Years		6
Each Five (5) Years		6
Each Six (6) Years		7
Each Seven (7) Years		7
Each Eight (8) Years		7
Each Ten (10) Years		8
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GENERAL

Piper Aircraft, Inc. (Piper) takes a continuing interest in having the owner get the most efficient use from his airplane, and keeping the airplane in the best mechanical condition. To that end, Piper publishes a recurring maintenance schedule which is supplemented with Service Bulletins, Service Letters and Service Spares Letters as required.

A. The recurring maintenance schedule for the PA-34-220T Seneca IV–V is provided in 5-20-00.

B. Piper Service Bulletins are of special importance and Piper considers compliance mandatory.

C. Service Letters deal with product improvements and service hints pertaining to the affected aircraft. Owners should give careful attention to service letter information so they can ensure their airplane is properly serviced and kept up to date with the latest changes.

D. Service Spares Letters offer improved parts, kits and optional equipment which were not originally available. These may be of interest to the owner.

E. Service Bulletins, Service Letters, and Service Spares Letters are also available by subscription. See the availability statement in Revision Status.

NOTE: Piper mails flight manual (AFM / POH) revisions to the registered owner's name and address as shown on the Aircraft Registration Certificate. If the aircraft is based and/or operated at a different location (or locations) and/or by a person (or persons) other than those recorded on the aircraft registration, then the registered owner(s) is responsible for forwarding these to the operating location(s) or person(s).

Changes in aircraft registration may take a substantial amount of time to be recorded by the Federal Aviation Administration and received by Piper to change the mailing address. Owners and operators should make arrangements to keep abreast of flight manual revisions during this interim period through their Piper Dealer/Service Center.

The Federal Aviation Administration (FAA) publishes Airworthiness Directives (AD's) that apply to specific aircraft. They are mandatory changes and are to be complied within a time limit set by the FAA. When an AD is issued, it is sent to the latest registered owner of the affected aircraft and also to subscribers of their service. The owner is solely responsible for being aware of and complying with airworthiness directives.

NOTE: A searchable database of AD's is available on the FAA website. See the "Airworthiness Directives" link at http://www.faa.gov.

Owners should periodically check with a Piper Dealer/Service Center to find out the latest information to keep his aircraft up to date.

Service Bulletins, Service Letters, and Service Spares Letters are also available by subscription. See the availability statement in Revision Status.

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TIME LIMITS

1. General

   A. Refer to 4-00-00 for the FAA-approved airworthiness limitations section. It sets forth each mandatory replacement time, structural inspection interval, and related structural inspection procedure required for type certification.

   B. Refer to 5-20-00 for Piper's recommended Inspection Programs. They include the frequency and extent of the inspections required for the continued airworthiness of these airplanes.

   C. Inspections required by Flight Hour or Calendar Year, if due, are included as part of the Annual / 100 Hour Inspection and/or the Progressive Inspection Event cycles, and are listed individually in 5-30-00.

2. Life Limited Parts Marking and Disposition

   14 CFR Part 43.10, Disposition of Life-Limited Aircraft Parts requires that proper procedures are followed when removing life limited parts with time and/or cycles remaining on them as well as the disposition of life limited parts with no time and/or cycles left. Life limited parts defined by Type Certificate (TC) are listed in 4-00-00. Other parts which are replaced or rebuilt after having accumulated cycles, hours, or other replacement limit are specified in Chapter 5.

   A. Parts that are removed prior to accumulating their life limit, are to be marked with indelible ink or marker with the part number, serial number and accumulated life status as defined in 14 CFR Part 43.10 in a manner that does not affect part structural integrity, i.e. - no surface deformation such as vibration/etching allowed.

   B. Parts that have accumulated the life limit shall be disposed of in accordance with the applicable FARs. Piper recommends life limited parts with no time and/or cycles remaining be completely destroyed.

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SCHEDULED MAINTENANCE

This section provides instructions for conducting inspections. Repair or replacement instructions for those components found to be unserviceable during inspections will be found in the applicable airplane system chapter. (See Chapter/Section Index Guide, Introduction.)

WARNING: GROUND THE MAGNETO PRIMARY CIRCUIT (P LEAD), BEFORE PERFORMING ANY MAINTENANCE OPERATION ON THE ENGINE.

1. Description

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

The recurring maintenance schedule for the PA-34-220T Seneca IV / V is provided herein as an Annual/ 100 Hour Inspection. A Progressive Inspection Program is available in a separate manual form. See the availability statement in Revision Status.

Piper inspection programs comply with the F.A.A. Federal Aviation Regulations Parts 43, 91 and 135. The owner/operator is primarily responsible for maintaining the airplane in an airworthy condition, including compliance with all applicable Airworthiness Directives and conformity with the requirements in FAR 91.409, 91.411 and 91.413.

The first overhaul or replacement of components should be performed at the given periods. The condition of various components can then be used as criteria for determining subsequent periods applicable to the individual airplane, depending on usage, providing the owner/operator has an established Part 91 Progressive Inspection Program (see 91.409(d)) or Part 135 Approved Aircraft Inspection Program (see 135.419).

The time periods given for inspections of various components are based on average usage and environmental conditions.

NOTE: The listed inspection, overhaul and replacement schedules do not guarantee that a particular item or component will reach the listed time without malfunction. Unique operating conditions encountered by individual airplanes cannot be controlled by the manufacturer.

2. Definitions

A. Inspections - Must be performed only by persons authorized by the FAA or appropriate National Aviation Authority who are qualified on these aircraft, using acceptable methods, techniques and practices to determine physical condition and detect defects.

(1) Routine Inspection - Consists of a visual examination or check of the aircraft and its components and systems without disassembly.

NOTE: This includes examination of all fasteners for broken torque stripe. If broken torque stripe is found, loosen fastener, retorque, apply torque stripe, and safety, as required. Please report broken torque stripe via the FAA Service Difficulty Reporting (SDR) system.

(2) Detailed Inspection - Consists of a thorough examination of the aircraft, appliance, component, or system; with disassembly as necessary to determine condition.

(3) Special Inspection - Involves those components, systems or structure which by their application or intended use require an inspection peculiar to, more extensive in scope or at a time period other than that which is normally accomplished during an event or annual inspection.

B. Checks - Can be performed by pilots and/or mechanics who are qualified on this aircraft and consist of examinations in the form of comparisons with stated standards for the purpose of verifying condition, accuracy and tolerances.

C. Approved Inspection - Means a continuing airworthiness inspection of an airplane and its various component and systems at scheduled interval in accordance with procedures approved by the FAA under FAR Part 91.409(d) or Part 135.419.

D. Tests - Operation of aircraft components, appliances or systems to evaluate functional performance.

(1) Operational Test - A task to determine that an item is fulfilling its intended purpose. The task does not require quantitative tolerances. This is a fault finding task.

(2) Functional Test - A quantitative check to determine if one or more functions of an item performs within specified limits. This test may require the use of supplemental bench test equipment.

(3) In addition, each of the above tests must be performed by an FAA Certified Repair Station with appropriate ratings or by a Certified Mechanic who is qualified on this aircraft. The recording of the above function must be made in the permanent aircraft records by the authorized individual performing the test.

E. Bench Test - Means removal of component from the aircraft to inspect for cleanliness, impending failure, need for lubrication, repair or replacement of parts and calibration to at least the manufacturers specifications using the manufacturers recommended test equipment or standards or the equivalent.

Each bench test will be performed by a Piper Service Center, FAA Certified Repair Station with appropriate rating or by a certified mechanic. This test will be performed at the scheduled interval regardless of any bench test performed on a particular component while being repaired/overhauled before scheduled interval bench test. After the component is installed into the aircraft, an operational test of the component and its related system should be performed to ensure proper function. Serviceable parts that were issued to the component will be filed in the aircraft permanent records. The person performing the test must make appropriate entries in the aircraft's permanent maintenance record.

F. Maintenance - The word maintenance as defined by FAR Part 1, means "inspection, overhaul, repair, preservation and the replacement of parts, but excludes preventive maintenance."

G. On Condition Maintenance - A primary maintenance process having repetitive inspections or tests to determine the condition of units, systems, or portions of structure with regard to continued serviceability (corrective action is taken when required by item condition.)

H. Time - as used in this manual.

(1) Time-in-service for aircraft components, unless otherwise specified, is a cumulative total of flight hours or calendar time calculated from the time a new or overhauled component was first installed in any aircraft, and including:

(a) the aircraft time that elapses from the initial installation to the first removal, if any; and,

(b) the aircraft time that elapses from each subsequent installation to each subsequent removal, if any; or,

(c) the calendar time elapsed since the installation.

NOTE: Dates stamped on individual components at the time of manufacture are typically applied to determine shelf life - i.e., the maximum time allowed from manufacture/assembly/cure until actually installed in an aircraft and are not relevant.

Do not, however; ignore markings applied to life-limited parts when removed with time and/or cycles remaining on them.

(2) Aircraft time, flight hours, or aircraft hours are the "Hobbs Time" shown on, or calculated from, the installed "Hour Meter."

3. Inspection Requirements

WARNING: INSTRUCTIONS FOR CONTINUED AIRWORTHINESS (ICA) FOR ALL NON-PIPER APPROVED STC INSTALLATIONS ARE NOT INCLUDED IN THIS MANUAL. WHEN A NON-PIPER APPROVED STC INSTALLATION IS INCORPORATED ON THE AIRPLANE, THOSE PORTIONS OF THE AIRPLANE AFFECTED BY THE INSTALLATION MUST BE INSPECTED IN ACCORDANCE WITH THE ICA PUBLISHED BY THE OWNER OF THE STC. SINCE NON-PIPER APPROVED STC INSTALLATIONS MAY CHANGE SYSTEMS INTERFACE, OPERATING CHARACTERISTICS AND COMPONENT LOADS OR STRESSES ON ADJACENT STRUCTURES, THE PIPER PROVIDED ICA MAY NOT BE VALID FOR AIRPLANES SO MODIFIED.

Inspections must be accomplished by persons authorized by the FAA or appropriate National Aviation Authority. Checks may be performed by a pilot or owner who is checked out on the airplane.

A. Annual / 100 Hour Inspection

See Paragraph “Annual / 100 Hour Inspection Procedure” on page 05204.

Owners/operators may maintain the airplane solely under FAR 91.409 (a) and (b) inspection requirements. The 100 hour inspection cycle is a complete inspection of the airplane and is identical in scope to an annual inspection.

B. Progressive Inspection

The Progressive Inspection program is designed to permit the best utilization of the aircraft through the use of a planned inspection schedule. This schedule:

P/N 767-006, PA-34-220T, Seneca IV (S/N 3447001 thru 3447029) and Seneca V

is prepared in a manual form. See the availability statement in Revision Status.

Refer to Piper’s Customer Service Information File P/N 1753-755 for a checklist to ensure obtaining latest issue.

NOTE: The 50 Hour Progressive Inspection Manual (P/N 767-006) referenced above is not a stand-alone document. It constitutes a snapshot of the Airworthiness Limitations and Inspection sections of the Instructions for Continued Airworthiness (ICA) and is current only at the time of printing. Use it as follows:

(1) Owners/operators desiring to establish a Part 91 Progressive Inspection Program (PIP) (see 91.409(d)) or a Part 135 Approved Aircraft Inspection Program (AAIP) (see 135.419) should use the appropriate Progressive Inspection Manual as a template for submission to their regional FAA office.

(2) Service centers conducting Event Cycle inspections under a FAA-approved PIP or AAIP can use the appropriate Progressive Inspection Manual as a working check-off list/form, provided they verify its currency against the FAA-approved PIP or AAIP.

C. Overlimits Inspection

If the airplane has been operated so that any of its components have exceeded their maximum operational limits, special inspections may be required by Piper and/or the component manufacturer. See 5-50-00 and applicable vendor publications.

4. Annual / 100 Hour Inspection Procedure

A. Scheduled Maintenance

See “Scheduled Maintenance” on page 05205.

(1) The required periodic inspection procedures are listed in paragraph “5. Scheduled Maintenance”. These inspection procedures are broken down into major groups which include Propeller, Engine, Turbocharger (where applicable), Cabin and Cockpit, Fuselage and Empennage, Wing, Landing Gear, Special Inspections, Operational Inspection, and General. The first column in each group lists the inspection or procedure to be performed. The second column is divided into two sub-columns indicating the required inspection interval of 50 hours or 100 hours. Each inspection or operation is required at each of the inspection intervals indicated by a circle (O). When a vendor publication specifies a time outside the 50 / 100 hour cycle, it will be listed as a special inspection in “Special Inspections” on page 05301.

(2) Refer to the applicable chapter of this manual for instructions on how to gain access to remove any item that must be removed and is not completely accessible.

(3) Inspection Report Forms.

To help in the performance of periodic inspections, an Inspection Report form:

P/N 767-012, PA-34-220T, Seneca IV, (S/N’s 3447001 thru 3447029) and Seneca V

is available. See the availability statement in Revision Status.

NOTE: Service centers conducting Part 91 Annual / 100 Hour Inspections can use the Inspection Report Form (P/N 767-012), as a working check-off list, provided they verify its currency against an up-to-date copy of the ICA (i.e. - this Maintenance Manual, see 4-00-00 and 5-20-00).

(4) In addition to inspection intervals required in scheduled maintenance (i.e. - paragraph 5), preflight inspection must also be performed.

(5) References to maintenance manual applicable areas are per the “chapter - system/sub-system” assignment of subject material numbering system.

B. Special Inspections

See 5-30-00.

C. Unscheduled Maintenance Checks

See 5-50-00.

5. Scheduled Maintenance

Refer to Notes “1”, “2”, “3”, and “4” before performing the following inspections.

NATURE OF INSPECTION	Inspection Interval (Hrs) L	R	50	100
A. PROPELLER GROUP				
WARNING: USE EXTREME CAUTION WHEN ROTATING PROPELLER BY HAND; PROPELLER MAY KICK BACK. PRIOR TO ROTATING PROPELLER, ENSURE BOTH MAGNETO SWITCHES ARE OFF (GROUNDED). IF MAGNETOS ARE NOT GROUNDED, TURNING PROPELLER MAY START ENGINE.				
1. Inspect spinner and back plate for cracks	O	O	O	O
2. Inspect blades for nicks and cracks	O	O	O	O
3. Inspect for grease and oil leaks	O	O	O	O
4. Lubricate propeller (**Hartzell only**) per Lubrication Chart, 12-20-00	O	O		O
5. Inspect spinner mounting brackets for cracks	O	O		O
6. Inspect propeller mounting bolts for condition and security. If safety is broken, re-torque and safety	O	O		O
7. Inspect hub parts for cracks and corrosion	O	O		O
8. Inspect complete propeller and spinner assembly for security, chafing, cracks, deterioration, wear, and correct installation.	O	O		O
9. Check propeller air pressure. (**Hartzell only**.)	O	O	O	O
10. Inspect propeller deicers (if installed) for nicks, tears, or delamination from propeller	O	O	O	O
11. Inspect propeller deicer slip ring and brush block (if installed) for condition and security. Replace as required	O	O	O	O
12. Inspect propeller deice wiring and junction blocks (if installed) for security, cracks, chafing and damage	O	O	O	O
B. ENGINE GROUP				
WARNING: GROUND MAGNETO PRIMARY CIRCUIT BEFORE WORKING ON ENGINE.				
NOTE: Read Notes “5” and “6” prior to beginning this group.				
1. Remove engine and nacelle cowlings. (See Figure 1, 6-40-00.)	O	O	O	O
2. Clean and inspect cowling for cracks, distortion, and loose or missing fasteners.	O	O	O	O
3. Compression check while engine is warm. (Refer to Continental Aerospace Technologies Publication M-0, “Standard Practice Maintenance Manual,” latest revision.)	O	O		O
4. Drain oil sump while engine is warm	O	O	O	O
5. Change full flow (spin-on type) oil filter element (inspect element for foreign particles)	O	O	O	O
6. Fill engine with oil per Lubrication Chart, 12-20-00	O	O	O	O
7. Inspect oil temperature sender unit for leaks and security	O	O		O
8. Inspect oil lines and fittings for leaks, security, chafing, dents, and cracks. (See Note “14”.)	O	O	O	O
9. Clean and inspect oil radiator cooling fins	O	O		O

5. Scheduled Maintenance (continued)

NATURE OF INSPECTION	Inspection Interval (Hrs) L	R	50	100
B. ENGINE GROUP (CONTINUED)				
CAUTION: DO NOT CONTAMINATE VACUUM PUMP WITH CLEANING FLUID.				
10. Clean engine	O	O	O	O
11. Inspect condition of spark plug. Clean and adjust gap as required. (See Note “7”.)	O	O		O
12. Inspect ignition harness and insulators for high tension leakage and continuity. (See Notes “7” and “10”.)	O	O	O	O
13. Inspect magneto for oil seal leakage (See Note “10”.)	O	O		O
14. Inspect magnetos to engine timing (See Note “7” and “21”.)	O	O		O
15. Remove induction air filter and tap gently to remove dirt particles. Replace as required.	O	O	O	O
16. (Seneca V only) Remove and clean injector fuel inlet strainer. Replace strainer O-rings, if damaged. (See Note “13”.)	O	O	O	O
17. (Seneca V only) Inspect fuel injector servo linkages for adjustment and security.	O	O	O	O
18. Inspect fuel nozzles, upper deck and fuel injection nozzle reference lines, hoses, manifolds and fittings for proper routing, support and signs of fuel stains. Inspect manifold valve for security of installation, proper venting and signs of fuel stains. (See Note “18”.)	O	O		O
19. Inspect induction air box valve and inspect for excessive wear or cracks, replace defective parts	O	O		O
20. Inspect fuel injector attachments for loose hardware.	O	O		O
21. Inspect the fuel injection system to ensure that stainless steel control levers are installed on mixture and throttle control shafts. Inspect each installed lever for looseness or free play on the shaft - looseness or free play of the lever on the shaft is prohibited. Verify the retaining nut is torqued 100–120 inch-pounds	O	O		O
22. (Seneca IV only) Inspect the fuel screen and plug assembly. (See Note “23”.)	O	O		O
23. Inspect intake seals for leaks and clamps for tightness.	O	O		O
24. Inspect all air inlet duct hoses (replace as required)	O	O		O
25. Inspect condition of flexible fuel lines. (See Note “14”.)	O	O		O
26. Inspect fuel system for leaks.	O	O		O
27. Inspect engine-driven and electric fuel pumps for condition and operation. Replace as required	O	O		O
28. Inspect pneumatic pumps and lines. (See Note “17”.)	O	O		O
29. If installed, replace vacuum regulator filter element	O	O		O
30. Inspect throttle, alternate air, mixture, and propeller governor controls for security, travel, and operating condition.	O	O		O
31. Inspect exhaust stacks, connections, and gaskets (replace gaskets as required)	O	O	O	O
32. Inspect breather tubes for obstructions and security.	O	O		O
33. (Seneca V only) Inspect air/oil separator for adjustment, leakage, and security	O	O	O	O
34. Inspect crankcase for cracks, leaks, and security of seam bolts	O	O		O

NATURE OF INSPECTION	Inspection Interval (Hrs) L	R	50	100
B. ENGINE GROUP (CONTINUED)				
35. Inspect engine mounts for cracks and loose mountings	O	O		O
36. Inspect rubber engine mount bushings for deterioration. Replace as required	O	O		O
37. Inspect all engine baffles	O	O		O
38. Inspect firewall seals	O	O		O
39. Inspect condition of alternator and starter	O	O		O
40. Inspect all lines, air ducts, electrical leads, (wiring) and engine attachments for general condition (chafing, cracks or cracked insulation, deterioration) security, proper routing, and correct installation	O	O	O	O
41. Inspect condition/tension of compressor and/or alternator belts	O	O		O
42. Inspect security of compressor mounting	O			O
43. Inspect compressor clutch security and condition of wiring	O			O
44. Inspect compressor & refrigerant lines for condition & security	O			O
45. Lubricate per Lubrication Chart, 12-20-00	O	O		O
46. (Seneca IV only) Inspect engine set-up per TCM SID97-3C or later. See also 76-10-00, Seneca IV, Engine Setup Procedures	O	O		O
47. (Seneca V only) Inspect engine set-up per 76-10-00, Seneca V, Engine Setup Procedures	O	O		O
C. TURBOCHARGER GROUP				
1. Inspect all air inlet ducting and compressor discharge ducting for worn spots, loose clamps, or leaks	O	O	O	O
2. Inspect engine air inlet assembly for cracks, loose clamps, and screws	O	O	O	O
3. Inspect exhaust ducting and exhaust stacks for signs of leaks or cracks. Check all clamps for tightness	O	O	O	O
4. Inspect V-Band couplings per V-Band Coupling 100 Hour Inspection, 81-20-00	O	O		O
5. Inspect turbocharger rotor for excessive play, vane damage, carbon and dirt deposits	O	O		O
6. Inspect all turbo support brackets, struts for breakage, sagging or wear	O	O	O	O
7. Inspect all oil lines and fittings for wear, leakage, heat damage or fatigue	O	O	O	O
8. Inspect bypass valve for security and safety	O	O	O	O
9. Install engine and nacelle cowlings	O	O	O	O
10. Run up engines, check all instruments for smooth, steady response	O	O	O	O

5. Scheduled Maintenance (continued)

NATURE OF INSPECTION	Inspection Interval (Hrs) 50	100
D. CABIN AND COCKPIT GROUP		
1. Inspect cabin entrance doors, cargo, and baggage doors for damage and operation. Check condition and security of locks, latches and hinges		O
2. Inspect upholstery for tears		O
3. Inspect seats and attaching brackets and hardware for condition, security, and operation		O
4. Inspect seat belts and shoulder harnesses per Restraint System, Inspection, in 25-10-00		O
5. Inspect trim operation		O
6. Inspect operation and condition of rudder pedals. (See Note “20”.)		O
7. Inspect parking brake valve and toe brakes for operation and cylinder leaks		O
8. Inspect rudder pedals and toe brakes for travel, binding, and security		O
9. Ensure no hose clamps or ty-raps are attached to the torque tubes of the rudder pedal and bar assembly. Verify free and correct movement of the rudder pedal and bar assembly.		O
10. Ensure the routing of wiring, cables, plumbing, and adjacent system components do not present a snagging hazard or interfere with the rudder controls.		O
11. Inspect control wheels, column, pulleys, bobweight, and cables for condition and security. Inspect cables per Control Cable Inspection, 27-00-00		O
12. Inspect tee bar assembly for cracks. (See Note “22”.)		O
13. Inspect condition and security of electrical leads and installations		O
14. Cycle each circuit breaker with airplane power off	O	O
15. Inspect circuit breakers for condition.	O	O
16. Check landing, navigation, cabin and instrument lights		O
17. Inspect instruments, lines and attachments		O
18. In airplanes equipped with Garmin G1000/G1000 NXi, perform 100 hour inspections listed under Inspections in 34-25-01 or 34-25-02, as appropriate		O
19. Inspect gyro operated instruments and electric turn and bank. (Overhaul or replace as required.)		O
20. Replace filters on gyro horizon and directional gyro		O
21. Inspect static system, altimeter (and Avidyne or Garmin PFD, or Aspen Avionics Evolution Backup Display (EBD), if installed) and transponder for installation/certification per latest revision of AC43.13-1 and current test/inspection per FAR’s 91.411 and 91.413, respectively.		O
22. Inspect operation of fuel selector valves		O
23. Inspect operation of fuel drains		O
24. Inspect condition of heater controls and ducts		O
25. Inspect condition and operation of air ducts		O
26. Inspect condition of air conditioning ducts		O
27. Remove and clean air conditioning evaporator filter		O
28. If installed, inspect disposable-type (non-gauged) fire extinguisher minimum weight as specified on nameplate		O

NATURE OF INSPECTION	Inspection Interval (Hrs) 50	100
E. FUSELAGE AND EMPENNAGE GROUP		
1. Remove inspection plates and panels. (See Figure 1, 6-40-00.)		O
2. Inspect battery, box and cables. Flush box as required and fill battery per instructions on box	O	O
3. Inspect cabin heater. (See Note “19”.)		O
4. Inspect the heater fuel pressure regulator and shutoff valve for signs of leakage per Fuel Regulator and Shutoff Valve, 100 Hour Inspection, 21-40-00		O
5. Conduct a general visual inspection of electrical and electronic installations (mounting, wiring, harnesses, shields, connectors, etc.) for condition and security. For Garmin G600 or G1000/G1000 NXi equipped aircraft, see respective 100 hour Inspections in 34-23-00, 34-25-01 or 34-25-02		O
6. Inspect skins, bulkheads, frames, stringers, for damage, irregularities, or structural defects (i.e., skin cracks, distortion, dents, corrosion and loose or missing rivets)		O
7. Inspect antenna mounts for electric wiring security and condition		O
8. Check hydraulic pump fluid level. (Fill as required.)	O	O
9. Inspect hydraulic pump lines for damage and leaks		O
10. Check air conditioning system for refrigerant leaks. (See Note “11”.)		O
11. Check refrigerant level in sight gauge of receiver-dehydrator. (See Note “11” and 21-50-00.)	O	O
12. Inspect fuel lines, valves and gauges for damage and operation		O
13. Inspect security of all lines		O
14. Inspect vertical fin for surface damage or irregularities (i.e. - skin cracks, distortion, dents, and corrosion); structural defects (i.e. - loose or missing rivets); misrigging; and attachment points for missing or worn hardware		O
15. Inspect vertical stabilizer forward spar attachment steel bracket for corrosion; clean and prime with Dinitrol AV8, if required		O
16. Inspect rudder and tab hinges, horn and attachments for damage and operation		O
17. Inspect ELT battery mount for condition and security		O
18. Inspect ELT antenna for condition, security, and operation (check operation per Antenna Test in 25-60-00). Replace antenna if bent or damaged.		O
19. Inspect rudder and tab for surface damage or irregularities (i.e. - skin cracks, distortion, dents, and corrosion); structural defects (i.e. - loose or missing rivets); misrigging; hinge damage, excessive wear, freedom of movement and proper lubrication; and attachment points for missing or worn hardware		O
20. Inspect rudder and tab hinge bolts and pins for excess wear and free play. Replace as required. (See Note “8”.)		O
21. Inspect rudder trim mechanism.		O
22. Inspect stabilator and tab for surface damage or irregularities (i.e. - skin cracks, distortion, dents, and corrosion); structural defects (i.e. - loose or missing rivets); misrigging; hinge damage, excessive wear, freedom of movement and proper lubrication; and attachment points for missing or worn hardware		O
23. Inspect stabilator and tab hinges, horn and attachments for damage and operation		O
24. Inspect stabilator attachments. (See Note “16”.)		O

5. Scheduled Maintenance (continued)

NATURE OF INSPECTION	Inspection Interval (Hrs) 50	100
E. FUSELAGE AND EMPENNAGE GROUP (CONTINUED)		
25. Inspect stabilator and tab hinges, bolts, bearings, and pins for excess wear. (Replace as required.)		O
26. Inspect condition and operation of stabilator trim mechanism		O
27. Inspect wing rear attach fittings for corrosion, general condition and security. (See Aft Wing Attach Fittings 100 Hour Inspection, 53-20-00.)		O
28. Inspect rudder, stabilator, and stabilator trim cable tensions per 27-00-00, Chart 2. Use a tensiometer.		O
29. Inspect aileron, rudder, stabilator, and stabilator trim cables and pulleys for routing, safety, condition, and operation. Inspect cable terminals, turnbuckles, guides, and fittings for safety and condition. Inspect per Control Cable Inspection, 27-00-00		O
30. Inspect all control cables, air ducts, electrical leads, harnesses, and attaching parts for security, routing, chafing, deterioration, wear and correct installation. (Perform 100 Hour Inspection, 51-80-00.)		O
31. Lubricate per Lubrication Chart, 12-20-00	O	O
32. Inspect strobe lights for security and operation.		O
33. Inspect autopilot system, if installed. Inspect servos, bridle cables, clamps, and autopilot controls for proper installation, function, safety, and condition. Check for damage and wear. Verify for correct bridle cable tension and unbroken torque stripes on cable clamp fasteners. See 22-10-00 for details. Cables are to be inspected for condition per Control Cables Inspection in 27-00-00.		O
34. If installed, inspect deice system (i.e. - pneumatic valves and lines, electrical switches, wiring and components) for condition and security.		O
35. Inspect static wicks per Static Wicks, Inspection, 23-60-00. (Replace as required.).		O
36. Reinstall inspection plates and panels		O
F. WING GROUP		
1. Remove inspection plates and fairings. (See Figure 1, 6-40-00.)		O
2. Inspect exterior surfaces, skins, and tips for damage and loose or missing fasteners, and condition of walk-way as applicable	O	O
3. Inspect visible interior structural components (skins, spars, ribs, stringers, etc.) for condition, security, distortion, or failure. (See Note “15”.)		O
4. Inspect wing carry-through structure for condition, security, distortion, or failure		O
5. Inspect ailerons for surface damage or irregularities (i.e. - skin cracks, distortion, dents, and corrosion); structural defects (i.e. - loose or missing rivets); misrigging; hinge damage, excessive wear, freedom of movement and proper lubrication; and attachment points for missing or worn hardware		O
6. Inspect aileron cables, pulleys, and bellcranks for safety, condition, and operation. Inspect aileron cable terminals, turnbuckles, fittings, and guides for safety and condition. Inspect per Control Cable Inspection, 27-00-00		O
7. Inspect aileron cable tension per 27-00-00, Chart 2. Use a tensiometer		O
8. Inspect flaps for surface damage or irregularities (i.e. - skin cracks, distortion, dents, and corrosion); structural defects (i.e. - loose or missing rivets); misrigging; hinge damage, excessive wear, freedom of movement and proper lubrication; and attachment points for missing or worn hardware		O

NATURE OF INSPECTION	Inspection Interval (Hrs) 50	100
F. WING GROUP (CONTINUED)		
9. Inspect condition of bolts used with hinges (replace as required)		O
10. Lubricate per Lubrication Chart, 12-20-00	O	O
11. Inspect the wing forward and aft attach fittings and bolts for security, corrosion, and condition. Take special care to gain access to the area local to each fitting to inspect fitting, rivets, and adjacent structures for evidence of corrosion.		O
12. Retorque wing aft spar attach bolts per Wing Aft Spar-to-Fuselage Attachment Hardware 100 Hour Inspection, 57-40-00		O
13. Inspect all control cables, electrical leads, air ducts, lines, and attaching parts for security, routing, chafing, deterioration, wear, and correct installation		O
14. Inspect air conditioner condenser air scoop rigging	O	O
15. If installed, inspect deice system pneumatic valves and lines for condition and security.		O
16. Inspect fuel tanks and lines for leaks and water		O
17. Remove, drain, and clean fuel filter bowls and screens	O	O
18. Flush fuel filter bowl and screens completely		O
19. Inspect fuel caps, cap gaskets, fuel gauge transmitter gaskets, gauge transmitter access covers, and upper surface inspection covers for condition, proper sealing, security, alignment, etc. Ensure to service and clean these areas, replacing parts as necessary		O
20. Inspect the interior of metal fuel tanks for signs of corrosion, which may indicate water contamination. If signs of contamination are found, alert the owner and fuel supplier of your findings for corrective action		O
21. Inspect the interior of bladder tanks for wrinkles, broken or missing hangers, etc. If signs of contamination are found, alert the owner and fuel supplier of your findings for corrective action		O
22. Fuel tanks marked for minimum octane rating		O
23. Inspect fuel tank vents. (See Note “14”.)		O
24. Reinstall inspection plates and fairings		O
25. Inspect static wicks per Static Wicks, Inspection, 23-60-00. (Replace as required.)		O
G. LANDING GEAR GROUP		
NOTE: The following inspections incorporate the requirements of Piper Service Bulletin (SB) No. 1123D. Any later revisions to SB No. 1123 must be complied with separately. See also AD 2005-13-16.		
1. Check oleo struts for proper extension and evidence of fluid leakage. (See Landing Gear, 12-10-00.)	O	O
2. Inspect wheel alignment, if required		O
3. Put airplane on jacks. (See 7-10-00.)		O
4. Inspect tires for cuts, uneven or excessive wear and slippage		O
5. Remove wheels, clean, check and repack bearings		O
6. Inspect wheels for cracks, corrosion and broken bolts		O
7. Check tire pressures. (See Tires, 12-10-00, and Chart 1, 6-00-00.)	O	O
8. Inspect brake linings and discs for condition and wear. (See 32-40-00.)		O
9. Inspect brake backing plates		O

5. Scheduled Maintenance (continued)

NATURE OF INSPECTION	Inspection Interval (Hrs) 50	100
G. LANDING GEAR GROUP (CONTINUED)		
10. Inspect brake lines and retaining clamps		O
11. Inspect condition of nose gear centering spring and bungees. (See Note “12”.)		O
12. Inspect gear forks for damage		O
13. Inspect nose and main gear actuators.		O
14. Inspect main gear trunnion pin. Replace if necessary		O
15. Inspect hydraulic lines, electrical leads, and attaching parts for condition and security (i.e. – routing, chafing, damage, wear, etc.)		O
16. Inspect drag and side brace link bolts. (Replace as required.)		O
17. Inspect gear doors and attachments for condition and security		O
18. Inspect gear warning horn and light for operation		O
19. Retract gear - check operation. Observe nose gear tiller roller as it moves from/to the steering arm channel to the tiller track and up and down the tiller track		O
20. Retract gear – inspect doors for clearance and operation		O
21. Inspect operation of squat switch		O
22. Inspect down lock switches, up switches, and electrical leads for security		O
23. Inspect down locks for operation and adjustment		O
NOTE: In the following inspections, refer to Nose Landing Gear; Cleaning, Inspection, and Repair; 32-20-00.		
24. Inspect nose gear steering control and travel. See Nose Landing Gear, Alignment, 32-20-00		O
25. Inspect nose and main gear struts, attachments, torque links, retraction links, bolts, and bushings for condition and security		O
26. Inspect nose gear upper drag link AN7-35 attach bolt or the alternate NAS6207-50D bolt per Nose Landing Gear, 100 Hour Upper Drag Link Attach Bolt Inspection, 32-20-00		O
27. Inspect the nose gear retraction link retention spring (P/N 96178-0) for damage, distortion, or corrosion		O
28. Remove triangular shaped, nose gear strut servicing access panel located in the forward baggage compartment.		
(a) Inspect nose tiller roller, steering arm channel and tiller track for condition		O
(b) Examine the tiller, tiller roller, and steering arm channel, turn-stop bosses for damage caused by exceeding nose wheel turn limits when towing with power equipment		O
(c) Inspect the AN4-10A bolts attaching the P/N 95393-00 arm to the steering channel for proper torque (50–70 in. lbs). If found loose, replace bolts and re-torque		O
29. Inspect the nose gear drag link center pivot and attachment bolts for condition and security. (Replace as required.)		O
30. Inspect the nose gear down lock link assembly for binding, worn spring retention pin, and any noticeable elongation of the hole associated with the spring retention pin. Inspect the down lock spring for damage, distortion, or corrosion. Clean and lubricate using MIL-PRF-7870C oil.		O
31. Inspect the actuator mounting bracket per Nose Landing Gear, 100 Hour Actuator Mounting Bracket Inspection, 32-20-00		O

NATURE OF INSPECTION	Inspection Interval (Hrs) 50	100
G. LANDING GEAR GROUP (CONTINUED)		
32. Inspect the bolt and bushing associated with the attachment of the P/N 95712-004 retraction link assembly to the actuator mounting bracket. Replace if “wear grooves” are noted in either the bolt or bushing		O
33. Inspect the AN23-25 stop bolt in the actuator mounting bracket for condition and security		O
34. Lubricate per Lubrication Chart, 12-20-00	O	O
35. Verify proper adjustment of nose gear down lock link by rigging per Nose Landing Gear, Installation and Rigging, 32-20-00		O
36. Inspect the Tunnel Bracket (P/N 95554-000) installation per Nose Landing Gear, 100 Hour Tunnel Bracket Installation Inspection, 32-20-00		O
WARNING: DO NOT REMOVE JACKS UNTIL IT HAS BEEN DETERMINED THAT THE LANDING GEAR IS DOWN AND LOCKED.		
37. Remove airplane from jacks		O
H. SPECIAL INSPECTION		
Review inspections in 5-30-00. Perform all special inspections applicable to your aircraft and currently due per the given inspection interval.		
I. OPERATIONAL INSPECTION		
NOTE: Refer to Note “9” prior to making operational checks.		
1. Check fuel pump and fuel tank selector	O	O
2. Check fuel quantity, pressure and flow readings	O	O
3. Check oil pressure and temperature	O	O
4. Check alternator output	O	O
5. Check manifold pressure	O	O
6. Check alternate air	O	O
7. Check parking brake	O	O
8. Check vacuum gauge (See Gyro Filter, 37-10-00)	O	O
9. Check gyros for noise and roughness	O	O
10. Check cabin heater operation	O	O
11. Check magneto switch operation	O	O
12. Check magneto RPM variation	O	O
13. Check throttle and mixture operation	O	O
14. Check propeller smoothness	O	O
15. Check propeller governor action	O	O
16. Check engine idle	O	O
17. Check electronic equipment operation. (See 25-60-00 for ELT check.)	O	O
18. Check air conditioner compressor clutch operation	O	O
19. Check air conditioner compressor scoop operation	O	O
20. Check operation of flaps and flap indicator	O	O
21. Check operation of autopilot, including pitch trim and manual electric trim.	O	O
22. Inspect free and correct operation of flight controls	O	O

5. Scheduled Maintenance (continued)

NATURE OF INSPECTION	Inspection Interval (Hrs) 50	100
J. GENERAL		
1. Aircraft conforms to FAA Specifications	O	O
2. Latest revision of applicable FAA Airworthiness Directives complied with	O	O
3. Current and correct Pilot's Operating Handbook (POH) is in the airplane	O	O
4. Check airplane for required placards as specified in Section 2 of the POH		O
5. Appropriate entries made in the Aircraft and Engine Log books	O	O
6. Airworthiness & Registration Certificates in the aircraft and properly displayed	O	O
7. Aircraft Equipment List, Weight and Balance and FAA Form(s) 337 (if applicable) are in the aircraft and in proper order	O	O
8. Operational inspection and run-up completed	O	O
9. Aircraft cleaned and lubricated after wash (as required)	O	O

K. NOTES

1. Refer to Piper's Customer Service Information File (see Availability in Revision Status) for latest revision dates to Piper Inspection Reports/Manuals and this maintenance manual. References to Chapter/Section are to the appropriate Chapter/Section in this manual.

   WARNING: INSTRUCTIONS FOR CONTINUED AIRWORTHINESS (ICA) FOR ALL NON-PIPER APPROVED STC INSTALLATIONS ARE NOT INCLUDED IN THIS MANUAL. WHEN A NON-PIPER APPROVED STC INSTALLATION IS INCORPORATED ON THE AIRPLANE, THOSE PORTIONS OF THE AIRPLANE AFFECTED BY THE INSTALLATION MUST BE INSPECTED IN ACCORDANCE WITH THE ICA PUBLISHED BY THE OWNER OF THE STC. SINCE NON-PIPER APPROVED STC INSTALLATIONS MAY CHANGE SYSTEMS INTERFACE, OPERATING CHARACTERISTICS AND COMPONENT LOADS OR STRESSES ON ADJACENT STRUCTURES, THE PIPER PROVIDED ICA MAY NOT BE VALID FOR AIRPLANES SO MODIFIED.

2. Inspections or operations are to be performed as indicated by a "O" at the 50 or 100 hour inspection interval. Inspections or operations (i.e., component overhauls/replacements, etc.) required outside the 100 hour cycle are listed as special inspections in section 5-30-00. Inspections must be accomplished by persons authorized by the FAA or appropriate National Aviation Authority. Checks may be performed by a pilot or owner who is checked out on the airplane.
   (a) The 50 hour inspection accomplishes preventive maintenance, lubrication and servicing as well as inspecting critical components.
   (b) The 100 hour inspection is a complete inspection of the airplane, identical to an annual inspection.

   NOTE: A logbook entry should be made upon completion of any inspections.

3. Piper Service Bulletins are of special importance and Piper considers compliance mandatory. In all cases, see Service Bulletin/Service Letter Index (see Availability, in Revision Status) to verify latest revision. See also "Chart 1" on page 052018.
4. Piper Service Letters are product improvements and service hints pertaining to servicing the airplane and should be given careful attention.
5. Inspections given for the power plant are based on the engine manufacturer's operator's/maintenance manual (Seneca IV – X30553 or Seneca V – X30645A) for this airplane. Any changes issued to the engine manufacturer's operator's manual shall supersede or supplement the inspections outlined in this report.
6. Refer to the latest revision of Continental Aerospace Technologies Service Documents MSB94-8 and MSB96-10. Also, in the latest revision of Publication M-0, "Standard Practice Maintenance Manual," refer to Chapter 6 (supercedes SB-96-11 and SIL-98-9) and Appendix C (supercedes SB97-6).
7. For high altitude operations (12,000 feet and up), more frequent ignition system maintenance is required. In Seneca V airplanes equipped with Slick Magnetos, see Champion Aerospace/Slick SB1-88B.
8. Refer to 27-20-00, for allowable rudder tab and trim free play.
9. Refer to Pilot's Operating Handbook for preflight and flight check, and for intended function in all modes.

5. Scheduled Maintenance (continued)

K. NOTES (CONTINUED)

10. Bendix only: Refer to latest revision of Bendix Service Bulletin 612 for inspection of Bendix magnetos and ignition harness.

    CAUTION: ENVIRONMENTAL REGULATIONS MAY REQUIRE SPECIAL EQUIPMENT AND PROCEDURES BE UTILIZED WHEN CHARGING AIR CONDITIONING SYSTEM.

11. Compressor oil level need not be checked unless a refrigerant leak has developed and recharging is required.
12. Ensure that the nose gear centering spring is secure and correctly installed, with the bolt head down, and washer installed properly as shown in 32-20-00.
13. Remove and clean the injector fuel inlet strainer at the first 25 hour inspection and each 50 hour inspection thereafter. Damaged strainer O-rings should be replaced.
14. Flexible hose replacement times are in-service times. In-service must be determined by (1) the date the aircraft was licensed, if new or (2) the date entered in the logbook for the replacement hose placed in service. Do not use the date stamped on the hose, as time may be included for shelf life, and not in-service use.
15. In S/N's 3447001 thru 3447029 and 3449001 thru 3449508, for airplanes which have not installed Kits No. 767-397 (LH) and 767-398 (RH), (with kit instructions marked as drawing number (Dwg. No.) 88373 revision "J" (or later)) each 100 hours time-in-service: conduct 100 Hour Wing Rib Inspection in 57-10-00.
16. Perform Attach Brackets Corrosion Control Inspection, 55-20-00.
17. For airplanes equipped with Parker Hannifin / Airborne vacuum pump(s), verify compliance with Parker Hannifin / Airborne Service Letter No. 72.
18. At the first 100-hour inspection on new, rebuilt or overhauled engines, remove fuel injection nozzles. Clean nozzles by soaking in lacquer thinner, acetone or methyl ethyl ketone (MEK).
19. Heater Inspections:
    (a) In airplanes equipped with a Model B-3040, P/N 91E88-1 or a Model B-3500, P/N 91E88-1(EL) Heater: for heaters with 500 heater operating hours or twenty-four (24) months time-in-service since new, or overhauled with a new combustion tube assembly, each 100 heater operation hours or twenty-four (24) months, whichever comes first; conduct the 100 Hour Inspection under Heater in 21-40-00.
    (b) In airplanes equipped with a Model I-3500, P/N 91E88-1(IS) Heater, inspect as required by the appropriate Hartzel Engine Technologies Owner's Manual.
20. In airplanes with a rudder pedal and bar assembly which has accumulated 1,200 or more hours time-in-service, each 100 hours: conduct Inspection, under Rudder Pedal Assembly, in 27-20-00.
21. Use of magneto manufacturer's service publication is recommended:
    (a) For airplanes equipped with Slick Magnetos: inspect magneto(s) per the appropriate 100 Hour Inspection in the Slick F1100 Master Service Manual.
    (b) For airplanes equipped with TCM/Bendix Magnetos: inspect magneto(s) per the procedures in the Periodic Maintenance section of the applicable Continental/Bendix Service Support Manual.

K. NOTES (CONTINUED)

22. In airplanes with a tee bar assembly which has accumulated 3,500 or more hours time in service, each 100 hours, complete the Control Column Tee Bar Assembly Inspection in 27-10-00.
23. For Seneca IVs only, refer to the latest revision of Continental Aerospace Technologies Service Bulletin SB20-01 per Piper Vendor Service Publication No. 271.

6. Service Publications List

"Chart 1" is a cumulative list of Piper service publications (i.e., Service Bulletins, Service Letters, etc.) applicable to the airplane model covered by this manual, with the following exceptions:

A. The following service publications have been incorporated into this manual and are not listed below: Service Bulletins 990, 1048, 1123D, 1161D, 1242A, 1244C, 1245A, 1280A, 1355B, and 1366; Service Letters 1041A, 1052, 1069, 1074, 1086, 1087, 1095C, 1108, 1116A, 1135, 1165, 1197, 1210, 1265, 1274, 1280, and 1337.

B. Nor are service publications which have become obsolete.

Kits are listed when installation of that single kit indicates compliance with the associated service publication. Kits listed may be no longer available or may have been replaced.

Effectivity is listed by airplane model and year. See the individual service publication for specific serial number applicability.

CHART 1 (Sheet 1 of 4)
SERVICE PUBLICATIONS LIST

Airplane Model	Model Year	Pub No.	Kit No.	Subject
Seneca IV	1995	SB 988		Pneumatic Deice Switch Relocation
		SB 989		Stabilator Trim Screw Assembly Attachment Hardware
		SB 993		KLN-90B GPS Wiring Modification
		SB 996		Missing Rivet in Upper Door Frame, Forward and Aft
	1995-1996	SB 1014		Replacement of Battery Power Fuse
		SB 1017		Alternator Regulator Wiring Modification
		SB 1020		Inspection of Artex 110-4 ELT and Attachment Bracket
		SB 1022		Inspection of Induction Air Filter; Purolator Service Bulletin SB090298.01
		SB 1023		Inspection of Landing Gear Cylinder; Parker Hannifin Service Bulletin SB7063
		SB 1026		Parker Hannifin Airborne Service Letter No. 48
		SB 1041		Airborne Service Letter No. 56
		SB 1100A		Stabilator Trim Link Assembly
		SB 1113		Nose Gear Down Lock Pin Inspection
		SB 1127B		Combustion Heater Fuel Pump Leakage
		SB 1130	767-369	Vertical Fin Contour / Rivet Check
		SB 1146		Vertical Fin Aft Spar Reinforcing Angle Low E/D Inspection
		SB 1197E		Control Wheel Shaft Inspection
		SB 1285		Propeller Deice Timer Inspection
		SB 1297		Altimeter Inspection and Replacement; United Instruments SB No. 13
		SB 1368		Heater Fuel Pump Leak Inspection and Replacement
		SL 1156		Propeller Feather Stop Product Improvement
		SL 1093B		Electrical Contactor Replacement Kit
		SL 1218A		Crew Seat Control Cable
		VSP 137A		Cleveland SB 7073A Inspect/Replace Gear Actuator
		VSP 149		Kelly Aerospace SB A-107A
		VSP 155		Cleveland SB 7076 Inspect/Replace Gear Actuator
		VSP 172		Avidyne SA 05-001 PFD Crosscheck Attitude
		VSP 175		S-TEC SB 06-001 Pitch Servos 0107 and 0108

CHART 1 (Sheet 2 of 4)
SERVICE PUBLICATIONS LIST

Airplane Model	Model Year	Pub No.	Kit No.	Subject
Seneca IV (cont.)	1995-1996	VSP 191		Avidyne SA 08-001 Primary Flight Display
		VSP 182		Avidyne SB 601-00006-067 PFD Crosscheck Attitude
		VSP 183		Avidyne SB 601-00006-075 PFD & Garmin with WAAS
		VSP 207		Hartzell Service Bulletin No. 040 A
		VSP 267A		Continental CSB 19-01-A, “Fuel (Primer) Diverter Valve Assembly”
		VSP 268		Continental SB19-02 (Champion Aerospace Slick Service Bulletin SB1-19)
		VSP 271		Continental SB20-01, “Fuel Screen Assembly, Scheduled Maintenance”
		VSP 273		Continental SB20-03, “Engine Preservation during COVID-19 Pandemic”
		VSP 274		Parker Hannifin, PRM110-A
Seneca V	1997	SB 1005		Addition of Drain Holes to Induction Air Inlet Scoops
		SB 1010		Replacement of Fuel Selector Levers
		SB 1014		Replacement of Battery Power Fuse
		SB 1017		Alternator Regulator Wiring Modification
	1997-1998	SB 1013		Replacement of Crankcase Breather System
		SB 1020		Inspection of Artex 110-4 ELT and Attachment Bracket
		SB 1022		Inspection of Induction Air Filter; Purolator Service Bulletin, SB090298.01
		SB 1023		Inspection of Landing Gear Cylinder; Parker Hannifin Service Bulletin SB7063
		SB 1031		Horizon Engine Instruments Repair
		SL 1020		Relocation of Signal Ground for Cylinder Head and Oil Temp. Sensors
		SL 1021		TCM Service Bulletin SB 98-8, Number One and Three Cylinder Intake Tube Modification
	1997-1999	SB 1038		Aft Spinner Bulkhead (Superseded by VSP-145)
		SB 1039		Manifold Pressure Gauge (MAP) Wiring Modification
		SL 1025		Replacement of Cylinder Head Temperature Wires
	1997-2000	SB 1041		Airborne Service Letter Number 56
		SB 1042		Heated Deice Panel Inspection
		SB 1043		Deice Control Valve Replacement
		SL 1031		Fuel Pump Cooling Inlet Installation
	1997-2001	SB 1081A		Horizon Aerospace Analog Instrument Pointer Calibration
		SB 1100A		Stabilator Trim Link Assembly
		SL 1036		TCM Service Information Letter SIL 01-2
	1997-2002	SB 1113		Nose Gear Down Lock Pin Inspection
		VSP 145		McCauley SL 2002-24 Inspection of Bulkhead for Cracks
	1997-2003	SB 1130	767-369	Vertical Fin Contour / Rivet Check
		SB 1146		Vertical Fin Aft Spar Reinforcing Angle Low E/D Insp.
		SL 1048A		DDMP High Altitude Erroneous Information
		SL 1078		Improved Flap Potentiometer
		VSP 137A		Cleveland SB 7073A Inspect/Replace Gear Actuator
		VSP 155		Cleveland SB 7076 Inspect/Replace Gear Actuator

CHART 1 (Sheet 3 of 4)
SERVICE PUBLICATIONS LIST

Airplane Model	Model Year	Pub No.	Kit No.	Subject
Seneca V (cont.)	1997-2004	SB 1127B		Combustion Heater Fuel Pump Leakage
		SB 1157A		Field Upgrade of Pressurized Magnetos & Tachometer Sensor Assemblies
	1997-2008	SB 1197E		Control Wheel Shaft Inspection
	1997-2010	SB 1225		Inaudible TIS or TAS Aural Alerts
	1997-2012	SL 1154		Engine Starter Motor Replacement
		SL 1156		Propeller Feather Stop Product Improvement
	1997-2013	SB 1254		Stabilator Drain Hole Inspection
	1997-2015	SB 1285		Propeller Deice Timer Inspection
		SL 1218A		Crew Seat Control Cable
	1997 and Up	SB 1297		Altimeter Inspection and Replacement; United Instruments SB No. 13
		SB 1368		Heater Fuel Pump Leak Inspection and Replacement
		SL 1164	88537-001	Engine Front Baffle Replacement
		SL 1093B		Electrical Contactor Replacement Kit
		VSP 172		Avidyne SA 05-001 PFD Crosscheck Attitude
		VSP 175		S-TEC 06-001 Pitch Servos 0107 and 0108
		VSP 182		Avidyne SB 601-00006-067 PFD Crosscheck Attitude
		VSP 183		Avidyne SB 601-00006-075 PFD & Garmin with WAAS
		VSP 191		Avidyne SA 08-001 Primary Flight Display
		VSP 207		Hartzell Service Bulletin No. 040A
		VSP 268		Continental SB19-02 (Champion Aerospace Slick Service Bulletin SB1-19)
		VSP 273		Continental SB20-03, “Engine Preservation during COVID-19 Pandemic”
		VSP 274		Parker Hannifin, PRM110-A
	1998-2004	SB 1139A		Control Wheel Attachment Inspection and Retainer Clip Installation
	2000-2001	SB 1083C		S-TEC System 55X Autopilot
	2000-2002	SB 1135		GMA-340 Audio Panel Wiring
	2001-2002	SB 1097A		Pneumatic Deicer Deflate Valve Assembly Replacement
		SB 1187	88432-002	Voltage Suppressor Replacement
	2003	SB 1137		Stabilator Balance Weight Tube Assy.
		SB 1141		Clip Nut Replacement
	2004	SB 1151E		4130N Steel Hardness Discrepancy
	2004 and Up	SL 1089		Aircraft Painting Requirement
	2005-2008	SL 1126A		Voltage Suppressor Advisory
	2005-2010	VSP 210A		Avidyne SB 601-00004-106, Rev. 01
	2005 and Up	SL 1123		Avidyne EXP5000 Software Upgrade (MOD 57)
	2006-2007	SB 1184		Engine Mount / Oil Line Fitting Interference / Contact (TCM CSB07-2)
	2006-2011	SB 1241		Wastegate Clearance Inspection
	2009 and Up	VSP 219		Garmin Service Alert 1047
	2010	SB 1223		Wire Screen Replacement
	2010-2012	SB 1247B		Garmin SafeTaxi Limitation
	2010-2013	SB 1252		Garmin G600 Software Upgrade Ver. 942.40

CHART 1 (Sheet 4 of 4)
SERVICE PUBLICATIONS LIST

Airplane Model	Model Year	Pub No.	Kit No.	Subject
Seneca V (cont.)	2010-2020	VSP 278		Garmin SA 2111-C, “3rd Party Wireless SD Cards”
	2012-2016	SB 1303A		Diode Module Installation
	2012-2017	SB 1337		Emergency Power Supply Modification
	2012-2020	SB 1402		Airworthiness Directive 2021-08-15
		SL 1284A		Aspen Avionics EFD1000 Max Software Update Version 2.10.2
		VSP 288		Garmin SB 22117-B, “Cable Retention Pins”
	2013	SB 1270A		Re-enablement of GDL-69, GSR 56, and/or GTS 825 for EASA Importation
	2014-2015	SB 1274	68273-002	Control Wheel Bushing

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SPECIAL INSPECTIONS

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

The following inspections are in addition to those listed in 5-20-00. These inspections are performed at intervals of:

- ✈ Flight hours;
- ✈ Heater Operation hours;
- ✈ Calendar Year; or
- ✈ the specific operation being conducted or the environment being operated in.

Unless otherwise indicated, these inspections are to be repeated at each occurance of the specified interval. Note that the items listed herein are guidelines based on past operating experience. Each operator should closely monitor his own unique operating conditions/environment and react accordingly to keep his aircraft airworthy.

NOTE: A log book entry should be made upon completion of any inspections.

1. Per Flight Hour

A. Each 200 Hours

[ ] For airplanes with wing flap(s) which have accumulated ten (10) years time-in-service: each 200 hours, remove the flap(s) and conduct the following special inspection: Inspect the interior of the wing flap for evidence of dissimilar metal corrosion where aluminum sheet metal is in contact with steel flap brackets. Use a bore scope or other suitable tool. Installation of a new wing flap will relieve this inspection requirement until such time as the replacement wing flap reaches ten (10) years time-in-service.

B. Each 300 Hours

WARNING: DO NOT USE GREASE OR ANY TYPE OF GREASE FITTING ON ANY OXYGEN SYSTEM. WHEN WORKING WITH AN OXYGEN SYSTEM MAKE SURE HANDS, CLOTHING, TOOLS, AND IMMEDIATE AREA ARE FREE OF GREASE.

[ ] (1) If installed, each 300 hours time-in-service, inspect the oxygen system regulator, pressure gauge, high and low pressure lines, and outlets per Inspections, 35-10-00.

[ ] (2) For airplanes equipped with Aero Accessories, Inc. Tempest Dry Air Pumps with Wear Indicator Ports; beginning at 300 hours time-in-service, and each 100 hours thereafter, inspect per Vacuum Pump(s), Inspection, 37-10-00.

NOTE: For airplanes with frequent deice cycles, begin this inspection at 200 hours time-in-service.

[ ] (3) Clean fuel nozzles every 300 hours or at the annual inspection. Clean nozzles by soaking in lacquer thinner, acetone or methyl ethylketone (MEK).

C. Each 500 Hours

[ ] (1) For airplanes equipped with Slick Magnetos: inspect and clean magneto(s) per 500 Hour Inspection in the Slick F1100 Master Service Manual. See also Continental Motors SB19-02, per Piper Vendor Service Publication No. 268.

[ ] (2) For airplanes equipped with TCM/Bendix Magnetos: inspect and clean magneto(s) per the procedures in the Periodic Maintenance section of the applicable Service Support Manual.

[ ] (3) Replace induction air filter each 500 hours time-in-service, annually, or after 10 cleanings, whichever comes first.

[ ] (4) Remove propellers; remove sludge from crankshaft. Clean any residual sludge clinging to the exterior of propeller / crankshaft mating surfaces.

[ ] (5) Clean and lubricate stabilator trim drum screw.

[ ] (6) Replace the drag link bolt AN7-35 or NAS6207-50D every 500 hours.

[ ] (7) In S/N's 3447001–3447029 and 3449001–3449508, for airplanes which have not installed Kits No. 767-397 (LH) and 767-398 (RH), (with kit instructions marked as drawing number (Dwg.No.) 88373 revision "J" (or later)), each 500 hours time-in-service: conduct 500 Hour Wing Rib Inspection, 57-10-00.

[ ] (8) If installed, replace the vacuum system inlet air filter (i.e., central air filter, gyro filter, etc.) element each 500 hours time-in-service, annually, or at pump replacement, whichever comes first.

[ ] (9) Lubricate per 500 hour requirements in Lubrication Charts, 12-20-00.

[ ] (10) If installed inspect the Garmin G1000/G1000 NXi system per the 500 hour inspection in 34-25-01 or 34-25-02, as appropriate.

D. Each 600 Hours

[ ] For airplanes equipped with Aero Accessories, Inc. vacuum pump(s) model AA441CC / AA442CW series without Wear Indicator Ports, replace each pump as it accumulates 600 hours time-in-service.

E. Each 1000 Hours

[ ] (1) Replace engine compartment flexible fuel and oil hoses as required; but not to exceed 1000 hours time-in-service, eight (8) years, or engine overhaul, whichever comes first; except for TSO-C53a - Type D hoses which are replaced on-condition.

[ ] (2) Remove all turbocharger components from the engine. Inspect and repair or replace as necessary. Inspect turbocharger rotor for excessive play, carbon and dirt deposits. Remove and inspect turbine and compressor housings. Inspect turbine wheel and impeller for physical damage and excessive build up of deposits. If excessive, replace turbocharger assembly.

[ ] (3) Beginning at 2000 hours and each 1000 hours thereafter, inspect the nose gear trunnion (P/N 95723-005/-006) for cracks per 1000 Hour Nose Gear Trunnion Inspection, 32-20-00.

[ ] (4) In flight, with the landing gear retracted, check that gear warning horn sounds with either throttle retarded to 14 ± 2 in.Hg.

[ ] (5) For airplanes equipped with Garmin G600 Flight Display System: each 1000 hours conduct the G600 Equipment Electrical Bonding Check in 51-80-00.

E. Each 1000 Hours (cont.)

[ ] (6) If installed, inspect the Garmin G1000/G1000 NXi system per the 1000 hour inspection in 34-25-01 or 34-25-02, as appropriate.

[ ] (7) If installed, the first 1,000 hours or three (3) years, whichever comes first, clean and apply Mobilgrease 33 grease to output gear of the GSA 8X Servo Actuator. (See 22-10-00.)

F. Each 1800 Hours

[ ] Overhaul each engine each 1,800 hours or twelve (12) years time-in-service, whichever comes first, or as specified in Continental Aerospace Technologies Publication M-0, “Standard Practice Maintenance Manual,” latest revision.

G. Each 2000 Hours

[ ] (1) Each 2000 hours or seven (7) years, whichever occurs first, remove interior cabinets, panels, and headliner and conduct detailed inspection of aircraft structure (skin, bulkheads, stringers, etc.) for condition and security. Inspection of structure concealed by headliner may be accomplished by alternate means (i.e. – through the use of a bore scope) without removing the headliner, providing access is obtained to all concealed areas and bore scope provides sufficient detail to adequately accomplish the inspection. At F.S. 73, examine the two steel Upper Forward Cockpit Fittings (one each side), P/N’s 79553-000/-001 or 62522-000/68257-000, as applicable, for signs of corrosion.

[ ] (2) At each 2000 hours, twelve (12) years, or engine overhaul whichever comes first replace alternators.

[ ] (3) Overhaul or replace McCauley propellers each six (6) years or 2000 hours, whichever comes first. See McCauley MPC26, Owner/Operator Information Manual, latest revision.

[ ] (4) Overhaul or replace Hartzell propellers each five (5) or six (6) years or each 2000 or 2400 hours. (Refer to latest revision of Hartzell Service Letter No. 61 to determine specific requirements for individual airplanes.)

[ ] (5) Each 2000 operating hours or seven (7) calendar years time-in-service, whichever occurs first, inspect the aft wing attach fittings and adjacent structure for corrosion for signs of corrosion as specified in 57-40-00, Wing Aft Attach Fitting Inspection.

[ ] (6) In airplanes that have accumulated fifteen (15) calendar years time-in-service or more, each two thousand (2,000) hours or seven (7) calendar years time-in-service, whichever occurs first, perform the Stabilator Control System Special Inspection in 27-00-00.

H. Each 2200 Hours

[ ] For airplanes equipped with the Evolution Backup Display (EBD): replace the internal back-up battery in the EBD every 3 years or 2200 hours, or if it fails the operational test.

I. Each 2400 Hours

[ ] (1) Overhaul or replace Hartzell propellers each five (5) or six (6) years or each 2000 or 2400 hours. (Refer to latest revision of Hartzell Service Letter No. 61 to determine specific requirements for individual airplanes.)

[ ] (2) Overhaul or replace Hartzell propeller governors each 2400 hours or at engine overhaul. (Verify TBO in latest revision of Hartzell Service Letter No. 61.)

2. Per Heater Operation Hour

A. Each 100 Hours

[ ] Inspect Janitrol combustion heater each 24 months or 100 heater operating hours, whichever comes first, per AD 2004-21-05 and latest revision of Aircraft Heater Overhaul and Maintenenance Manual.

B. Each 200 Hours

[ ] In S/N's 3449414 and up (and in other Seneca Vs and all Seneca IVs with service replacement Heater Fuel Pump Kit P/N 461-771 or Heater Fuel Pump P/N 461-773 installed): Replace heater fuel pump filter (P/N 601-815) each 200 hours heater operation. See Figure 12, 21-40-00.

C. Each 1500 Hours

[ ] In airplanes equipped with a Model B-3040, P/N 91E88-1 or a Model B-3500, P/N 91E88-1(EL) Heater: overhaul combustion heater each 1,500 heater operation hours or whenever a Pressure Decay Test is failed. See latest revision of AD 2004-21-05 and the appropriate Hartzel Engine Technologies Aircraft Heater Overhaul and Maintenance Manual.

D. Each 2000 Hours

[ ] In airplanes equipped with a Model I-3500, P/N 91E88-1(IS) Heater: overhaul combustion heater each 2,000 heater operation hours or whenever a Pressure Decay Test is failed. See appropriate Hartzel Engine Technologies Overhaul Manual.

3. Per Calendar Year

A. Each Thirty (30) Days

NOTE: Pilots/owners may perform and sign-off the following checks.

[ ] (1) For airplanes equipped with Hartzell propellers, check propeller air pressure.

[ ] (2) If installed, check portable fire extinguisher for condition and charge. Verify nozzle is not obstructed and safety seal is intact. Determine charge by "hefting" extinguisher.

[ ] (3) If installed, the pilot should check gauged rechargeable fire extinguisher for condition and charge. Check to see that the extinguisher is undamaged; that the nozzle is unobstructed; that the gauge pressure is in the operable (green) range; that the lockpin and tamper seal is in place, and that the operating instructions are clearly visible.

B. Each Ninety (90) Days

[ ] (1) For airplanes equipped with Avidyne Entegra IFDS or Garmin G600 Flight Display System: if the Standby Attitude Indicator has not been operated in the previous 90 days, charge the battery of the Standby Attitude Indicator. See Standby Attitude Indicator, 34-20-00.

NOTE: Pilots/owners may perform and sign-off the following checks.

[ ] (2) Remove, drain, and clean fuel strainer bowl and screen located in bottom of fuel selector valve.

[ ] (3) Inspect battery, box or shelf, and cables at least every 90 days. Flush box as required and fill battery per instructions on box. (Not required if maintenance-free battery installed.)

B. Each Ninety (90) Days (cont.)

[ ] (4) Operationally test ELT per the Test procedure in 25-60-00 for the appropriate installed ELT at least every 90 days.

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[ ] (5) If equipped, at least every 90 days, visually check oxygen cylinder installation for leakage, corrosion, bulges, gouges, distortion, and security of mounting.

C. Each Four (4) Months

[ ] Change the engine oil and full-flow cartidge type oil filter each four (4) months or every 50 hours time-in-service, whichever comes first.

D. Each Six (6) Months

[ ] For Hartzell propellers only: If annual usage is significantly less than 100 Hours, lubricate Hartzell propellers, each six (6) months. See Hartzell Standard Practices Manual No. 202A.

E. Each Twelve (12) Months

[ ] (1) For Hartzell propellers only: Lubricate propellers every 100 Hours or annually, whichever comes first.

[ ] (2) For airplanes equipped with Avidyne Entegra IFDS or Garmin G600 Flight Display System: each twelve months, perform a full capacity test of the Standby Attitude Indicator battery. See Standby Attitude Indicator, 34-20-00.

[ ] (3) If installed, replace the vacuum system inlet air filter (i.e., central air filter, hyro filter, etc.) element each 500 hours time-in-service, annually, or at vacuum pump replacement, whichever comes first.

[ ] (4) For airplanes equipped with Airborne 1H5 series check valve manifolds and/or 1H37 series check valves: beginning at five years from date of check valve/manifold manufacture and each 12 months thereafter, inspect per the latest revision of Airborne SL 39A.

[ ] (5) For airplanes equipped with Garmin G600 Flight Display or G1000/G1000 NXi Integrated Avionics System: each twelve (12) months conduct the Annual Inspection as specified in 34-23-00, 34-25-01 or 34-25-02.

[ ] (6) If equipped, visually inspect the GFC 700 system per the twelve month inspection under Garmin GFC 700 AFCS, Components, Garmin Servos - General, Inspection in 22-10-00.

[ ] (7) For airplanes equipped with Aspen Avionics EBD: every 12 months inspect per the annual inspection in 34-24-00 under Aspen Avionics EBD, Inspections.

[ ] (8) Replace induction air filter each 500 hours time-in-service, annually, or after 10 cleanings, whichever comes first.

[ ] (9) Clean fuel nozzles every 300 hours or at the annual inspection. Clean nozzles by soaking in lacquer thinner, acetone or methyl ethylketone (MEK).

[ ] (10) If a rechargeable (gauged) fire extinguisher is installed perform annual maintenance per “Gauged, Class 2B:C Fire Extinguisher Model No. A344T” in 26-20-00.

F. Each Two (2) Years

[ ] (1) Swing the magnetic compass and, if equipped, recalibrate the magnetometer/flux detector every two (2) years. See Magnetic Heading Systems, 34-20-00 and for airplanes equipped with Garmin G1000/G1000 NXi: see Magnetometer Calibration under LRU Test Procedures in 34-25-01 or 34-25-02.

[ ] (2) Test and inspect the transponder. Ensure compliance with the requirements of FAR 43, Appendix F. (See FAR 91.413.)

[ ] (3) Test and inspect the static pressure system and altimeters (including ADAHRS (Meggitt), AHRS (in Avidyne PFD) or GDC XX (Garmin) or EBD, if installed). Ensure compliance with the requirements of FAR 43, Appendix E. (See FAR 91.411.)

NOTE: For airplanes equipped with Aspen Avionics Evolution Backup Display (EBD) System, See 34-24-00, EBD, Inspections.

NOTE: For airplanes equipped with Garmin G600 Flight Display, see Tests under Post-Installation Procedures in 34-23-00.

NOTE: For airplanes equipped with Garmin G1000/G1000 NXi Integrated Avionics System, see GDC XX Tests in 34-25-01 or 34-25-02 before beginning the above test and inspection.

[ ] (4) Inspect Janitrol combustion heater each 24 months or 100 heater operating hours, whichever comes first, per AD 2004-21-05 (if applicable) and latest revision of Aircraft Heater Overhaul and Maintenenance Manual.

G. Each Three (3) Years

[ ] (1) For airplanes equipped with Avidyne Entegra IFDS or Garmin G600 Flight Display System: replace the Standby Attitude Indicator battery as required, but at least every three (3) years. See Standby Attitude Indicator, 34-20-00.

[ ] (2) For airplanes equipped with the Evolution Backup Display (EBD): replace the internal back-up battery in the EBD every 3 years or 2200 hours, or if it fails the operational test.

[ ] (3) If installed the first 3 years or 1,000 hours, whichever comes first, clean and apply Mobilgrease 33 grease to output gear of the GSA 8X Servo Actuator. (See 22-10-00).

H. Each Four (4) Years

[ ] For airplanes equipped with TCM/Bendix Magnetos: overhaul or replace TCM/Bendix magnetos at engine overhaul, or each four (4) years time-in-service, whichever comes first.

I. Each Five (5) Years

[ ] (1) If installed:

WARNING: DO NOT USE GREASE OR ANY TYPE OF GREASE FITTING ON ANY OXYGEN SYSTEM. WHEN WORKING WITH AN OXYGEN SYSTEM MAKE SURE HANDS, CLOTHING, TOOLS, AND IMMEDIATE AREA ARE FREE OF GREASE.

[ ] (a) In Seneca IV only: remove and hydrostatically test standard weight oxygen cylinders (i.e., cylinders marked ICC or DOT “3AA”) every five (5) years or if also marked with a five-pointed star, every ten (10) years (or as specified in the latest applicable national regulations).

[ ] (b) In Seneca V only: remove and hydrostatically test light weight composite oxygen cylinders every five (5) years (i.e. - cylinders marked US DOT E 8162 or SP 8162 or Transport Canada SU 4237-127 or SU 9209) (or as specified in the latest applicable national regulations). No lightweight composite oxygen cylinder may exceed fifteen (15) years total time-in-service.

[ ] (c) For each oxygen system outlet, inspect outlet and especially rubber components for condition and operation. Replace rubber components or entire outlet on condition.

[ ] (d) Overhaul the regulator. Replacement of the fill valve is recommended but not required.

[ ] (e) Overhaul or replace oxygen system masks each five (5) years time-in-service.

[ ] (2) Overhaul or replace Hartzell propellers each five (5) or six (6) years or each 2000 or 2400 hours. (Refer to latest revision of Hartzell Service Letter No. 61 to determine specific requirements for individual airplanes.)

[ ] (3) For airplanes equipped with Garmin G600 Flight Display or G1000/G1000 NXi Integrated Avionics System: update the GRS 77 AHRS “Earth magnetic field model” each five (5) calendar years. The updated “Earth magnetic field model” is expected to be available from Garmin on July 1st of each fifth year - i.e. - 2020, 2025, etc.

[ ] (4) In S/N’s 3449509 and up, the CO Pulse Oximeter/Detector must be returned to CO Guardian at the end of Service Life (Five (5) years) for replacement and calibration of the CO sensor to maintain airworthiness of the unit.

J. Each Six (6) Years

[ ] (1) Overhaul or replace Hartzell propellers each five (5) or six (6) years or each 2000 or 2400 hours. (Refer to latest revision of Hartzell Service Letter No. 61 to determine specific requirements for individual airplanes.)

[ ] (2) Overhaul or replace McCauley propellers each six (6) years or 2000 hours; whichever comes first. See McCauley MPC26, Owner/Operator Information Manual, latest revision.

[ ] (3) For airplanes equipped with Aero Accessories, Inc. vacuum pump(s), replace the shear coupling each six (6) years time-in-service.

[ ] (4) If installed, completely discharge the rechargeable (gauged) fire extinguisher and have it professionally inspected and recharged each six years time-in-service.

K. Each Seven (7) Years

[ ] (1) Each 2000 hours or seven (7) years, whichever occurs first, remove interior cabinets, panels, and headliner and conduct detailed inspection of aircraft structure (skin, bulkheads, stringers, etc.) for condition and security. Inspection of structure concealed by headliner may be accomplished by alternate means (i.e. – through the use of a bore scope) without removing the headliner, providing access is obtained to all concealed areas and bore scope provides sufficient detail to adequately accomplish the inspection. At F.S. 73, examine the two steel Upper Forward Cockpit Fittings (one each side), P/N’s 79553-000/-001 or 62522-000/68257-000, as applicable, for signs of corrosion.

[ ] (2) Each 2000 operating hours or seven (7) calendar years time-in-service, whichever occurs first, inspect the aft wing attach fittings and adjacent structure for corrosion for signs of corrosion as specified in 57-40-00, Wing Aft Attach Fitting Inspection.

[ ] (3) In airplanes that have accumulated fifteen (15) calendar years time-in-service or more, each two thousand (2,000) hours or seven (7) calendar years time-in-service, whichever occurs first, perform the Stabilator Control System Special Inspection in 27-00-00.

L. Each Eight (8) Years

[ ] Replace engine compartment flexible fuel and oil hoses as required; but not to exceed 1000 hours time-in-service, eight (8) years, or engine overhaul, whichever comes first; except for TSO-C53a - Type D hoses which are replaced on-condition.

M. Each Ten (10) Years

[ ] (1) Each ten (10) years time-in-service, test fuselage and wing fluid hoses to system pressure. Visually inspect for leaks. Hoses that pass inspection may remain in service, but must be rechecked each five (5) years additional time-in-service. No fluid hose may exceed twenty (20) years total time-in-service.

[ ] (2) For airplanes equipped with Airborne 1H5 series check valve manifolds and/or 1H37 series check valves, replace manifold or check valve at 10 years from check valve/ manifold date of manufacture. See revision of Airborne SL 39A.

WARNING: DO NOT USE GREASE OR ANY TYPE OF GREASE FITTING ON ANY OXYGEN SYSTEM. WHEN WORKING WITH AN OXYGEN SYSTEM MAKE SURE HANDS, CLOTHING, TOOLS, AND IMMEDIATE AREA ARE FREE OF GREASE.

[ ] (3) In Seneca IV only, if installed: for standard weight oxygen cylinders (ICC or DOT “3AA”) marked with a five-pointed star, remove and hydrostatically test every ten (10) years (or as specified in the latest applicable national regulations).

N. Each Twelve (12) Years

[ ] (1) Replace engine-driven fuel pumps at engine overhaul or each twelve (12) years, whichever comes first.

[ ] (2) Overhaul each engine each 1,800 hours or twelve (12) years time-in-service, whichever comes first, or as specified in Continental Aerospace Technologies Publication M-0, “Standard Practice Maintenance Manual,” latest revision.

[ ] (3) If installed, replace disposable-type (non-gauged) fire extinguishers at twelve (12) years from date of manufacture.

[ ] (4) If installed, hydrostatically test rechargeable (gauged) fire extinguishers each twelve (12) years time-in-service.

[ ] (5) At twelve (12) years, each 2000 hours, or engine overhaul, whichever comes first, replace alternators.

O. Each Fifteen (15) Years

WARNING: DO NOT USE GREASE OR ANY TYPE OF GREASE FITTING ON ANY OXYGEN SYSTEM. WHEN WORKING WITH AN OXYGEN SYSTEM MAKE SURE HANDS, CLOTHING, TOOLS, AND IMMEDIATE AREA ARE FREE OF GREASE.

[ ] In Seneca V only: If installed, no lightweight composite oxygen cylinder (i.e., US DOT E 8162 or SP 8162 or Transport Canada SU 4237-127 or SU 9209) may exceed fifteen (15) years total time-in-service.

P. Each Twenty (20) Years

[ ] No fluid hose may exceed twenty (20) years total time-in-service.

4. Per Specific Operation / Operating Environment

A. Operation in High Dust or Industrial Pollution Environment

CAUTION: DISCONNECT LINES FROM PITOT/STATIC SYSTEM BEFORE CONDUCTING THIS INSPECTION.

	Item	Inspection	Inspection Interval
☐	Engine Air Filter.	Clean and inspect.	Daily.
☐	Cabin Environmental and Instrument Air Filters.	Inspect and replace if necessary.	100 Hours.
☐	Pitot/Static system.	Check for obstruction. Reverse flow to lines.	100 Hours or as required.
☐	Landing Gear Oleos	Clean.	Before each flight.
		Inspect.	100 Hours.
☐	Landing Gear Wheel Bearings.	Clean, inspect and repack.	50 Hours.
☐	Windows.	Inspect for cracks, erosion, crazing, visibility, and cleanliness.	Daily.
☐	Structure drain holes.	Clean with pipe cleaner.	Before each flight.

B. Operation in High Salt or High Humidity Environment

	Item	Inspection	Inspection Interval
☐	Fuselage, Empennage, Wings, and Control Surfaces.	Remove floor panels and exterior access plates; inspect for corrosion. using a borescope or other suitable tool.	200 Hours.
☐	Landing Gear.	Inspect for corrosion and lubrication.	200 Hours.

WARNING: ENSURE BOTH MAGNETO SWITCHES ARE OFF (GROUNDED), BEFORE TURNING PROPELLER. ENGINE MAY START IF BOTH SWITCHES ARE NOT OFF. USE EXTREME CAUTION WHEN ROTATING PROPELLER BY HAND; PROPELLER MAY KICK BACK.

☐	Engines with more than 50 hours total time.	Each five days, pull prop through five complete revolutions. Each 30 days, fly aircraft for 30 minutes or, ground run until oil temperature is in the green arc. Avoid excessive ground run.	Each 5 days and each 30 days.
☐	Engines with less than 50 hours total time.	Each day, pull prop through five complete revolutions. Each 30 days, fly aircraft for 30 minutes or, ground run until oil temperature is in the green arc. Avoid excessive ground run.	Daily and each 30 days.
☐	Instruments and Wiring.	Inspect for proper seal of cases and corrosion.	100 Hours.
☐	Interior.	Inspect upholstery, seat belts, seats and rugs for corrosion and integrity.	100 Hours.

NOTE: Do not use metallic tie downs (i.e. - chains, cables, etc.) in high salt or high humidity environments.

C. Operation in Extreme Cold

	Item	Inspection	Inspection Interval
☐	Hydraulic, Pneumatic and Environmental.	Check all fittings and attachments for security and leaks.	First 100 Hour, then as required.

D. Operation from Soft or Unusual Terrain

	Item	Inspection	Inspection Interval
☐	Landing Gear.	Inspect for cracks, attachment, damage, cleanliness and lubrication.	100 Hours.
☐	Wheels.	Inspect for cracks, damage, chipped rims; bearings for damage, corrosion and lubrication.	100 Hours.
☐	Tires.	Inspect for cuts, wear, inflation and deterioration.	Daily.
☐	Wheel Wells.	Inspect for foreign material, damage and corrosion.	100 Hours.
☐	Brakes.	Inspect for damage, foreign material, cracks and overheating.	Daily.
☐	Flaps, Lower Fuselage and Wing.	Inspect for damage, cracks and corrosion.	100 Hours.

E. Each Flight

	Item	Inspection	Inspection Interval
☐	H3R Fire Extinguishers, Model RT-A600, with date codes 2012 and later only. NOTE: See 26-20-00	Weigh unit. Weight must be above the minimum specified on the nameplate.	Each Flight.
☐	In S/N's 3447001 thru 3447008, for those airplanes which have not complied with Piper Service Bulletin No. 990, inspect per Restraint System, Inspection, additional inspection requirements for S/N's 3447001 thru 3447008.		

F. At Engine Overhaul

	Item	Inspection	Inspection Interval
☐	Oil Cooler.	Replace or overhaul at engine overhaul.	Each Occurrence.
☐	Alternator.	Replace at engine overhaul.	Each Occurrence.

UNSCHEDULED MAINTENANCE CHECKS

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

The following inspections are required in response to specific anomalies encountered during aircraft operation. Note that the items listed herein are guidelines based on past operating experience. Each operator should closely monitor his own unique operating conditions/environment and react accordingly to keep his aircraft airworthy.

NOTE: A log book entry should be made upon completion of any inspections.

1. Lightning Strike

	Item	Inspection	Inspection Interval
☐	Propeller.	Hartzell Propellers - refer to the inspection requirements in the latest revision of Hartzell Owner's Manual No.115. McCauley Propellers - see latest revision of McCauley SB 177.	Each occurrence, before further flight.
☐	Engine.	See latest revisions of appropriate Continental Service Bulletins and Overhaul Manual.	Each occurrence, before further flight.
☐	Electrical and Avionics Systems.	Inspect and check harness, connections, and equipment for high voltage damage, burns and insulation degradation. Replace or overhaul as required. Consult with appropriate avionics vendor(s) for inspections and operational checks. Bench test alternator and voltage regulator(s), see 24-30-00. If equipped, replace lightning suppressors.	Each occurrence, before further flight.
	Antenna	Inspect all antenna connections for condition and security.	Each occurrance, before further flight.
☐	All exterior surfaces, skins, and structure.	Inspect for burns, evidence of arcing, and damage on surfaces and bearings. Check for correct material properties in the area of the strike path. Degauss engine mount. Replace or repair affected areas/parts. If equipped, replace lightning diverters.	Each occurrence, before further flight.
☐	System Components.	Inspect instrumentation, vacuum, pitot/static, and fuel systems, for damage and correct operation.	Each occurrence, before further flight.

1. Lightning Strike (continued)

	Item	Inspection	Inspection Interval
☐	Static Wicks.	Replace.	Each occurrence, before further flight.
☐	Bearings.	Inspect all control surface hinges and bearings, and landing gear and wheel bearings for pitting and damage. Replace as required.	Each occurrence, before further flight.

2. Engine Overspeed, Overtemp, Loss of Oil, or Sudden Stoppage

	Item	Inspection	Inspection Interval
☐	Engine.	See latest revisions of appropriate Continental Service Bulletins and Overhaul Manual.	Each occurrence, before further flight.
☐	Propeller.	Hartzell Propellers - refer to the inspection requirements in the latest revision of the Hartzell Owner's Manual No. 115. McCauley Propellers - see latest revision of: McCauley SB 176 for stoppages/strikes; McCauley SL 1998-23 for overspeed.	Each occurrence, before further flight.
☐	Engine Mount and Attachments.	Inspect for distortion and damage. Replace or repair as required.	Each occurrence, before further flight.

3. Severe Turbulence, Hard or Overweight Landing

CAUTION: MINOR OR APPARENTLY SUPERFICIAL DAMAGE MAY INDICATE A MORE SEVERE CONDITION SOMEWHERE ELSE IN THE STRUCTURE.

A. Place aircraft in a normal level attitude.

B. Make a preliminary inspection of checking alignment and out-of-track condition of engine, wings, tail, landing gear and doors.

C. Follow Piper and Continental Maintenance Manual procedures. If there are any questions regarding repairs or procedures, contact your Piper Dealer's Service Advisor (DSA).

D. Inspect the following items closely to determine the extent of damage:

	Item	Inspection	Inspection Interval
☐	Landing Gear Struts. (Not required for severe turbulence.)	Cracks, signs of overstress deformation, loose or damaged strut housings. Axles for cracks, bending or flat spots. Damaged oleos and seals, hydraulic leaks and landing gear alignment.	Each occurrence, before further flight.

3. Severe Turbulence, Hard or Overweight Landing (continued)

	Item	Inspection	Inspection Interval
☐	Wheels, Tires, Brakes. (Not required for severe turbulence.)	Cracks, chips, loose or cracked mounting bolts, alignment of slippage marks, sidewall distress, hydraulic or air leaks. Inspect the wheels (dye penetrant method) and wheel bolts (magnetic particle method).	Each occurrence, before further flight.
☐	Wheel Wells and Landing Gear attach points. (Not required for severe turbulence.)	Buckling, cracks, overstress, wing skin buckling, and side brace for damage and condition. Inspect landing gear attachment bolts (magnetic particle method).	Each occurrence, before further flight.
☐	Wings.	Wing attach bolts for slippage, damage and overstress. Fasteners securing wing attachment fittings to fuselage frames and wing spars for condition and security. Upper and lower wing skins for wrinkles, cracks, popped or loose rivets. Remove access plates and inspect for internal damage to ribs, stringers and sparwebs; and fuel tanks for damage, attachment, and leaks.	Each occurrence, before further flight.
☐	Engine.	Engine mounts for distortion and damage to elastomeric parts. Propeller for evidence of ground strike (i.e. - hard or overweight landing).	Each occurrence, before further flight.
☐	Fuselage.	Loose or missing rivets, door alignment, windows and attachments for overstress, cracks or damage. Wing carry through member for overstress damage. Stringers, bulkheads, keel beams for buckling, cracks, or damage. Avionics, instruments and accessories installation for security and operation.	Each occurrence, before further flight.
☐	Empennage.	Skins for buckling wrinkles, loose or missing rivets. Stabilator, rudder, and vertical fin for security of attachment and overstress of bolts. Ribs, stringers for buckling, cracks and damage.	Each occurrence, before further flight.

4. Flaps Extended Above Maximum Flap Extension Speed (V_{FE})

	Item	Inspection	Inspection Interval
☐	Flap torque tube/pushrod.	Inspect for distortion. Replace as required. (See Flap Torque Tube/Pushrod Distortion Inspection, 27-50-00.)	Each occurrence, before further flight.
☐	Flaps.	Inspect for damage to the skin and attach points. Replace as required.	Each occurrence, before further flight.

5. Flood Damage, Immersion in Water

A. These guidelines are general in nature and should be applied or varied to fit the individual aircraft according to water level, length of time of exposure and other variables. Only those areas that might not be obvious to the mechanic are addressed.

CAUTION: MAKE ALL REPAIRS AND/OR ADJUSTMENTS IN ACCORDANCE WITH THE APPROPRIATE PIPER MAINTENANCE MANUAL, THE COMPONENT MANUFACTURER'S MAINTENANCE MANUAL, AND FAR PART 43. PAY PARTICULAR ATTENTION TO SILT, CORROSION AND CONTAMINANTS.

B. Follow Piper and Continental Maintenance Manual procedures. If there are any questions regarding repairs or procedures, contact your Piper Dealer's Service Advisor (DSA).

C. Determine the water level on the aircraft. Determine which operating and/or electrical components have been exposed to the water.

D. If the following items were immersed, inspect them closely to determine the extent of damage:

	Item	Inspection	Inspection Interval
☐	Airframe.	Clean silt and contaminants from airframe.	If immersed, each event, before further flight.
☐	Tubular Structures. (i.e. - Engine Mounts, etc.)	Check for internal corrosion. Clean and represerve as required. (See 71-20-00 - Engine Mount Corrosion Inspection, Immersion in Water.)	If immersed, each event, before further flight.
☐	Wings.	Inspect to ensure that contaminants are cleaned from fuel cell areas.	If immersed, each event, before further flight.
☐	Landing Gear and associated Bearings, Torque Links, Shimmy Dampeners, etc.	Jack airplane and cycle landing gear oleos and torque links to ensure proper operation.	If immersed, each event, before further flight.
☐	Control Surfaces.	Remove surface, clean and check all bearings - relube or replace as necessary. Rebalance before installation.	If immersed, each event, before further flight.
☐	Flight Control System.	Clean and inspect all cables, pulleys, and bearings for evidence of corrosion. Replace corroded cables. Re-preserve galvanized cable with MIL- C-11796 Class 2 (hot).	If immersed, each event, before further flight.

5. Flood Damage, Immersion in Water (continued)

	Item	Inspection	Inspection Interval
☐	Trim Control System.	Clean and inspect all trim system cables, pulleys, drums, bearings, jack screws, etc. Do not apply preservation to trim cables.	If immersed, each event, before further flight.
☐	Actuating Cables.	Inspect “push-pull” actuating cables for powerplant, heating and ventilating system, fuel system, etc. for proper operation.	If immersed, each event, before further flight.
☐	Engine.	Remove, disassemble, and inspect. Examine all parts paying particular attention for evidence of corrosion, rust or contaminants imbedded on bearing surfaces, piston, mounting flanges or any aluminum, magnesium or bronze surface that may be porous. Remove evidence of rust or corrosion. If pitting in stressed areas is found, the part should not be reused. Silt imbedded in porous surfaces may be removed. Be certain oil passages, dowel holes and similar hidden openings and recesses are thoroughly free from contaminants. Test electrical components and fuel metering devices in accordance with manufacturer’s instructions to determine fitness for future use. Reassemble engine using new seals, gaskets, stressed bolts, nuts and crankshaft sludge tubes.	If immersed, each event, before further flight.
☐	Engine Accessories.	Inspect. Aircraft systems that supply either fuel or oil to the engine must be thoroughly cleaned, including oil cooler, lines, valves, etc. to prevent contamination of the engine after reinstallation.	If immersed, each event, before further flight.
☐	Propeller.	Inspect and repair as necessary in an authorized propeller shop.	If immersed, each event, before further flight.

5. Flood Damage, Immersion in Water (continued)

	Item	Inspection	Inspection Interval
☐	Electrical Systems.	Replace all circuit breakers and switches.	If immersed, each event, before further flight.
		Replace all solenoids, relays and master contactors.	
		Replace battery.	
		Disassemble all connectors; clean and inspect for corrosion. Replace all corroded or pitted connectors. Inspect for wire corrosion at connector.	
		Check all harness assemblies for entrapped contaminants. Clean and check for short circuits.	
		Remove electric motors and electric pumps.	
		Remove all potted solid state electrical equipment such as alternator inop. switches, low fuel warning switches, etc. Clean, dry and bench test per appropriate maintenance manual.	
		Clean and check voltage regulators and overvoltage relays. Replace as necessary	
		Clean and check all strobe light power supplies. Refer to appropriate maintenance manual.	
		Replace all fuel senders, etc.	
		Clean, inspect and check heated pitot systems.	
☐	Autopilot System. (If Installed.)	Bench test in accordance with appropriate maintenance manual. Pay particular attention to clutch settings.	If immersed, each event, before further flight.

5. Flood Damage, Immersion in Water (continued)

	Item	Inspection	Inspection Interval
☐	Vacuum and Pitot-Static Systems.	Replace gyros.	If immersed, each event, before further flight.
		Replace filters.	
		Clean and inspect all lines, and pitot and static vents.	
		Clean and check all regulating valves.	
		Remove and inspect engine driven and auxiliary vacuum pumps.	
☐	Induction System.	Clean and inspect for silt and corrosion. Check all ducts and gaskets. Replace as necessary.	If immersed, each event, before further flight.
		Clean and inspect all heat shrouds and ducting.	
☐	Fuel System.	Remove and clean fuel cells and fuel cells wing area. Clean all associated lines and pumps.	If immersed, each event, before further flight.
		Clean and inspect all fuel tank vents, cap vents and vent lines.	
☐	Instruments.	Clean and inspect instruments. Bench test per appropriate maintenance manual.	If immersed, each event, before further flight.
☐	Heating and Ventilating Systems.	Replace blower.	If immersed, each event, before further flight.
		Clean and inspect all distribution boxes, ducting and valves.	
		Inspect and check system control cables. Replace corroded or binding cables.	
		If installed, clean and inspect air conditioning evaporator, condenser, and compressor.	

5. Flood Damage, Immersion in Water (continued)

	Item	Inspection	Inspection Interval
☐	Oxygen System. (If installed.)	Disconnect all lines from source and outlets; clean all fittings and lines per MIL-I-5585A. Remove and clean regulator valve per appropriate Scott publication. Replace pressure gauge.	If immersed, each event, before further flight.
☐	Avionics Systems.	Replace avionics. Clean and inspect antennas and connectors.	If immersed, each event, before further flight.
☐	Insulation and Upholstery.	Remove all wet insulation and upholstery. Thoroughly clean and dry (or replace) to ensure corrosion is not promoted in adjacent structures.	If immersed, each event, before further flight.

6. Hail Damage

	Item	Inspection	Inspection Interval
☐	Metallic skin of the wings, fuselage, or empennage; control surfaces; or composite components.	Classify and disposition per Hail Damage, 51-70-00.	Each occurrence, before further flight.

7. Overspeed/Overload Event

	Item	Inspection	Inspection Interval
☐	Wings (skin, internal web structure, attach points) and Landing Gear assemblies.	Perform the Overspeed/Overload Wing Inspection, 57-10-00.	Each occurrence, before further flight.

CHAPTER

6

DIMENSIONS AND AREAS

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CHAPTER 6

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
6-List of Effective Pages	1	Oct 10/23			
	2	Sep 30/18			
6-Table of Contents	1	Oct 10/23			
	2	Sep 30/18			
6-00-00	1	Oct 10/23			
	2	Oct 10/23			
	3	Oct 10/23			
	4	Oct 10/23			
6-10-00	1	Oct 10/23			
	2	Oct 10/23			
6-20-00	1	Oct 10/23			
	2	Oct 10/23			
6-40-00	1	Oct 10/23			
	2	Oct 10/23			
	3	Oct 10/23			
	4	Oct 10/23			

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CHAPTER 6 - DIMENSIONS AND AREAS

TABLE OF CONTENTS

SUBJECT	SECTION	PAGE
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Leading Particulars		1
PRINCIPAL DIMENSIONS	**6-10-00**	1
REFERENCE LINES	**6-20-00**	1
ACCESS PROVISIONS	**6-40-00**	1

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GENERAL

1. Leading Particulars

See “Chart 1” for Seneca IV or “Chart 2” on page 06003 for Seneca V.

CHART 1 (Sheet 1 of 2)
LEADING PARTICULARS - SENECA IV

MODEL	**PA-34-220T SENECA IV**
ENGINE	
Manufacturer	Continental
Model - Left (24V)	TSIO-360-KB (CW)
Model - Right (24V)	LTSIO-360-KB (CCW)
FAA Type Certificate	E9CE
Rated Horsepower (Sea Level)	
Max. Takeoff - 5 minute maximum	220 HP
Max. Continuous	200 HP
Rated Speed - RPM	
Max. Takeoff - 5 minute maximum	2800 RPM
Max. Continuous	2600 RPM
Oil SAE Number	See Lubrication Chart
Oil Sump Capacity	8 U.S. quarts
Fuel: Aviation Grade - Minimum Octane	100 or 100LL
Fuel Injector	Continental
Magnetos: (TCM Pressurized)	79020-118
Left (Left Engine)	S6LSC-25P
Right (Right Engine)	S6RSC-25P
Magneto Timing	20° BTC
Magneto Point Clearance	.018 + .006
Spark Plugs (Shielded):	Refer to latest revision of Continental Aircraft Engine Service Bulletin M77-10
Spark Plug Gap Setting	.015 to .019
Firing Order:	
Left Engine	1-6-3-2-5-4
Right Engine	1-4-5-2-3-6
Starter - Prestolite (28-volt)	
Left Engine	646275
Right Engine	646275
Alternator - Continental (60 amp):	
Left Engine	653344
Right Engine	653344
Alternator Control Unit - Lamar (28V)	B-00382-1

CHART 1 (Sheet 2 of 2)
LEADING PARTICULARS - SENECA IV

MODEL	PA-34-220T SENECA IV	
PROPELLER		
Manufacturer	Hartzell	McCauley
Hub Model:		
Left Engine	BHC-C2YF-2CKUF (Left Eng.) (1)	3AF32C508
Right Engine	BHC-C2YF-2CLKUF (Right Eng.) (1)	3AF32C509
Blade Model:		
Left Engine	FC8459-8R	82NFA-6
Right Engine	FJC8459-8R	L82NFA-6
Diameter, Minimum	75 in.	75 in.
Blade Angle, Low Pitch (High RPM)	12.6° ± 0.2°	11.0° ± 0.2°
Blade Angle, High Pitch (Low RPM)	80° to 81.5°	81.0° to 83.5°
Governor Models (Hartzell):		
Left Engine	E-3-7	
Right Engine	E-3-7L (E-8-7L) 2	
FUEL SYSTEM		
Fuel Tank	64 gal./wing	
Total Capacity (Both Wings)	128 gal.	
Total Usable Fuel	123 gal.	
LANDING GEAR		
Tread (Width From Each Tire Center)	11.1 ft.	
Turning Radius	60.4 ft.	
Nose Gear Strut	Combination Air-Oil	
Nose Tire Pressure	40 PSI	
Nose Gear Travel	27 degrees LEFT or RIGHT	
CAUTION: DO NOT OVERSTEER THE NOSE GEAR WHEN TOWING AIRCRAFT.		
Main Gear Strut	Combination Air-Oil	
Main Tire Pressure	55 PSI	
Brakes	30-83	
Tires:		
Main	6.00 x 6, 8 ply	
Nose	6.00 x 6, 6 or 8 ply	
Wheels:		
Main	Cleveland 40-120C	
Nose	Cleveland 40-56B	

(1) Propellers to be mounted in pairs only. Do not mix with other propellers.
(2) With synchrophaser installation only.

CHART 2 (Sheet 1 of 2)
LEADING PARTICULARS - SENECA V

MODEL	PA-34-220T SENECA V
ENGINE	
Manufacturer	Continental
Model - Left (24V)	TSIO-360-RB (CW)
Model - Right (24V)	LTSIO-360-RB (CCW)
FAA Type Certificate	E9CE
Rated Horsepower (Sea Level)	
Max. Continuous	220 HP 200 HP
Rated Speed - RPM	
Max. Continuous	2600 RPM
Oil SAE Number	See Lubrication Chart
Oil Sump Capacity	8 U.S. quarts
Fuel: Aviation Grade - Minimum Octane	100 or 100LL
Fuel Injector	Precision RSA-5AD2
Magnetos: (Pressurized)	
Left (Left Engine)	Slick 6324, TCM S6LSC-25P
Right (Right Engine)	Slick 6320, TCM S6RSC-25P
Magneto Timing	22° BTC
Magneto Point Clearance	.008 to .012
Spark Plugs (Shielded):	Refer to latest revision of Continental Aircraft Engine Service Bulletin M77-10
Spark Plug Gap Setting	.015 to .019
Firing Order:	
Left Engine	1-6-3-2-5-4
Right Engine	1-4-5-2-3-6
Starter - Prestolite (28-volt), S/N's 3449001–3449448 (1)	
Left Engine	646275
Right Engine	646275
Starter - Iskra (24-volt), S/N's 3449449–3449463 (1)	
Left Engine	655566
Right Engine	655566
Starter - Sky-Tec (24-volt), S/N's 3449464–3449507 (1)	
Left Engine	657596
Right Engine	657596
Starter - Energizer (24-volt), S/N's 3449508 and up (1)	
Left Engine	646275-1
Right Engine	646275-1
Alternator - Continental (85 amp):	
Left Engine	654200
Right Engine	654200
Alternator Control Unit - Lamar (28V)	B-00382-1

(1) Factory installed in these serial numbers. All starters are interchangeble for service.

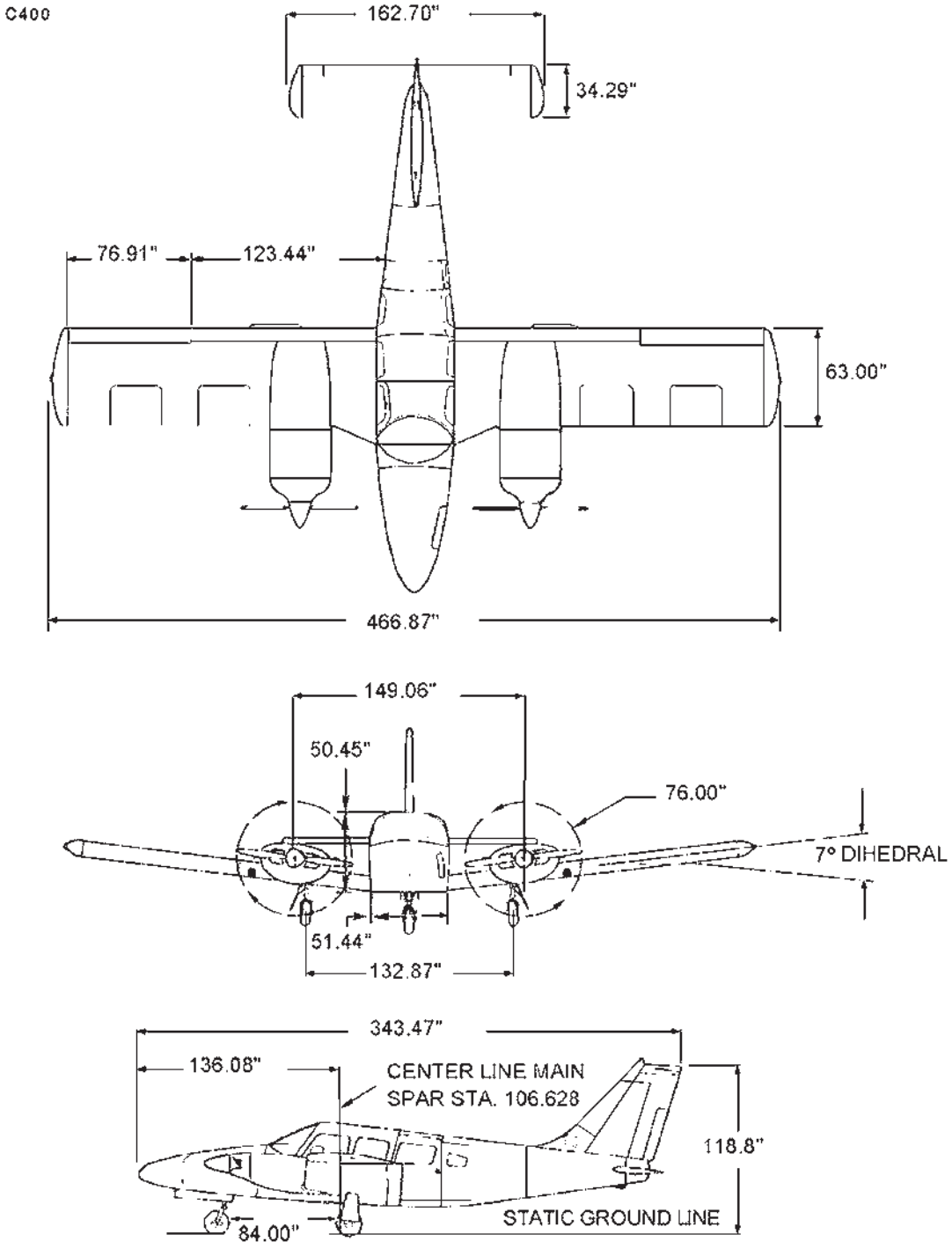
CHART 2 (Sheet 2 of 2)
LEADING PARTICULARS - SENECA V

MODEL	PA-34-220T SENECA V	
PROPELLER		
Manufacturer	Hartzell	McCauley
Hub Model:		
Left Engine	BHC-J2YF-2CUF (Left Eng.) (1)	3AF32C522
Right Engine	BHC-J2YF-2CLUF (Right Eng.) (1)	3AF32C523
Blade Model:		
Left Engine	FC8459(B)-8R	82NJA-6
Right Engine	FJC8459(B)-8R	L82NFA-6
Diameter, Minimum	75 in.	75 in.
Blade Angle, Low Pitch (High RPM)	14.6° ± 0.2°	12.6° ± 0.2°
Blade Angle, High Pitch (Low RPM)	80° to 81.5°	81.6° to 82.6°
Governor Models (Hartzell):		
Left Engine	E-3-9(U-3-19) (2)	
Right Engine	E-3-9L (U-3-19L) (2) (U-8-9L) (3) (U-8-19L) (4)	
FUEL SYSTEM		
Fuel Tank	64 gal./wing	
Total Capacity (Both Wings)	128 gal.	
Total Usable Fuel	122 gal.	
LANDING GEAR		
Tread (Width From Each Tire Center)	11.1 ft.	
Turning Radius	60.4 ft.	
Nose Gear Strut	Combination Air-Oil	
Nose Tire Pressure	40 PSI	
Nose Gear Travel	27 degrees LEFT or RIGHT	
CAUTION: DO NOT OVERSTEER THE NOSE GEAR WHEN TOWING AIRCRAFT.		
Main Gear Strut	Combination Air-Oil	
Main Tire Pressure	55 PSI	
Brakes	30-83	
Tires:		
Main	6.00 x 6, 8 ply	
Nose	6.00 x 6, 6 or 8 ply	
Wheels:		
Main	Cleveland 40-120C	
Nose	Cleveland 40-56B	

(1) Propellers to be mounted in pairs only. Do not mix with other propellers.
(2) With optional unfeathering installation.
(3) With optional syncrophaser installation.
(4) With optional unfeathering and syncrophaser installation.

PRINCIPAL DIMENSIONS

The principal airplane dimensions are shown in “Figure 1”. The airplane serial number is located on the Manufacturers Aircraft Association (MAA) plate located on the left side of the fuselage at approximately F.S. 278.6. The engine serial number plate is located on the right side of the engine oil sump just below cylinder number 5.

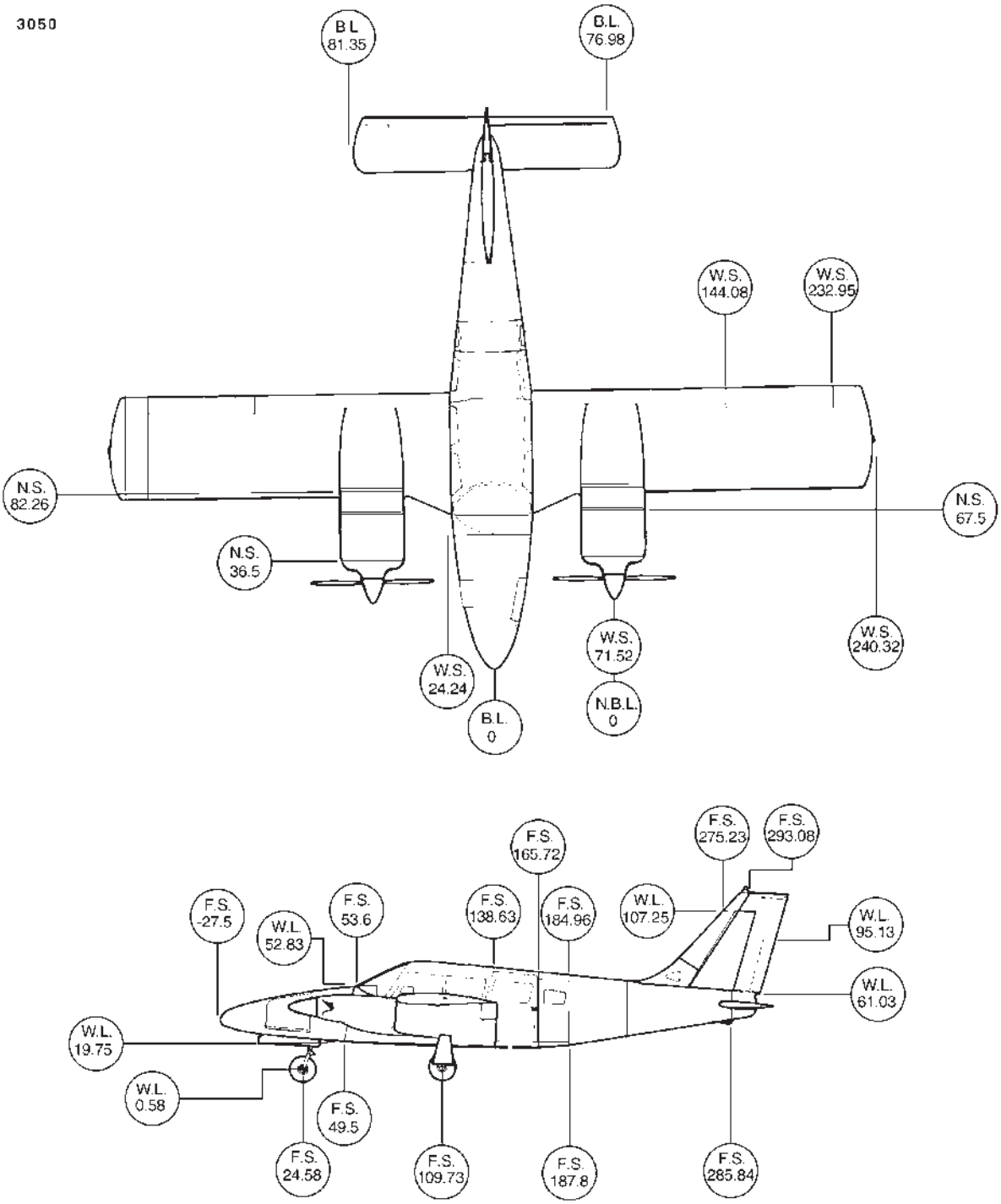


Three View
Figure 1

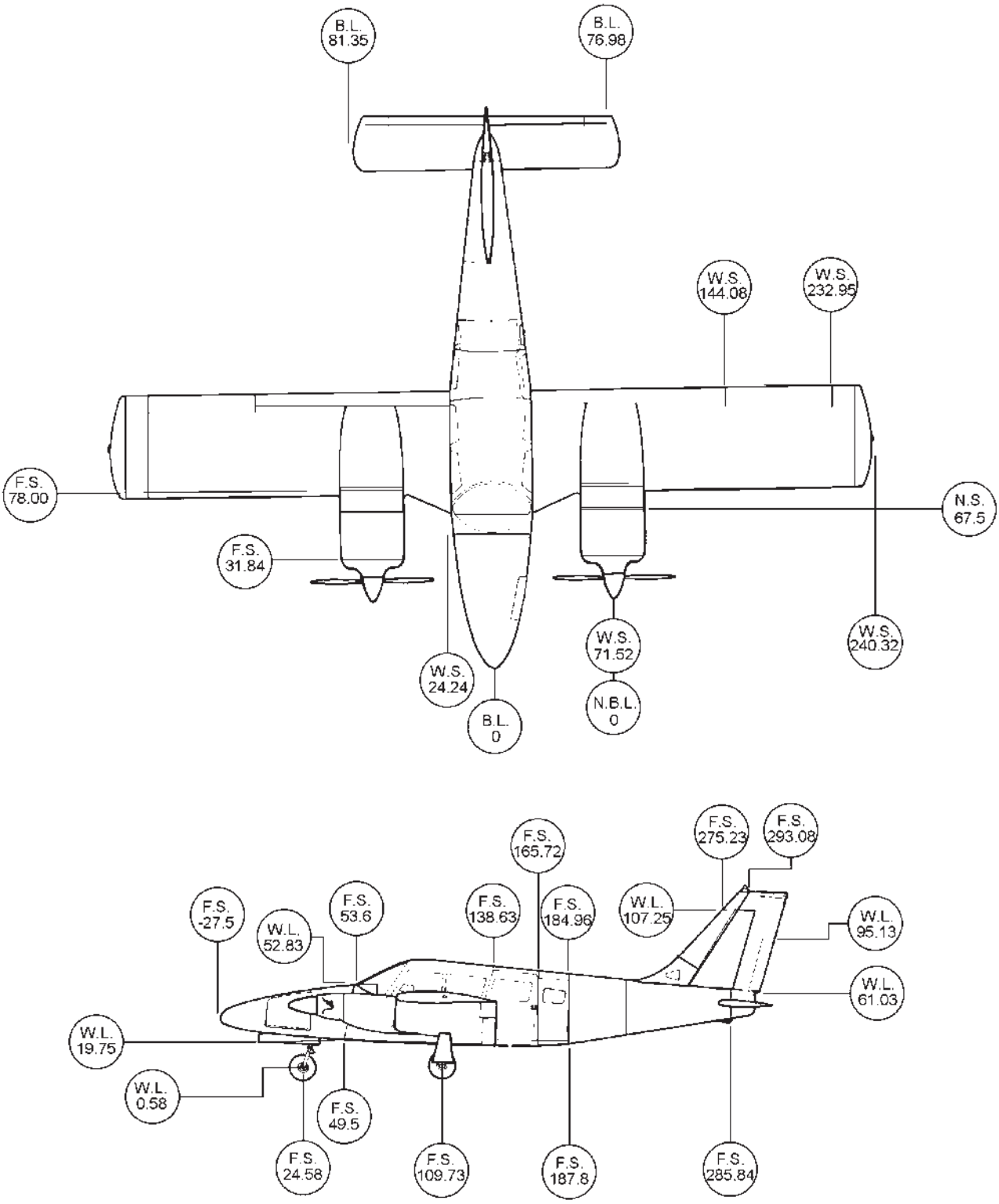
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REFERENCE LINES

To facilitate locating various components and structural members in the airplane, a method using fuselage station (FS), wing station (WS), buttock line (BL), or waterline (WL) designations is frequently employed in this manual. (See Figure 1.) Fuselage and wing stations and buttock and water lines are reference points measured in inches (vertically or horizontally) from a given reference (or zero) line.



Station Reference Lines SENECA IV
Figure 1 (Sheet 1 of 2)



Station Reference Lines SENECA V
Figure 1 (Sheet 2 of 2)

ACCESS PROVISIONS

CAUTION: BEFORE ENTERING THE AFT SECTION OF THE FUSELAGE, BE SURE THE AIRPLANE IS SUPPORTED AT THE TAIL SKID.

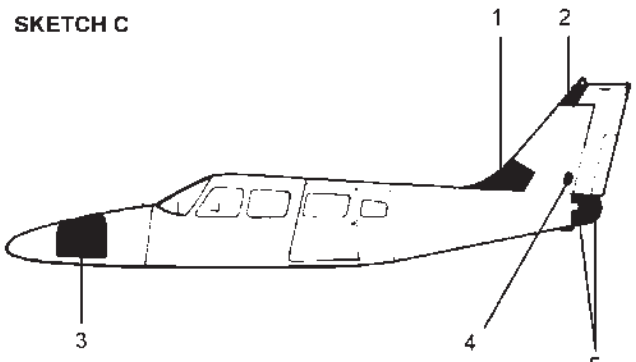
The access and inspection provisions for the airplane are shown in “Figure 1” on page 06402. The component to be serviced or inspected through each opening is identified in the illustration. All access plates and panels are secured by either metal fasteners or screws.

Access plates and panels include all forms of covers, cowlings, fairings, plates, panels, tips, etc. identified in “Figure 1” on page 06402, which may be removed for access to any component or space.

All access plates and panels should be inspected for damage or excessive wear, including all mounting holes, each time they are removed. If damage or excessive wear is detected, installation of a new access plate or panel is recommended.

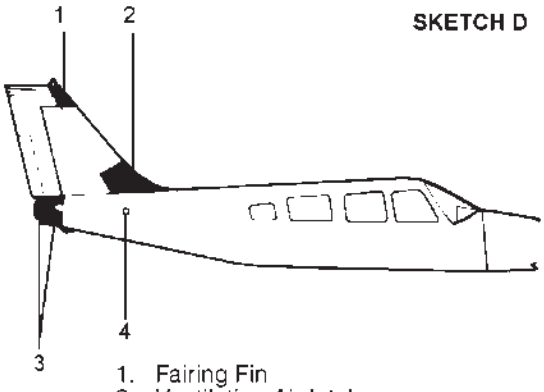
To enter the aft section of the fuselage, open the baggage compartment door and remove the rear trim panel.

SKETCH C

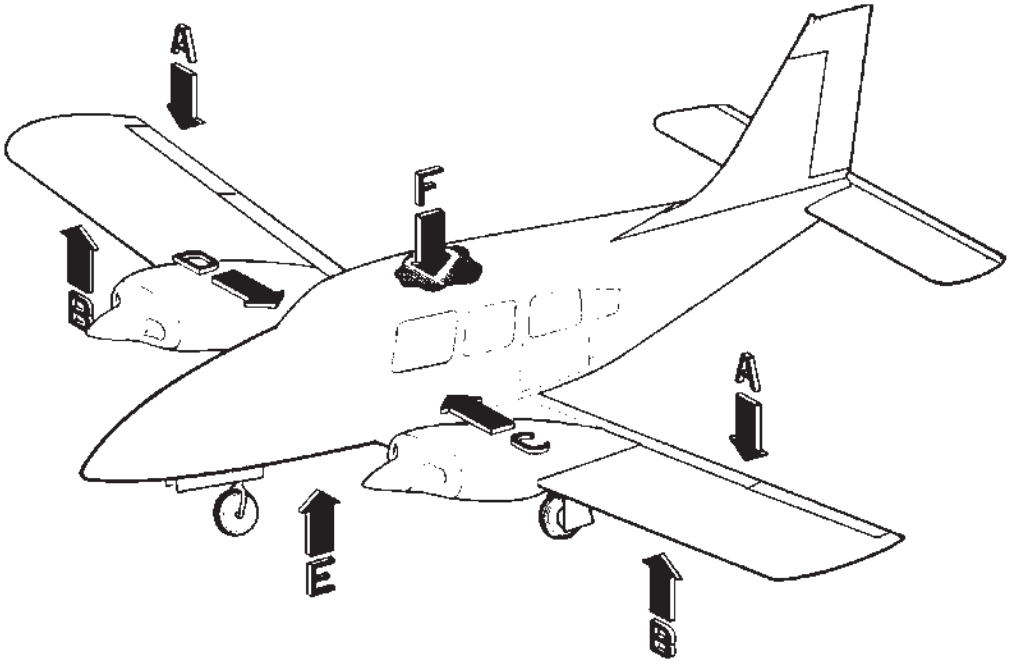


1. Ventilating Air Intake
2. Fairing Fin
3. Battery, Brake Reservoir, Electronic Equipment, Hydraulic Power Pack
4. Rudder Trim Tab Screw
5. Stabilator and Rudder Stops, Stabilator Trim Tab Screw

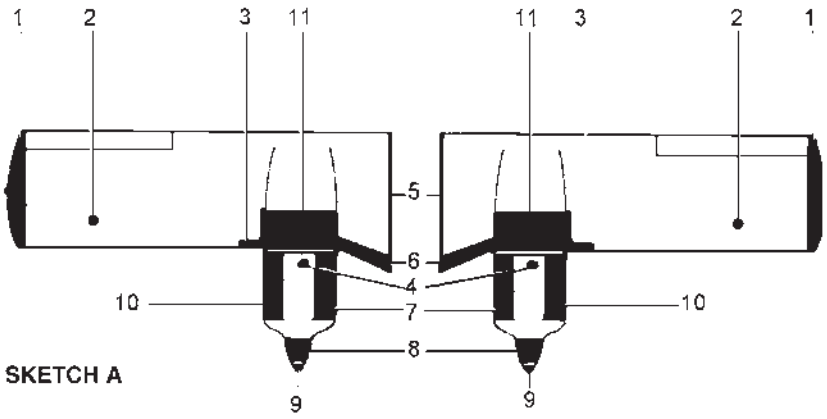
SKETCH D



1. Fairing Fin
2. Ventilating Air Intake
3. Stabilator and Rudder Stops, Stabilator Trim Tab Screw
4. Combustion Heater, Combustion Air Intake



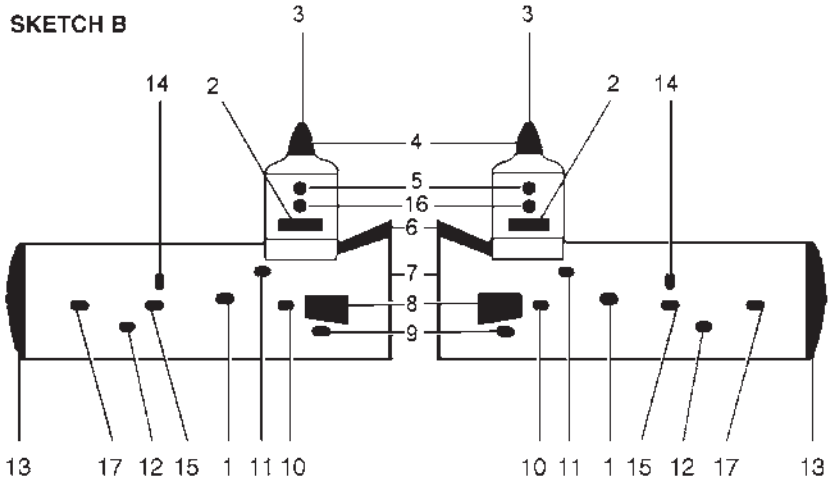
Access Plates and Panels
Figure 1 (Sheet 1 of 2)



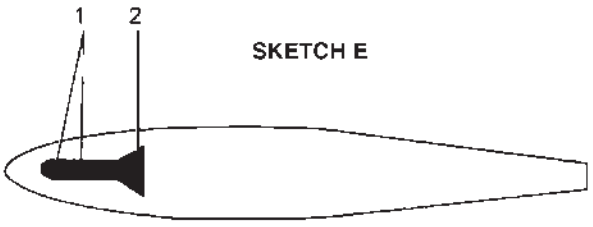
SKETCH A

1. Wing Tip
2. Cap, Fuel Tank Assembly
3. Fairing
4. Oil Filler Door
5. Fairing
6. Lines, Wires, Cables
7. Cowl, Inboard (Seneca IV only)
8. Spinner, Propeller
9. Propeller Recharge Valve
10. Cowl, Outboard (Seneca IV only)
11. Hatch, Nacelle

SKETCH B

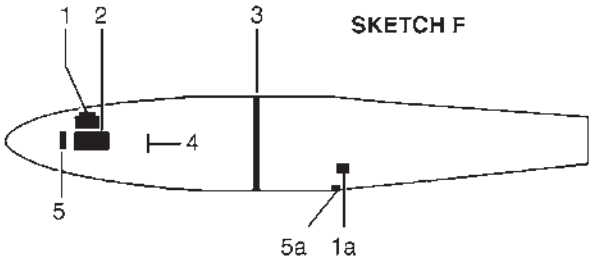


1. Fuel Gauge Sender
2. Cowl Flap
3. Propeller Recharge Valve
4. Spinner, Propeller
5. Engine Oil Drain
6. Lines, Wires, Cables
7. Fairing
8. Door, Main Gear, Landing
9. Cover, Gear Attachment Fitting
10. Electric Wiring
11. Fuel Valve
12. Cover, Bellcrank, OAT probes and Control Cable
13. Wing Tip
14. Stall Warning & Deicer Boots
15. Pitot Heat Wiring - Aileron Cables, Deicer Lines
16. Oil Filter Access Door (Seneca V only)
17. Magnetometer Cover(s) (Seneca V only - Avidyne, single - pilot side, Garmin, dual - both sides)



SKETCH E

1. Door, Nose Gear, Landing
2. Hydraulic Pressure Switch, Nose Gear Actuating Cylinder and Limit Switch, Voltage Regulators



SKETCH F

1. Battery (Seneca IV)
1a. Battery (Seneca V)
2. Nose Landing Gear
3. Main Spar
4. Brake Reservoir
5. External Power Plug (Seneca IV)
5a. External Power Plug (Seneca V)

Access Plates and Panels
Figure 1 (Sheet 2 of 2)

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CHAPTER

7

LIFTING AND SHORING

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CHAPTER 7

LIST OF EFFECTIVE PAGES

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7-Table of Contents	1	Sep 30/18			
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CHAPTER 7 - LIFTING AND SHORING

TABLE OF CONTENTS

SUBJECT	SECTION	PAGE
JACKING	**7-10-00**	1

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JACKING

Jacking the airplane is necessary to service the landing gear and other operations. The jacking operation is normally performed using tripod jacks. In other situations (i.e. - emergency, post-accident lifting, etc.), slings or air bags may be more appropriate.

If wing or fuselage shoring is required, make sure the support is contoured to conform to the surface it is supporting.

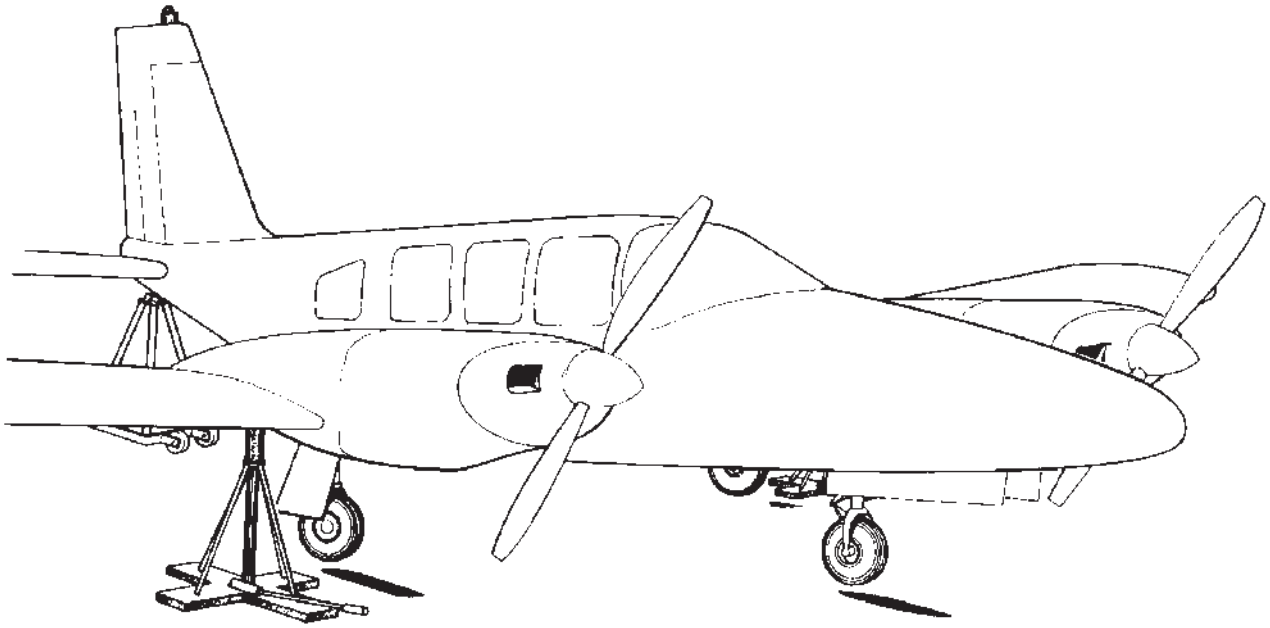
Jacking

A. Align jacks under the wing respective pads on the wing front spar.

CAUTION: BE SURE TO APPLY SUFFICIENT TAIL SUPPORT BALLAST. OTHERWISE, THE AIRPLANE WILL SLIP FORWARD AND FALL ON THE FUSELAGE NOSE SECTION.

B. Attach a tail stand with approximately 600 pounds ballast to tail skid.

C. Carefully raise jacks until all three wheels are clear of the surface.



Jacking
Figure 2

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CHAPTER

8

LEVELING AND WEIGHING

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CHAPTER 8

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CHAPTER 8 - LEVELING AND WEIGHING

TABLE OF CONTENTS

SUBJECT	SECTION	PAGE
WEIGHING	**8-10-00**	1
LEVELING	**8-20-00**	1

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WEIGHING

The airplane may be weighed by the following procedure (see “Figure 1”):

A. Position a scale and ramp in front of each of the three wheels.

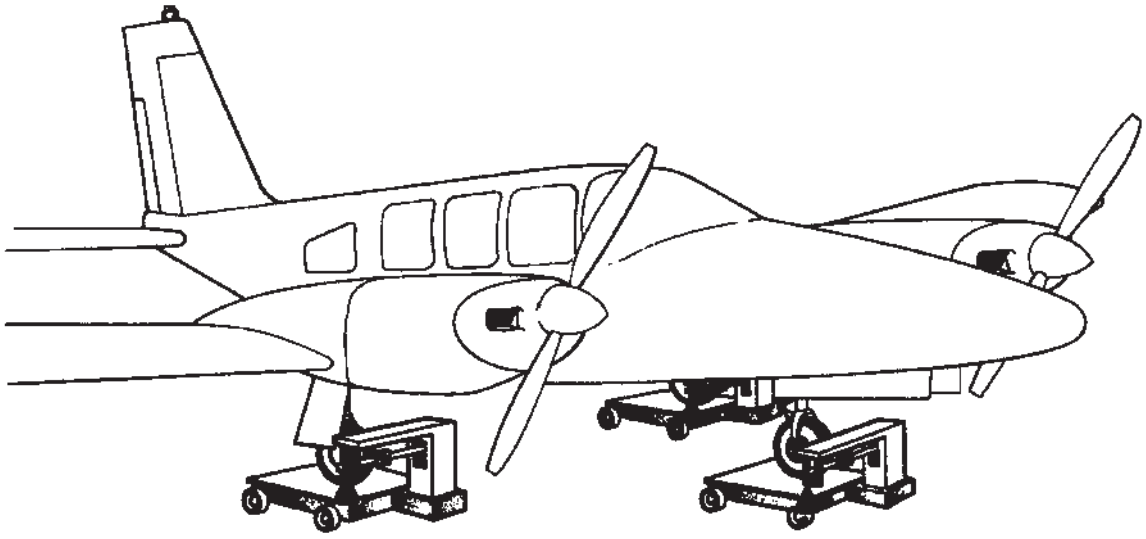
B. Secure the scales from rolling forward and tow the airplane up onto the scales. (See Towing, 9-10-00.)

C. Remove the ramp so as not to interfere with the scales.

D. If the airplane is to be weighed for weight and balance computations, level the airplane per 8-20-00.

NOTE: See Chapter 6 in the applicable Pilot’s Operating Handbook for specific instructions on weighing the aircraft for weight and balance computations.

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Weighing Airplane
Figure 1

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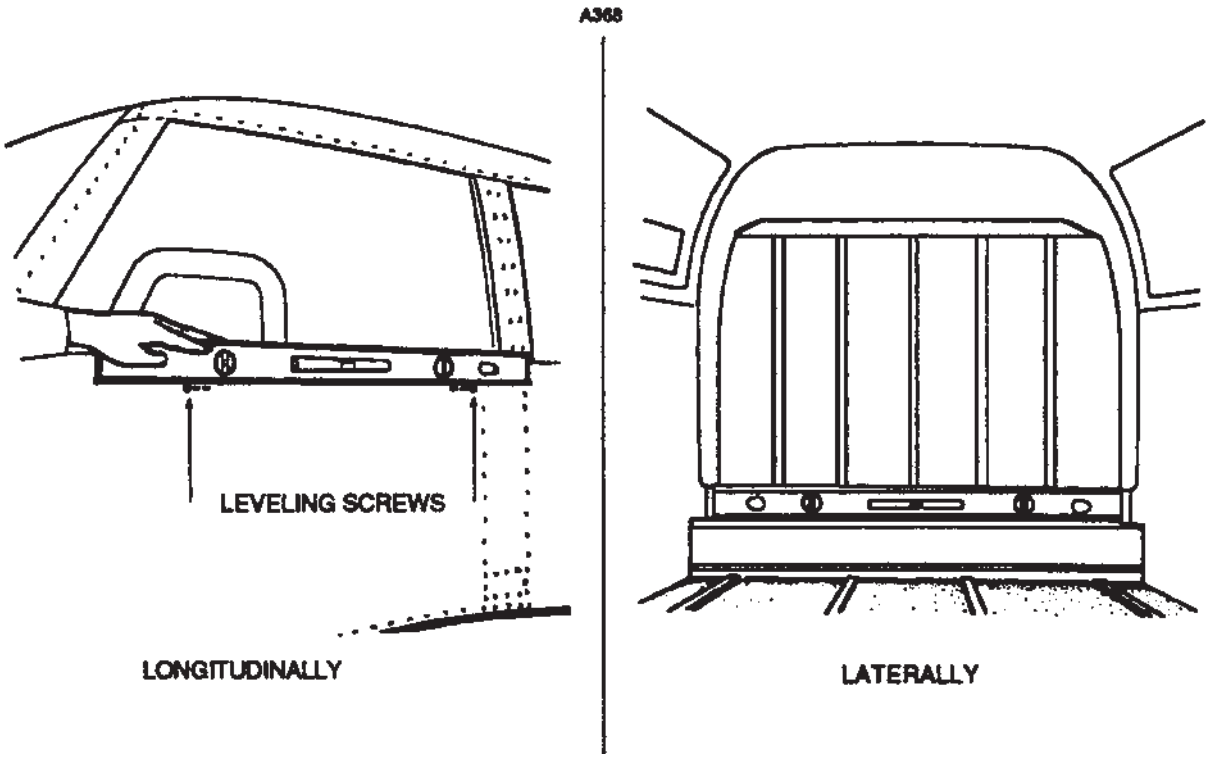
LEVELING

All configurations of the airplane are provided with a means for longitudinal and lateral leveling. The airplane may be leveled while on jacks, during the weighing procedure while the wheels are on scales, or while the wheels are on the ground. To level the airplane for purposes of weighing or rigging, the following procedures may be used:

NOTE: Always level the airplane laterally first, then level it longitudinally.

A. To laterally level the airplane, place a spirit level across the baggage compartment floor along the rear bulkhead (See “Figure 1”) and deflate the tire on the high side of the airplane or adjust either jack until the bubble of the level is centered.

B. To longitudinally level the airplane, partially withdraw the two leveling screws located immediately below the left front side window. (See “Figure 1”) Place a spirit level on these screw heads and deflate the nose wheel tire or adjust the jacks until the bubble of the level is centered.



Leveling Airplane
Figure 1

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CHAPTER

9

TOWING AND TAXIING

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CHAPTER 9

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CHAPTER 9 - TOWING AND TAXIING

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TOWING	**9-10-00**	1
TAXIING	**9-20-00**	1

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TOWING

Before towing airplane, ground personnel must be informed by a qualified pilot or other qualified personnel about tow turning limits of nose gear and any other system functions required to properly and safely move the airplane.

CAUTION: WHEN TOWING WITH POWER EQUIPMENT, TURNING THE NOSE GEAR IN EITHER DIRECTION BEYOND THE STEERING RADIUS LIMITS WILL RESULT IN DAMAGE TO THE NOSE GEAR AND STEERING MECHANISM.

CAUTION: WHEN TOWING WITH POWER EQUIPMENT, USE ONLY EQUIPMENT THAT IS APPROPRIATE FOR THE AIRPLANE CLASS AND WEIGHT.

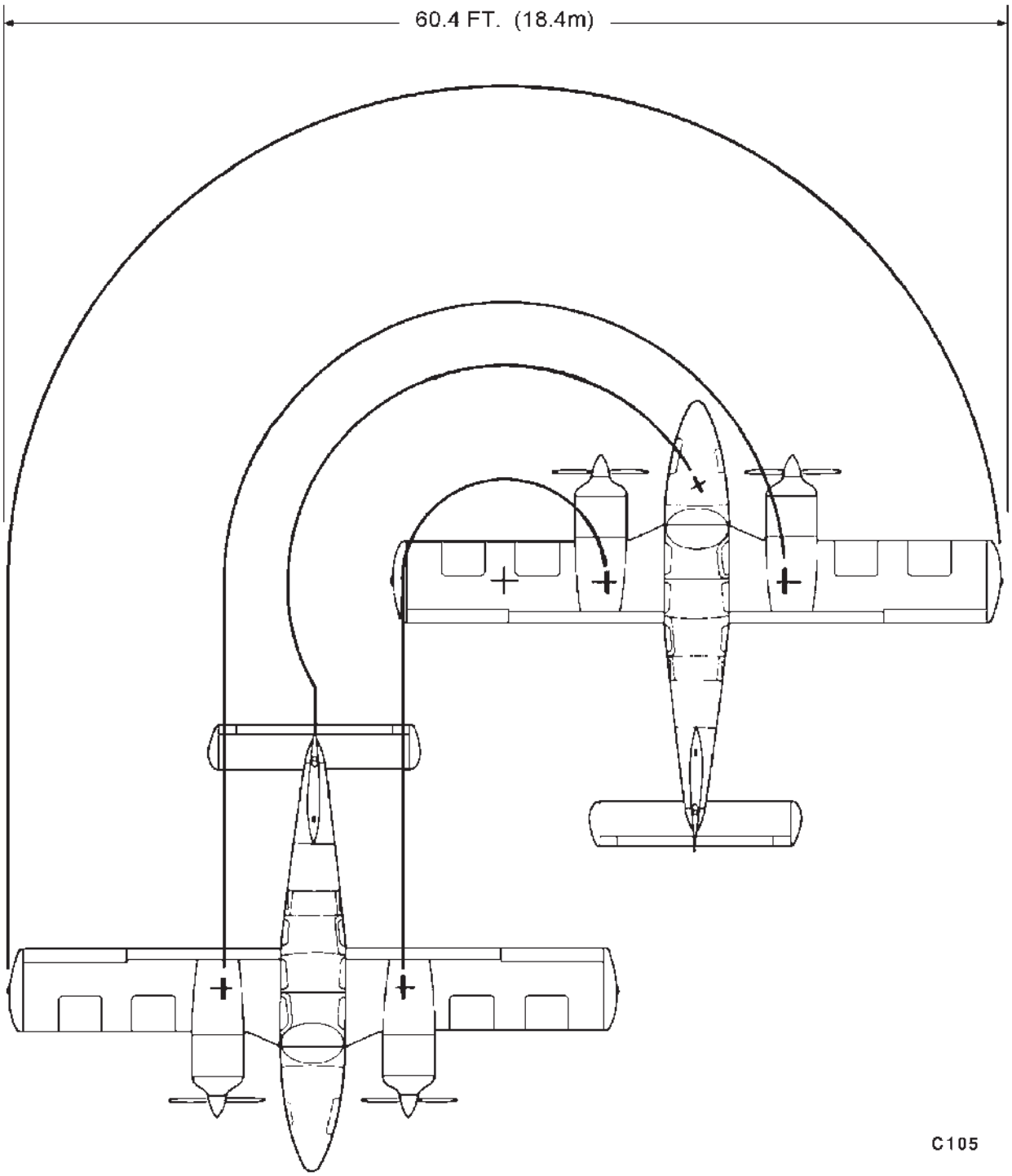
CAUTION: IF AIRPLANE IS STUCK (OR THE WHEELS ARE OTHERWISE NOT FREEWHEELING), DO NOT TOW THE AIRPLANE FROM THE NOSE. ATTACH TOW LINES TO THE MAIN GEAR STRUTS AS DESCRIBED BELOW.

CAUTION: PUSHING ON THE TRAILING EDGE OF THE AILERONS, WHEN MOVING THE AIRCRAFT FORWARD BY HAND, WILL CAUSE THE AILERON CONTOUR TO CHANGE RESULTING IN AN OUT-OF-TRIM CONDITION.

The airplane may be moved by using the nose wheel steering bar or by using power equipment appropriate for the airplane class and weight (i.e., that will not damage or cause excess strain to the nose gear, steering assembly, engine mount, etc.). The nose wheel steering bar is stowed in the forward baggage compartment and engages the front axle inside the fork.

In the event towing lines are necessary, lines (rope) should be attached to both main gear struts as high up on the tubes as possible. Lines should be long enough to clear the nose and/or tail by not less than 15 feet, and a qualified person to ride in the pilot's seat to maintain control by use of the brakes.

NOTE: See "Figure 1" on page 09102 for aircraft turning distance.



Airplane Turning Radius
Figure 1

TAXIING

Before attempting to taxi the airplane, ground personnel should be checked out by a qualified pilot or other responsible person. Engine starting and shutdown procedures should be covered as well. When it is ascertained that the propeller back blast and taxi areas are clear, apply power to start the taxi roll and perform the following checks:

CAUTION: DO NOT OPERATE THE ENGINE AT HIGH RPM WHEN RUNNING UP OR TAXIING OVER GROUND CONTAINING LOOSE STONES, GRAVEL OR ANY LOOSE MATERIAL THAT MAY CAUSE DAMAGE TO THE PROPELLER BLADES.

CAUTION: OBSERVE WING CLEARANCES WHEN TAXIING NEAR BUILDINGS OR OTHER STATIONARY OBJECTS. IF POSSIBLE, STATION A GUIDE OUTSIDE THE AIRPLANE TO OBSERVE.

CAUTION: WHEN TAXIING ON UNEVEN GROUND, AVOID HOLES AND RUTS.

A. Taxi forward a few feet and apply brakes to determine their effectiveness.

B. Taxi with propeller set in low pitch, high rpm setting.

C. While taxiing, make slight turns to ascertain the effectiveness of steering.

D. Observe wing clearances when taxiing near buildings or other stationary objects. If possible, station a guide outside the airplane to observe.

E. When taxiing on uneven ground, look for and avoid holes and ruts.

F. Do not operate the engines at high rpm when running up or taxiing over ground containing loose stones, gravel, or any loose material that may cause damage to the propeller blades.

NOTE: See 9-10-00, “Figure 1” on page 09102 for aircraft turning distance.

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CHAPTER

10

PARKING AND MOORING

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CHAPTER 10

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CHAPTER 10 - PARKING AND MOORING

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Parking		1
Locking Airplane		1
MOORING	**10-20-00**	1

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PARKING

1. Parking

When parking the airplane, ensure that it is sufficiently protected against adverse weather conditions and presents no danger to other aircraft. When parking the airplane for any length of time or overnight, it is recommended that it be moored.

A. To park the airplane, head it into the wind, if possible.

B. Set the parking brake by pulling back the brake lever and depressing the knob attached to the left side of the handle.

C. To release the parking brakes, pull back on the brake lever to disengage the catch mechanism. Then allow the handle to swing forward.

NOTE: Take care when setting brakes that are overheated or during cold weather when accumulated moisture may freeze the brakes.

D. The aileron and stabilator controls may be secured with the pilot's seat belt.

2. Locking Airplane

The right front cabin door, left aft cabin door and the nose baggage compartment door are provided with a key lock on the outside. All doors and the locking gas cap (optional) use the same key, 1995 - 2003.

In S/N's 3449301 and up, and for service replacement 2004 and later, the locking gas cap uses a separate key.

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MOORING

CAUTION: WHEN MOORING, USE SQUARE OR BOWLINE KNOTS. DO NOT USE SLIP KNOTS.

The airplane is moored to ensure its immovability, protection, and security under various weather conditions.

A. Head the airplane into the wind, if possible.

B. Block the wheels.

C. Lock the aileron and stabilator controls by looping the pilot's seat belt around wheel.

CAUTION: WHEN USING ROPE CONSTRUCTED OF NON-SYNTHETIC MATERIAL, LEAVE SUFFICIENT SLACK TO AVOID DAMAGE TO THE AIRPLANE WHEN THE ROPES CONTRACT DUE TO MOISTURE.

D. Secure tie-down ropes to the wing, tie-down rings and the tail skid at approximately 45 degree angles to the ground.

NOTE: Additional preparations for high winds include using tie-down ropes from the landing gear forks, and securing the rudder.

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CHAPTER

11

PLACARDS AND MARKINGS

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CHAPTER 11

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CHAPTER 11 - PLACARDS AND MARKINGS

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Self-Adhesive Installation		1
Application of Edge Sealer (Decals only)		1
Without Adhesive - Installation with Tape		2
Meyercord / Ink Transfers (Decals)		3
Removal		3
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EXTERIOR PLACARDS AND MARKINGS	**11-20-00**	1
INTERIOR PLACARDS	**11-30-00**	1

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GENERAL

1. Placards, Nameplates, and Decals

(PIR-PPS65103, Rev. K.)

A. Self-Adhesive Installation

NOTE: For proper adhesion of decals, apply when the temperature is between 55° and 90° F.

(1) Wipe the area where the item is to be installed, using a clean cloth soaked with a 50:50 mixture of isopropyl alcohol and water. Dry with a clean, dry cloth.

NOTE: Dry the area before the alcohol evaporates, to avoid discoloration and deposits at the edges of the wetted area.

(2) Lift a corner or an edge of backing from the item, without touching the adhesive.

(3) Position as specified by the applicable figure in 11-20-00 or 11-30-00 and secure a free corner or edge.

(4) Using a squeegee or equivalent item to hold the item to the surface, draw the rest of the backing from the item.

(5) Press the item in place with the squeegee by wiping across it from end to end, around edges and corners, and into recessess using firm, deliberate strokes and heavy pressure.

(6) Remove the premask from face of item by using an eraser or finger nail to lift a corner. Then peel back as parallel to the item as possible.

NOTE: Any attempt to squeegee the item after the premask is removed will cause shiny marks, blemishing the item's background.

(7) If any portion of the item tends to lift off, remove and discard it. Repeat this procedure to install a replacement item.

B. Application of Edge Sealer (Decals only)

Decals intended for use in harsh environments have an overlay film applied during manufacturing. This overlay film typically extends 0.25 inch around the perimeter of the decal. If any of the overlay film is cut or otherwise damaged during installation, then an additional sealer must be applied along all edges of the decal. 3M 3950 Edge Sealer (P/N 279-474) is an acceptable sealer to be used in this application. Apply Edge Sealer as follows:

(1) Apply edge sealer to the edges of the decal by straddling the decal and the aircraft structure with a ¼-inch brush or dauber.

(2) Wipe this interface in a smooth continuous motion to provide a fillet-type seal that extends onto the decal and aircraft structure by aprroximately 0.1 inch.

(3) Ensure that the entire edge is sealed without gaps or air bubbles.

(4) Touch up as required with additional brush coats of edge sealer within 2 hours.

(5) Wipe all excess sealer with a dry wiper.

(6) Air dry per manufacturer's recommendation to achieve a tack-free condition of the edge sealer.

C. Without Adhesive - Installation with Tape

To bond to the aircraft any placards, etc., that do not have an adhesive backing, use 3M™ VHB™ pressure sensitive acrylic foam tape, as follows.

(1) Confirm the correct location to adhere the double-sided tape. Place location marks (using masking tape) on the aircraft surface to precisely position the tape.

(2) Clean the surface using acetone or a 50:50 mixture of isopropyl alcohol and water. Wipe the area dry with a clean, lint-free, dry cloth.

(3) Wipe both faying surfaces with a brush or swab saturated with 3M Tape Primer 94 (P/N 279-127, one quart). Use the minimum amount of primer that will coat the desired surface areas. Allow 10 minutes for it to dry.

NOTE: Use a minimum coating of the tape primer. More is not better.

CAUTION: THE ADHESION OF THE TAPE IS ALMOST IMMEDIATE, SO USE EXTREME CARE AND APPLY VERY LITTLE CONTACT PRESSURE. AVOID THE NEED TO REPOSITION THE TAPE.

(4) Handle the double-sided acrylic foam tape by its edges only; carefully position the tape using the location marks. Adhesion is almost immediate, so use extreme care and as little contact pressure as possible. Avoid the need to reposition the tape, but if necessary, move the tape immediately and cautiously.

(5) When the placement of the tape is completed, use a roller, squeegee, or other smoothing tool to apply even pressure against the tape. Pay close attention to the tape edges.

CAUTION: DO NOT TRY TO REPOSITION THE TAPE AFTER USING SMOOTHING TOOL. DO NOT PULL ON THE TAPE OR ATTEMPT TO LIFT OR PEEL ITS EDGES.

(6) Keep the outward liner intact on the tape and position the item to be bonded with the tape over the location. Remove the tape liner using the pull tabs, folding them back at an approximately 45-degree angle from under the item, exposing the tape to the item.

(7) Once bonded, if there is some bowing or deformation of the adhered item, apply pressure not exceeding 5 psi with a shot bag (approximately 6 inches wide, 36 lbs per foot is only 1 psi) for 30 minutes.

2. Meyercord / Ink Transfers (Decals)

(PIR-PPS65104, Rev. J.)

Transfers (decals) installed on the instrument panel are Meyercord / Ink type. The following procedures should be followed in the event one or more of these transfers (decals) must be replaced.

A. Removal

CAUTION: DO NOT USE LACQUER THINNER ON ANY PANEL THAT HAS BEEN PAINTED WITH ENAMEL OR LACQUER. INSTRUMENT PANELS ARE PAINTED AT THE FACTORY WITH POLYURETHANE PAINTS.

Remove placard to be replaced with a clean cloth dampened with lacquer thinner.

CAUTION: BUTYL GLYCOL ETHER/ISOPROPYL ALCOHOL SOLVENT SOLUTION WILL REMOVE ENAMEL, LACQUER, AND POLYUTHERANE BASED PAINT PRODUCTS IF LIQUID IS DROPPED ONTO PAINTED SURFACE AND NOT REMOVED IMMEDIATELY.

If panel is painted with enamel or lacquer use a clean cloth dampened with Butyl Glycol Ether/Isopropyl Alcohol Solvent Solution to remove placard to be replaced.

B. Installation

(1) If required, clean surface to receive transfer (decal) using alcohol. Newly painted surfaces need no preparation.

(2) Mix a solution consisting of six (6) parts water and one (1) part Butyl Glycol Ether/Isopropyl Alcohol Solvent Solution (P/N 179-497).

(3) Submerge transfer (decal) in the mixed Butyl Glycol Ether/Isopropyl Alcohol Solvent Solution for approximately 3 to 5 seconds.

(4) Remove transfer (decal) from mixed solvent solution and lay in position.

(5) Using a plastic squeegee, squeegee out from center to edges to remove excess solution.

(6) Wait approximately 30 to 60 seconds, then slide the backing paper off and wipe up the excess solution with a damp cloth.

(7) Wait at least 30 minutes at room temperature before wiping the face of the transfer (decal) with a damp cloth to remove excess solvent residue.

(8) Allow the transfer (decal) to dry thoroughly, (tack free in 2 hours at room temp) before handling.

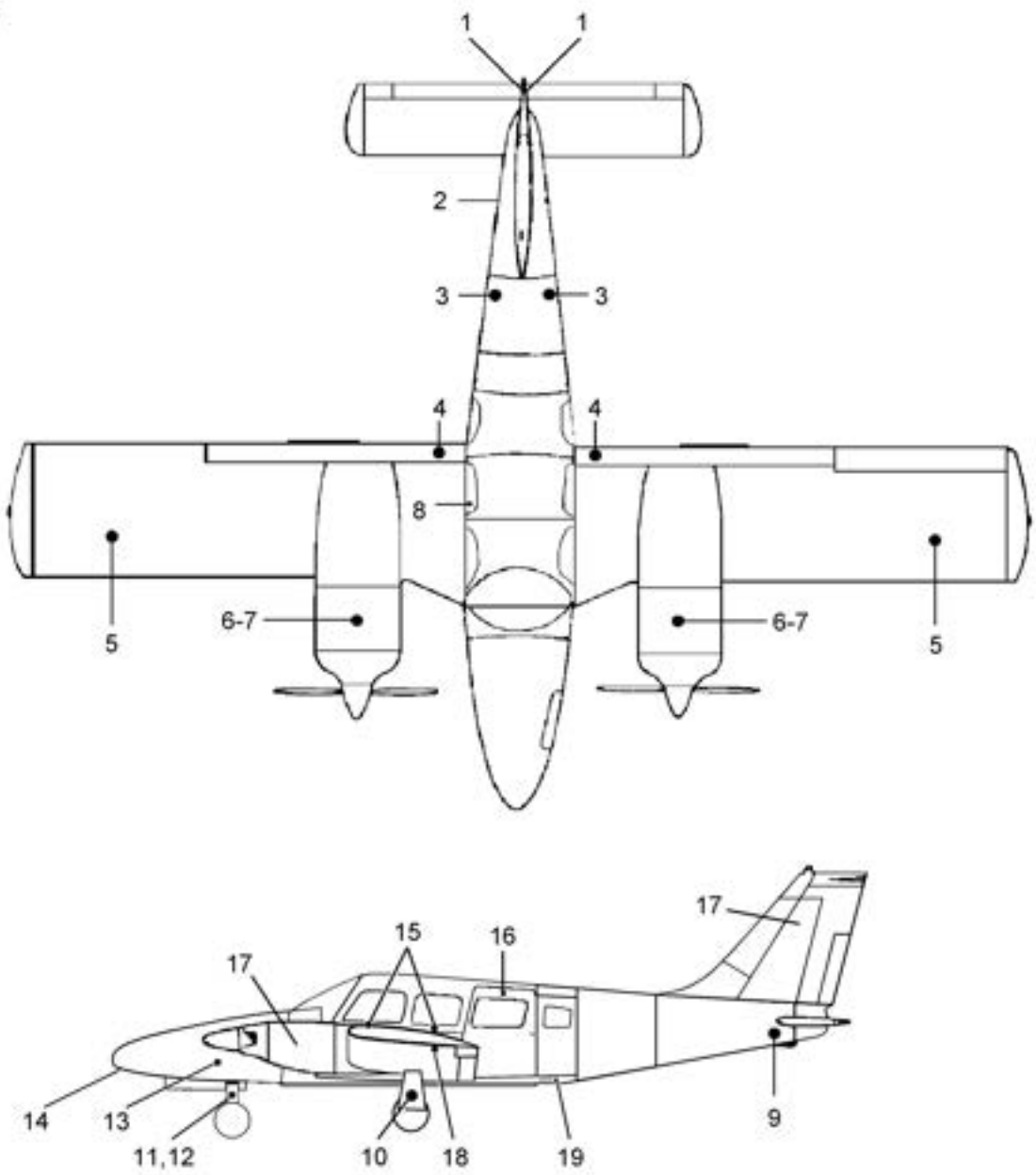
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EXTERIOR PLACARDS AND MARKINGS

The airplane nameplate placard (“Figure 1” on page 11202, Item 9) is located on the left side of the fuselage near the stabilator leading edge at approximately F.S. 278.60. The placard identifies the airplane by its model number and serial number. Should a question arise concerning the care of the airplane, it is important to include the airplane serial number in any correspondence to your Piper Dealer’s Service Advisor (DSA).

NOTE: Any time an airplane is repainted or touched up, inspect all placards to ensure that they are not covered with paint, are legible, and securely attached.

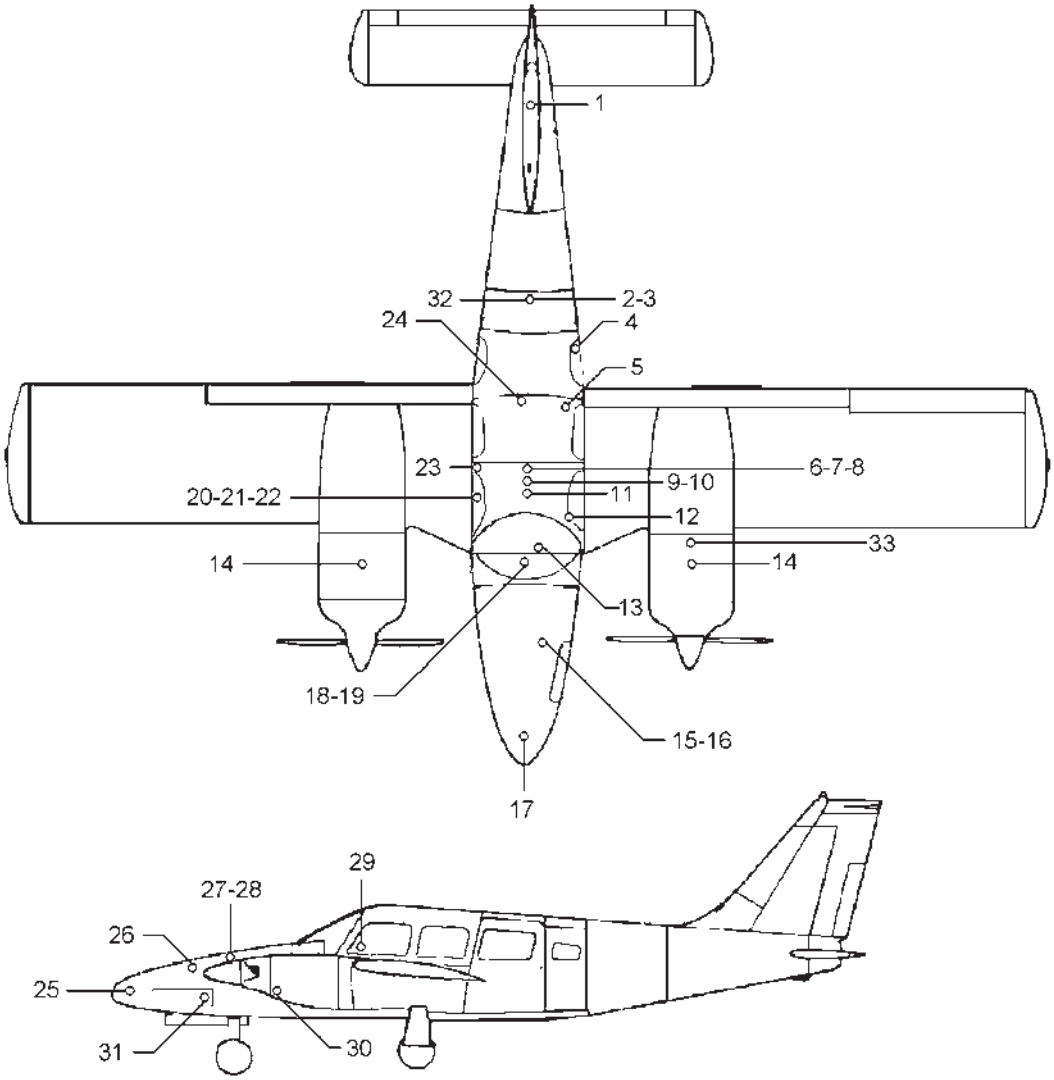
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1. SILKSCREEN - DO NOT PUSH
2. PLACARD - ELT WARNING
3. SILKSCREEN - STATIC VENT
4. SILKSCREEN - NO STEP
5. DECAL - AVGAS
6. PLACARD - OIL SPECIFICATIONS
7. PLACARD - WINTERIZATION INFORMATION
8. SILKSCREEN - PIPER AIRE (OPTIONAL)
9. PLATE - AIRPLANE NAMEPLATE
10. PLACARD - OLEO SERVICE INSTRUCTIONS, MAIN GEAR
11. PLACARD - OLEO SERVICE INSTRUCTIONS, NOSE GEAR
12. PLACARD -TOW LIMITATIONS
13. SILKSCREEN - ALIGNMENT ALERT BAR
14. PLACARD - 28 VOLT EXTERNAL POWER (SENECA IV)
15. SILKSCREEN - LEVEL POINTS
16. SILKSCREEN - DOOR RELEASE
17. SILKSCREEN OR DECAL - MODEL NAME OR PIPER LOGO
18. DECAL - USE ONLY NON-MAGNETIC FASTENERS .....
    (used when magnetometer installed on access cover with Avidyne Entegra or Garmin G600/G1000, up to two per side)
19. DECAL - 28 VOLT EXTERNAL POWER (SENECA V)

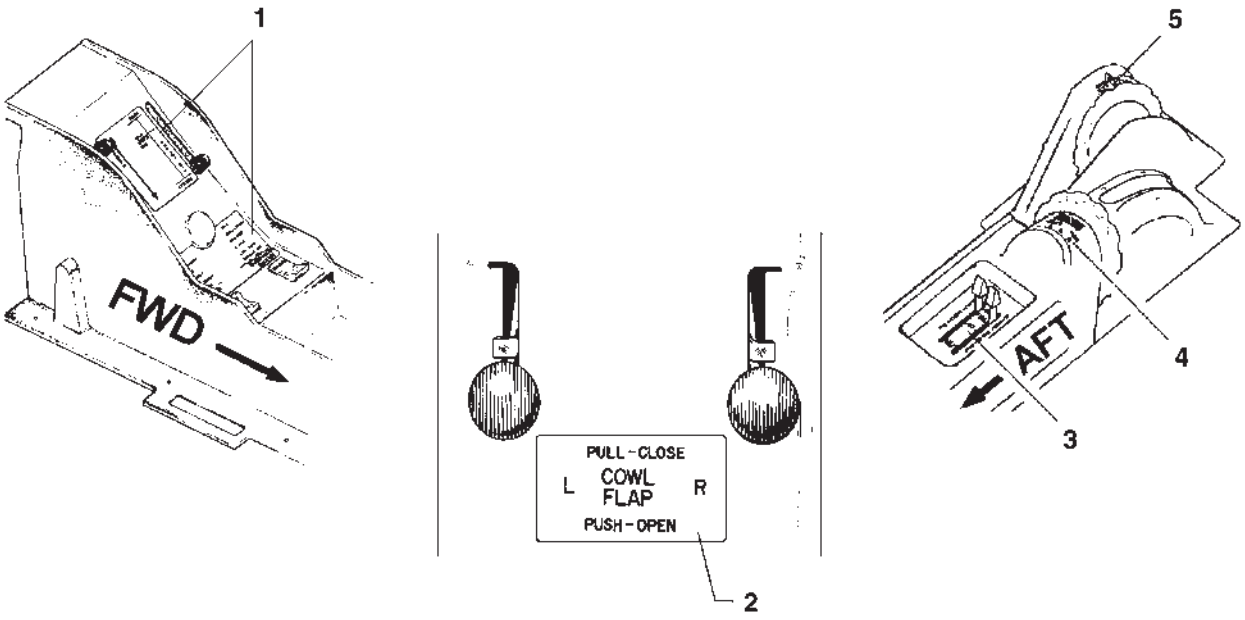
Exterior Placards and Markings
Figure 1

INTERIOR PLACARDS



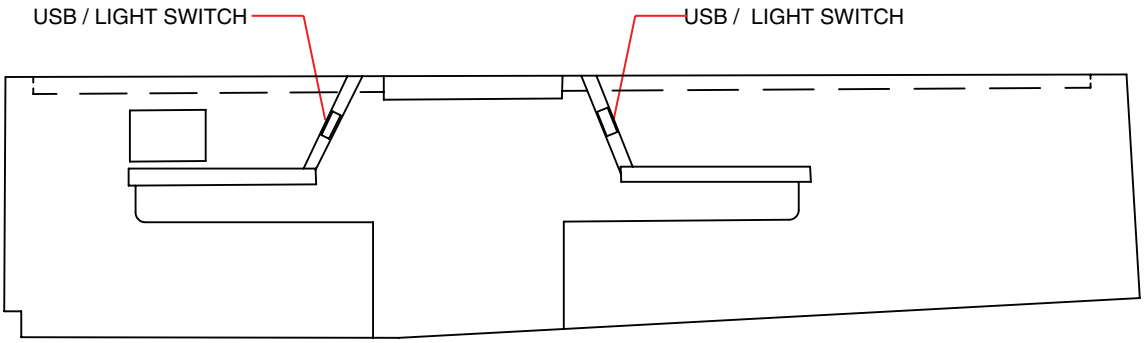
1. PLACARD - STABILATOR BALANCE WEIGHT
2. PLACARD - BAGGAGE LIMITATIONS
3. PLACARD - CABIN BULKHEAD
4. PLACARD - AFT CABIN DOOR RELEASE
5. PLACARD - TABLE STOWAGE
6. PLACARD - FUEL SELECTOR (GALLONS PER SIDE)
7. PLACARD - HEATER CONTROL
8. PLACARD - HEATER, FLIGHT AND GROUND OPERATION
9. PLACARD - CABIN AIR, PULL OFF
10. PLACARD - VENT FAN (3 POSITION)
11. PLACARD - FLAP LEVER
12. PLACARD - PITOT DRAIN
13. PLACARD - CONTROL WHEEL (TRANSPONDER IDENT)
14. PLACARD - OIL DRAIN PROBE LOCATION
15. PLACARD - BRAKE RESERVOIR
16. PLACARD - FUEL LEVEL LINE, RESERVOIR
17. PLACARD - HYDRAULIC FLUID SPECIFICATION
18. PLACARD - PROPELLER SYNCHROPHASER
19. PLACARD - TAKEOFF AND LANDING OPERATION
20. PLACARD - DOOR, OPEN
21. PLACARD - DOOR, LATCH
22. PLACARD - FORWARD CABIN DOOR RELEASE
23. PLACARD - SUMP DRAIN
24. PLACARD - OXYGEN BOTTLE INSTALLATION
25. PLACARD - EXTERNAL POWER
26. PLACARD - BAGGAGE LIMITATIONS
27. PLACARD - LIGHT SWITCH, BAGGAGE DOOR
28. PLACARD - UNLATCH
29. PLACARD - STORM WINDOW
30. PLACARD - OIL DRAIN PROBE
31. PLACARD - PRESS BALL TO UNLATCH
32. PLACARD - MAX. CABIN STRUCTURAL LOAD
33. DECAL - A/C SERVICE

Interior Placards
Figure 1 (Sheet 1 of 2)

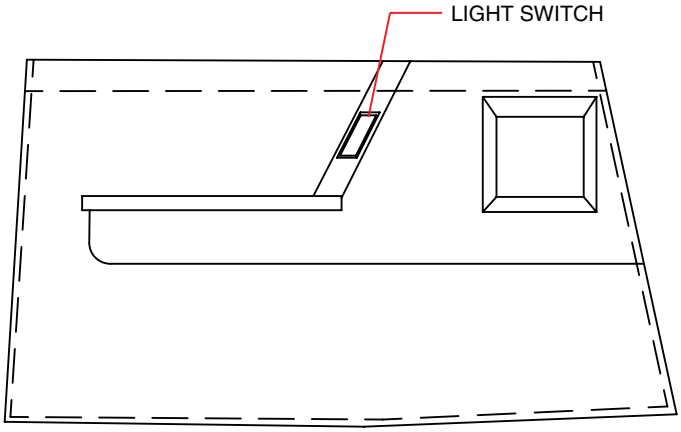


1. PLACARD - HEATER / FAN OPERATION
2. PLACARD - COWL FLAPS
3. PLACARD - FUEL CONTROLS
4. PLACARD - RUDDER TRIM
5. PLACARD - ELEVATOR TRIM

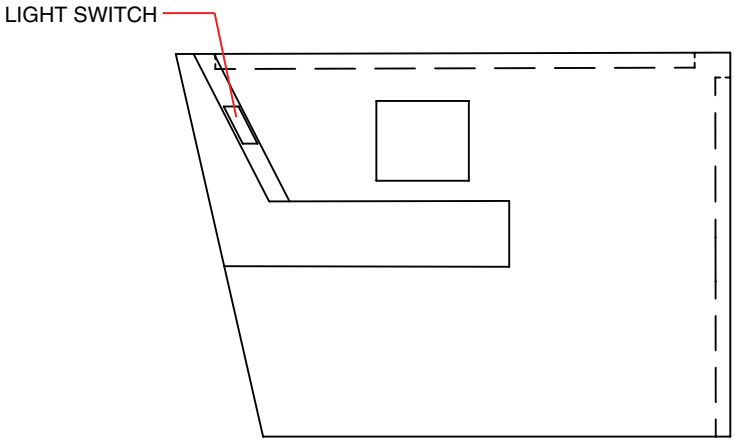
Interior Placards
Figure 1 (Sheet 2 of 2)



RH - LOWER PANEL ASSY



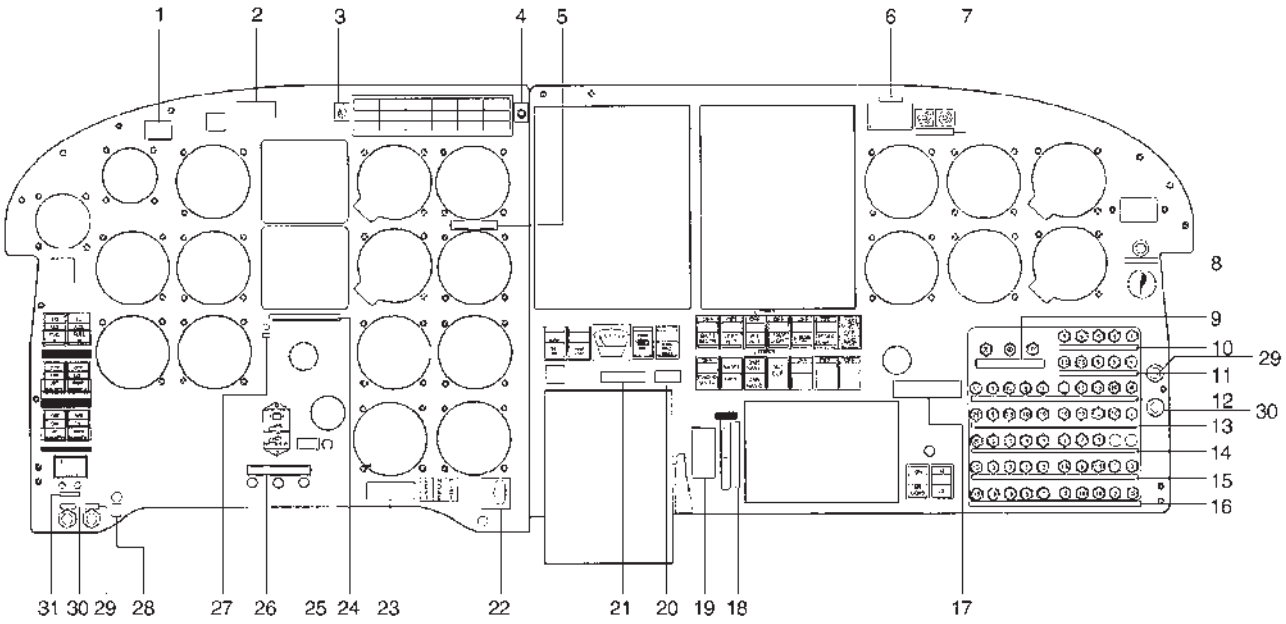
LH - FWD - PANEL ASSY



RH - FWD - PANEL ASSY

Panel Assy - Cabin
Figure 2

Effectivity
Seneca V
with Garmin G1000 NXi

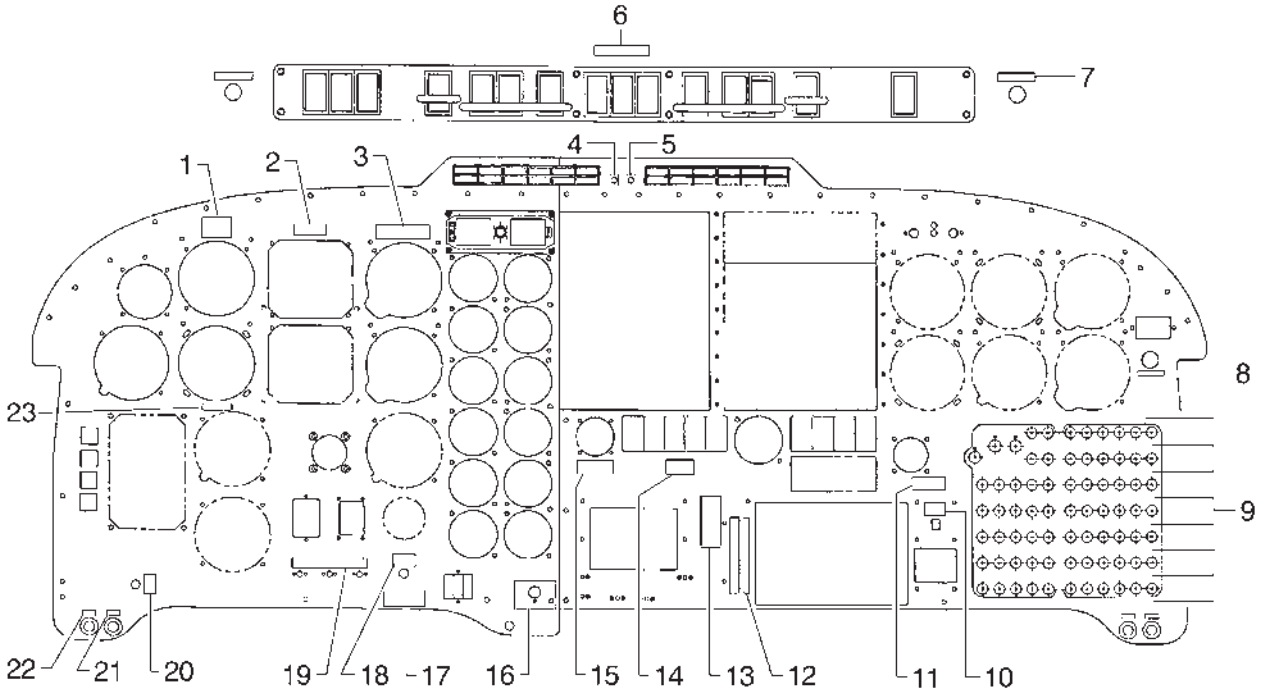


1. PLACARD - SPEED LIMITATIONS
2. PLACARD - AIRPLANE REGISTRATION
3. PLACARD - DAY - NIGHT
4. PLACARD - PRESS TO TEST
5. PLACARD - AVOID CONTINUOUS GROUND OPERATION . . . . .
6. PLACARD - AMMETER
7. PLACARD - L ALT. AMPS - R.ALT AMPS
8. PLACARD - KLN-90 DATALOADER
9. PLACARD - CIRCUIT BREAKER
10. PLACARD - CIRCUIT BREAKER
11. PLACARD - CIRCUIT BREAKER
12. PLACARD - CIRCUIT BREAKER
13. PLACARD - CIRCUIT BREAKER
14. PLACARD - CIRCUIT BREAKER
15. PLACARD - CIRCUIT BREAKER
16. PLACARD - CIRCUIT BREAKER
17. PLACARD - WARNING - AIR CONDITIONER MUST BE OFF .....
18. PLACARD - FLAP TRAVEL
19. PLACARD - ALTERNATE STATIC SOURCE
20. PLACARD - WINDSHIELD PANEL HEAT
21. PLACARD - WARNING -THIS AIRCRAFT IS NOT APPROVED .....
22. PLACARD - GEAR UP / DOWN MAXIMUM SPEEDS
23. PLACARD - EMERGENCY GEAR EXTENSION
24. PLACARD - GPS LIMITED TO VFR USE ONLY
25. PLACARD - OXYGEN - PULL ON
26. PLACARD - DIMMING - SWITCH - PANEL - AVIONICS
27. PLACARD - GPS
28. PLACARD - PARK BRAKE PULL
29. PLACARD - PHONE
30. PLACARD - MIC
31. PLACARD - L PRIMER R

Effectivity
Seneca IV

Instrument Panel Placards
Figure 3 (Sheet 1 of 8)

87290 BV

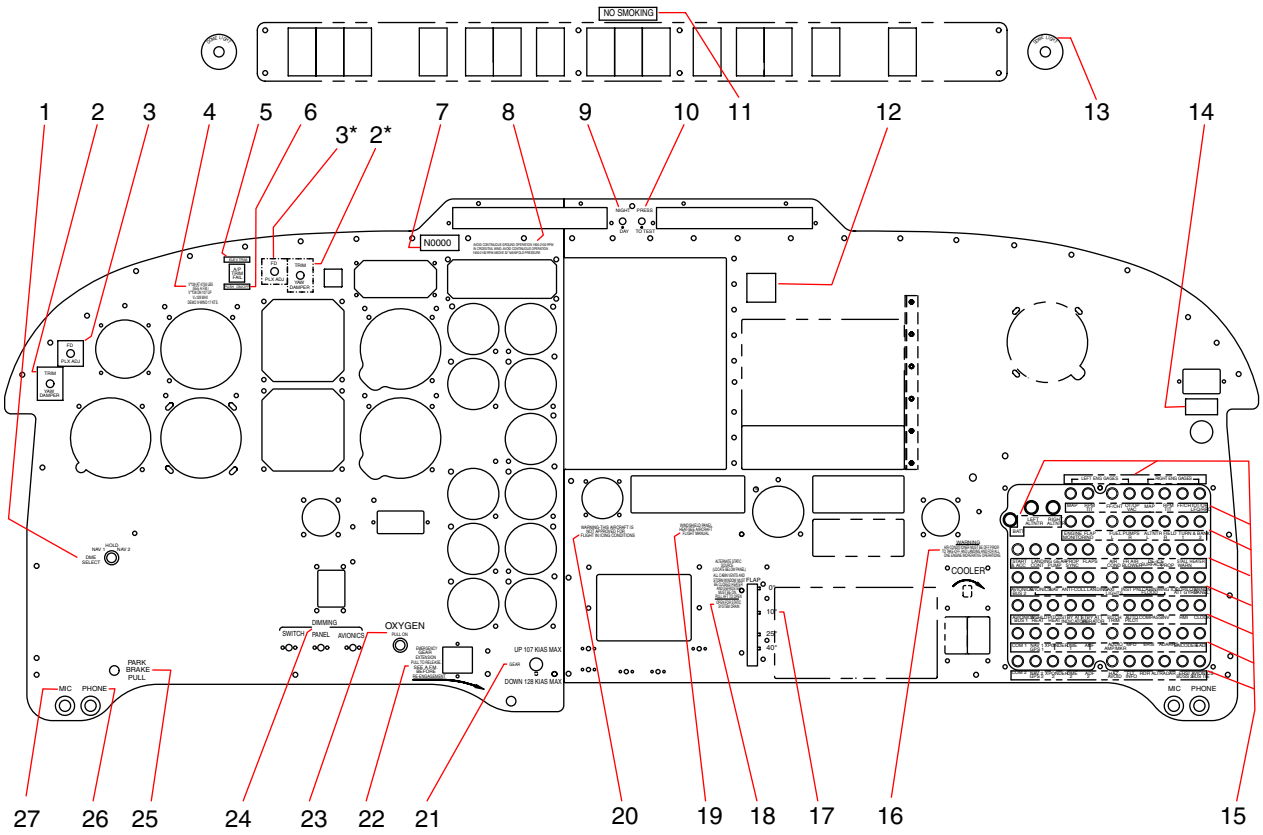


1. PLACARD - SPEED LIMITATIONS
2. PLACARD - AIRPLANE REGISTRATION NUMBER
3. PLACARD - AVOID CONTINUOUS GROUND OPERATION ...
4. PLACARD - NIGHT / DAY
5. PLACARD - PRESS TO TEST
6. PLACARD - NO SMOKING
7. PLACARD - DOME LIGHT
8. PLACARD - KLN-90 DATALOADER
9. PLACARD - CIRCUIT BREAKER
10. PLACARD - COOLER
11. PLACARD - WARNING - AIR CONDITIONER MUST BE OFF ...
12. PLACARD - FLAP TRAVEL
13. PLACARD - ALTERNATE STATIC SOURCE
14. PLACARD - WINDSHIELD PANEL HEAT
15. PLACARD - WARNING - THIS AIRCRAFT IS NOT APPROVED ...
16. PLACARD - GEAR UP / DOWN MAXIMUM SPEEDS
17. PLACARD - EMERGENCY GEAR EXTENSION
18. PLACARD - OXYGEN PULL ON
19. PLACARD - DIMMING - SWITCH, PANEL, AVIONICS
20. PLACARD - PARK BRAKE PULL
21. PLACARD - PHONE
22. PLACARD - MIC
23. PLACARD - ALTITUDE CORRECTION CARD
(AIRPLANES WITHOUT A/C - S/N'S 3449094 AND UP)

Instrument Panel Placards
Figure 3 (Sheet 2 of 8)

Effectivity
3449001 thru 3449151

87290



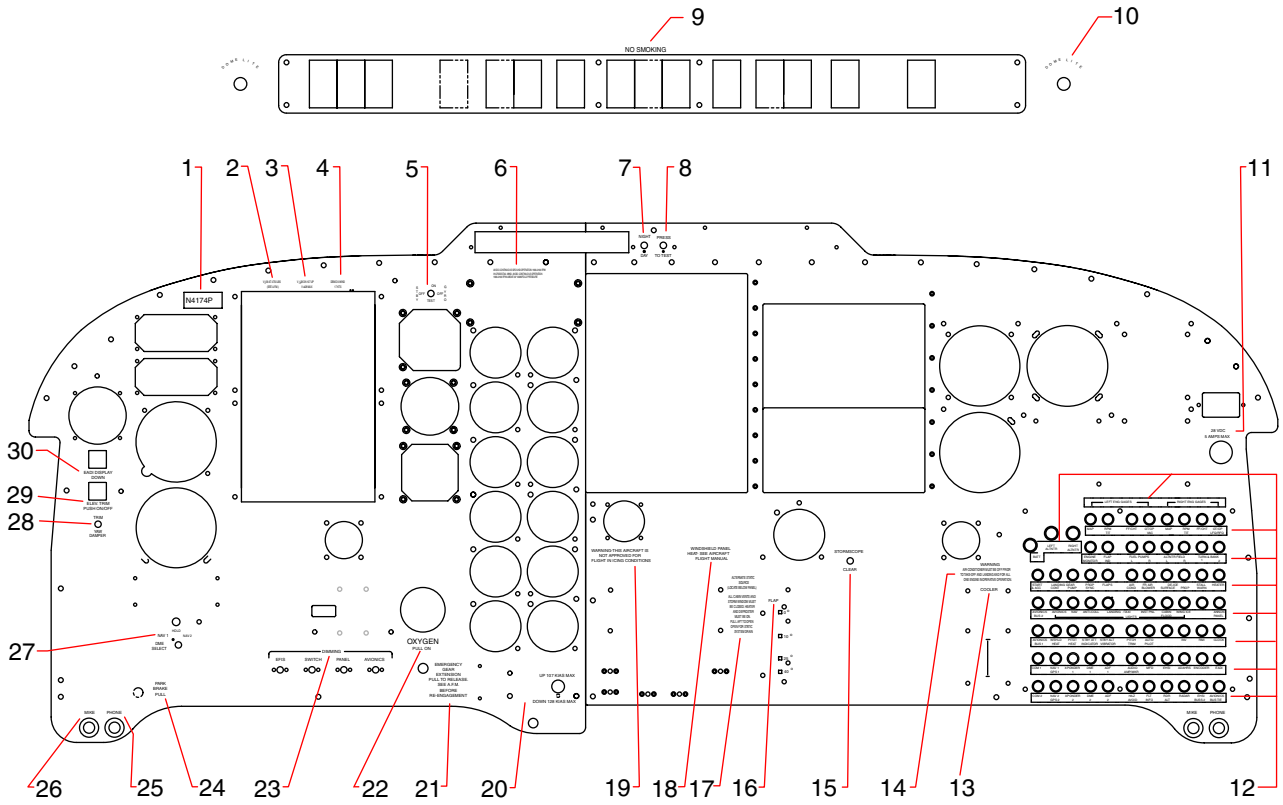
* LOCATED HERE ONLY ON S/N'S 3449152 THRU 3449177

1. PLACARD - DME SELECT
2. PLACARD - TRIM - YAW DAMPER
3. PLACARD - FD - PLX ADJ
4. PLACARD - V_A 139 AT 4750 LBS (SEE A.F.M.)
5. PLACARD - ELEV. TRIM
6. PLACARD - PUSH ON / OFF
7. PLACARD - AIRPLANE REGISTRATION NUMBER
8. PLACARD - AVOID CONTINUOUS GROUND OPERATION...
9. PLACARD - NIGHT / DAY
10. PLACARD - PRESS TO TEST
11. PLACARD - NO SMOKING
12. PLACARD - STORMSCOPE (OPTIONAL)
13. PLACARD - DOME LIGHT
14. PLACARD - 28V DC 5 AMPS
15. PLACARD - CIRCUIT BREAKER
16. PLACARD - WARNING - A/C MUST BE OFF...
17. PLACARD - FLAP TRAVEL
18. PLACARD - ALTERNATE STATIC SOURCE
19. PLACARD - WINDSHIELD PANEL HEAT
20. PLACARD - WARNING - THIS AIRCRAFT IS NOT APPROVED FOR...
21. PLACARD - GEAR UP / DOWN MAX SPEEDS
22. PLACARD - EMERGENCY GEAR EXTENSION
23. PLACARD - OXYGEN PULL ON
24. PLACARD - DIMMING - SWITCH, PANEL, AND AVIONICS
25. PLACARD - PARK BRAKE PULL
26. PLACARD - PHONE
27. PLACARD - MIC

Effectivity
3449152 thru 3449310,
3449312 thru 3449322,
3449324, 3449325,
3449410 & up

Instrument Panel Placards
Figure 3 (Sheet 3 of 8)

87290

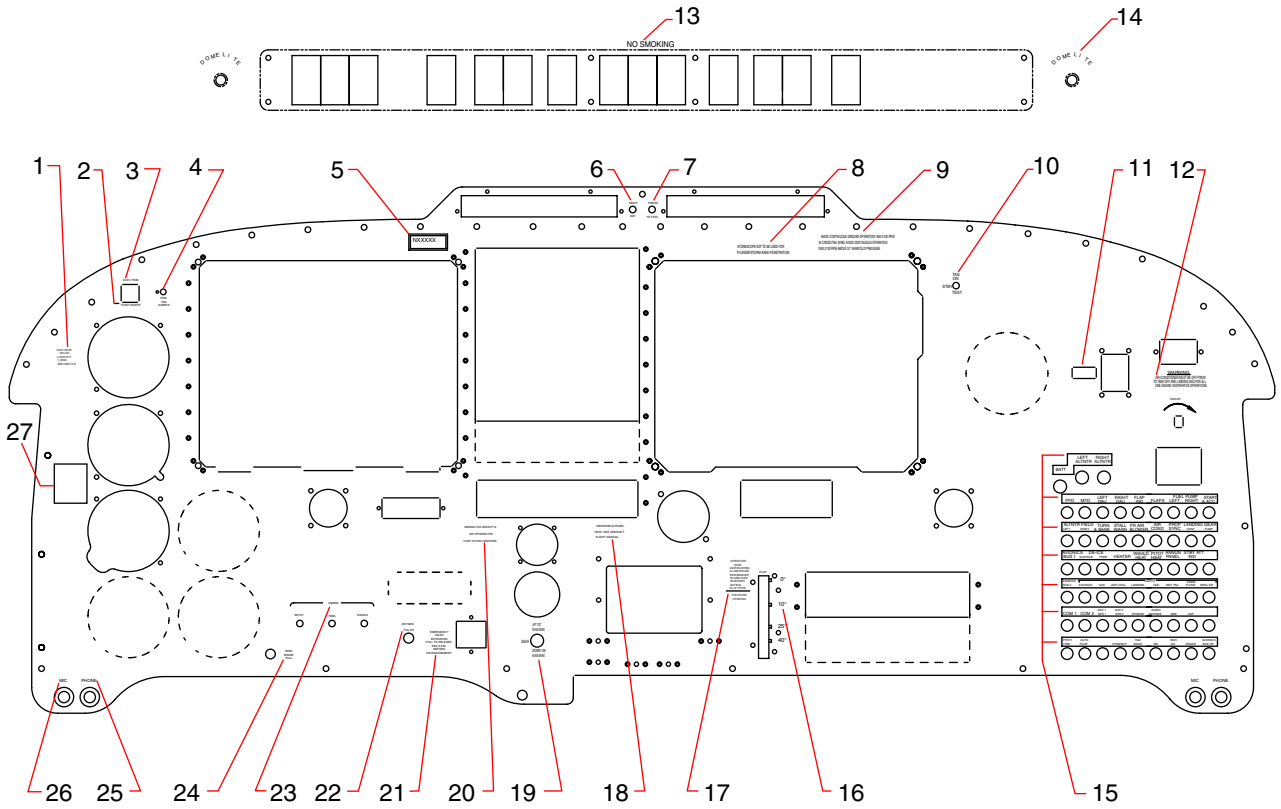


1. PLACARD - AIRPLANE REGISTRATION NUMBER
2. PLACARD - V_A 139 AT 4750 LBS (SEE A.F.M.)
3. PLACARD - V_{LO} 128 DN 107 UP - V_{LE} 128 MAX
4. PLACARD - X-WIND
5. PLACARD - GYRO
6. PLACARD - AVOID CONTINUOUS GROUND OPERATION...
7. PLACARD - NIGHT / DAY
8. PLACARD - PRESS TO TEST
9. PLACARD - NO SMOKING
10. PLACARD - DOME LIGHT
11. PLACARD - 28 VDC 5 AMPS MAX
12. PLACARD - CIRCUIT BREAKER
13. PLACARD - A/C COOLER
14. PLACARD - WARNING - A/C MUST BE OFF...
15. PLACARD - STORMSCOPE (OPTIONAL)
16. PLACARD - FLAP HANDLE
17. PLACARD - ALTERNATE STATIC SOURCE
18. PLACARD - WINDSHIELD PANEL HEAT
19. PLACARD - WARNING - THIS AIRCRAFT IS NOT APPROVED...
20. PLACARD - GEAR UP / DOWN MAX SPEEDS
21. PLACARD - EMERGENCY GEAR EXTENSION
22. PLACARD - OXYGEN PULL ON
23. PLACARD - DIMMING - EFIS, SWITCH, PANEL, AVIONICS
24. PLACARD - PARK BRAKE PULL
25. PLACARD - PHONE
26. PLACARD - MIC
27. PLACARD - DME SELECT
28. PLACARD - TRIM - YAW DAMPER
29. PLACARD - ELEV. TRIM - PUSH ON / OFF
30. PLACARD - EADI DISPLAY - DOWN

Instrument Panel Placards
Figure 3 (Sheet 4 of 8)

Effectivity
Seneca V
with Meggitt EFIS

87290

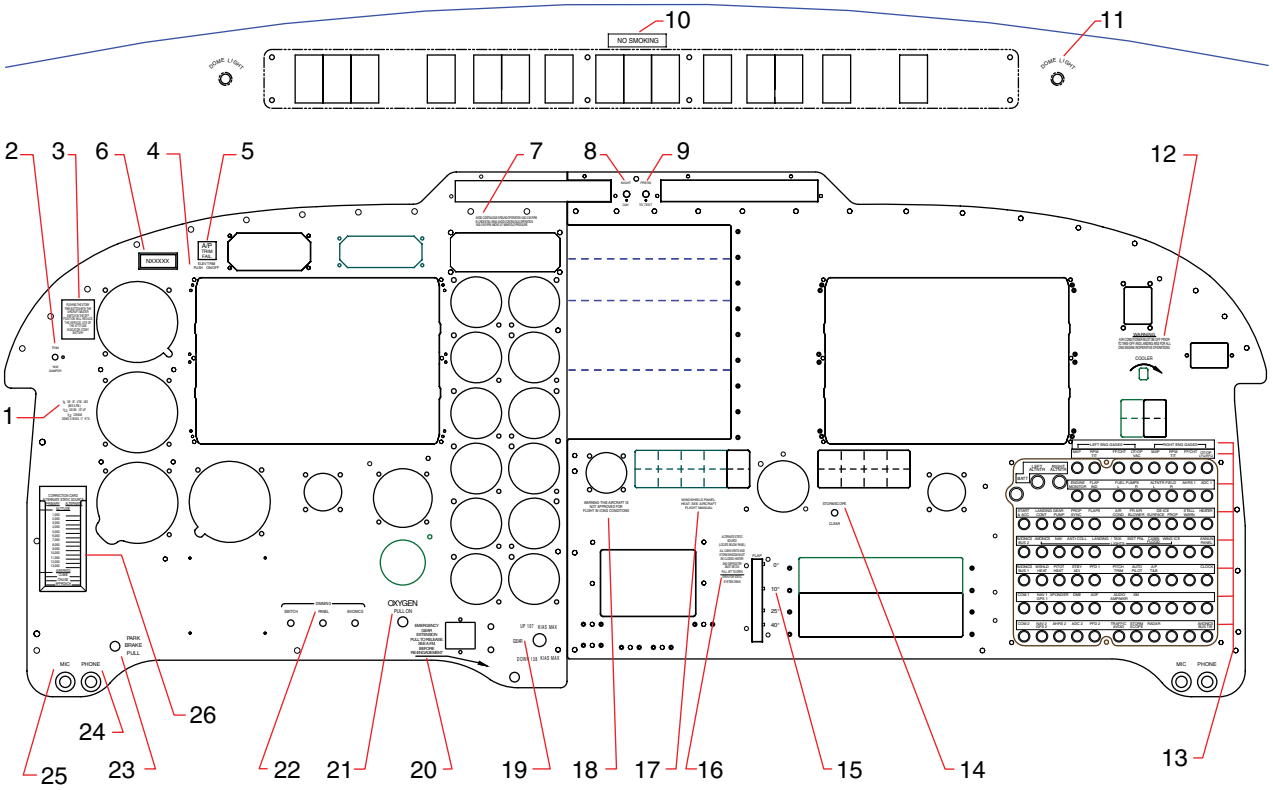


1. PLACARD - V_A 139 AT 4750 LBS (SEE A.F.M.)
2. PLACARD - PUSH ON / OFF
3. PLACARD - ELEV. TRIM
4. PLACARD - TRIM- YAW DAMPER (OPTIONAL)
5. PLACARD - AIRPLANE REGISTRATION NUMBER
6. PLACARD - NIGHT / DAY
7. PLACARD - PRESS TO TEST
8. PLACARD - STORMSCOPE (OPTIONAL)
9. PLACARD - AVOID CONTINUOUS GROUND OPERATION...
10. PLACARD - TAS
11. PLACARD - ELT CAUTION (OPTIONAL IN S/N'S 3449359 & UP)
12. PLACARD - WARNING - A/C MUST BE OFF...
13. PLACARD - NO SMOKING
14. PLACARD - DOME LIGHT
15. PLACARD - CIRCUIT BREAKER
16. PLACARD - FLAP TRAVEL
17. PLACARD - ALTERNATE STATIC SOURCE
18. PLACARD - WINDSHIELD PANEL HEAT
19. PLACARD - GEAR UP / DOWN MAX SPEEDS
20. PLACARD - WARNING - THIS AIRCRAFT IS NOT APPROVED FOR...
21. PLACARD - EMERGENCY GEAR EXTENSION
22. PLACARD - OXYGEN PULL ON
23. PLACARD - DIMMING - SWITCH, PANEL, AND AVIONICS
24. PLACARD - PARK BRAKE PULL
25. PLACARD - PHONE
26. PLACARD - MIC
27. DECAL - STANDBY ATTITUDE INDICATOR BATTERY (S/N'S 3449388 & UP, OR WITH KIT NO. 88435-002 INSTALLED.)

Effectivity
Seneca V
with Avidyne EFIS

Instrument Panel Placards
Figure 3 (Sheet 5 of 8)

107370 A
107333 B

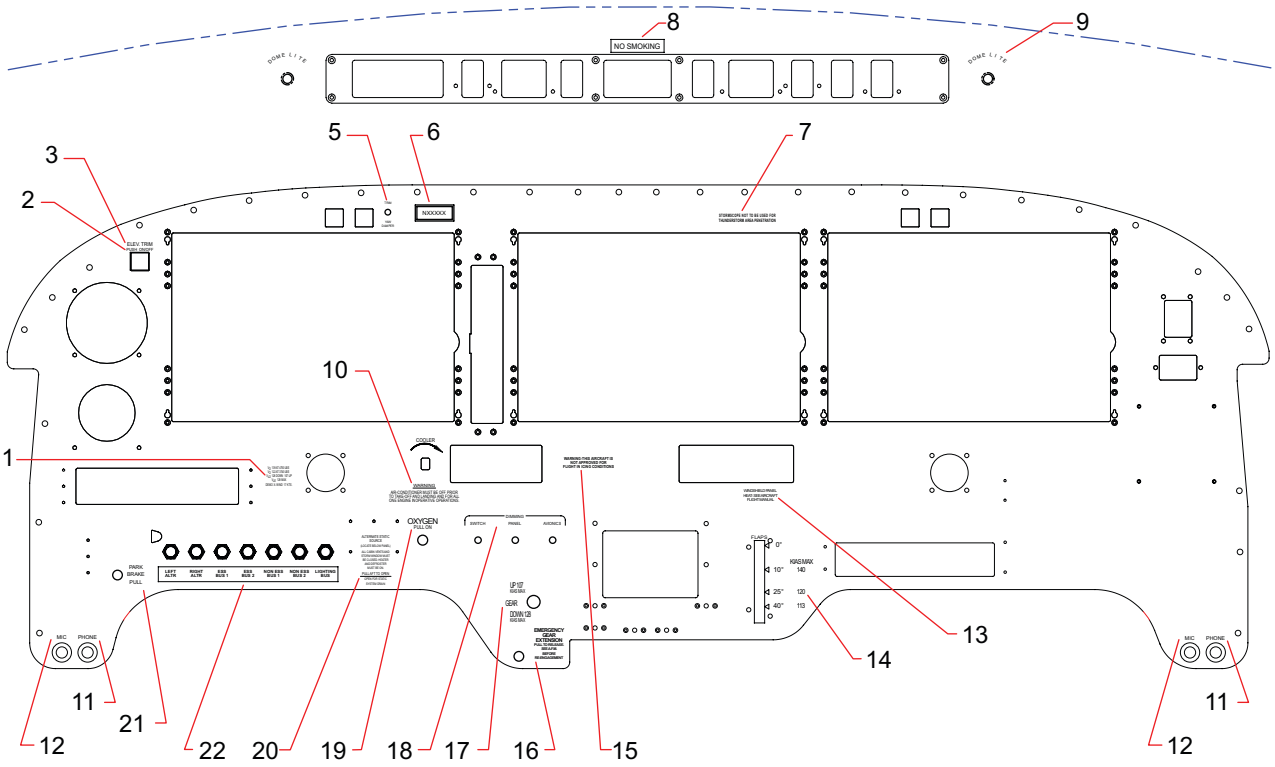


1. PLACARD - V_A 139 AT 4750 LBS (SEE A.F.M.)
2. PLACARD - TRIM- YAW DAMPER (OPTIONAL)
3. DECAL - STANDBY ATTITUDE INDICATOR BATTERY
4. PLACARD - PUSH ON / OFF
5. PLACARD - ELEV. TRIM
6. PLACARD - AIRPLANE REGISTRATION NUMBER
7. PLACARD - AVOID CONTINUOUS GROUND OPERATION...
8. PLACARD - NIGHT / DAY
9. PLACARD - PRESS TO TEST
10. PLACARD - NO SMOKING
11. PLACARD - DOME LIGHT
12. PLACARD - WARNING - A/C MUST BE OFF...
13. PLACARD - CIRCUIT BREAKER
14. PLACARD - STORMSCOPE (OPTIONAL)
15. PLACARD - FLAP TRAVEL
16. PLACARD - ALTERNATE STATIC SOURCE
17. PLACARD - WINDSHIELD PANEL HEAT (OPTIONAL)
18. PLACARD - WARNING - THIS AIRCRAFT IS NOT APPROVED FOR...
19. PLACARD - GEAR UP / DOWN MAX SPEEDS
20. PLACARD - EMERGENCY GEAR EXTENSION
21. PLACARD - OXYGEN PULL ON
22. PLACARD - DIMMING - SWITCH, PANEL, AND AVIONICS
23. PLACARD - PARK BRAKE PULL
24. PLACARD - PHONE
25. PLACARD - MIC
26. PLACARD - ALTERNATE STATIC (AS REQUIRED)

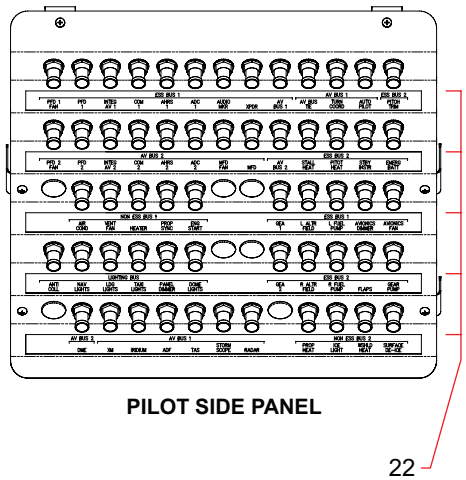
Instrument Panel Placards
Figure 3 (Sheet 6 of 8)

Effectivity
Seneca V
with Garmin G600

87290 CD 10
107250 L 2
107274 B

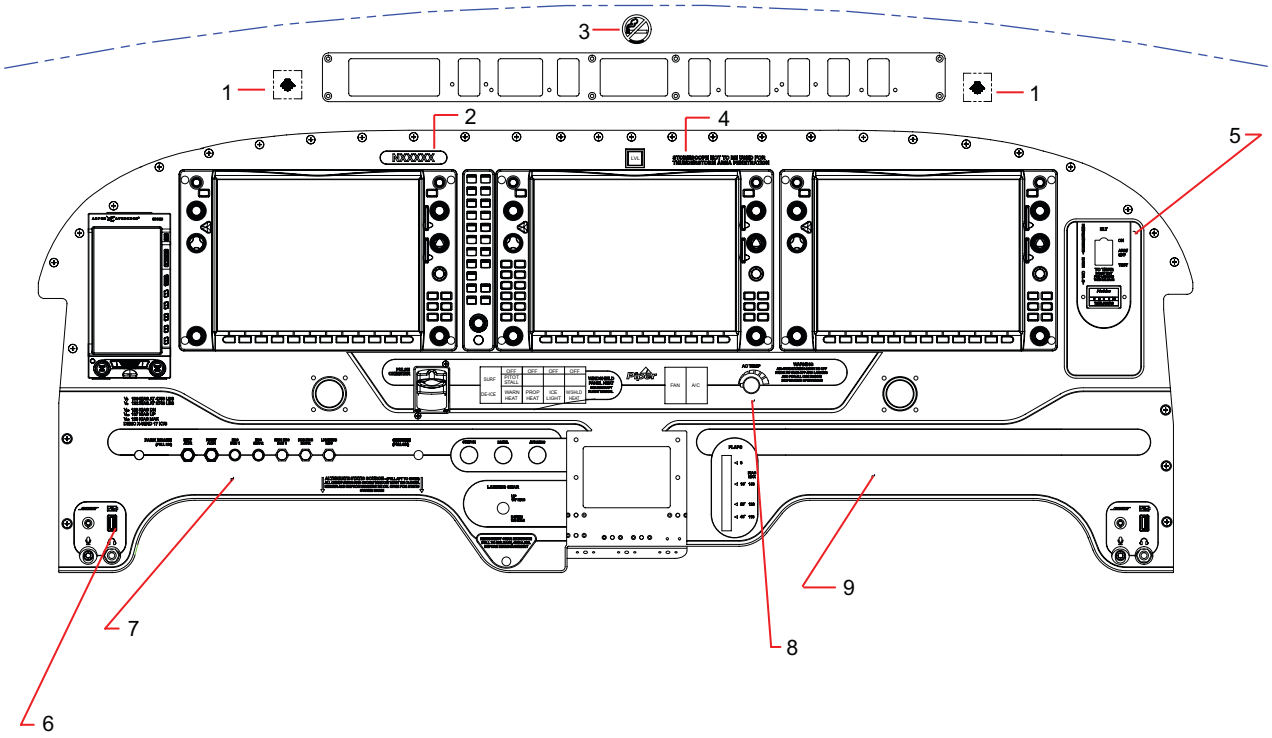


1. TRANSFER - V_O 139 AT 4750 LBS (SEE A.F.M.)
2. PLACARD - PUSH ON / OFF (AS REQUIRED)
3. PLACARD - ELEV. TRIM (AS REQUIRED)
4. DELETED
5. PLACARD - TRIM - YAW DAMPER (OPTIONAL, S-TEC ONLY)
6. PLATE - AIRPLANE REGISTRATION NUMBER
7. TRANSFER - STORMSCOPE (OPTIONAL)
8. DECAL - NO SMOKING
9. DECAL - DOME LITE
10. TRANSFER - WARNING - A/C MUST BE OFF…
11. DECAL - PHONE
12. DECAL - MIC
13. TRANSFER - WINDSHIELD PANEL HEAT (OPTIONAL)
14. LED PLACARD - FLAP TRAVEL WITH AIRSPEED
15. DECAL - WARNING - THIS AIRCRAFT IS NOT APPROVED FOR…
16. LED PLACARD - EMERGENCY GEAR EXTENSION
17. LED PLACARD - GEAR UP / DOWN MAX SPEEDS
18. TRANSFER - DIMMING - SWITCH, PANEL, AND AVIONICS
19. DECAL - OXYGEN PULL ON
20. LED PLACARD - ALTERNATE STATIC SOURCE
21. LED PLACARD - PARK BRAKE PULL
22. PLACARD - CIRCUIT BREAKER



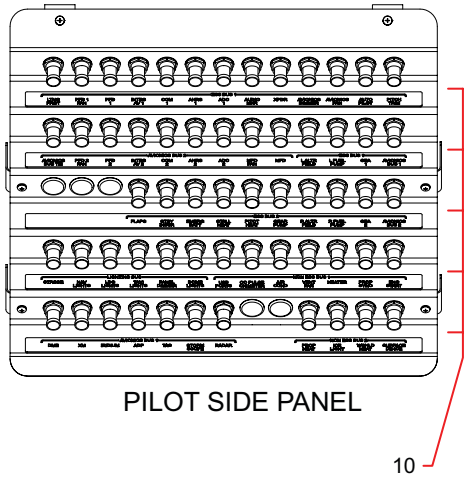
PILOT SIDE PANEL

34F24A003 A 2.0
34F24A000 NEW 6.0
87290 CD 11.0



INSTRUMENT PANEL VIEW LOOKING FWD

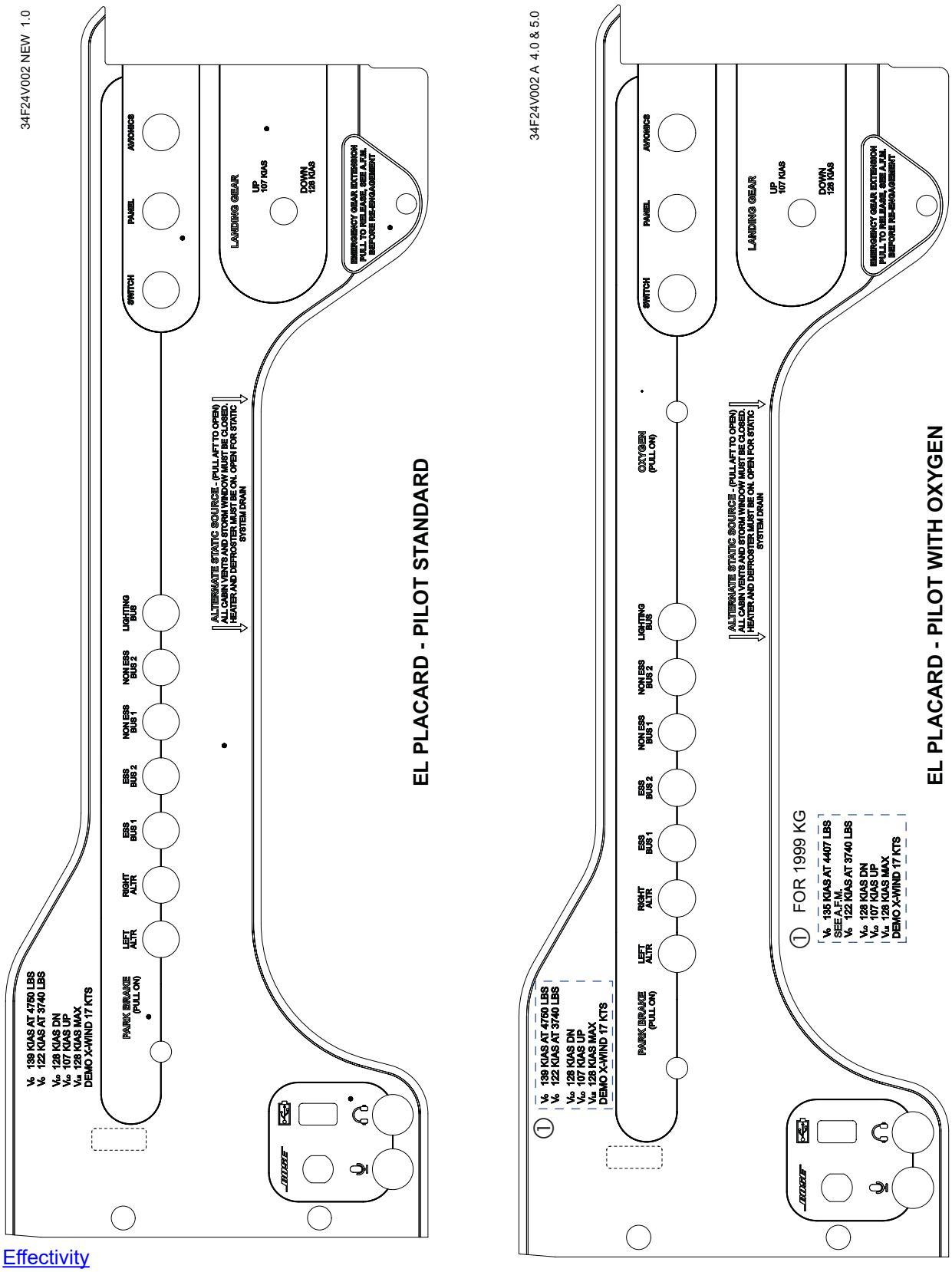
1. DECAL - DOME LIGHT
2. PLATE ASSY
3. DECAL - NO SMOKING
4. TRANSFER - STORMSCOPE (OPTIONAL)
5. PLACARD ASSY - ELT
6. SYSTEM INSTL - USB PORTS
7. PLACARD ASSY - PILOT STANDARD
   OR
   *PLACARD ASSY - PILOT W/OXYGEN
8. PLACARD ASSY - STANDARD
   OR
   *PLACARD ASSY - ALL OPTIONS
   OR
   *PLACARD ASSY - W/A.C.
   OR
   *PLACARD ASSY - W/DE-ICE
9. PLACARD ASSY - CO-PILOT
10. PLACARD - CIRCUIT BREAKER



PILOT SIDE PANEL

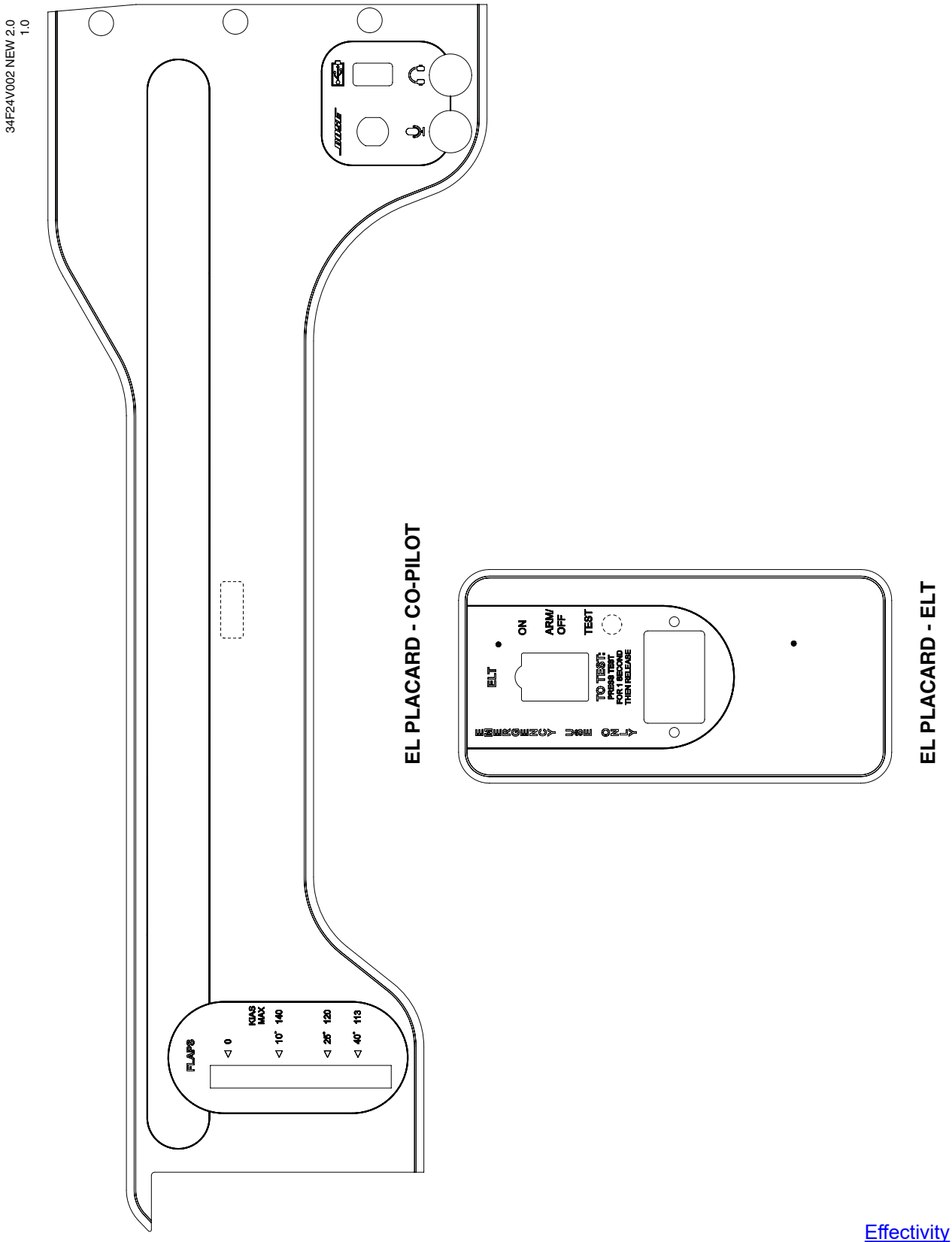
Instrument Panel Placards
Figure 3 (Sheet 8 of 8)

Effectivity
Seneca V
with Garmin G1000 NXi



Effectivity
Seneca V
with Garmin G1000 NXi

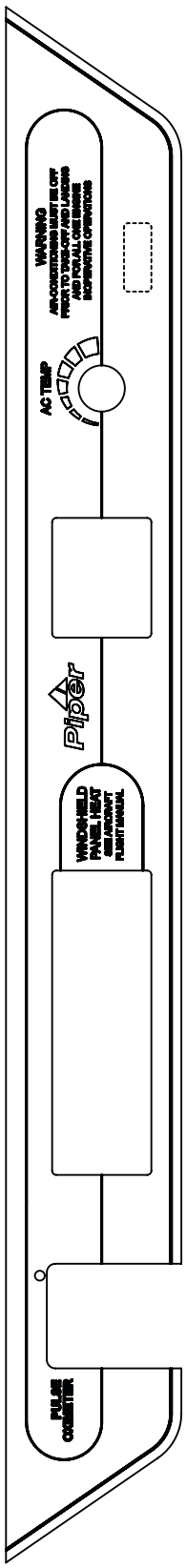
EL Placards - Instrument Panel
Figure 4 (Sheet 1 of 4)



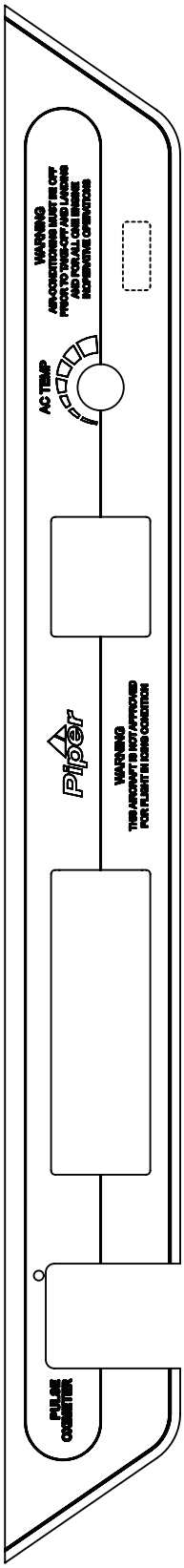
EL Placards - Instrument Panel
Figure 4 (Sheet 2 of 4)

Effectivity
Seneca V
with Garmin G1000 NXi

34F24V002 NEW 2.0
3.0



EL PLACARD - ALL OPTIONS

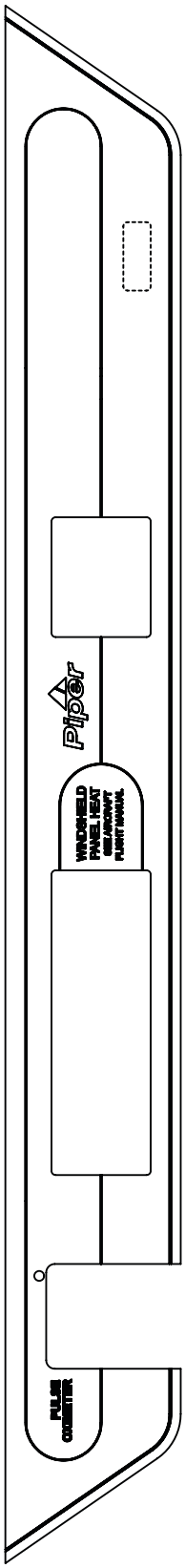


EL PLACARD - PILOT WITH AC

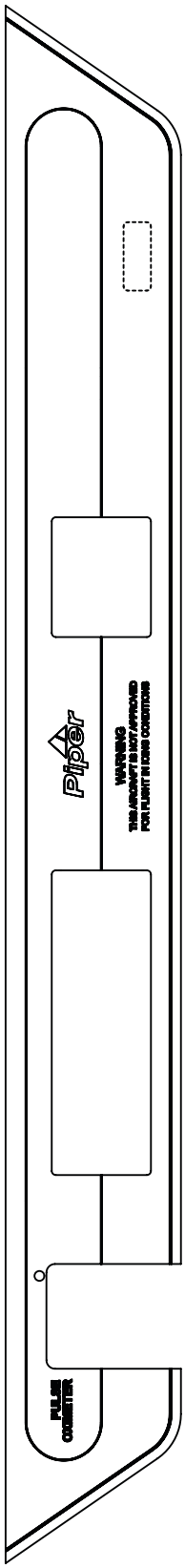
Effectivity
Seneca V
with Garmin G1000 NXi

EL Placards - Instrument Panel
Figure 4 (Sheet 3 of 4)

34F24V002 NEW 3.0



EL PLACARD - WITH DE-ICE



EL PLACARD - STANDARD

EL Placards - Instrument Panel
Figure 4 (Sheet 4 of 4)

Effectivity
Seneca V
with Garmin G1000 NXi

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CHAPTER

12

SERVICING

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CHAPTER 12

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	3	Sep 30/18
	4	Sep 30/18
	5	Sep 30/18
	6	Sep 30/18
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	2	Oct 10/23
	3	Oct 10/23
	4	Oct 10/23
	5	Oct 10/23
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	7	Oct 10/23
	8	Oct 10/23
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	10	Oct 10/23
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	7	Sep 30/18
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	11	Oct 10/23
	12	Oct 10/23
	13	Oct 10/23
	14	Oct 10/23

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CHAPTER 12 - SERVICING

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CHAPTER 12 - SERVICING

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Application of Oil		6
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GENERAL

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

This chapter covers all routine servicing of airplane, scheduled and non-scheduled, including replenishment of fuel, oil, brake fluid, oxygen (if installed), tire pressure, lubrication requirements, servicing of oleo struts with air and oil, etc. Pay special attention to all WARNINGS and CAUTIONS.

1. Aircraft Finish Care

WARNING: DO NOT USE GASOLINE, ALCOHOL, BENZENE, CARBON TETRACHLORIDE, THINNER, ACETONE OR WINDOW CLEANING SPRAYS TO CLEAN AIRPLANE.

The entire airplane is carefully finished inside and out to assure maximum service life. The external surfaces are coated with durable polyurethane enamel.

A. Axalta Coating System Imron 6000 Paint System

CAUTION: FAILURE TO OBSERVE THE PROPER "FINISH CARE" GUIDELINES MAY RESULT IN DAMAGE OR LOSS OF SHINE OF THE AIRCRAFT PAINT. IMPROPER CARE MAY ALSO VOID THE WARRANTY REGARDING THE AIRCRAFT FINISH.

Piper aircraft delivered in 1999 and later (Seneca V S/N's 3449097 and up), use the Axalta Coating System Imron 6000 paint system. The guidelines outlined below must be followed to prevent damage to the finish and ensure long paint life.

(1) For the first 30 days after painting:
   (a) Hand wash the aircraft often. Use fresh water only.
   (b) Avoid parking under trees or places where birds roost. If sap, bird droppings, or insect remains are discovered, rinse them off immediately. (Sap, bird droppings, or insect remains will damage the paint during this period.)

(2) For the first 120 days after painting:
   (a) To remove heavy soil, use mild liquid soap. Never use detergent.
   (b) DO NOT WAX THE AIRCRAFT WITHIN 120 DAYS OF PAINTING!

(3) For long term paint finish protection:
   (a) Park in a sheltered area whenever possible.
   (b) Never use a scraper to remove ice or snow from painted surfaces.
   (c) Never let avgas, oil, or hydraulic fluid stand on painted surfaces. (This will permanently damage the finish.)
   (d) Never wash the aircraft in the hot sun.
   (e) Never wipe the finish with a dry cloth, always use fresh water.
   (f) Avoid abrasive cleaners, chemicals, abrasive wax, or brushes.
   (g) Have paint nicks or scratches touched up as soon as possible to maintain the aircraft's corrosion protection.

To summarize, Piper airplanes using the Axalta Coating System paint system need special attention in the early days of ownership.

B. Cleaning

WARNING: DO NOT USE GASOLINE, KEROSENE, ALCOHOL, BENZENE, CARBON TETRACHLORIDE, THINNER, ACETONE OR WINDOW CLEANING SPRAYS TO CLEAN AIRPLANE.

CAUTION: IF PAINT IS LESS THAN SIX MONTHS, SEE "AXALTA COATING SYSTEM IMRON 6000 PAINT SYSTEM" ON PAGE 12001.

CAUTION: DO NOT DIRECT ANY STREAM OF WATER OR CLEANING SOLUTION AT THE OPENINGS IN THE PITOT HEAD, STATIC PORTS, ALTERNATE STATIC PORTS OR FUSELAGE DRAINS.

The airplane should be washed with a mild soap and water. Harsh abrasives or detergents used on painted or plastic surfaces could make scratches or cause corrosion of metal surfaces. Cover areas where cleaning solution could cause damage. To wash the airplane, the following procedure may be used:

(1) Flush away loose dirt with water.
(2) Apply cleaning solution with a rag, sponge or soft bristle brush.
(3) To remove stubborn oil and grease, use cloth dampened with naphtha.
(4) Where exhaust stains exist, allow solution to remain on the surface longer.
(5) Any good automotive wax may be used to preserve the painted surfaces. Soft cleaning cloths or a chamois should be used to prevent scratches when cleaning or polishing. A heavier coating of wax on the leading surfaces will reduce the abrasion problems in these areas.

2. Cleaning

A. Exterior Surfaces

See "Aircraft Finish Care" on page 12001, and "Windshield and Windows" on page 12003.

B. Engine Compartment

Before cleaning the engine compartment, place strips of tape on the magneto vents to prevent any solvent from entering these units.

(1) Place a pan under the engine to catch waste.

CAUTION: DO NOT SPRAY SOLVENT INTO THE ALTERNATOR, STARTER, VACUUM PUMP(S), AIR INTAKE AND ALTERNATE AIR INLETS.

(2) With the engine cowling removed, spray or brush the engine with solvent or a mixture of solvent and degreaser, as desired. It may be necessary to brush areas that were sprayed where heavy grease and dirt deposits have collected in order to clean them.
(3) Allow the solvent to remain on the engine from five to ten minutes; then rinse the engine clean with additional solvent and allow to dry.

CAUTION: DO NOT OPERATE ENGINE UNTIL EXCESS SOLVENT HAS EVAPORATED OR OTHERWISE BEEN REMOVED.

(4) Remove the protective covers from the magnetos.
(5) Lubricate controls, bearing surfaces, etc., per Lubrication Charts,12-20-00.

C. Landing Gear

(1) Struts and Torque Links

Before cleaning the landing gear struts and torque links, place a plastic cover or similar material over the wheel and brake assembly.

(a) Place a pan under the gear to catch waste.

(b) Spray (low pressure only) or brush the gear area with solvent or a mixture of solvent and degreaser.

(c) Allow the solvent to remain on the gear for five to ten minutes. Rinse gear with additional solvent and allow to dry.

(d) Remove cover from wheel and remove the catch pan.

(e) Lubricate gear per Lubrication Charts, 12-20-00.

(2) Wheels and Brakes

CAUTION: DO NOT USE HIGH PRESSURE SPRAY WASH EQUIPMENT. ITS USE CAN INJECT SOAP SOLUTION AND WATER INTO THE WHEEL BEARINGS AND OTHER INTERNAL CAVITIES RESULTING IN CORROSION AND REDUCED SERVICE LIFE.

(a) Hand wash wheels and brakes with a mild soap and water solution.

(b) Rinse with low-pressure spray.

(c) Lubricate gear per Lubrication Charts, 12-20-00, if not already done, above.

D. Windshield and Windows

WARNING: DO NOT USE GASOLINE, ALCOHOL, BENZENE, CARBON TETRACHLORIDE, THINNER, ACETONE, STRONG SOLVENTS OR WINDOW CLEANING SPRAYS. DO NOT USE PLASTIC CLEANER ON HEATED GLASS WINDSHIELDS.

(1) Remove dirt, mud, etc., from exterior surfaces with clean water.

(2) Wash with mild soap and warm water, or an aircraft plastic cleaner using a soft cloth or sponge and a straight rubbing motion. Do not rub surfaces harshly.

(3) Remove oil and grease with a cloth moistened with kerosene.

(4) After cleaning plastic surfaces, apply a thin coat of hard polishing wax. Rub lightly with a soft cloth. Do not use a circular motion.

(5) A superficial scratch or mar in plastic can be removed by using jeweler’s rouge to rub out the scratch. Smooth both sides and apply wax.

(6) To improve visibility through windshield and windows during flight through rain, a rain repellent such as REPCON should be applied to windshield and windows. The surfaces of the windshield and windows treated becomes so smooth that water beads up and readily flows off the surface. Apply this product in accordance with the manufacturer’s instructions. (See 91-10-00, Consumable Materials and Vendor Contact Information.)

E. Cleaning Interior

WARNING: SOLVENT CLEANERS REQUIRE ADEQUATE VENTILATION.

(1) Vinyl interior surfaces may be cleaned with a damp cloth and mild soap and water solution.

(2) Leather may be cleaned with a mild hand soap and water solution or with a saddle soap. Follow the precautions which apply to the cleaning of any fine leather product. Avoid saturation and never use detergents or harsh cleaning solutions on leather.

CAUTION: USE OF COMMON HOUSEHOLD CLEANERS AND POLISHES ON WOOD LAMINATED SURFACES COULD BE VERY HARMFUL.

(3) Wood laminated surfaces should be maintained using only a high grade furniture wax.

(4) All upholstery fabrics are Scotchgard treated and may be cleaned as follows:

(a) Spilled oily and watery liquids will generally bead up on the fabric and can be blotted away leaving little or no stain. Blot spills up as quickly as possible with an absorbent cloth, tissue or sponge. If the material is a solid or semi-solid, such as butter, remove the excess by gently scraping with a table knife. Often, blotting will remove all traces of stain but if the staining agent is not completely removed by blotting, the following techniques are suggested:

(b) Water-based stains such as ketchup, milk, ice cream, coffee:

1) Wipe the stain with a cloth wet with water containing a detergent or ammonia (4 fluid ounces of ammonia to one gallon of water). Repeat if necessary.

2) Oil based stains such as salad dressing, butter or mayonnaise may be removed by either of the following procedures:

a) Apply “Texize K-2R Spot Remover” by spraying or rubbing into the fabric and let dry. Vacuum off the residual powder. Repeat if necessary.

b) Wet a cloth with a solvent type spot cleaner such as “Energine” or “Renuzit” and wipe or gently rub the stained area. Turn cloth and rewet with solvent often. Repeat until stain disappears.

3) To remove residual detergent left on the fabric, wipe the entire fabric surface with a cloth dampened with water. The cloth should be rinsed in clean water several times. This procedure will ensure that the treatment will continue to function.

NOTE: Test the cleaner on an inconspicuous portion of the fabric to test for discoloration. Also avoid soaking or harsh rubbing.

(5) Headliner, Side Panels and Seats

Clean headliner, side panels, and seats with a stiff brush and vacuum where necessary.

(6) Carpets

WARNING: SOLVENT CLEANERS REQUIRE ADEQUATE VENTILATION.

Use a small whisk broom or vacuum cleaner to remove dirt. For soiled spots, use a nonflammable dry cleaning fluid. Floor carpets may be removed and cleaned like any household carpet.

F. Cleaning Surface Deicing Equipment

The deice boots should be cleaned when the aircraft is washed using a mild soap and water solution.

In cold weather, wash the boots with the airplane inside a warm hangar if possible. If the cleaning is to be done outdoors, heat the soap and water solution before taking it out to the airplane. If difficulty is encountered with the water freezing on boots, direct a flow of warm air along the region being cleaned, using a portable type ground heater.

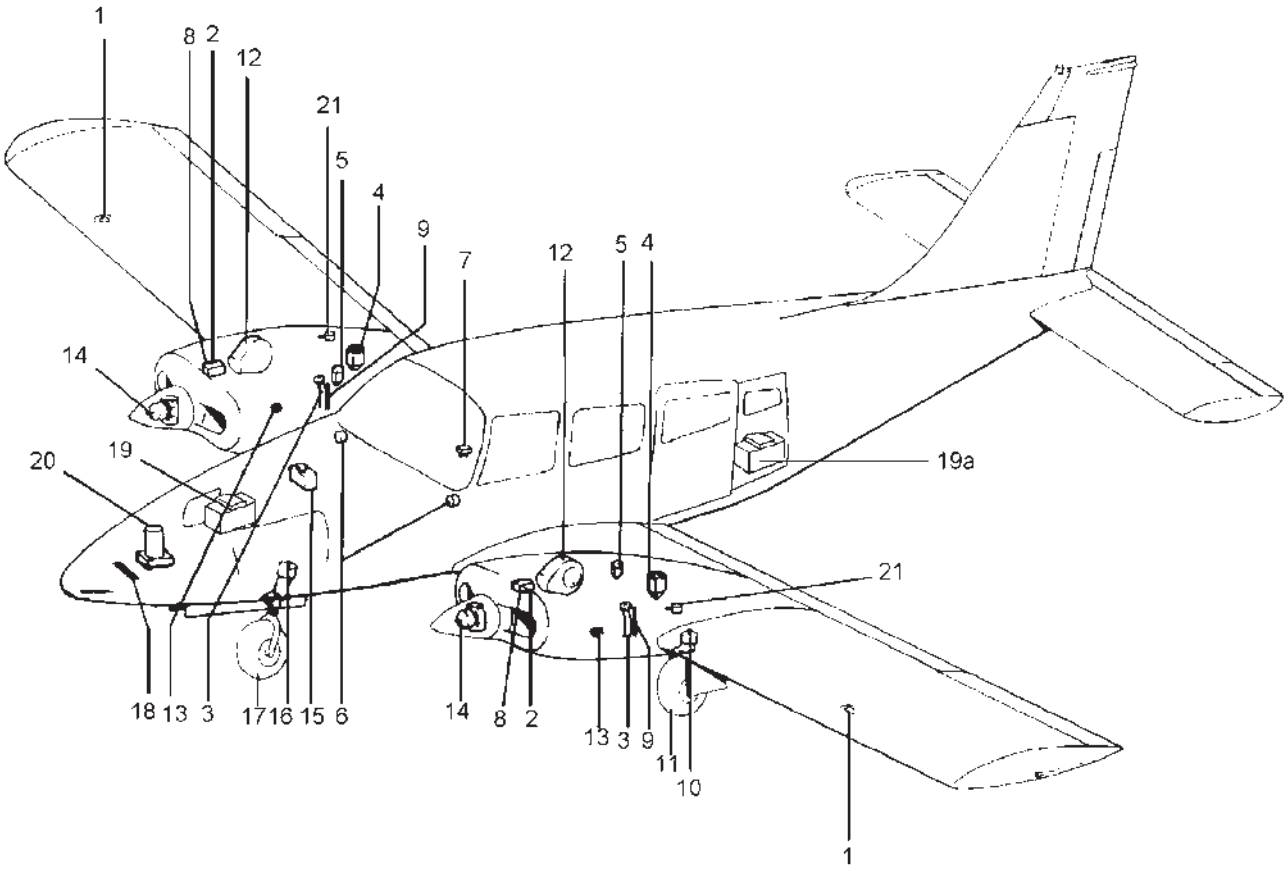
CAUTION: PETROLEUM PRODUCTS ARE INJURIOUS TO RUBBER AND THEREFORE SHOULD BE USED SPARINGLY IF AT ALL.

As an alternate cleaning solvent, use benzol or non-leaded gasoline. Moisten the cleaning cloth in the solvent, scrub lightly and then with a clean dry cloth, wipe dry so that the cleaner does not have time to soak into the rubber.

When deice boots are clean, a coating of B.F. Goodrich Icex should be applied. Icex is compounded to lower the strength of adhesion between ice and rubber surface of the deice boots.

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REPLENISHING



1. FUEL FILLERS
2. FUEL INJECTORS & STRAINERS
3. ENGINE OIL FILL
4. ENGINE OIL FILTERS
5. FUEL FILTERS
6. INSTRUMENT AIR FILTERS
7. FUEL SYSTEM DRAINS
8. FUEL METERING CONTROL UNIT
9. OIL DIPSTICK
10. MAIN GEAR STRUT
11. MAIN GEAR TIRES
12. INDUCTION AIR FILTERS
13. ENGINE OIL SUCTION SCREEN
14. PROPELLER AIR CHARGE
15. BRAKE RESERVOIR
16. NOSE GEAR STRUT
17. NOSE GEAR TIRE
18. EXTERNAL POWER RECEPTACLE
19. BATTERY (Seneca IV)
19a. BATTERY (Seneca V)
20. HYDRAULIC RESERVOIR
21. VACUUM REGULATOR FILTER

Service Points
Figure 5

1. Fuel System

A. Fuel Filter

At intervals of 50 hours or 90 days, whichever comes first, clean the fuel screen/filter (i.e. - strainer) per Fuel Filters in 28-20-00.

B. Fuel Tanks

(1) Fuel Additives

CAUTION: WHEN USING ADDITIVES, MAKE SURE THE CORRECT PROCEDURES ARE FOLLOWED. WHEN REFUELING, OBSERVE ALL SAFETY PRECAUTIONS AND USE THE FUEL SPECIFIED ON THE FILLER PLACARD.

CAUTION: MAKE SURE THAT THE ANTI-ICING ADDITIVE IS DIRECTED INTO THE FLOWING FUEL STREAM. ADDITIVE FLOW SHOULD START AFTER AND STOP BEFORE FUEL FLOW. DO NOT PERMIT THE CONCENTRATED ADDITIVE TO COME INTO DIRECT CONTACT WITH PAINTED SURFACES OR INTERIOR SURFACES OF THE FUEL TANKS.

CAUTION: SOME FUELS HAVE BEEN PRE-BLENDED WITH ANTI-ICING ADDITIVES. NO FURTHER BLENDING SHOULD BE PERFORMED. FUEL ADDITIVES DO NOT ELIMINATE PREFLIGHT FUEL DRAINING.

Ice contamination can be prevented by using an anti-icing additive that meets the requirements of MIL-DTL-85470.

When an additive is used, it must be uniformly blended with the fuel while refueling, and not exceed 0.15% by volume of the refueled quantity. The blend should not be less than 0.10% by volume. A good example would be 1-1/2 liquid ounces per 10 gallons of fuel. For the best results, follow the manufacturer's mixing or blending instructions. A blender supplied by the manufacturer should be used. Except for the information contained in this section, the manufacturer's mixing or blending instructions should be carefully followed.

(2) Filling

Each fuel system is filled through a single filler neck in its outboard tank. The standard fuel system involves two 24.5 gallon tanks per side and one 15 gallon bladder cell per side comprising a total of 64 gallons per side.

(3) Water Contamination

WARNING: PILOTS, OWNERS, OPERATORS, MAINTENANCE, AND SERVICE PERSONNEL SHOULD ASSUME SOME WATER EXISTS IN THE FUEL SYSTEM.

Water may enter the fuel tank system via any penetration in the wing fuel tank and from moisture condensation inside the tank. Water in the fuel may come out of solution, settle and make its way to a drain location in the form of a blob, pea, or BB-shaped translucent mass found at the bottom of the sampler cup. Water suspended in the fuel may lead to a cloudy or hazy appearance in the sampler cup. Water may have dissolved in the fuel, but conditions have not yet occurred to cause the water to come out of solution and perhaps adhere to the dry tank upper surface or walls (similar to condensation).

See FAA Special Airworthiness Information Bulletin (SAIB) No. CE-12-06 for additional information.

(4) Draining Moisture

CAUTION: WHEN DRAINING ANY AMOUNT OF FUEL, ENSURE THAT NO FIRE HAZARD EXISTS BEFORE STARTING ENGINE.

The fuel systems are designed to allow moisture and foreign matter to be drained at the systems' lowest points. Two drains are located under the wing (see placards) from the inboard tank and fuel filter. Quick drains for the two systems are mounted underneath the fuselage.

CAUTION: AFTER EACH USE OF THE QUICK DRAIN VALVE, CHECK THE FUEL SELECTOR VALVE DRAIN TO ENSURE THAT THE QUICK DRAIN VALVE HAS PROPERLY SEATED AND THAT THERE IS NO LOSS OF FUEL FROM THE DRAIN.

The fuel system should be drained daily prior to first flight and after refueling to avoid the accumulation of water sediment. Each aluminum fuel tank is equipped with an individual quick drain located at the lower inboard rear corner of the tank. This allows each wing to be drained individually. The fuel selector valve is provided with a quick drain valve located on the forward face of the spar box.

NOTE: There is no single point of drainage that can be used to check for all fuel system contaminants simultaneously.

Drain fuel tanks and selector valve per the following:

With the airplane in the normal ground attitude and starting at the highest drain location, check all drain locations for contaminants before every flight, whether or not refueling has occurred. Have fuel sample disposal provisions and proper lighting at your disposal to properly check for fuel tank system contamination.

(a) Drain at least one cup of fuel (using a clear sampler cup) from each drain location:
    1) Drain each tank through its individual quick drain located at the lower inboard rear corner of the tank;
    2) Drain the fuel strainer as required to completely flush its contents in each of the fuel selector positions. Drain the fuel strainer by opening the quick drain on the strainer.

(b) Check for water, clarity, cloudiness, haze, proper fuel type/grade (i.e.; 100LL is light blue in tint, jet fuel is clear or yellowish), odor, or other contaminants.

(c) Allow time between fueling and draining. It takes time for any contaminates to settle to sump area prior to draining tanks.

(d) If any contamination is detected in the fuel tank system, thoroughly drain all drain locations again.

(e) If contamination is observed, take further samples until the fuel appears clear, and gently rock the airplane in both the roll and pitch axis to move any additional contaminants to the drain points.

(f) Take repeated samples from all drain locations until all contamination has been removed.

(g) If contaminants are still present, do not fly the airplane. Have qualified maintenance personnel drain and purge the fuel tank system. Remove all evidence of contamination prior to further flight.

(5) Draining Entirely

CAUTION: WHEN DRAINING ANY AMOUNT OF FUEL, ENSURE THAT NO FIRE HAZARD EXISTS BEFORE STARTING ENGINE.

Fuel may be drained from the system by opening the valve at the inboard end of each aluminum fuel tank. The flush type drain valve requires the drain cup pin to hold the valve open. The remaining fuel in the system may be drained through the filter bowl. Either wing may be drained by closing the selector valve and then draining as desired.

C. Flushing Tanks and Selector Valve

(1) To flush the fuel tanks and selector valve, disconnect the fuel line at the carburetor (Seneca IV) or fuel injector (Seneca V).

(2) Select a fuel tank, turn on the electric fuel pump and flush fuel through the system until it is determined there is no dirt and foreign matter in the fuel valve or tank. During this operation, agitation of the fuel within the tank will help pick up and remove any dirt.

(3) Repeat this procedure for each tank.

(4) When all tanks are flushed, clean all filters.

2. Oil System

CAUTION: DO NOT INTRODUCE ANY TRADE ADDITIVE TO THE BASIC LUBRICANT UNLESS RECOMMENDED BY THE ENGINE MANUFACTURER.

CAUTION: THE ENGINE MANUFACTURER DOES NOT RECOMMEND OILS BY BRAND NAMES. USE A QUALITY BRAND AVIATION GRADE OIL OF THE PROPER SEASON VISCOSITY.

Engine oil level should be checked before each flight. The engine oil and full flow cartridge filter should be changed every 50 hours or four months, whichever occurs first. Use a quality brand Aviation Grade Oil of the proper season viscosity.

A. Oil Sump

NOTE: Warm the engine to operating temperature before draining to ensure complete drainage.

(1) Draining

To drain the oil sump, provide a suitable container with a minimum capacity of that required to fill the sump. Remove the engine cowl and open the oil drain located on the underside of the engine by pushing the arms of the drain up and turning counterclockwise. This will hold the drain in the open position. It is recommended the engine be warmed to operating temperature to ensure complete draining of the old oil.

(2) Filling

NOTE: See the latest revision of Continental Aerospace Technologies Publication M-0, “Standard Practice Maintenance Manual,” Chapter 3, for oil specifications for new or overhauled engine break-in.

The oil sump should normally be filled with oil to the mark on the engine dipstick. The quantity of oil required for the engine may be found in Chart 1, 6-00-00. The specified grade of oil may be found in the Power Plants and Propellers Lubrication Chart, 12-20-00, on the inside surface of the engine oil filler access door; or in the appropriate vendor publication. To service the engine with oil, open the quick release oil filler access door on top of the cowl, and remove the oil filler cap with dipstick.

(3) Oil Screen (Suction)

The oil suction screen, located on the bottom aft end of the accessory case is installed horizontally. To remove, cut the safety wire and remove the hex head plug. The screen should be cleaned at each oil change to remove any accumulation of sludge and to examine for metal filings or chips. If metal particles are found in the screen, the engine should be examined for internal damage. To avoid possible damage to the screen, after cleaning and inspection, place the screen inside the recess in the hex head plug, and insert the screen into the housing. When certain that the screen is properly seated, tighten and safety the plug with MS-20995-C41 safety wire.

(4) Recommendations for Changing Oil

NOTE: Change the oil and filter each 50 hours of operation or every four months, whichever occurs first.

Always start and warm the engine to operating temperature before performing an oil change. While draining the oil, the screens should be removed from the crankcase cover and cleaned thoroughly. If sludge deposits are heavy, subsequent oil changes should be made at shorter intervals. Use type and weight of oil shown in the Power Plants and Propellers Lubrication Chart, 12-20-00. When the average ambient air temperature is approximately at the dividing line, use the lighter oil.

(a) A change to ashless dispersant oil should be made with a degree of caution in engines that have been operating on straight mineral oil for several hundred hours, since the cleaning action of some additive oils will tend to loosen sludge deposits and cause plugged oil passages. On any engine that has been operating on straight mineral oil, and is known to be in excessive dirty condition, do not switch to an additive or compounded oil until the engine has been overhauled.

(b) When changing from straight mineral oil to compounded oil, the following precautionary steps should be taken:

1) Do not add additive oil to straight mineral oil. Drain the straight mineral oil from the engine and fill with additive oil.

2) Do not operate the engine longer than five hours before the first oil change.

3) Check all oil screens for evidence of sludge or plugging and change oil every ten hours if sludge conditions are evident. Resume normal oil drain periods after sludge conditions improve.

B. Oil Filter (Full Flow)

(1) The oil filter should be replaced after each 50 hours of engine operation. This is accomplished by removing the lockwire from the bolt head at the end of the filter housing, loosening the bolt, and removing the filter assembly from the adapter.

(2) Before discarding the throwaway filter, remove the element for inspection by using a Champion cutter tool, CT-470, available from Champion Spark Plug Co., Toledo, Ohio 43601. It will cut open any spin-on type oil filter for inspection. Examine the material trapped in the filter for evidence of internal engine damage, such as chips or particles from bearings. In new or newly overhauled engines, some small particles of metallic shavings might be found; these are generally of no consequence and should not be confused with particles produced by impacting, abrasion or pressure. Evidence of internal engine damage found in the oil filter justifies further examination to determine the cause.

(3) After the filter has been replaced, tighten the cartridge to 18 to 20 foot-pounds of torque. Lockwire the bolt through the loops on the side of the housing to the drilled head of the thermostatic valve. Be sure the lockwire is replaced at both the attaching bolt head and the thermostatic oil cooler bypass valve. Use MS-20995-C41 safety wire.

3. Landing Gear

The landing gear consists of tires, brakes and oleo strut assemblies. These should be inspected for scored piston tubes, possible hydraulic fluid leakage and security and condition of all connection points. Check the brake linings for wear and frayed edges, and brake discs for scoring. Replace if necessary. Minor servicing is described in the following paragraphs. For more servicing, repair, and overhaul instructions, see 32-10-00 and 32-20-00.

A. Oleo Struts

(PIR-PPS50045, Rev. T.)

Air-oil struts are incorporated in each landing gear oleo to absorb the shock resulting from the impact of the wheels on the runway during landing. For landing gear and/or oleo strut repair procedures, see 32-10-00 (Main Gear) and 32-20-00 (Nose Gear).

WARNING: DO NOT RELEASE AIR BY REMOVING THE STRUT VALVE CORE OR FILLER PLUG. DEPRESS THE VALVE CORE PIN UNTIL STRUT CHAMBER PRESSURE HAS DIMINISHED.

CAUTION: CHECKING VISIBLE PISTON EXTENSION IS NOT AN ACCEPTABLE ALTERNATIVE TO VERIFYING THE PRESSURE IN THE LANDING GEAR ASSEMBLIES.

CAUTION: DIRT AND FOREIGN PARTICLES ACCUMULATE AROUND THE FILLER PLUGS OF THE LANDING GEAR STRUTS. THEREFORE, BEFORE ATTEMPTING TO REMOVE THESE PLUGS, THE TOPS OF THE STRUTS SHOULD BE CLEANED WITH COMPRESSED AIR AND/OR WITH A DRY SOLVENT.

(1) Fluid and Nitrogen Levels Check

Check levels for each oleo strut as follows:

(a) Raise the airplane on jacks (see 7-10-00).

(b) With the strut extended, remove the cap from the air valve at the top of the housing.

(c) Depress the valve core to allow air to escape from the strut piston until it is fully compressed.

(d) Allow the foam from the air-oil mixture to settle and then determine if oil is visible up to the bottom of the filler plug hole.

(e) If fluid is visible up to the bottom of the plug hole, then check the valve for unsatisfactory conditions, and proceed to step (g).

(f) If, however, the fluid is below the bottom of the filler plug hole, then the oleo should be checked for leaks or other conditions. If no condition exists, then, as applicable, add oil as described in “Filling Nose Gear Oleo Strut” or “Filling Main Gear Oleo Struts”, below.

(g) Proceed with nitrogen (or dry air) servicing in “Inflating Oleo Struts” on page 12109.

NOTE: Acceptable pressure levels are 250 ± 25 psi for the main gear struts; 120 ± 12 psi for the nose gear strut.

(2) Filling Nose Gear Oleo Strut

Fill the nose gear oleo strut with hydraulic fluid (MIL-PRF-5606), whether it be the addition of a small or large amount, as follows:

(a) Place the airplane on jacks (see 7-10-00) and raise the nose gear completely off the ground.

(b) Place a pan under the gear to catch spillage.

(c) If not previously accomplished, remove the engine cowl and relieve air from the strut housing chamber by removing the cap from the air valve and depressing the valve core.

(d) Fill the nose gear strut chamber by one of the following two methods:

1) Method 1 (Addition of small amount of fluid.):

a) Remove the valve core from filler plug at the top of strut housing. Do not remove the filler plug.

b) Attach one end of a clear plastic hose to the valve stem of the filler plug and submerge the other end into a container of hydraulic fluid. Ascertain that the end of the hose on the valve stem is tight and the fluid container is approximately equal in height to the top of the strut housing.

NOTE: An airtight connection is necessary between the plastic tube and the valve stem. Without such a connection, a small amount of air will be sucked into the oleo strut during each sequence, resulting in an inordinate amount of air bubbles and prolonged filling operations.

c) Fully compress and extend strut to draw fluid from the fluid container and expel air from strut chamber. By watching the fluid pass through the plastic hose, determine when the strut is full and no air is present in the chamber.

d) When air bubbles cease to flow through hose, compress the strut fully and remove the hose from the valve stem.

e) With strut compressed, remove filler plug to determine that the fluid level is visible up to the bottom of filler plug hole.

f) Reinstall the filler plug core. Apply thread lubricant (MIL-PRF-907) to threads of filler plug. Reinstall the filler plug in the top of strut housing. Torque plug from 350 to 400 in-lb.

2) Method II (Filling completely empty struts.):

a) Remove the filler plug from top of strut housing.

b) Raise the strut piston until fully compressed.

c) Pour fluid from a clean container through the filler opening until it reaches the bottom of the filler plug hole.

d) Install the filler plug finger tight. Extend and compress the strut two or three times to remove any air that may be trapped in housing.

e) Remove the filler plug. Fully compress the strut. Fill with fluid if needed.

f) Apply thread lubricant (MIL-PRF-907) to threads of filler plug and install filler plug in the top of strut housing. Torque plug from 350 to 400 in-lb.

(e) With airplane raised, compress and extend the gear strut several times. Ensure strut actuates freely. The weight of the gear fork and wheel should extend strut.

(f) Clean off any fluid overflow, and inflate strut with nitrogen (or dry air) as described in “Inflating Oleo Struts” on page 12109.

(g) Check that fluid is not leaking from around strut piston at bottom of housing.

(3) Filling Main Gear Oleo Struts

Fill the nose gear oleo strut with MIL-PRF-5606 fluid, whether it be the addition of a small or large amount, as follows:

(a) Place the airplane on jacks (see 7-10-00) and raise the main wheels off the ground.

(b) Place a pan under the gear to catch spillage.

(c) If not previously accomplished, remove caps on top wing to gain access to the top of strut housings.

(d) Release air from the strut housing chambers by removing the caps from the air valves and depressing the valve cores.

(e) Fill the main gear strut housings by one of the following two methods:

1) Method I (Addition of small amount of fluid.):

a) Remove valve core from filler plug at top of strut housing. Do not remove plug.

b) Attach one end of a clear plastic hose to valve stem of filler plug and submerge the other end in a container of hydraulic fluid.

NOTE: An air tight connection is necessary between the plastic tube and the valve stem. Without such a connection, a small amount of air will be sucked into the oleo strut during each sequence, resulting in an inordinate amount of air bubbles and prolonged filling operations.

c) Fully compress and extend strut to draw fluid into the strut. By watching fluid pass through plastic hose, determine when the strut is full and no air is present.

d) When air bubbles cease to flow through hose, compress strut fully and remove hose from valve stem.

e) With strut fully compressed, remove filler plug to determine that fluid level is visible up to bottom of filler plug hole.

f) Reinstall the core in filler plug. Apply thread lubricant (MIL-PRF-907) to threads of filler plug. Reinstall filler plug in the top of the strut housing. Torque the plug from 350 to 400 in-lb.

2) Method II (Filling completely empty struts.):

a) Remove the filler plug from the top of the strut housing.

b) Raise the strut until fully compressed.

c) Pour fluid from a clean container through the filler opening until it is reaches at the top of the strut chamber.

d) Lower the gear until the wheel touches the ground and then fully compress and extend the strut three or four times to remove any air from the housing.

NOTE: Gear assemblies with the air valve/filler plug on the side of the strut housing may be serviced in the horizontal position with the air valve/filler plug hole vertical.

e) Raise the strut to full compression and if needed, fill with fluid to the bottom of the filler plug.

f) Apply thread lubricant (MIL-PRF-907) to the threads of the filler plug. Reinstall the filler plug and torque from 350 to 400 in-lb.

(f) With the airplane raised, retract and extend gear strut several times to ascertain that the strut actuates freely. The weight of gear fork and wheel should extend strut.

(g) Clean off the overflow of fluid. Inflate the strut with nitrogen, or dry air, as described in “Inflating Oleo Struts”, below.

(h) Check that fluid is not leaking around the strut piston at the bottom of the housing.

(4) Inflating Oleo Struts

NOTE: Dry air may be used in lieu of nitrogen gas on a temporary basis; however, the strut should be deflated and serviced with nitrogen gas as soon as possible.

(a) Ensure that the struts have a sufficient amount of fluid.

(b) With the aircraft on jacks (see 7-10-00), ensure that the torque links are properly connected, and fully extend the struts.

(c) Remove the air valve cap. Attach a strut pump to the air valve. Use nitrogen gas (or dry air) to inflate the struts to the appropriate pressures: 250 ± 25 psi for the main gear struts; and 120 ± 12 psi for the nose gear strut.

(d) Check for valve core leakage before capping valve.

B. Brake System

The brake system incorporates a hydraulic fluid reservoir. Fluid is drawn from the reservoir by the brake cylinders to maintain the volume of fluid required for maximum braking efficiency. Spongy brake pedal action is often an indication that the brake fluid reservoir is running low on fluid. Instructions for filling the reservoir are given in Filling Brake Cylinder Reservoir. When found necessary to accomplish repairs to any of the brake system components, or to bleed the system, these instructions may be found in 32-40-00.

(1) Filling Brake Cylinder Reservoir

The brake cylinder reservoir is mounted on the aircraft centerline to a support bracket located on forward side of the nose gear mount tubes at F.S. 44.67 and W.L. 44.85. Access to the reservoir is through a cover assembly at the upper rear of the nose baggage compartment. Fluid level can be seen through the window in the cover assembly. It should be checked at every 50 hour inspection and replenished as necessary. Remove four (4) screws to remove the cover assembly and fill with MIL-PRF-5606 fluid to the level marked on reservoir. No brake adjustment is necessary, though they should be checked periodically per instructions given in 32-40-00.

(2) Draining Brake System

(a) Connect a hose to bleeder fitting on the bottom of the cylinder.

(b) Place other end of hose in a suitable container.

(c) Open bleeder fitting and slowly pump hand brake lever and appropriate brake pedal until fluid ceases to flow.

(d) Clean brake system by flushing with denatured alcohol.

C. Tires

Discounting tire growth after initial inflation, once the tire has been inflated, the maximum permissible pressure drop is 5% in any 24 hour period. When the loss rate exceeds 5% in 24 hours, check tires for damage.

Some aircraft tires are designed to permit air or nitrogen that is trapped in the cords or that diffuses through the liner or tube to escape through special sidewall vents. This venting prevents pressure build-up within the cord body which might cause tread, sidewall or ply separation. Tires requiring lower sidewall vents will have either a green or white paint dot applied to the area of each vent hole. Tires not needing lower sidewall vents will have no color dot in this zone.

When water or a soap solution is brushed over the outside of a vented tire, bubbles form. Some vents may emit a continuous stream of bubbles. Others may produce intermittent bubbles. And some may not bubble at all. This variety is normal and does not mean that there is anything wrong with the tire. In fact, as long as a vented tire is inflated, there will be some diffusion from the vents. Vents should remain open, so check periodically to make sure they have not been covered over or closed by tire paint or spilled solvent. And since vents may be covered during retreading, check for evidence that your retreads have been revented.

However, as stated above, any drop in pressure exceeding 5% in any 24 hour period is not acceptable.

(1) Several basic characteristics of aircraft tires may be mistaken for problems:

(a) Tire growth in the first 12 to 24 hours after inflation will result in a seemingly severe pressure drop. Simply inflate, wait for another 24 hours, then check pressure. It will probably be within specs.

(b) Make sure that initial inflation is to recommended operating pressure to ensure full tire growth.

(c) It is normal for tires to show a small amount of pressure leakage throughout the life of the tires.

(2) Maintain tires at pressure specified in 6-00-00, Chart 1. When checking tire pressure, examine tires for wear, cuts, bruises, and slippage. Apply Age-Master #1 to tires to protect against ozone attack and weathering as follows:

(a) Clean oil and grease from all tire surfaces.

(b) Apply single heavy coat using brush at 0.4–0.5 fluid ounces per square foot. Cover surface completely and evenly; allow to dry for 5–10 minutes.

(c) Apply second coat per step (2); allow to dry for 20–30 minutes before handling.

(d) Remove agent on wheel assembly with cleaning solvent.

(e) Re-apply as conditions dictate.

4. Hydraulic System

The hydraulic pump and landing gear actuating cylinders should be checked for leaks, tightness of line fittings and general condition. The cylinder rods are to be free of all dirt and grit. To clean the rods, use an oil soaked rag and carefully wipe them. All the hydraulic lines should also be checked for leaks, kinks, corrosion and attachment fittings for tightness and security.

Maintenance procedures for the hydraulic pump, cylinders, and various components are in Chapter 29, while the actuating cylinders and freefall valve procedures are in 32-30-00.

5. Battery

On Seneca IV models, the battery is located in a compartment located on the right side of the nose section of the airplane. On Seneca V models, the battery (or batteries) is located under the floor in the aft baggage compartment. Check battery for proper fluid level, if applicable. Do not fill battery above the baffle plates. Do not fill the battery with acid - use water only. A hydrometer check will determine the percent of charge in the battery.

Check for spilled electrolyte and corrosion at each 50 hour inspection or every 90 days, whichever comes first. Should corrosion be found on or around the battery, remove the battery and clean it and the surrounding area in accordance with the instructions in Chapter 24.

6. Oxygen System

See 35-10-00.

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SCHEDULED SERVICING

Routine cleaning and lubrication of the airplane and its component parts will significantly extend its service life and reduce the frequency of repairs.

1. Servicing Steering Bungees

   At the specified frequency according to the Lubrication Chart, the steering bungees must be serviced as follows:

   A. Remove the access panels located in the forward baggage compartment.

   B. Clamp the rudder pedals in the neutral position as shown in Chapter 27.

   C. Remove the nut, washers, and bolt that secures the steering bungee and the steering arm.

   D. Remove the clamp that secures the boot, on the frame at station 49.50, to the bungee.

   E. Within the fuselage, disconnect the bungee from the rudder pedal arm by removing the nut, washer and bolt.

   F. Remove the steering bungee from the aircraft.

   G. Cut the safety wire from the bungee retainer.

   H. Carefully remove the retainer and release the spring.

   I. Apply Aero Lubriplate to the spring and mounting hardware as specified in the Lubrication Chart.

   J. Compress the spring into the bungee tube and install the retainer securing with MIL-W-6713 Type 316 safety wire.

   K. Ascertain that the measurement taken between the facing sides of the washers at the rod end is 13.71 inches.

   L. With the nose gear in the neutral position, install the steering bungee into position. The web must be in the vertical position. (See 32-20-00.)

   M. Install the bolt, washers, and nut that secures the bungee to the steering arm.

   N. Install the bolt, washer, and nut that secures the bungee to the rudder pedal arm.

   O. Install the boot clamp.

   P. Repeat this procedure for the other steering bungee.

   Q. Align the nose gear. (See 32-20-00).

   R. Remove the rudder pedal clamps and check the operation of the steering bungees.

   S. Install the access panels in the forward baggage compartment with the attachment hardware.

2. Induction Air Filter

Check induction air filter each 50 hour maintenance inspection. Clean or replace if found to be dirty. Replace the filter after one year, ten cleanings or 500 flight hours, whichever comes first.

A. Removal

The induction air filter is located on the right rear side of the engine compartment and may be removed by the following procedure:

(1) Seneca IV

(a) Remove the upper cowling.

(b) Release the three stud fasteners, remove filter cover.

(c) Remove the filter.

(2) Seneca V

(a) Remove the upper cowling.

(b) Disconnect alternate air linkage/cable.

(c) Remove induction air hose.

(d) Remove lower cowl.

(e) Remove six screws securing cover.

(f) Remove filter.

B. Cleaning

(1) When returning existing filter to service, rap gently on a hard flat surface to remove embedded debris. Be careful not to damage sealing ends.

(2) Inspect filter housing for damage.

(3) The filter housing may be cleaned by wiping with a clean cloth soaked in a suitable quick drying type solvent.

C. Installation

After cleaning or replacing the filter, install the filter in the reverse order of removal.

3. Alternate Air Door

The alternate door is located in the air induction box to provide a source of air to the engine should there be an air stoppage through the filter system. The following should be checked during inspection:

A. Check that air door seals are tight and that the hinge is secure.

B. Check that when the cockpit control is in the closed position, the door is properly seated in the closed position.

C. Actuate the door by operating the control lever in the cockpit to determine that it is not sticking or binding.

D. Check the cockpit control cable for free travel.

4. Electrical System

Servicing the electrical system involves adding distilled water to the battery to maintain correct electrolyte level, and checking for any spilled electrolyte that would lead to corrosion. The security of all electrical connections should be checked as well as the operation of all lights, general condition of the generator or alternator and starter. All electrical wires should be inspected for chafing and bare wires. For detailed information on this system, see 24-30-00.

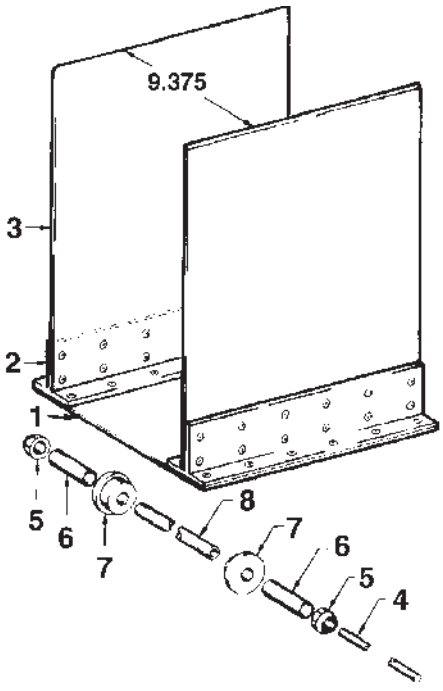
5. Tire Balance

Proper balancing is critical for the life of aircraft tires. If a new tire is balanced upon installation it will usually remain balanced for the life of the tire without having any shimmy or flat spots. An inexpensive balancer can be made that will balance almost any tire for light aircraft. See below for construction details.

A. Balance the tire as follows: (Text keyed to “Figure 1”.)

(1) Mount tire and tube (if one is used) on wheel, but do not install the securing bolts. Install wheel bearings in wheel; then, using the -7 bushings, -6 spacers, and -5 nuts, (refer to “Figure 1”) install the wheel-tire assembly on the pipe. Secure the nuts finger-tight so that the wheel halves touch each other. Be sure the bolt holes are aligned. Insert the axle through the pipe and place the wheel in the center of the balancer. Make sure the axle is only on the chamfered edges of the balancer and that it is at 90° to the sides of the balancer.

(2) Release the tire. If it is out of balance it will rotate, coming to rest with the heaviest point on the bottom. Tape a one-half (1/2) ounce patch across top center of the tire. Rotate the tire 45° and release it again. If the tire returns to the same position, add a one (1) ounce patch and again rotate the tire and release it. Continue this procedure until the tire is balanced.



USE THE FOLLOWING LIST OF MATERIALS TO MAKE THE BALANCER

- 1. 1 EA BASE	12 X 11	0.190 2024 T3 CLAD ALUMINUM ALLOY
- 2. 2 EA TEE	2.5 X 2 X 11	0.190 2024 T4 EXTRUDED ALUMINUM ALLOY
- 3. 2 EA SIDES	14 X 11	0128 2024 T3 CLAD ALUMINUM ALLOY
- 4. 2 EA AXLE	0.125 X 10.25	4130 STEEL, NOMALIZED
- 5. 2 EA NUTS	AN 365-624	
- 6. 2 EA SPACER	0.50 X 2.25	5052-0 ALUMINUM TUBING
2 EA SPACER	0.50 X 1.25	5052-0 ALUMINUM TUBING
- 7. 2 EA BUSHING	1.480 X 1.625 X 1.00	PHENOLIC OR ALUMINUM
2 EA BUSHING	2.240 X 1.37 X 1.00	PHENOLIC OR ALUMINUM
- 8. 1 EA PIPE	1/8 X 9.3	BLACK STEEL PIPE
* 2 EA BEARINGS	SAVE TWO OF EACH SIZE WORN WHEEL BEARINGS FROM PREVIOUS INSPECTIONS.	

Tire Balancer
Figure 1

(3) When balance is attained, put a chalk mark on the sidewall directly below the patch. Use one mark for each half ounce of weight needed. Mark the valve stem location on the tire and the opposite wheel half to assure reassembly in the same position. Remove the wheel from the balance stand, break it down and clean the inside of the tire with toluol. Apply a coat of patch cement to both the patch and the inside center of the tire in line with the chalk marks. When the cement has dried, install the patches making certain they are on the center line of the tire and aligned with the chalk marks on the sidewall. Burnish the patches to remove trapped air, etc.

(4) When reassembling the wheel, powder the inside of the tire. Mount the tire on the valve side of the wheel in the same position it was in when it was balanced. Install the other wheel half, aligning the chalk marks. Install the bolts and tighten to required torque, then air the tire and recheck the balance. The wheel should not be more than one-half (1/2) ounce out of balance.

B. Building the Tire Balancer

A useful tire balancing fixture can be built as follows: (Text keyed to “Figure 1” on page 12203.)

(1) Chamfer top edges of -3 sides, leaving 1/16 inch flat on top of the inboard edge. Rivet -2 tee’s to -3 sides using AN 470-AD5 rivets, with 2 inch spacing, and using AN 426-AD5 rivets (2 inch center to center) to secure -2 tee’s to -1 base. If tee extrusion is unavailable, heavy angle extrusion could be used. -3 sides must be vertical.

(2) The -4 axle must slide through the -8 pipe, the -5 nuts are made by reaming the existing threads in the AN 365-624 nuts with an R drill, then tapping them with a 1/8-27 pipe tap.

(3) The -6 spacers were made from 1/2 inch aluminum tubing, the two lengths of spacers are suitable for balancing most any aircraft wheel.

(4) The -7 bushings may be made from one inch phenolic or aluminum using a 1-1/2 inch hole saw to cut out the smaller bushing and a 1-3/4 hole saw to cut out the larger. By inserting a 1/4 inch long threaded bolt through the pilot hole and securing with a washer and nut, a drill press and file may be used to make the off-set on the bushing. The turned-down part should just slide inside the bearing race and then ream the pilot hole to slide over the -8 pipe threads.

(5) The -8 pipe was made from a piece of 1/8 inch black pipe and threaded with a 1/8-27 pipe die, this will be thread 3 inches in from each end of the pipe.

6. Lubrication

Proper lubrication procedures are valuable both as a means of prolonging the service life of the airplane and reducing the frequency of repairs. The periodic application of recommended lubricants to their relevant bearing surfaces, together with cleanliness, ensures the maximum efficiency and utmost service life of all moving parts. Instructions regarding the locations, time intervals, and types of lubricant used are found in the Lubrication Charts. See also 91-10-00, Consumable Materials charts.

NOTE: If the airplane is inactive for long periods of time, lubricate in accordance with Lubrication Charts at least every 90 days.

A. Precautions

To ensure the best possible results from the application of lubricants, observe the following precautions:

CAUTION: MIL-PRF-23827 AND MIL-PRF-81322, CONTAIN CHEMICALS WHICH MAY BE HARMFUL TO PAINTED SURFACES.

CAUTION: DRY LUBRICANT (I.E. - PTFE BASED MS-122) WILL ATTACK ANY ACRYLIC BASED PLASTIC (LUCITE), POLYCARBONATES (LEXAN), POLYSTYRENE AND ITS COPOLYMERS (ABS), AND CELLULOSE ACETATE.

CAUTION: AFTER THOROUGHLY WASHING AIRPLANE, ENSURE LANDING GEAR, FLIGHT CONTROLS, FLAP TRACKS, STABILATOR TRIM SCREW, AND ENGINE COMPARTMENT ARE STILL PROPERLY LUBRICATED.

(1) Use recommended lubricants. Where general purpose lubricating oil is specified, but unavailable, clean engine oil may be used.

(2) Check components to be lubricated for evidence of excessive wear and replace as required.

(3) Remove excess lubricant from components to prevent collecting dirt and sand in quantities capable of causing excess wear or damage to bearing surfaces.

B. Application of Grease

Before using a grease gun, ensure that gun is filled with new, clean grease of the grade specified for the particular application.

(1) Where a reservoir is not provided around a bearing, apply the lubricant sparingly and wipe off any excess.

(2) Remove wheel bearings from the wheel hub and clean thoroughly with a suitable solvent.When repacking with grease, be sure the lubricant enters the space between the rollers in the retainer ring. Do not pack the grease into the wheel hub.

(3) Use extra care when greasing the constant speed propeller hub to avoid blowing the clamp gaskets. Remove one grease fitting and apply grease to the other fitting until fresh grease appears at the hole of the removed fitting.

C. Application of Oil

Whenever specific lubrication instructions for individual components are not available, observe the following precautions:

(1) Apply oil sparingly, never more than enough to coat the bearing surfaces.

(2) Do not oil control cables.

(3) For Bendix magnetos only: Squeeze the magneto cam follower felts at regular inspection periods. If oil appears on fingers, do not add oil. If the felt is dry, moisten with light oil.

D. Lubrication Charts

(PIR-101138, Rev F.)
(PIR-PPS65102-134, Rev D.)

The lubrication charts consist of individual illustrations for the various aircraft systems. Each component to be lubricated is indicated by a number, the type of lubricant and the frequency of application. Special instructions are listed in Chart 2 before the lubrication charts.

NOTE: When the average ambient air temperature is approximately at the dividing line, use the lighter oil.

While the specified lubricant should be used if available, lubricants listed in the Lubrication Charts should all be considered “or equivalent.” Lubricant specifications become problematic over time. Where a specific product is called out, that manufacturer may go out of business, may be sold, or renamed. The named product may subsequently be no longer available, or renamed itself. Many lubricant military specifications have been superceded over the last several years. Accordingly, a cross-reference chart (“Chart 1”) is provided for lubricants where specification or product changes have been identified.

CHART 1 (Sheet 1 of 2)
LUBRICANT SPECIFICATION CROSS-REFERENCE

Old Spec / Product	superceded by	**New Spec / Product**	**Product Type**
Lubriplate 907		Lubriplate Aero	Grease, Lithium Base, Multipurpose.
MIL-A-907		MIL-PRF-907 (aka Kopr-Kote *)	Anti-Seize Thread Compound, High Temp. (up to 566 Degrees C) (up to 1050 Degrees F)
MIL-A-5092		MMM-A-1617	Adhesive, (Rubber Base) General Purpose.
MIL-C-16173		MIL-PRF-16173	Corrosion Preventative Compound, Solvent Cutback, Cold Application.
MIL-DTL-27686		See MIL-I-27686 below.	
MIL-G-3278		MIL-PRF-23827	Grease, Aircraft & Instrument, Gear and Actuator Screw.
MIL-G-3545		MIL-PRF-81322	Grease Aircraft, General Purpose, Wide Temp. (-54 to 177 Degrees C) (-65 to 350 Degrees F).
MIL-G-6032		SAE-AMS-G-6032	Grease, Plug Valve, Gasoline & Oil Resistant.

CHART 1 (Sheet 2 of 2)
LUBRICANT SPECIFICATION CROSS-REFERENCE

Old Spec / Product	superceded by	New Spec / Product	Product Type
MIL-G-7711		MIL-PRF-81322	See MIL-PRF-81322 above.
MIL-G-18709		DOD-G-24508	Grease, High Perfomance, Multipurpose.
MIL-G-23827		MIL-PRF-23827	See MIL-PRF-23827 above.
MIL-G-81322		MIL-PRF-81322	See MIL-PRF-81322 above.
MIL-H-5606		MIL-PRF-5606	Hydraulic Fluid, Petroleum Base, Aircraft, Missile, and Ordnance.
MIL-I-27686		MIL-DTL-85470	Inhibitor, Icing, Fuel System, High Flash, NATO Cold Number S-1745.
MIL-L-6082		SAE J 1966	Oil, Lubricating, Aircraft Piston Engine (Non-Dispersant Mineral Oil).
MIL-L-7870		MIL-PRF-7870	Oil, Lubricating, Low Temp.
MIL-L-22851		SAE J 1899	Oil, Lubricating, Aircraft Piston Engine (Ashless Dispersant).
MIL-L-25567		MIL-PRF-25567	Leak Detection Compound, Oxygen Systems.
MIL-L-60326		MS-122AD *	Dry-Lubricant.
MIL-M-7866		SAE-AMS-M-7866	Molybdenum Disulfide, Technical, Lubrication Grade.
MIL-S-11031B		A-A-59293	Adhesive (Curing), Sealing Compound (Polysulfide Base).
MIL-S-22473		ASTM-D-5363	Adhesive, Anaerobic Single-Component.
MIL-S-8660		SAE-AS-8660	Silicone Compound, Nato S-736, (-54 to 204 Degrees C) (-65 to 400 Degrees F)
MIL-T-5544		SAE-AMS-2518	Thread Compound, Anti-Seize, Graphite-Petrolatum
MIL-T-27730		A-A-58092	Tape, Anti-Seize, Polytetrafluoroethylene
MS-122 *		MS-122AD *	Dry-Lubricant
MS-122-6075 *		MS-122AD *	Dry-Lubricant
Parker O-Ring Lube *		Parker O-LUBE *	O-Ring Lubricant
Parker 6PB * or 6PB Parker *		MIL-PRF-907 (aka Kopr-Kote *)	See MIL-PRF-907 above.
TT-A-580 (JAN-A-669)		TT-A-580 (TT-S-1732)	Sealing Compound, Pipe Joint and Thread, Lead Free, General Purpose.
		* Product Nomenclature	

CHART 2
SPECIAL INSTRUCTIONS

1. BEARINGS AND BUSHINGS - Clean exterior with a quick-drying solvent before lubricating.
2. LUBRICATION POINTS - Wipe all lubrication points clean of old grease, oil, dirt, etc., before lubricating.
3. WHEEL BEARINGS - For nose and main wheels, see bearing installation procedure in 32-40-00, under Main Wheel Assembly, Assembly and Installation, for grease requirements and packing instructions. Wheel bearings also require cleaning and repacking after exposure to any abnormal quantity of water.
4. OLEO STRUTS, HYDRAULIC PUMP RESERVOIR, AND BRAKE RESERVOIR - Fill per instructions on unit or container, or refer to applicable chapter in this manual.
5. DOOR SEALS - Apply release agent/dry lubricant to door seals at least once a month to improve sealing characteristics and to prevent the seal from sticking.
6. CONTROL CABLES - Do not oil control cables. Grease control cables where they pass over a pulley or through a fairlead.
7. AIR FILTER - To clean filter, tap gently to remove dirt particles or wash in warm water and mild detergent and dry. Do not blow out with compressed air. Do not use oil. Replace filter if damaged.
8. OIL AND FILTER - Continental recommends changing the oil and filter every 50 hours or four months, whichever comes first. Refer to the latest revision of Continental Aerospace Technologies Publication M-0, “Standard Practice Maintenance Manual,” Chapters 3 and 6.
9. NOT USED.
10. O-RING, CONTROL WHEEL SHAFT BUSHING - Disassemble the retainer plates and lubricate the O-ring around the control wheel shaft bushing.

    CAUTION: GREASE MUST BE APPLIED TO ALL BLADES OF A PROPELLER ASSEMBLY AT THE TIME OF LUBRICATION.

    CAUTION: DO NOT MIX GREASES. MIXING NYCO GN 3058 WITH OTHER GREASES IS NOT PERMITTED. IF NYCO GN 3058 IS INADVERTENTLY MIXED WITH ANOTHER GREASE, REMOVE THE PROPELLER FROM THE AIRCRAFT: DISASSEMBLE, CLEAN, AND SERVICE WITH ALL NEW GREASE.

    NOTE: Hartzell only.

11. PROPELLER - See the appropriate Hartzell Propeller Owner’s Manual for lubrication procedure. If annual usage is significantly less than 100 hours, increase lubrication frequency to every six months. See also latest revision of Hartzell Manual No. 202A. If initial lubrication after assembly or overhaul, use procedure in Hartzell Manual No. 202A.
12. NOT USED.
13. BUNGEES - Lubricate springs if bungee is disassembled.
14. COWL FLAP LINKAGE - Lubricate outer surface of inner spacer.

CHART 3
LUBRICATION - MAIN LANDING GEAR

	COMPONENT	LUBRICANT	FREQUENCY
1.	MAIN GEAR PIVOT POINTS (See Spec. Instr. 1 & 2)	MOBILGREASE 33 or MOBILGREASE 28	100 HRS
2.	MAIN GEAR DOOR HINGE (See Spec. Instr. 2)	MIL-PRF-7870C	100 HRS
3.	MAIN GEAR TORQUE LINKS (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
4.	EXPOSED OLEO STRUT MAIN (See Spec. Instr. 2)	FLUOROCARBON RELEASE AGENT / DRY LUBRICANT MS-122	100 HRS
5.	MAIN GEAR WHEEL BEARINGS (See Spec. Instr. 1 & 3)	MOBIL AVIATION GREASE SHC 100	100 HRS
6.	MAIN GEAR DOOR CONTROL ROD ENDS (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
7.	MAIN GEAR SIDE BRACE LINK ASSEMBLY (See Spec. Instr. 1 & 2)	MOBILGREASE 33 or MOBILGREASE 28	100 HRS
8.	UPPER SIDE BRACE SWIVEL FITTING (See Spec. Instr. 1 & 2)	MOBILGREASE 33 or MOBILGREASE 28	100 HRS
9.	CYLINDER ATTACHMENT POINTS (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
10.	OLEO STRUT FILLER POINT (MAIN GEAR) (See Spec. Instr. 2 & 4)	MIL-PRF-5606	AS REQUIRED
11.	HYDRAULIC PUMP RESERVOIR (See Spec. Instr. 4)	MIL-PRF-5606	100 HRS
12.	BRAKE RESERVOIR (See Spec. Instr. 4)	MIL-PRF-5606	100 HRS
13.	MAIN GEAR DOWNLOCK ASSEMBLY (See Spec. Instr. 1 & 2)	MOBILGREASE 33 or MOBILGREASE 28	100 HRS

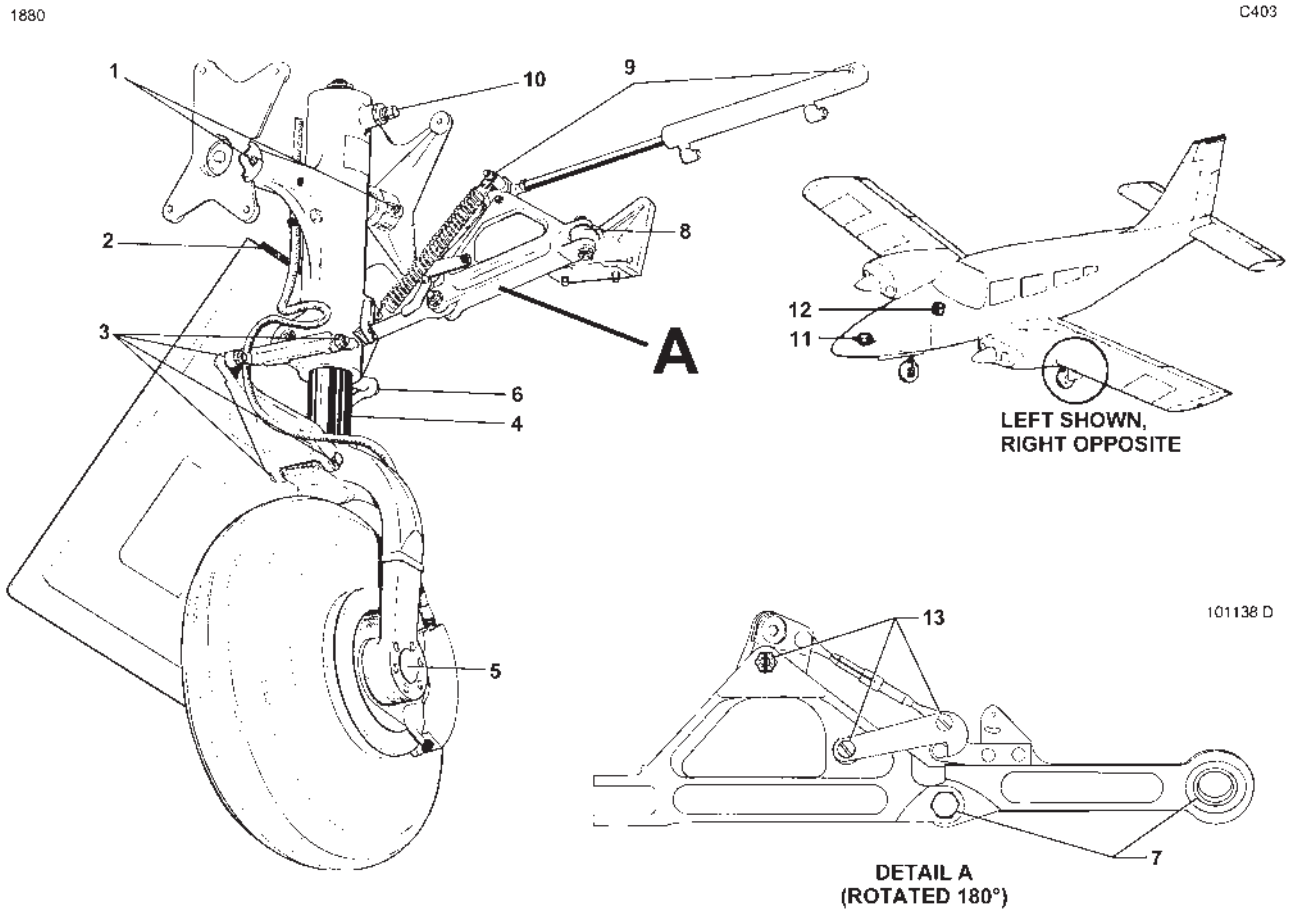


CHART 4
LUBRICATION - NOSE LANDING GEAR

	COMPONENT	LUBRICANT	FREQUENCY
1.	NOSE GEAR STRUT HOUSING (See Spec. Instr. 2)	MOBILGREASE 33 or MOBILGREASE 28	100 HRS
2.	NOSE GEAR PIVOT POINT AND HYDRAULIC CYLINDER ROD END (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
3.	NOSE GEAR DOOR RETRACTION MECHANISM (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
4.	NOSE GEAR DOOR HINGES (See Spec. Instr. 2)	MIL-PRF-7870C	100 HRS
5.	EXPOSED OLEO STRUT (See Spec. Instr. 2)	FLUOROCARBON RELEASE AGENT / DRY LUBRICANT MS-122	100 HRS
6.	NOSE WHEEL BEARINGS (See Spec. Instr. 1 & 3)	MOBIL AVIATION GREASE SHC 100	100 HRS
7.	NOSE GEAR TORQUE LINK ASSEMBLY (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
8.	NOSE GEAR TORQUE LINK ASSEMBLY AND STRUT HOUSING (See Spec. Instr. 1 & 2)	MOBILGREASE 33 or MOBILGREASE 28	100 HRS
9.	NOSE GEAR PIVOT POINT, DRAG LINK ASSEMBLY, DOWNLOCK AND CYLINDER ASSEMBLY, STEERING ROLLER AND CENTERING SPRING PIVOT POINTS (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
10.	LINK BUSHING (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
11.	BUNGEES (See Spec. Instr. 13)	MOBILGREASE 33 or MOBILGREASE 28	100 HRS
12.	STEERING BELLCRANK PIVOT POINTS AND ROD ENDS (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
13.	NOSE GEAR ROLLER TRACK (See Spec. Instr. 2)	MOBILGREASE 33 or MOBILGREASE 28	100 HRS
14.	NOSE GEAR OLEO STRUT FILLER POINT (See Spec. Instr. 2 & 4)	MIL-PRF-5606	AS REQUIRED

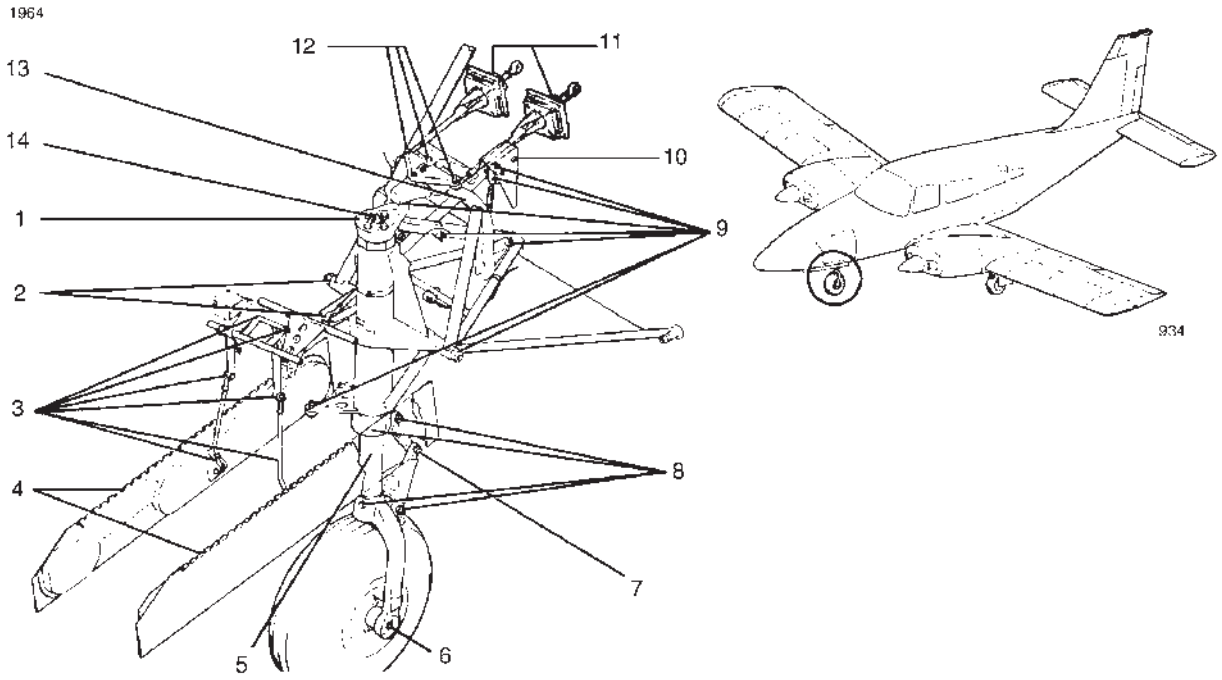
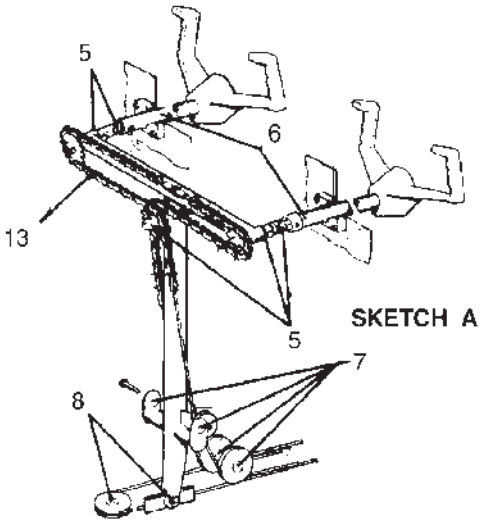


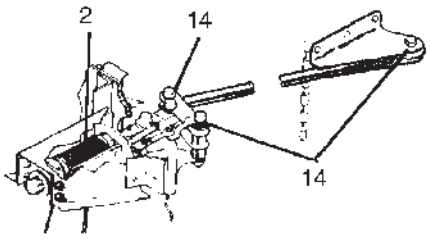
CHART 5 (Sheet 1 of 2)
LUBRICATION - CONTROL SYSTEM

	COMPONENT	LUBRICANT	FREQUENCY
1.	FLAP HINGE BEARINGS (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
2.	RUDDER TRIM SCREWS (See Spec. Instr. 2)	MOBILGREASE 33 or MOBILGREASE 28	100 HRS
3.	CONTROL CABLE PULLEYS (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
4.	TRIM CONTROL WHEELS, STABILATOR AND RUDDER (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
5.	CONTROL COLUMN FLEX JOINT AND SPROCKET (See Spec. Instr. 2)	MIL-PRF-7870C	100 HRS
6.	O-RING CONTROL SHAFT BUSHING (See Spec. Instr. 2 & 10)	FLUOROCARBON RELEASE AGENT / DRY LUBRICANT MS-122	100 HRS
7.	TEE BAR PIVOT POINTS, AILERON AND STABILATOR CONTROL PULLEYS (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
8.	STABILATOR CONTROL ROD AND IDLER PULLEY (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
9.	FLAP CONTROL ROD END BEARINGS (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
10.	TORQUE TUBE (See Spec. Instr. 2)	MIL-PRF-7870C	100 HRS
11.	FLAP SERVO (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
12.	FLAP TORQUE TUBE BEARING BLOCK (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
13.	AILERON AND STABILATOR CONTROL CHAIN (See Spec. Instr. 2)	MIL-PRF-7870C	500 HRS
14.	RUDDER TAB ROD END BEARINGS (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
	* AILERON, ELEVATOR, RUDDER AND TRIM CABLES (See Spec. Instr. 6)	MOBILGREASE 33 or MOBILGREASE 28	100 HRS
	(* Not Shown) (Do not oil control cables. Grease cables where they pass over a pulley or through a fairlead.)		

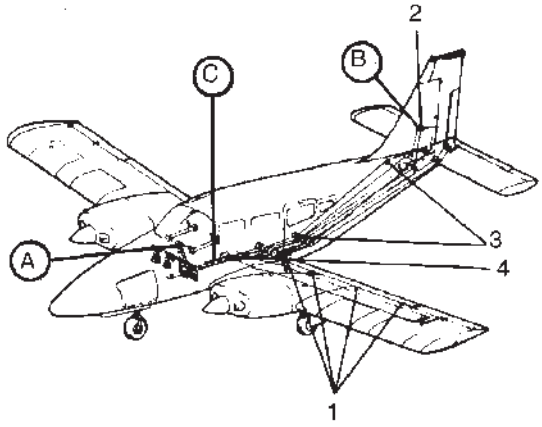
CAUTION: DO NOT LUBRICATE CONTROL WHEEL SHAFT OR BUSHING. CLEAN ONLY WITH ALCOHOL OR OTHER SUITABLE SOLVENT.

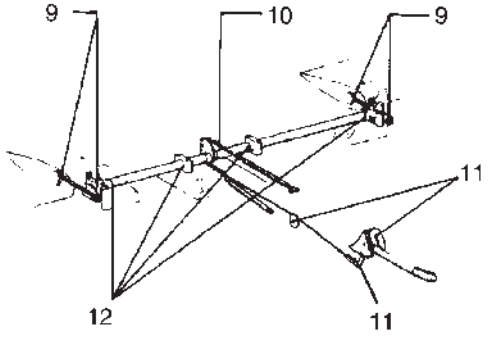


SKETCH A



SKETCH B



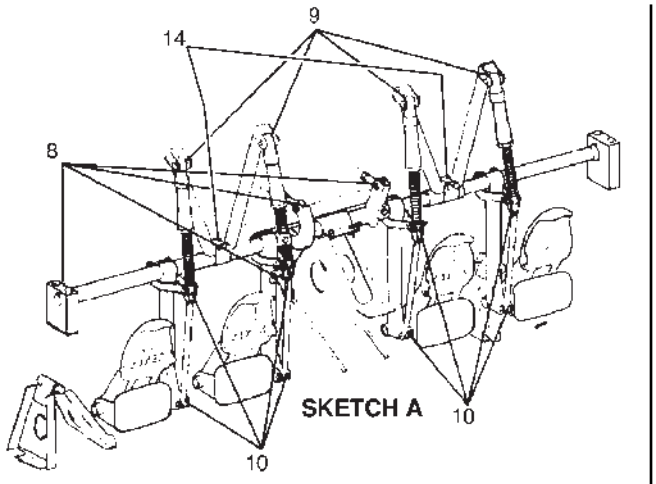


SKETCH C

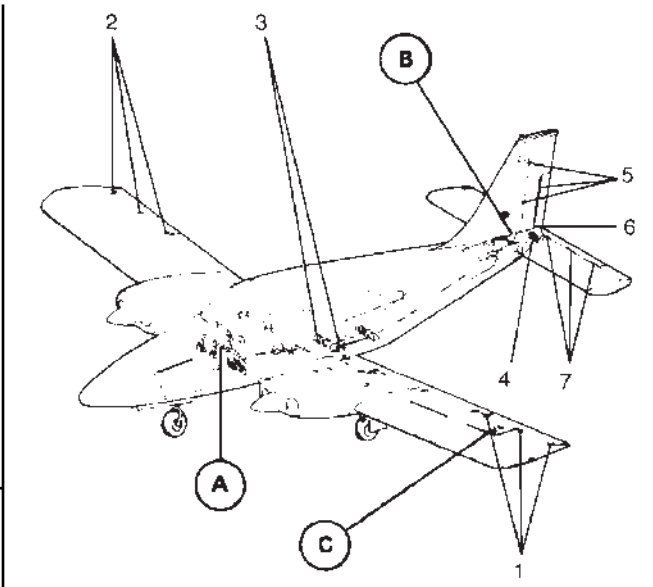
CHART 5 (Sheet 2 of 2)
LUBRICATION - CONTROL SYSTEM

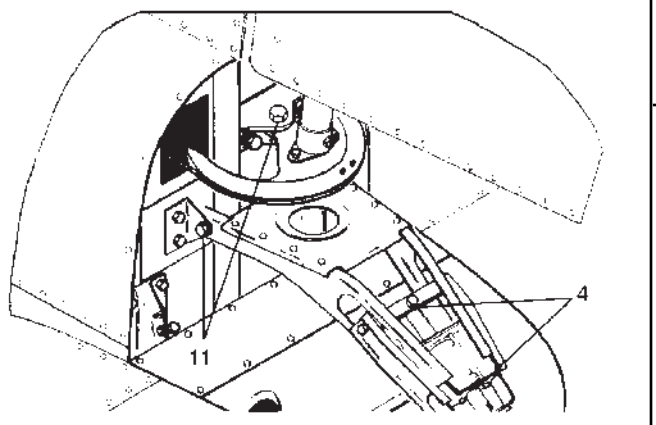
	COMPONENT	LUBRICANT	FREQUENCY
1.	AILERON HINGE PINS (See Spec. Instr. 2)	MIL-PRF-7870C	100 HRS
2.	AILERON HINGE PINS (See Spec. Instr. 2)	MIL-PRF-7870C	100 HRS
3.	CONTROL CABLE PULLEYS (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
4.	STABILATOR TRIM SCREW (See Spec. Instr. 2)	MOBILGREASE 33 or MOBILGREASE 28	100 HRS
5.	RUDDER HINGE AND TAB HINGE BEARINGS (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
6.	ARM BUSHING (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
7.	STABILATOR TRIM TAB HINGE PINS (See Spec. Instr. 2)	MIL-PRF-7870C	100 HRS
8.	RUDDER TUBE CONNECTIONS, TUBE CABLE ENDS AND STEERING ROD ENDS (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
9.	TOE BRAKE ATTACHMENTS (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
10.	BRAKE ROD ENDS (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
11.	RUDDER SECTOR AND STABILATOR TRIM PIVOT POINTS (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
12.	AILERON CONTROL ROD END BEARINGS (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
13.	AILERON BELLCRANK CABLE ENDS (See Spec. Instr. 2)	MIL-PRF-7870C	100 HRS
14.	TOE BRAKE BRACES (See Spec. Instr. 2)	MOBILGREASE 33 or MOBILGREASE 28	100 HRS
	* AILERON, ELEVATOR, RUDDER AND TRIM CABLES (See Spec. Instr. 6)	MOBILGREASE 33 or MOBILGREASE 28	100 HRS

(* Not Shown) (Do not oil control cables. Grease cables where they pass over a pulley or through a fairlead.)

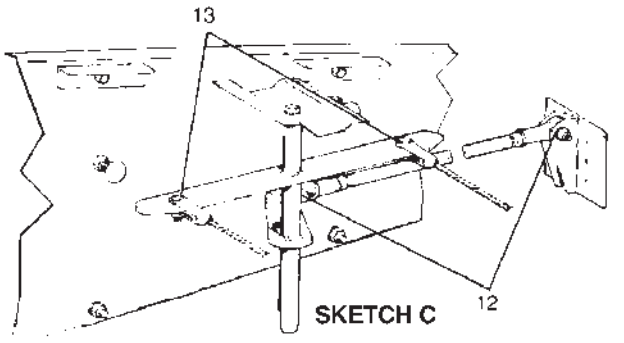


SKETCH A





SKETCH B

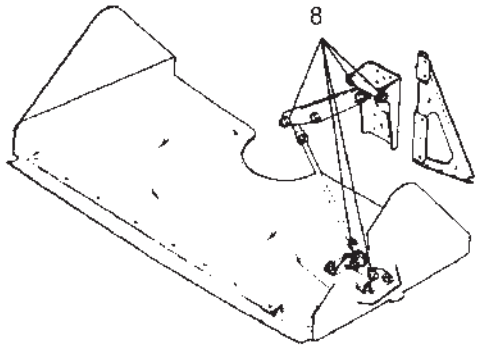


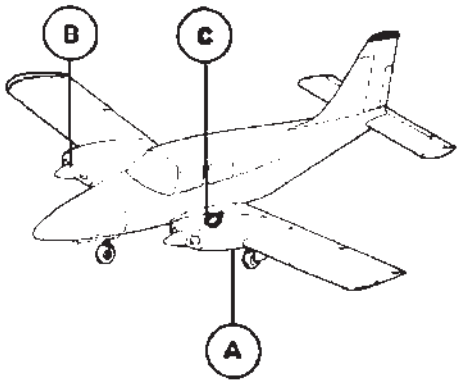
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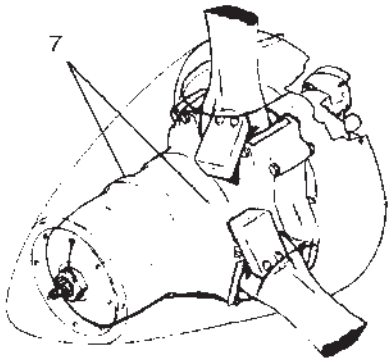
CHART 6
LUBRICATION - POWER PLANTS AND PROPELLERS

	COMPONENT		LUBRICANT	FREQUENCY
1.	ENGINE OIL (See Spec. Instr. 8)	AIR TEMP ABOVE 40°F BELOW 40°F	SAE-J-1899 SAE 50 OR 20W-50 SAE 30 OR 10W-30	50 HRS
2.	OIL FILTERS (See Spec. Instr. 8)		REPLACE	50 HRS
3.	INDUCTION AIR FILTERS (See Spec. Instr. 7)		CLEAN	As Required
4.	ALTERNATE AIR DOORS (See Spec. Instr. 2)		MIL-PRF-7870C	100 HRS
5.	GOVERNOR CONTROLS (See Spec. Instr. 2)		MIL-PRF-7870C	100 HRS
6.	CONTROL QUADRANT CONTROLS (See Spec. Instr. 2)		MIL-PRF-7870C	100 HRS
7.	PROPELLER ASSEMBLY (HARTZELL ONLY) (See Spec. Instr. 11)		USE GREASE PER LABEL ON PROPELLER	100 HRS/12 MOS
8.	COWL FLAP ACTUATING MECHANISM (See Spec. Instr. 14)		MOBILGREASE 33 or MOBILGREASE 28	500 HRS

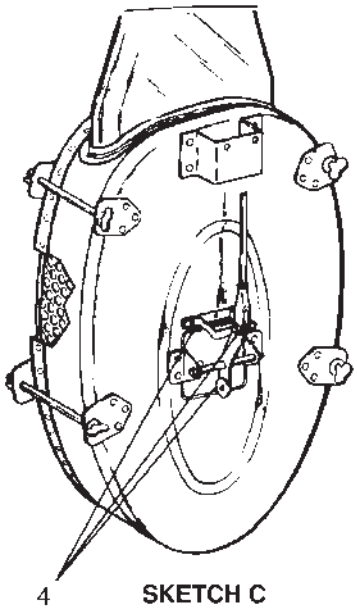
SKETCH A







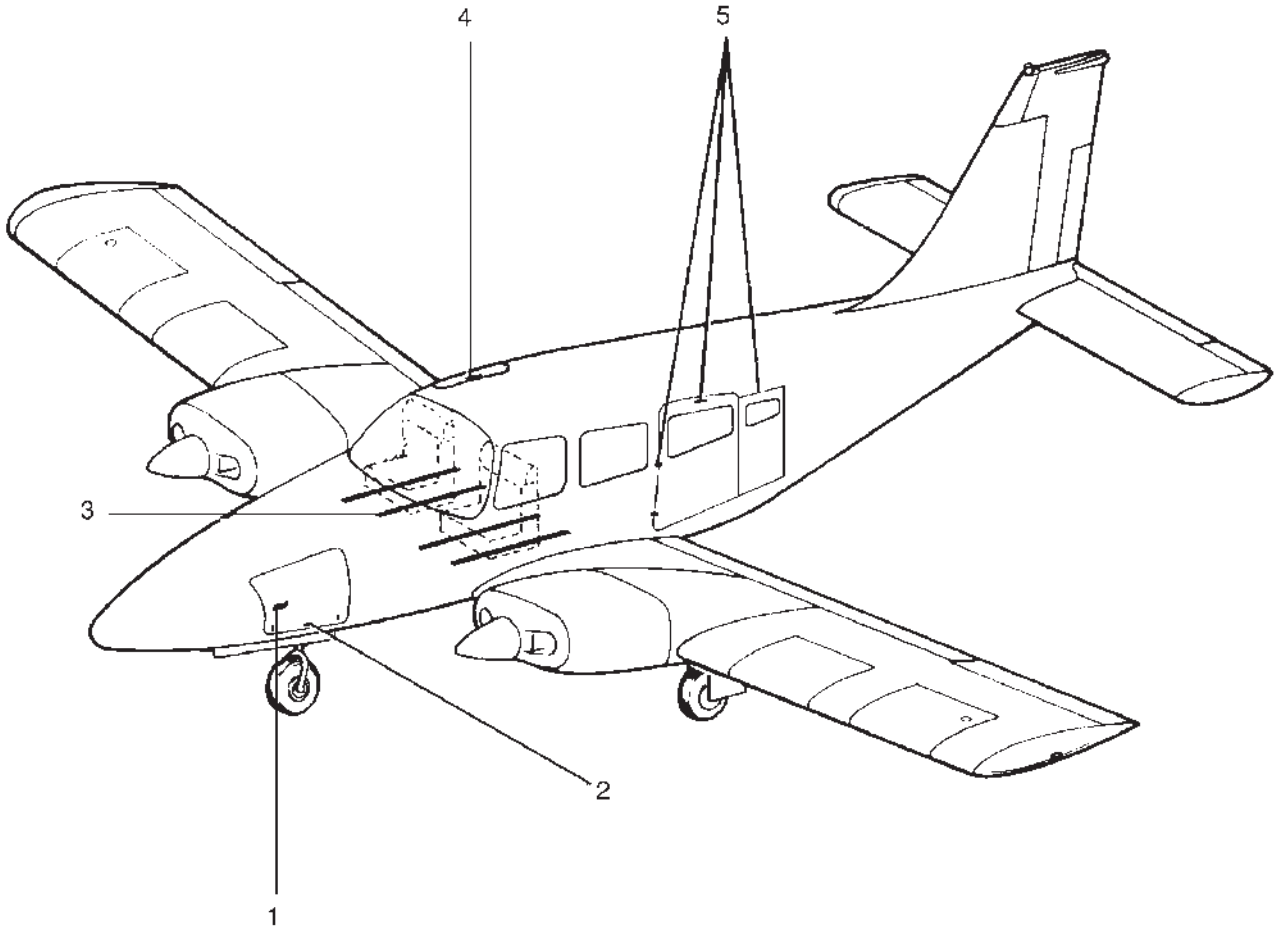
SKETCH B



SKETCH C
(SENECA IV SHOWN)

CHART 7
LUBRICATION - CABIN DOORS, BAGGAGE DOOR AND SEATS

	COMPONENT	LUBRICANT	FREQUENCY
1.	FORWARD BAGGAGE DOOR (See Spec. Instr. 2)	MIL-PRF-7870C	100 HRS
2.	LATCH MECHANISM (See Spec. Instr. 1 & 2)	MOBILGREASE 33 or MOBILGREASE 28	500 HRS
3.	PILOT AND COPILOT SEAT ADJUSTMENT (See Spec. Instr. 2)	MIL-PRF-7870C	100 HRS
3A.	SEAT TRACK ROLLERS AND STOP PINS (Not shown.) (See Spec. Instr. 2)	MIL-PRF-7870C	AS REQUIRED
4.	MAIN DOOR HINGES AND LATCH MECHANISM (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
5.	BAGGAGE AND REAR DOOR HINGES AND LATCH MECHANISM (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
6.	DOOR SEALS (MAIN,REAR, & BAGGAGE) (See Spec. Instr. 5)	FLUOROCARBON RELEASE AGENT/ DRY LUBRICANT MS-122	50 HRS



CHAPTER

20

STANDARD PRACTICES – AIRFRAME

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CHAPTER 20

LIST OF EFFECTIVE PAGES

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	2	Sep 30/18
20-Table of Contents	1	Oct 10/23
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	28	Oct 10/23
20-20-00	1	Sep 30/18
	2	Sep 30/18
	3	Sep 30/18
	4	Sep 30/18
20-97-00	1	Sep 30/18
	2	Sep 30/18
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	6	Sep 30/18

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CHAPTER 20 - STANDARD PRACTICES - AIRFRAME

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GENERAL

1. Description

This chapter contains general information pertaining to standard aircraft hardware installation and removal practices.

For standard repair practices of a minor nature, refer to AC 43.13-1 (latest revision).

If non-destructive testing is needed after repair of 4130 steel, use a magnetic particle inspection method such as Magnaflux.

Testing and inspecting of aluminum castings and machined aluminum parts may be done by the dye penetrant method.

Usually, a good visual inspection with a 10X magnifying glass will show any damage or defect in a repair that is of a significant nature.

2. Torque Wrenches

Torque wrenches should be checked daily and calibrated by means of weights and a measured lever arm to make sure that inaccuracies are not present. Checking one torque wrench against another is not sufficient and is not recommended. Some wrenches are quite sensitive as to the way they are supported during a tightening operation. Any instructions furnished by the manufacturer must be followed explicitly.

When it is necessary to use a special extension or adapter wrench together with a torque wrench, a simple mathematical equation must be worked out to arrive at the correct torque reading. Following is the formula to be used: See “Figure 1” on page 20002.

T = Torque desired at the part.

A = Basic lever length from center of wrench shank to center of handle or stamped on wrench or listed for that model wrench.

B = Length of adapter extension, center of bolt to center of shank.

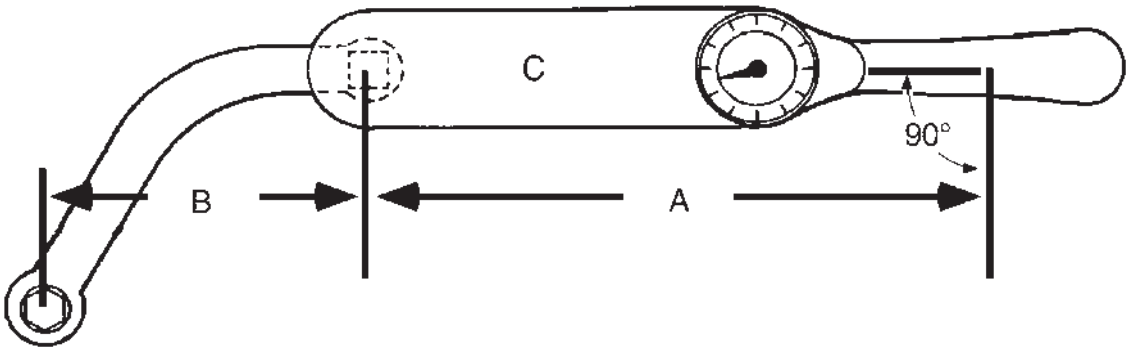
C = Scale reading needed to obtain desired torque (T).

The formula: $C = \frac{A \times T}{A + B}$

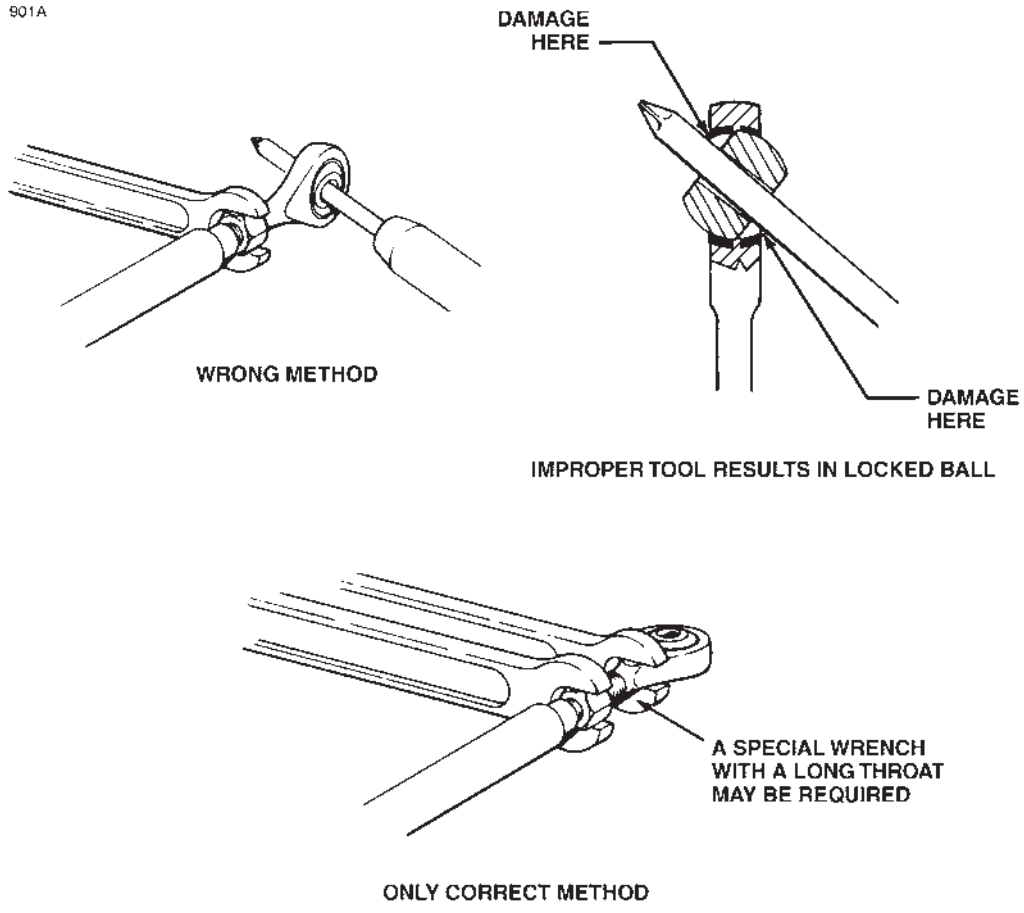
EXAMPLE: A bolt requires 30 foot pounds and a 3 inch adapter (one-quarter of a foot or 0.25’) is needed to get at it. You want to know what scale reading it will take on a one-foot lever arm wrench to obtain the 30 foot pounds at the bolt.

$$C = \frac{1 \times 30}{1 + 0.25} \text{ or } C = \frac{30}{1.25}$$

Remember, the 3 inch adapter must be projecting 3 inches straight along the wrench axis. In general, avoid all complex assemblages or adapters and extensions of flex joints.



Torque Wrench Formula
Figure 1



Installing Rod End Bearings
Figure 2

3. Installing Rod End Bearings

Install rod end bearings as shown in “Figure 2” on page 20002.

4. Removing Cherrylock Rivets

Use following procedure to remove cherrylock rivets:

A. To remove from thick material, use a tapered steel drift pin to drive out rivet stem. See “Figure 3”, View 1.

CAUTION: DRIVING OUT THE LOCKED STEM OF RIVETS INSTALLED IN THIN MATERIAL MAY DAMAGE THE MATERIAL.

NOTE: Drilling completely through the rivet sleeve, when removing rivets, tends to enlarge hole.

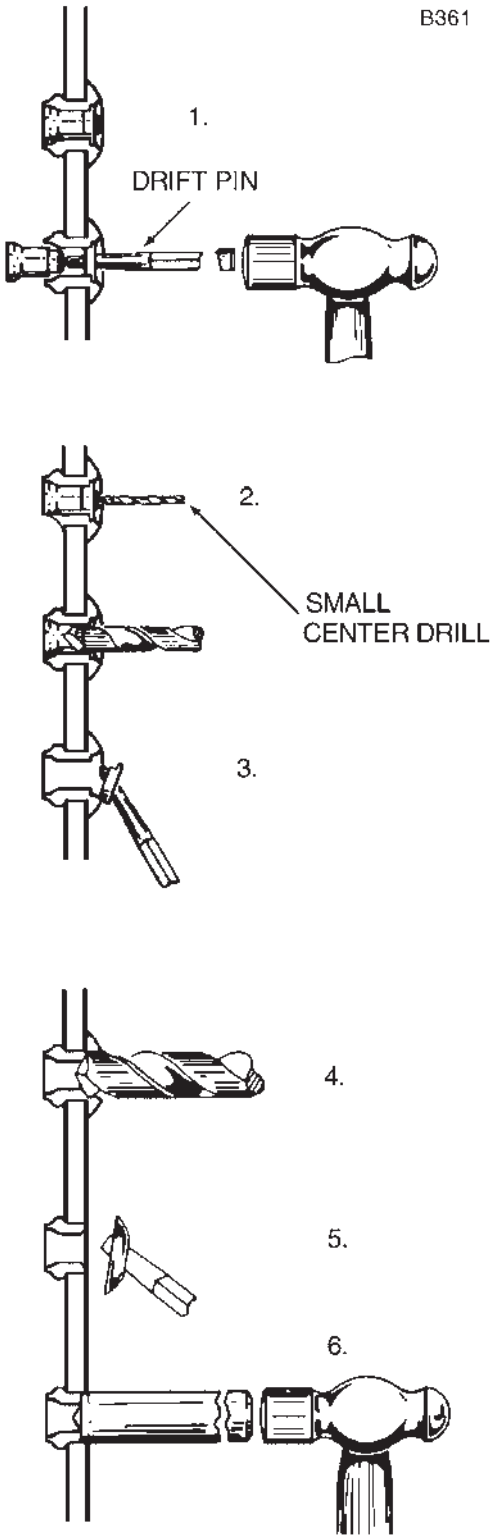
B. To remove from thin material, drill away tapered portion of stem to destroy the lock. Use a small center drill bit on top of the rivet stem to provide a guide for a larger bit. See “Figure 3”, Views 2 and 3.

C. Pry remainder of locking collar out of rivet head with a drift pin. See “Figure 3”, View 3.

D. Drill almost, but not completely, through head of rivet. Use a drill bit the same size as the rivet shank. See “Figure 3”, View 4.

E. Use a drift pin as a lever to break off rivet head. See “Figure 3”, View 5.

F. Drive out remaining rivet shank with a pin having same diameter as rivet shank. See “Figure 3”, View 6.



Removing Cherrylock Rivets
Figure 3

5. Identification of Fluid Lines

See “Figure 4” on page 20005.

Aircraft fluid lines are identified by color code markers, words and geometric symbols. The markers identify each line`s function, content, primary hazard, and the direction of fluid flow.

Most fluid lines are marked with 1-inch tape or decals. Paint is used on lines in the engine induction system.

Certain lines may also be identified as to the specific function within a system. For example: DRAIN, VENT, PRESSURE or RETURN.

Lines conveying fuel may be marked FLAM. Lines containing toxic materials are marked TOXIC. Line containing physically dangerous materials, such as oxygen, nitrogen, or freon, are marked PHDAN.

The aircraft and engine manufacturer is responsible for the original installation of identification markers, Aircraft maintenance personnel are responsible for their replacement when it becomes necessary.

Tapes, paint, tags and decals are placed on both ends of a line and at least once in each compartment through which the line runs. Identification markers are also placed immediately adjacent to each valve, regulator, filter or other accessory within a line.

6. Inspection of Flexible Hoses

NOTE: During the manufacturing process, a condition known as “rubber strike-through” occasionally occurs. This condition is such that rubber material protrudes through the wire braid cover. This condition has no effect on hose quality.

It is recommended that flexible hoses be inspected every 100 hours, especially those in the engine compartments. When inspecting hoses, look for the following conditions:

A. Check each installation to be sure the hose is not kinked, twisted, or distorted. Check for evidence of abrasion, cuts, and broken wires. Random broken wires are acceptable since wire breaks sometimes occur during manufacture. Discard hose if two or more broken wires are found per plait (braid) or more than six broken wires per lineal foot. Broken wires in an area where kinking is evident is also a cause for rejection.

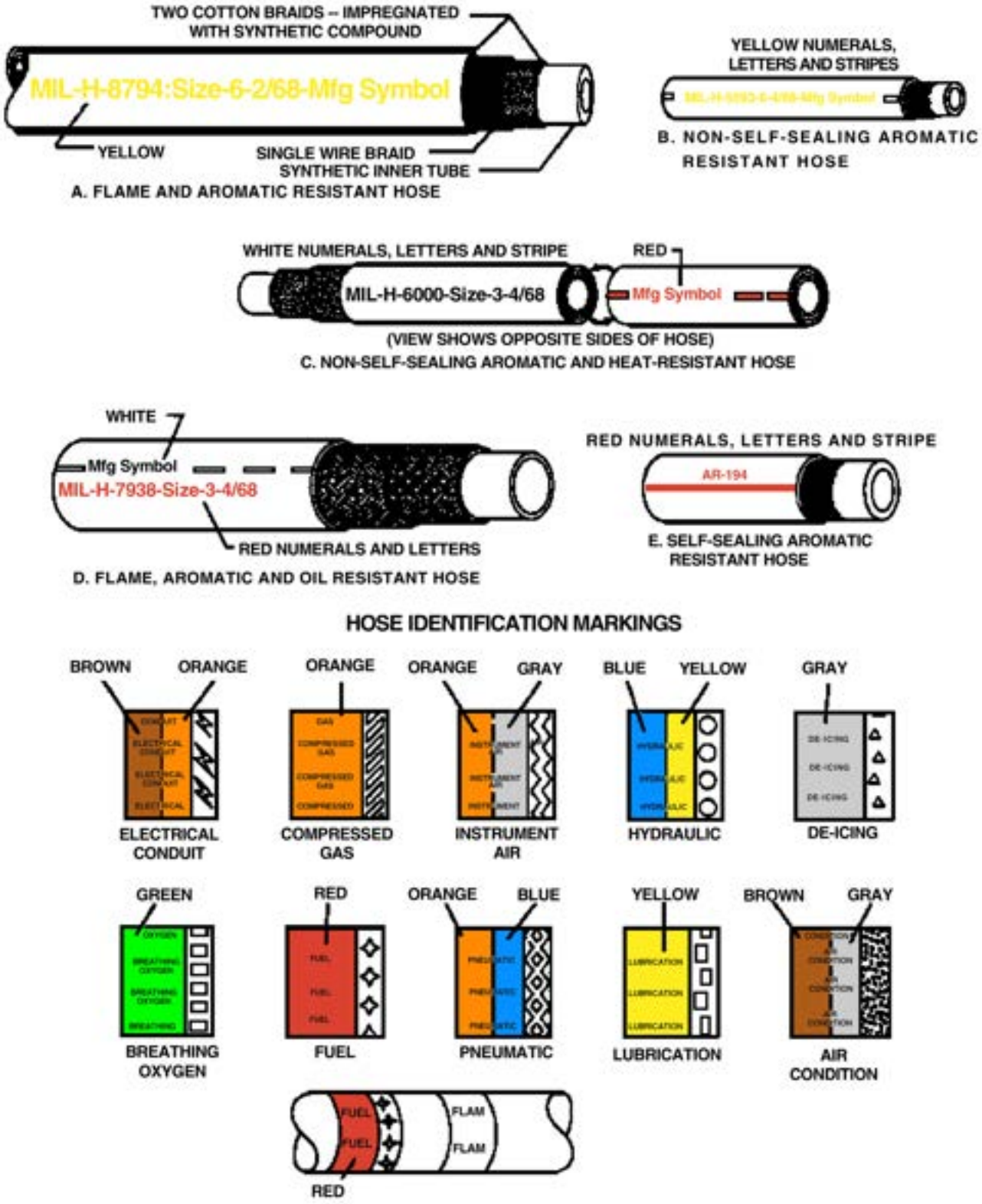
CAUTION: PUNCTURING THE OUTER COVER OF THE HOSE MAY CAUSE DAMAGE TO THE HOSE.

B. Check each assembly for deterioration, ply separation of cover or braid, cracks, weather checking, lack of flexibility, blisters or bulging, collapse, or sharp bending. Blisters on the outer synthetic cover do not necessarily indicate a faulty hose.

C. Remove hose from assembly if hose shows any visible wear. Inspect hose interior and check for signs of deterioration, tube collapse, cut rubber, wire braid puncture, or restriction. To inspect hoses with elbow fittings, use flexible inspection light and viewer, or inspection ball as described in “Chart 1”. Replace hose if any deterioration exists.

CHART 1
BALL DIAMETERS FOR TESTING HOSE RESTRICTIONS

Hose Size	Ball Size
- 4	5/64
- 5	9/65
- 6	13/64
- 8	9/32
- 10	3/8
- 12	1/2
- 16	47/64
- 20	61/64



Hose, Tube, and Line Markings
Figure 4

7. Flareless Tube Assemblies

The use of flareless tube fittings eliminates all tube flaring. An operation, referred to as presetting, is necessary prior to installing a new flareless tube assembly. Presetting is performed as follows:

A. Cut tube to correct length. Ensure ends are perfectly square. Deburr inside and outside of tube. Slip nut, then sleeve, over the tube. (See “Figure 5” on page 20007**,** Step 1.)

B. Lubricate fitting and nut threads as specified in table contained in “Figure 5”.

C. Place fitting in a vise (See “Figure 5”, Step 2). Hold tubing firmly and squarely on seat in fitting. (Tube must bottom firmly in the fitting.) Tighten nut until cutting edge of sleeve grips tube. This point is determined by slowly turning tube back and forth while tightening nut. When tube no longer turns, nut is ready for final tightening.

D. Final tightening depends upon type and size of tubing. On aluminum alloy tubing up to and including half inch outside diameter, tighten nut from 1 to 1-1/6 turns. On aluminum alloy tubing over half inch outside diameter, or steel tubing, tighten nut from 1-1/6 to 1-1/2 turns.

E. After presetting the sleeve, disconnect tubing from fitting and check the following points (“Figure 5”, Step 3):

(1) Tube extends 3/32 to 1/8 inch beyond sleeve pilot to prevent blow off.

(2) Sleeve pilot contacts tube. A maximum clearance of 0.005 inch for aluminum alloy tubing, or 0.015 inch for steel tubing, is acceptable.

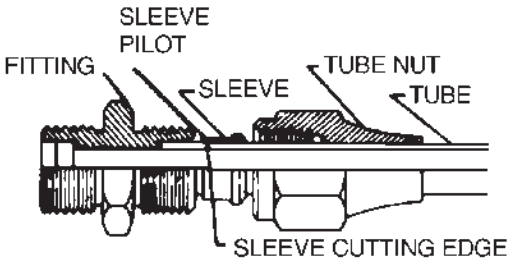
(3) A slight collapse of tube at sleeve cut is permissible. No movement of sleeve pilot, except rotation, is permissible.

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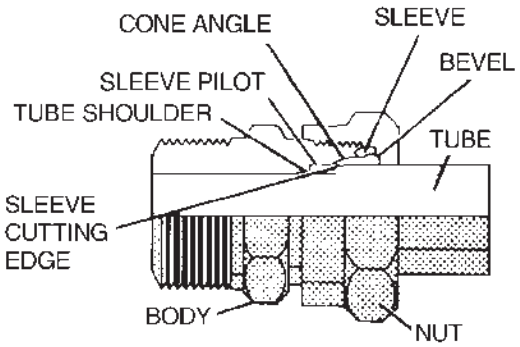
TUBING AND HOSE LUBRICANTS

TUBING SYSTEM	LUBRICANT
HYDRAULIC	MIL-H-5606
FUEL	MIL-H-5656
OIL	SYSTEM OIL
PNEUMATIC	MIL-L-4343
OXYGEN *	NONE

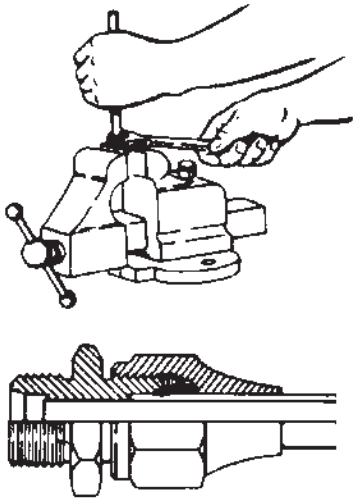
* SEE 35-10-00.



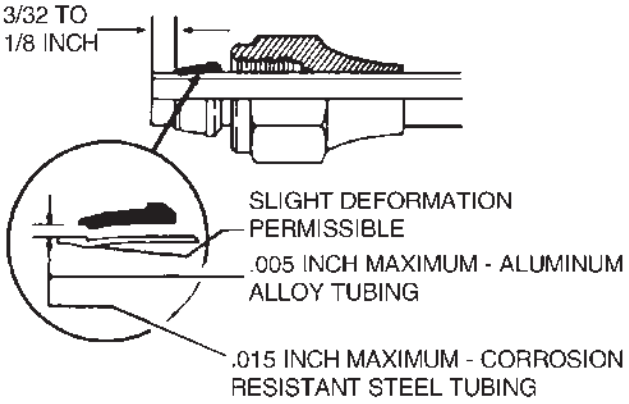
STEP 1



FLARELESS-TUBE FITTING



STEP 2



STEP 3

Presetting Flareless Tube Fittings
Figure 5

8. Support Clamps

CAUTION: MAKE CERTAIN THAT CLAMPS ARE OF THE CORRECT SIZE. CLAMPS OR SUPPORTING CLIPS SMALLER THAN THE OUTSIDE DIAMETER OF THE HOSE MAY RESTRICT THE FLOW OF FLUID THROUGH THE HOSE.

Support clamps are used to secure the various lines to the airframe or power plant assemblies. Several type of support clamps are used for this purpose. The rubber cushioned and plain are the most commonly used clamps. The rubber cushioned clamp is used to secure lines subject to vibration; the cushioning prevents chafing of the tubing. The plain clamp is used to secure lines in areas not subject to vibration.

A teflon cushioned clamp is used in areas where the deteriorating effects of hydraulic fluid or fuel is expected, however, because it is less resilient, it does not provide as good a vibration damping effect as other cushion materials.

Use bonded clamps to secure metal hydraulic, fuel and oil lines in place. Unbonded clamps should be used only for securing wiring. Remove any paint or anodizing from the portion of the tube at the bonding clap location.

All plumbing lines must be secured at specified intervals. The maximum distance between supports for rigid fluid tubing is shown in “Chart 2”.

CHART 2
MAXIMUM DISTANCE BETWEEN SUPPORTS FOR FLUID TUBING

Tube O.D. (IN.)	Distance Between Supports (IN.) Aluminum Alloy	Steel
1/8	9-1/2	11-1/2
3/16	12	14
1/4	13-1/2	16
5/16	15	18
3/8	16-1/2	20
1/2	19	23
5/8	22	25-1/2
3/4	24	27-1/2
1	26-1/2	30

9. Self-Lubricating Impregnated Bearing/Bushings

See “Figure 6” on page 20009.

Many systems and assemblies throughout the aircraft utilize bearings (bushings) that are self-lubricating or impregnated with oil. These parts are designed and built into the assemblies to provide lubricated bearing surfaces requiring little or no mechanical lubrication, or attention.

The most commonly used bearings are of the Garlock “DU” type. Unless absolutely specified, grease should not be used on these bearings although they can be oiled. It is significant to note that these bearings are designed to function wet or dry; however, with alternating conditions, greater bedding in will occur and greatly reduce dry operation wear resistance.

During normal operation, self-lubricating bearings generally bed in and deposit the overlay material on the mating surface of the bolt, etc., which forms a lubricant film. The bearing rubbing surfaces often acquire at this time, a grey green color, exposing a bronze matrix over about 10% of the bearing surface. During the run in period the surface layer may shed slightly exhibiting fine feathery particles. As the bronze surface is exposed and friction increases, the heat expands the bronze providing lubrication to the assembly. As shown in “Figure 6”, the level of bronze in time is slowly exposed to the point where the bearing should be replaced. When the bearing reaches the end of its useful life, about 70% of the bearing surface will be exposed.

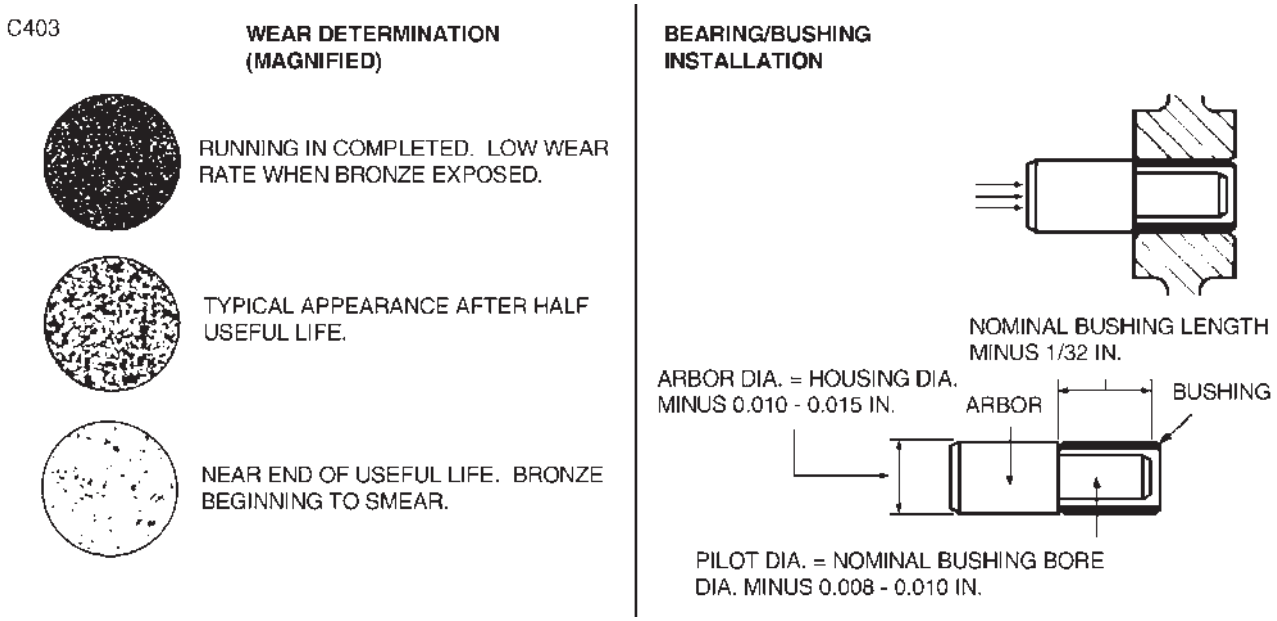
Removal of these bushings is relatively easy using a drift pin. Installation however, is critical to the life of the bearing. Care must be taken to ensure the bearing is squarely inserted in its housing to avoid damage to the lining material. The bearing should be installed as follows:

A. Apply a smear of oil to the outside surface of the bearing.

B. With an arbor press, obtain an arbor 0.010 to 0.015 of an inch smaller than the housing aperture.

C. Carefully align the bearing in the housing and press the bushing into the housing. MAKE SURE the bearing goes in squarely.

NOTE: For large type bushings or bearings where the previous procedure becomes impractical, other methods can be used provided care is taken to protect the edge of the bearing from being damaged. Steps must be taken to maintain alignment of the bearing during assembly. MAKE SURE THE LINING MATERIAL IS NOT SCRATCHED OR CHIPPED.



Self-Lubricating Impregnated Bearing/Bushings
Figure 6

10. Dye Penetrant Inspections

NOTE: The following procedure is general in nature. See manufacturer's instructions, included with dye penetrant kit, for specifics.

A. Using a volatile cleaner, thoroughly remove dirt, loose scale, oil and grease from surface to be inspected.

B. Heat surface to at least 70°F (21°C), but not exceeding 130°F (54°C).

C. Apply penetrant by brushing, spraying, or dipping. Let stand 2 to 15 minutes, depending on temperature.

D. Remove surplus penetrant by applying special cleaner recommended by penetrant manufacturer, or by rinsing with water. Allow housing to dry.

E. Apply a light, even coat of developer by spraying, brushing, or dipping. Cracks or other opening in surface being inspected will appear as bright red. An indication of size of the defect may be obtained by watching the size and rate of growth of red indication.

11. Threaded Fastener Installation

(PIR-PPS20015-1, Rev. AQ.)

A. General Requirements

Apply the following unless otherwise specified in the subject Chapter or Section:

(1) Fasteners installed in a vertical position must be installed with the fasteners head upward.

(2) Fasteners installed in a horizontal position must be installed with the fasteners head forward.

(3) Fasteners installed in a horizontal inboard-outboard position must be installed with the fastener head inboard.

B. Exceptions

The following exceptions to general installation practices are permissible:

(1) Fasteners installed on the aircraft exterior must be installed with the fasteners head on the outside.

(2) Fasteners which pierce the sides or cover of junction boxes, except the mounting side, must be installed with the head on the outside of the box. The length and location of the bolts must be such as to eliminate any damage to electrical wiring or equipment.

(3) Fasteners used for the final mounting of junction boxes must be installed with the fasteners head on the inside of the box.

(4) All fasteners must be installed in such a manner as to minimize injury to personnel or other damage (torn clothing, etc.).

(5) Fasteners centered across butt line zero or fasteners used for non-structural purposes (e.g., securing electrical or hydraulic lines), may be installed in any position to suit installation convenience unless specifically noted otherwise.

C. Thread Lubrication

Lubricants other than that supplied on the fastener as purchased must not be used unless specified in the subject Chapter or Section. See "Thread Lubrication" on page 200022 for general requirements.

D. Cotter Pin Installation

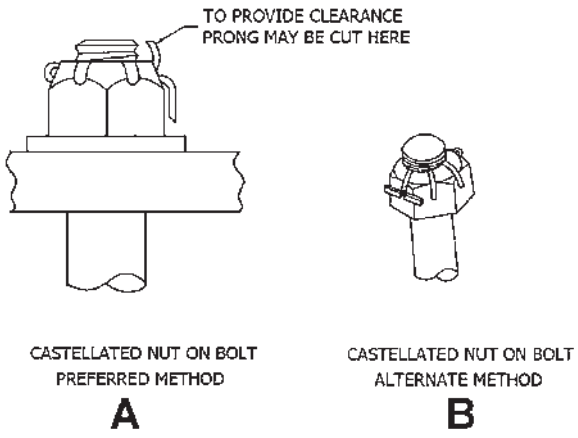
(PIR-PPS20023, Rev. H.)

(1) Installation of cotter pins in castellated nuts must conform to the methods shown in “Figure 7”, View A or B. In either case, the pin must be installed with the head parallel to the slot in the nut.

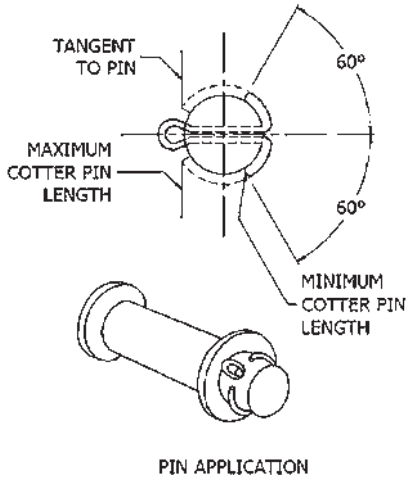
(2) Cotter pins in castellated nuts must not extend more than 50% of the cotter diameter above the top of the slot as indicated in “Figure 7”, View A. Prior to installation of the cotter pin, tighten the nut to the high side of the selected torque range and, if necessary, back off until the closest slot aligns with the hole. If axial hole alignment is not possible within the necessary grip range, then change/add washers per “Washers” on page 200012 until hole alignment is possible.

(3) Cotter pins must be installed in clevis pins as shown in “Figure 8” with the axis of the eye parallel to the shank of the clevis pin or rod end and bend the prongs around the shank of the pin or rod end.

(4) The largest nominal diameter cotter pin which the hole and slots will accommodate must be used, but in no application to a nut, bolt, or screw must the pin size be less than the sizes listed in “Chart 3”.



Cotter Pin Installation – Castellated Nut
Figure 7



Cotter Pin Installation – Clevis Pin
Figure 8

CHART 3
COTTER PIN SIZE

Thread Size (Inches)	Minimum Pin Size (Inches)
0.1380	0.028
0.1640	0.044
0.1900	
0.2500	
0.3125	
0.3750	0.072
0.4375	
0.5000	
0.5625	0.086
0.6250	
0.7500	
0.8750	
1.0000	
1.1250	0.116
1.2500	
1.3750	
1.5000	

E. Fastener Lengths

Fastener shank lengths must be long enough to prevent bearing loads on threads. In addition, the fastener or screw must extend through the nut such that at least one thread of the fastener or screw is protruding through the nut. A variation of specified fastener grip length by one size (i.e., longer or shorter) to meet requirements is allowed.

F. Washers

(1) A total of three NAS1149 (i.e., standard filler) washers are permitted unless otherwise specified.

(2) All standard filler washers must be of the correct diameter, material, and finish that matches the type of fastener being installed.

(3) A maximum of two standard filler washers may be added under fastener heads or nuts to correct for variations in material thickness within the tolerances permitted.

(4) A maximum of two standard filler washers may be used under the nut, if necessary, to adjust fastener length or alignment of cotter key hole. In this application, the replacement of the existing type design washers are also permitted to maintain the maximum of two filler washers and required torque range, provided that the requirements of step (2) are met.

(5) Where nutplates are used, a maximum of two standard filler washers may be used under the fastener head to adjust for a protruding head fastener length.

(6) One standard filler washer may be used under the nut or fastener head (whichever is being turned) to protect the surface from damage while tightening.

(7) Where standard filler washers are required in conjunction with insulating washers, the filler washer must be between the insulating washer and the bolt head or nut.

(8) Where lock washers are specified to be used in contact with aluminum parts, a standard filler washer must be used between the lock washer and the surface of the part to prevent damage.

(9) Where washers with radiused, beveled, or rounded outside diameters are specified, the radiused/beveled/rounded surface of the washer must be nested adjacent to the radiused or beveled surface of the mating structure to prevent damage.

G. Self-locking Fasteners

The limitations of self-locking nuts, fasteners, and screws, including fasteners with nonmetallic inserts, are as follows:

(1) Fasteners incorporating self-locking devices must not be reused if they can be run up using less than the required minimum torque values specified in “Torque Requirements” on page 200015. They may be reused if hand tools are required to run them up, providing there is no obvious damage to the self-locking device prior to installation.

(2) Fasteners 0.3125 (5/16) inch diameter and over with cotter pin holes may be used with self-locking nuts.

(3) Self-locking nuts (i.e., nonmetallic) must not be used on drilled bolts, or studs, 0.25 (1/4) inch or less in diameter.

(4) Self-locking nuts and fasteners must not be used at joints which subject either the nut or the fastener to rotation. Only self-locking nuts may be used with antifriction bearings and control pulleys provided the inner face of the bearing is clamped to the supporting structures by the nut and the fastener.

(5) Corrosion-resistant, steel, self-locking nuts must only be used with corrosion-resistant steel bolts.

(6) Self-locking fasteners must never be tapped or rethreaded. Nuts, fasteners, and screws with damaged threads or rough ends must not be used or rethreaded.

(7) Self-locking fasteners must be used only in applications which permit engagement of at least three threads past the locking device while the locking device is engaged.

(8) Self-locking fasteners must not be used in conjunction with those applications requiring controlled torques (i.e., molded rubber gaskets in fuel cells, etc.).

(9) Self-locking fasteners that have had the locking element reworked or reprocessed by anyone other than an approved bolt manufacturer must not be used.

(10) Nonmetallic insert self-locking bolts are intended only for use in temperatures ranging from -65°F to 250°F (-53°C to 121°C), and are designed to function satisfactorily in the specified range.

(11) The plating or replating of nonmetallic (e.g., fiber or nylon insert) self-locking nuts is not permitted because of the deterioration of the insert by the plating solution and post-plating baking.

H. Multiple Fastener Installations

Installations that require multiple fasteners are subject to the following steps, unless otherwise specified in the subject Chapter or Section:

(1) Finger-tighten all nuts first.

(2) Using an appropriate wrench, snug up all of the nuts.

(3) Torque opposite nuts or fasteners (i.e., torque hardware in a star pattern). Do not successively tighten adjacent nuts.

I. Fastener Head Shaving

Fastener heads must not be shaved unless otherwise specified in the subject Chapter or Section.

J. Inspection

(1) Cracks in Fastener Heads

Any bolt or bolt installation where open cracks are visible in the bolt head requires replacement of the bolt.

(2) Gaps Under Fastener Heads

Presence of gaps which allow for a 0.004-inch maximum feeler gage to be inserted between the fastener head and the bearing surface are acceptable provided these gaps cover less than 40 percent (40%) of the fastener head circumference.

12. Torque Requirements

(PIR-PPS20015-1, Rev. AQ.)

CAUTION: DO NOT OVERTORQUE FITTINGS.

"Chart 4" lists the torque values for flared fittings of various sizes and material.

NOTE: When installing flared fittings, verify that male threads are properly lubricated.

The torque values given in "Chart 5" on page 200018 are derived from oil-free cadmium-plated threads and are recommended for all airframe installation procedures where torquing is required, unless other values are specified in subject chapter/section. Engine torque values are found in the latest revision of Continental Motors SB96-7; and propeller torque values are found in 61-10-00.

NOTE: All torque values given are installation torques for installation purposes only and shall not be construed as retained torque.

NOTE: If normal operation requires movement between any of the components being clamped together, tighten the nut (or bolt) enough to insure intended operation of the assembly.

A. Calibrate the torque wrench periodically to assure accuracy, and recheck frequently. See "Torque Wrenches" on page 20001.

B. If the fastener, screw, or nut is listed in "Chart 5" on page 200018 but the mating fastener is not listed, tighten only to the low end of the torque range specified for the listed fastener.

CHART 4
FLARE FITTING TORQUE VALUES

(PIR-AC65-9A.)

Tubing OD Inches	Torque — Inch-Pounds					
	Aluminum Alloy Tubing		Steel Tubing		Hose End Fitting and Hose Assemblies	
	Minimum	Maximum	Minimum	Maximum	Minimum	Maximum
1/8	20	30	———	———	———	———
3/16	30	40	90	100	70	120
1/4	40	65	135	150	100	250
5/16	60	85	180	200	210	420
3/8	75	125	270	300	300	480
1/2	150	250	450	500	500	850
5/8	200	350	650	700	700	1150
3/4	300	500	900	1000	———	———
7/8	500	600	1000	1100	———	———
1	500	700	1200	1400	———	———
1-1/4	600	900	———	———	———	———
1-1/2	600	900	———	———	———	———

C. Unless otherwise specified by the engine manufacturer, in “AN386 Taper Pin Installation” on page 200017, or in the subject Chapter or Section, threaded fasteners must be torqued to final torque values as determined by adding friction drag to the torque values listed in Tables 3–6 in “Chart 5” on page 200018**.** The final torque values (with friction drag included) are shown in Tables 3–6 for some select fastener/nut combinations. If the final torque value is not listed, then it must be calculated per paragraphs “(2)” and “(3)”. In addition, the following limitations and exceptions apply:

(1) Fastener and nut threads must be clean and dry (free of lubricants). If the subject Chapter or Section requires the fastener and/or nut to be lubricated prior to tightening, but does not specify a torque requirement, the final torque range (including friction drag torque) must be 50% of the value shown in Tables 3–6.

(2) The friction drag torque can be determined as follows:

Run the nut down to near the contact (but not in contact) with the bearing surface. While still not contacting the bearing surface, the “friction drag torque” must be determined by observing the reading on face of “dial type” wrenches or observing the nearest detent adjustment setting for “snap-over” wrenches.

(3) The friction drag (if any) must be added to the desired torque specified by Tables 3–6. This is referred to as “final torque”, which should register on the indicator or be the setting for a snap-over wrench. The final torque values (with friction drag included) are shown in Tables 3–6 for some select fastener/nut combinations. If the value is not listed, then it must be determined per paragraph “(2)”.

(4) If the friction drag of the nut is significantly low, such that it cannot be accurately measured (free running nuts), then adding friction drag to determine final torque is not required.

(5) Torque requirements do not apply to cross-recessed screws, slotted screws, or fasteners listed in Tables 3–6 installed in nuts that are not designed to rotate for wrenching at their tail end (e.g., rivnuts, press nuts, swage nuts, speed nuts, etc.) excluding nutplates.

(6) When torque requirements do not apply (see “(5)” above), the fastener must be torqued until its head is fully seated and there is no play or looseness in the joint using hand pressure.

(7) Fasteners listed in Tables 3–6, installed into nutplates and are accessible to be torqued at the fastener, must be tightened to the low end of the torque range found for shear applications listed in Tables 3–6 unless otherwise specified in the subject Chapter or Section.

(8) When torque requirements do apply (i.e., not listed in “(5)” above) and fasteners are not installed in nutplates:

(a) If the fastener is stationary and the nut is torqued, use the lower limit of the torque range.

(b) If the nut is stationary and the fastener is torqued, use the higher limit of the torque range and ensure one (1) 0.032-inch thick NAS1149 standard filler washer is installed under the fastener head.

(9) Apply a smooth, even pull when applying torque pressure. If chattering or a jerking motion occurs during final torque, back off and retorque.

(10) For castellated nuts:

NOTE: Torque seal is not required in this application.

(a) Torque the nut onto the fastener joint to the lower limit of the torque range for that fastener/nut combination.

(b) Set the torque wrench to the upper limit of the torque range and slowly tighten until the cotter pin hole and nut castellation align.

If there is no alignment by the time maximum torque is reached, adjustments to the standard filler washers are permitted, in accordance with “Washers” on page 200012. Do not exceed the maximum torque to achieve alignment.

(c) Install the cotter pin.

(11) Unless otherwise specified in the subject Chapter or Section, when castellated nuts are used with a cotter pin on moving joints, the nuts must be tightened enough to remove any looseness in the joint and allow for cotter pin installation.

(12) Unless otherwise specified specified in the subject Chapter or Section, when parts are used on engines or other supplied assemblies using Piper furnished or existing supplier threaded fasteners, tightening torque limits shall be as specified by the latest applicable supplier specifications.

D. Marking with Torque Seal After Tightening

The application of torque seal is used to indicate that threaded fasteners have been properly torqued. Unless otherwise stated herein, nuts (or fasteners and screws if no nut is involved) that have been final tightened to torque values as specified per paragraph “(12)” on page 200017 and Tables 3–6 in “Chart 5” on page 200018**,** or as specified in the subject Chapter or Section, must be marked with torque seal.

Do not apply torque seal to fasteners or nuts used in the following installations, unless otherwise specified in the subject Chapter or Section:

(1) Where normal vibration may cause torque seal to fall off into openings of electronic devices.

(2) Direct contact with fuel, hydraulic, or de-ice fluids.

(3) Inside ducts and manifolds of induction, heating, and ventilation systems.

(4) Areas in which the torque seal will adversely affect the cosmetic appearance (e.g., instrument panel mounting screws or interior panels). This is applicable only if the hardware is without specific torque requirements – if hardware in these areas have specific torque requirements, then apply the torque seal in an unobtrusive way.

E. AN386 Taper Pin Installation

(1) If not already accomplished:

(a) Enlarge pilot holes with standard drill until an appropriate Brown & Sharpe tapered reamer (or equivalent) can be used.

(b) Ream the hole slowly, periodically inserting the specified taper pin as a guide, until the small end of the taper pin protrudes no more than 0.063 inches past the hole and the large end protrudes between 0.125 and 0.250 inches per NASM386. If the large end protrusion does not meet the requirements, adjust the grip length of the pin accordingly.

NOTE: Ream slowly, making sure to keep the axis of the reamer aligned as close as possible to the axis of the hole.

(2) Unless otherwise specified in the subject Chapter or Section, assemble the AN386 pin, AN975 washer, and castle nut as specified in the subject Chapter or Section, and torque nut per paragraph “(9)” on page 200016 until the cotter pin can be inserted.

F. Gap Conditions Between Parts Attached with Threaded Fasteners

If a gap condition exists between mating parts where a threaded fastener is to be installed, install the required fasteners and associated hardware except do not torque to final requirement. Instead, torque to a value 10% of the maximum required torque (from Tables 3–6, not including friction drag), and add the friction drag of the required (installed) nut per paragraph “(2)” on page 200016. If the final torque (including friction drag) is listed in Tables 3–6, the gap-closing torque may be calculated as follows:

$$T_{MAX} = T_D - (0.90 \times T_{ND})$$

Variable Definitions:

T_{MAX}: Permissible gap-closing torque

T_D: Final torque including friction drag (Tables 3–6 in “Chart 5”)

T_{ND}: Final torque not including friction drag (Tables 3–6 in “Chart 5”)

Review the example below referencing the AN6 fastener (3/8-24) from Table 3:

$$T_{MAX} = T_D - (0.90 \times T_{ND})$$
$$T_{MAX} = 270 \text{ in-lb} - (0.90 \times 190 \text{ in-lb})$$
$$T_{MAX} = 99 \text{ in-lb}$$

NOTE: If the torque requirement listed in Tables 3–6 exceeds the requirement specified in the subject Chapter or Section, the requirement specified in the subject Chapter or Section must be used to determine the permissible torque. Accomplish this for all fasteners common to gapped interface. If no gap exists after torqueing per the above procedure, finish torqueing to the final requirement. If a gap remains, consult your Piper Dealer’s Service Advisor (DSA) for further assistance.

CHART 5 (Sheet 1 of 4)
TORQUE TABLES

(1) The following tables are partially derived from FAA AC 43.13-1B.

(2) Some fasteners and nuts are listed in more than one group. Select torque from appropriate group depending on application (tension vs shear).

(3) If the chapter/section calls out a fastener and nut that are not in the same group, use the lower of the two torque values.

(4) NASM21042 contains wrenching torque values that are intended for testing purposes only, for aircraft assemblies use torque values shown herein.

(5) Although MS21042, NAS1291, and NAS679 nuts are reduced height, they are included in both tension and shear groups and may be used for both applications.

(6) Unless otherwise specified in the subject Chapter or Section, size #6 screws used with self-locking nutplates must be torqued to no greater than five (5) in-lb with an appropriate calibrated driver.

(7) If final torque (with friction drag included) is not listed it shall be calculated per paragraphs “(2)” and “(3)” on page 200016.

NOTE: Within Chart 5, Tables 1 and 2 are not used.

CHART 5 (Sheet 2 of 4)
TORQUE TABLES

Table 3: Standard Fasteners

Standard Fasteners		Fastener Size		Tension		Shear	
Fasteners	Nuts			Torque (in-lb)		Torque (in-lb)	
				No Friction Drag	W/ Friction Drag	No Friction Drag	W/ Friction Drag
AN3 thru 20 AN21 thru 37 AN42 thru 49 AN386 AN525 MS20033 thru 20046 MS20073 MS20074 MS21046 MS24694 MS27039	Tension AN310 AN315 MS14145 MS20365(1) MS21042 MS21044 MS21045 NAS679 NAS1291 Shear AN316 AN320 MS20364(1) MS21083 MS21245 NAS509	Fine Thread	8-36	12 –15		7 – 9	
			10-32	20 – 25	38 – 43	12 – 15	30 – 33
			1/4-28	50 – 70	65 – 100	30 – 40	60 – 70
			5/16-24	100 – 140	160 – 200	60 – 85	120 – 145
			3/8-24	160 – 190	240 – 270	95 – 110	175 – 190
			7/16-20	450 – 500	550 – 600	270 – 300	370 – 400
			1/2-20	480 – 690	-	290 – 410	-
			9/16-18	800 – 1000	-	480 – 600	-
			5/8-18	1100 – 1300	-	600 – 780	-
			3/4-16	2300 – 2500	-	1300 – 1500	-
			7/8-14	2500 – 3000	-	1500 – 1800	-
			1-14	3700 – 4500	-	2200 – 3300	-
			1 1/8-12	5000 – 7000	-	3000 – 4200	-
			1 1/4-12	9000 - 11000	-	5400 - 6600	-
		Coarse Thread	8-32	12 – 15	27 – 30	7 – 9	22 – 24
			10-24	20 – 25	38 – 43	12 – 15	30 – 33
			1/4-20	40 – 50	70 – 80	25 – 30	55 – 60
			5/16-18	80 – 90	140 – 150	48 – 55	108 – 115
			3/8-16	160 – 185	240 – 265	95 – 110	175 – 190
			7/16-14	235 – 255	335 – 355	140 – 155	240 – 255
			1/2-13	400 – 480	-	240 – 290	-
			9/16-12	500 – 700	-	300 – 420	-
			5/8-11	700 – 900	-	420 – 540	-
			3/4-10	1150 – 1600	-	700 – 950	-
			7/8-9	2200 – 3000	-	1300 – 1800	-
			1-8	3700 – 5000	-	2200 – 3000	-
			1 1/8-8	5500 – 6500	-	3300 – 4000	-
			1 1/4-8	6500 – 8000	-	4000 – 5000	-

(1) Superceded by MS21042.

CHART 5 (Sheet 3 of 4)
TORQUE TABLES

Table 4: High Strength Fasteners

High Strength Fasteners		Fastener Size		Tension		Shear	
Fasteners	Nuts			Torque (in-lb)		Torque (in-lb)	
				No Friction Drag	W/ Friction Drag	No Friction Drag	W/ Friction Drag
MS20004 thru 20024 NAS333 thru 340 NAS464 NAS624 thru 644 NAS1580 NAS6203 thru 6220 NAS6303 thru 6320(2) NAS6603 thru 6620 NAS6703 thru 6720	Tension AN310 AN315 MS20365(1) MS21042 MS21044 MS21045 NAS679 NAS1291 Shear AN316 AN320 MS20364(1) MS21083 MS21245 NAS509	Fine Thread	8-36	No Data			
			10-32	25 – 30	46 – 48	15 – 20	33 – 38
			1/4-28	80 – 100	110 – 130	50 – 60	50 – 90
			5/16-24	120 – 145	180 – 205	70 – 90	130 – 150
			3/8-24	200 – 250	280 – 330	120 – 150	200 – 230
			7/16-20	520 – 630	620 – 730	300 – 400	400 – 500
			1/2-20	770 – 950	-	450 – 550	-
			9/16-18	1100 – 1300	-	650 – 800	-
			5/8-18	1250 – 1550	-	750 – 950	-
			3/4-16	2650 – 3200	-	1600 – 1900	-
			7/8-14	3550 – 4350	-	2100 – 2600	-
			1-14	4500 – 5500	-	2700 – 3300	-
			1 1/8-12	6000 – 7300	-	3600 – 4400	-
			1 1/4-12	11000 – 13400	-	6600 – 8000	-
		Coarse Thread	All	No Data			

(1) Superceded by MS21042.

(2) Not applicable for aluminum coated fasteners.

CHART 5 (Sheet 4 of 4)
TORQUE TABLES

Table 5: Aluminum Fasteners

Aluminum Fasteners		Fastener Size		Tension		Shear	
Fasteners	Nuts			Torque (in-lb)		Torque (in-lb)	
				No Friction Drag	W/ Friction Drag	No Friction Drag	W/ Friction Drag
AN3DD	Tension AN310D AN315D Shear AN316D AN320D	Fine Thread	8-36	5 – 10	-	3 – 6	-
			10-32	10 – 15	28 – 33	5 – 10	23 – 28
			1/4-28	30 – 45	60 – 75	15 – 30	45 – 60
			5/16-24	40 – 65	100 – 125	25 – 40	85 – 100
			3/8-24	75 – 110	155 – 190	45 – 70	125 – 150
			7/16-20	180 – 280	280 – 380	110 – 170	210 – 270
			1/2-20	280 – 410	-	160 – 260	-
		Coarse Thread	All	No Data			

Table 6: MS17825 Castle Nuts

Aluminum Fasteners		Fastener Size		Tension		Shear	
Fasteners	Nuts			Torque (in-lb)		Torque (in-lb)	
				No Friction Drag	W/ Friction Drag	No Friction Drag	W/ Friction Drag
.190 .250 .3125 .375 .4375 .500	Tension MS17825 Shear MS17826	Fine Thread	8-36	No Data			
			10-32	28	N/A	16	N/A
			1/4-28	65		35	
			5/16-24	180		70	
			3/8-24	260		100	
			7/16-20	460		180	
			1/2-20	720		240	
			9/16-18	880		320	
			5/8-18	1300		480	
			3/4-16	2200		880	
			7/8-14	3700		1500	
			1-14	5400		2400	
			1 1/8-12	8000		4000	
			1 1/4-12	11000		5600	
		Coarse Thread	All	No Data			

NOTE: See paragraph “(10)” on page 200016 for castellated nut installation requirements.

13. Thread Lubrication

Lubricate all fittings on external lines, including attachment points at engine and other components, with proper lubricant as specified in “Chart 6”.

When applying thread lubricants, proceed as follows:

A. Thoroughly clean threads before applying lubricant.
B. Use thread lubricant sparingly.
C. Apply thread lubricant to male threads only.
D. Lubricate first three threads only on straight fittings.
E. Do not lubricate first two threads on tapered fittings; apply lubricant to next three threads only.
F. Ensure lubricant does not enter fittings or flared areas.
G. Lubricate any fittings going to engine with same fluid going through lines.

CHART 6
THREAD LUBRICANTS

Line	Lubricant
WARNING: DO NOT PERMIT SAE-AMS-2518 ANTI-SEIZE COMPOUND TO ENTER SYSTEM. APPLY TO FITTING THREADS ONLY.	
Air Conditioning Refrigerant	SAE-AMS-2518, Anti-Seize Compound, Graphite Petrolatum
Brakes	MIL-PRF-5606, Hydraulic Fluid
Fuel	SAE-AMS-2518, Anti-Seize Compound, Graphite Petrolatum
Landing Gear Air Valve	MIL-PRF-907E, Anti-Seize Thread Compound, High Temperature
Oil	SAE-AMS-G-6032, Grease (Gasoline and Oil Resistant)
Oxygen	A-A-58092, Ribbon Dope Thread Sealant, Anti-Seize and Sealing, Permacel 412
Pitot and Static	TT-A-580 (TT-S-1732), Anti-Seize Compound
Pneumatic Deicer	TT-A-580 (TT-S-1732), Anti-Seize Compound
CAUTION: LUBRICATE ENGINE FITTINGS ONLY WITH THE FLUID CONTAINED IN THE PARTICULAR LINE.	

14. Airframe Water Leak Check

(PIR-PPS60047, Rev.D.)

Use the following procedures to verify watertight integrity of the fuselage.

A. Required Equipment

(1) Choice of one of the following size/length combinations of standard garden hose:

3/4 inch inside diameter, maximum length 150 feet, or

5/8 inch inside diameter, maximum length 75 feet, or

1/2 inch inside diameter, maximum length 25 feet.

(2) A hose nozzle capable of adjustment. (Green Garden Nozzle 1220-C or equivalent is acceptable.)

B. Procedure

(1) Attach hose with nozzle to water tap at city water pressure. Check to insure that there are no leaks in the hose or fittings which would impair the check or cause other damage, with the tap full open.

(2) The check (except for baggage compartments located apart from the cabin) shall be conducted with an observer inside the aircraft.

(3) All cabin and baggage compartment doors and other controllable external closures shall be closed and latched during rainfall simulation.

(4) Spray water directly at opening around doors, windows, and at any other cockpit/cabin/separate baggage compartment closures. On multi-engine aircraft, spray wing root area (specifically the forward wing root seal) to ensure no water leaks into forward cabin floor area via engine control cable access port into the fuselage. Spray each door, window, and other closures for a period of 30 seconds to 1.5 minutes with the nozzle held between 3 to 10 feet from the point of water contact.

(5) If a leak allowing water to enter the cockpit/cabin/separate baggage compartment is detected, the seal or caulking (see Consumable Materials - General, in 91-10-00) shall be repaired, as required, and the check shall be repeated in the area where the leak was noted.

15. Hoses and Clamps

A. These items should be examined periodically and inspected carefully whenever maintenance activities cause hose disconnections.

B. Ends of hoses should be examined for rubber separation and slivers of rubber on inside diameter of hoses. These slivers can and do become detached. If this happens, the loose particles will migrate throughout the system and may eventually contribute to a failure.

C. Replace old, hard, cracked or brittle hose. Sections of the inner layers may separate.

D. Ensure hoses are clear and clean by blowing them out with shop air. Remove from aircraft as required.

CAUTION: DO NOT WIGGLE HOSE FROM SIDE TO SIDE DURING INSTALLATION. WIGGLING COULD CAUSE PARTICLES TO BE CUT FROM INNER WALL OF HOSE WHICH CAN LEAD TO DAMAGE OF OTHER COMPONENTS.

E. Where hose clearance is tight, making it difficult to reinstall it onto a fitting or barb, spray the fitting or barb with silicone. Let dry, then install hose by pushing it straight on.

CAUTION: WHEN REPLACING ANY THREADED FITTING, DO NOT USE PIPE DOPE, THREADLUBE, OR TAPE. USE ONLY SILICONE SPRAY, LETTING IT DRY BEFORE ASSEMBLY.

F. Hose clamps and fittings should be replaced when broken, damaged or corroded.

16. Solvent Cleaning

(PIR-PPS25000 PR 1001, Rev. R)

The following establishes the requirements for pre-cleaning and surface preparation by solvent cleaning prior to operations that require clean, water-break free surfaces.

A. Approved Solvents and Products

See “Chart 7”.

CHART 7
APPROVED SOLVENTS AND PRODUCTS

Solvent/Product	Piper P/N	Qty
Acetone	Procure locally	N/A
Methyl Ethyl Ketone (MEK)	Procure locally	N/A
Isopropyl Alcohol	Procure locally	N/A
Mineral Spirits	Procure locally	N/A
VM&P Naphtha	Procure locally	N/A
Kerosene	Procure locally	N/A
Socomore DS-108 (see 3.3.1)	279-561	1 gal
Citra-Safe	Procure locally	N/A
DuPont 2319S/Plastic Prep	Procure locally	N/A
Cloths (per SAE AMS 3819 Class 1 or 2, Grade A)	280-686	Roll of 12
Gloves	Gloves shall be lint-free, and of a material and quality sufficient to prevent transferring grease and oil.	
NOTE: Solvents shall be kept in approved EHS-approved containers. Contaminated solvents shall not be returned to clean solvent containers.		
NOTE: Solvents shall be used in well-ventilated areas only.		

B. Requirements

(1) General

(a) Metal surfaces, either primed or unprimed, shall be cleaned per paragraph “(2) Hand Wiping” on page 200026. Metal surfaces may be cleaned with any of the approved solvents listed in “Chart 7” on page 200024.

CAUTION: DO NOT USE SOCOMORE DS-108 ON NON-METALLIC PARTS.

(b) Windows shall be cleaned as follows.

WARNING: DO NOT USE THE FOLLOWING ON WINDOWS DUE TO CRAZING AND/OR ETCHING. THIS LIST IS NOT ALL-INCLUSIVE.

Acetone
Ammonia
Benzene
Carbon tetrachloride
Chlorinated hydrocarbons
Dry cleaning fluid
Fire extinguisher fluid
Gasoline
Isopropyl or Ethyl Alcohol in excess of 50% nominal concentration
Kitchen scouring compounds
Lacquer thinners
Methyl Ethyl Ketone (MEK)
Paint thinners
Toluene
Window sprays
Xylene

1) Polycarbonate Windows (PIR-PPS35101, Rev. P)

a) General Cleaning

Mild non-abrasive soap and water may be used for general cleaning.

b) Other Contaminates

1] Polycarbonate sheet should never be cleaned dry. The polycarbonate sheet shall be polished as stated in “Hand Polishing”, below.

2] To remove grease, oil, or other oil soluble contaminants, the only solvent that may be used is 50:50 mix of isopropyl alcohol and water. In all cases, polish per “Hand Polishing”, below.

c) Hand Polishing

Apply a suitable cleaner such as Novus 1 or 2, or Mirror Glaze M17 or M10, or Permatex Plastic Cleaner to a soft, grit-free cloth. Several applications may be necessary. Buff using a two-wheel system. The first wheel uses a buffing compound to remove shallow scratches. The second buffing wheel is used for restoring the gloss.

2) Acrylic / Extruded Acrylic Windows: (PIR-PPS35108, Rev. H)

a) General Cleaning

To clean acrylic, wash with plenty of non-abrasive soap and water using the bare hand to feel and dislodge any caked dirt or mud. A soft, grit free cloth, sponge or chamois may be used but only as a means of carrying water to the plastic. Dry with a clean damp chamois. Remove oil or grease using a 50/50 mixture of isopropyl alcohol and distilled water, mineral spirits, or VM&P aliphatic naphtha. The isopropyl alcohol mixture shall not be used in direct sunlight.

b) Other Contaminates

1] Excessive rubbing of acrylic sheet with a dry cloth may cause scratches or build-up of an electrostatic charge. Use a damp cloth or chamois to blot and remove dust and static charge. The acrylic sheet shall then be waxed as described in “Hand Polishing/Waxing” below.

2] To remove grease, oil, or other oil soluble contaminants, solvent clean with DuPont 2319S, VM&P Naphtha, or if not in direct sunlight, a 50/50 mixture of isopropyl alcohol and distilled water. In all cases, wax as described in “Hand Polishing/Waxing”, below.

3) Hand Polishing/Waxing

Polish with a fine abrasive in combination with a wax or tallow binder. The part shall be clean and dry before and after each polishing step. Apply a suitable cleaner such as Novus 1 or 2, or Mirror Glaze M17 or M10, or Permatex Plastic Cleaner to a soft, grit-free cloth. Several applications may be necessary. After cleaning and polishing, wax with a good grade of commercial wax. Wax shall be applied in a thin even coat and polished by rubbing lightly with a dry-soft cloth. Do not excessively rub in one spot.

CAUTION: DO NOT USE SOCOMORE DS-108 ON NONMETALLIC PARTS.

(c) Plastic surfaces, such as acrylic and other craze-prone materials as well as metal surfaces adjacent to such plastics, shall only be cleaned with DuPont 2319S, VM&P Naphtha, or if not in direct sunlight, a 50/50 mixture of isopropyl alcohol and distilled water.

CAUTION: DO NOT USE SOCOMORE DS-108 ON NONMETALLIC PARTS.

(d) Hoses, pipes, and lines shall be flushed with the appropriate solvent (see below). After flushing, remaining solvent shall be removed by blowing through the interior with clean, dry air.

(e) Cleaned surfaces shall be handled by gloved hands only.

(2) Hand Wiping

(a) Moisten a cloth with appropriate solvent (see “Chart 7” on page 200024 and “Chart 8” on page 200027). The solvent shall be poured onto the cloth. Cloths shall not be dipped into solvent containers unless a special dipping type container is used.

(b) Wipe the surface with the moistened cloth in one direction only, followed by wiping with a dry cloth in the same direction. The dry wipe shall occur while the surface is still wet and the applied solvent has not yet evaporated.

(c) After wiping dry, check the cloth for any signs of dirt, grease, or other contaminants. Repeat steps as necessary until all contaminants are removed as evidenced by a clean cloth after wiping dry.

(3) Solvent Selection

See “Chart 8” on page 200027. The ability of a solvent to dissolve a contaminant is dependant on the polarity of each material. Ideally, solvents should be selected so that the polarity of the solvent is as close as possible to the polarity of the contaminant. See “Chart 7” on page 200024 for approved solvents.

CHART 8
SOLVENT VS. CONTAMINANT

Relative Polarity	Representative Solvents	Representative Contaminants
Non-Polar Increasing Polarity Polar	Kerosene or Naphtha	Heavy grease, waxes, and oils. Safe for cured paint, most plastics, and silicones.
	Toluene, Xylene, Benzene	
	Mineral Spirits and Diethyl Ether	
	Trichloroethane and Tetrachloromethane	
	Ethyl Acetate	Medium-weight oils, adhesive residues. CAUTION: MAY DAMAGE ABS, ACRYLIC, AND OTHER PLASTICS USED IN INTERIORS.
	Methyl Ethyl Ketone (MEK) or Acetone	
	Pyridine and Triethylamine	
	Ethanol or Isopropyl Alcohol	General cleaning of surface oils, light residues, and ink. Safe for most plastics.
	Dimethylformamide	
	Ethanoic Acid	Salts and general soils. Full-spectrum cleaning with alkaline detergents.
	Distilled/Deionized Water	

(4) Cleaning of Tenacious Compounds and Removal of Cured Fuel Tank Sealants

CAUTION: DO NOT USE DS-108 ON ACRYLIC (WINDOWS) AS IT WOULD CAUSE CRAZING. ALL EXCESS AMOUNTS OF SOCOMORE DS-108 MUST BE REMOVED FROM AREAS THAT WOULD HAVE SEALANT APPLIED. USE APPROPRIATE PERSONAL PROTECTIVE EQUIPMENT (PPE) DURING THE USE OF THE SOCOMORE DS-108.

Tenacious (i.e., difficult to clean) compounds and cured fuel tank sealants on metal substrates can be cleaned using Socomore DS-108, followed by acetone cleaning to ensure no residual deposit of DS-108 remains on the surface. Special care needs to be taken to remove excess compound from metal substrates that would have fuel tank sealant applied to it with no intent on removing the sealant.

For removal of cured fuel tank sealants, apply Socomore DS-108 on the surface and allow to soak until sealant lifting is observed. The lifted sealant may be removed by something similar to a wooden tongue depresser followed by acetone cleaning to remove residual DS-108.

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PAINTING

1. Painting Safety

WARNING: OVERSPRAY FROM CERTAIN ENAMELS, IF PUT IN WATER, IS FLAMMABLE. STORE ALL OVERSPRAY IN COVERED CONTAINERS AWAY FROM BUILDINGS WHERE SPRAYING OPERATIONS ARE CONDUCTED.

WARNING: WASH ALL RAGS AND SPONGES USED TO APPLY ANY PHOSPHORIC ACID CONVERSION COATINGS (ALODINE) BEFORE DISPOSAL. IF MATERIAL DRIES ON RAG, THERE IS DANGER OF SPONTANEOUS COMBUSTION.

WARNING: MIX DOPES AND LACQUERS WITH AIR DRILL. DO NOT USE ELECTRIC DRILL. ARCING ELECTRIC DRILL MOTOR WILL IGNITE FUMES.

WARNING: VERIFY SPRAY ROOM IS WELL VENTILATED. A CONCENTRATION OF FUMES WILL CAUSE A DANGEROUS FIRE HAZARD OR INSUFFICIENT OXYGEN FOR THE OPERATOR.

CAUTION: DO NOT ALLOW PAINT STRIPPER TO CONTACT FIBERGLASS REINFORCED PARTS SUCH AS RADOMES, RADIO ANTENNAS, WING PARTS, OR WING TIPS. FIBERGLASS STRUCTURES MAY BE FINISHED WITH ACRYLIC LACQUER OR POLYURETHANE ENAMEL AND ARE DAMAGED BY THE STRIPPER.

2. Polyurethane Paint Safety

WARNING: POLYURETHANE PAINT MAY BE DANGEROUS TO YOUR HEALTH. SERIOUS INJURY WILL RESULT IF SAFETY PRECAUTIONS ARE NOT FOLLOWED.

WARNING: DURING TRANSIT AND STORAGE CHECK FOR SIGNS OF A BULGING CAN, OTHER THAN NORMAL ODOR, OR A CHANGE IN RESIN FROM A CLEAR TO A CLOUDY STATE. A SLOW CARBON DIOXIDE BUILDUP WILL CAUSE CAN TO BURST. REMOVE AND PROPERLY DISPOSE ANY DEFECTIVE CANS.

WARNING: ENSURE ADEQUATE VENTILATION AND WEAR APPROPRIATE BREATHING PROTECTION FACE MASK WHEN PAINTING.

WARNING: POLYURETHANE PAINTS CAN PRODUCE IRRITATION OF THE SKIN, EYES, AND RESPIRATORY TRACT DURING MIXING AND APPLICATION. EXPOSURE TO SPRAY VAPORS AND MISTS DURING SPRAY APPLICATION MAY CAUSE BREATHING DIFFICULTY, SHORTNESS OF BREATH, AND DRY COUGH. INDIVIDUAL SUSCEPTIBILITY IS A CONTROLLING FACTOR. ONCE SENSITIZED, MANY PEOPLE CANNOT TOLERATE ANY EXPOSURE AND MUST THEREAFTER AVOID EXPOSED WORK AREAS.

WARNING: PRODUCTION TYPE MIXING AND SPRAY PAINTING OPERATIONS MUST BE IN SPECIALLY DESIGNED, EXHAUST-VENTILATED AREAS.

WARNING: PAINTERS MUST BE FULLY CLOTHED WITH COLLARS BUTTONED AND SLEEVES TAPED AT THE WRIST. PAINTERS MUST WEAR FITTED, DOUBLE CARTRIDGE ORGANIC VAPOR RESPIRATOR WITH FRESH CARTRIDGE INSERTED DAILY, SOLVENT-RESISTANT GAUNTLET STYLE GLOVES, AND SAFETY GOGGLES.

3. Paint Application

WARNING: GROUND AIRCRAFT BEFORE PAINTING SO NO STATIC ELECTRICITY CHARGES BUILD UP AND DISCHARGE.

CAUTION: PROTECT WINDSHIELD WHEN MASKING AIRCRAFT. PAINT STRIPPERS, METAL BRIGHTENERS, AND SOLVENTS WILL DAMAGE WINDSHIELD.

CAUTION: BALANCE MOVABLE CONTROL SURFACES AFTER PAINTING. REFER TO APPROPRIATE MAINTENANCE MANUAL SECTIONS.

CAUTION: BEFORE FORCE DRYING AT ELEVATED TEMPERATURES, VERIFY THAT ALL FUEL TANK VENTS ARE UNOBSTRUCTED AND WILL NOT RESULT IN EXPANDED FUEL SPILLING ON NEWLY PAINTED SURFACES OR PAINT BOOTH FLOOR.

CAUTION: DO NOT PAINT PITOT TUBES, GAS CAPS, OR ANTENNA COVERS THAT WERE NOT FACTORY PAINTED.

CAUTION: DO NOT USE METALLIC PAINTS ON RADAR CONES OR ANTENNA COVERS.

CAUTION: DO NOT ALLOW SILICONE LUBRICANTS TO CONTACT ANY SURFACES TO BE PAINTED. SILICONE LUBRICANT IS VERY DIFFICULT TO REMOVE COMPLETELY.

4. Painting Sequence

For primer, tack, finish coats, and lacquer application:

A. Position airplane so airflow is from tail toward nose and overspray ahead of you.

B. To minimize overspray problems, have two painters work simultaneously on opposite sides of airplane.

C. Paint difficult areas such as landing gear, and wheel wells before flat surfaces. Paint the ends and leading edges of ailerons and flaps. Paint flap and aileron wells, wing tips, and leading and trailing edges.

D. Paint the bottom of the airplane first including bottom of horizontal tail surfaces. Starting at the root and working outward, spray chordwise. Work up fuselage and allow spray to cover sides. Work up to engine. Spray wing bottom. Start each painter at the root and work toward tip, spraying chordwise.

E. Lower airplane tail enough to reach fin top. When spraying fuselage top, tilt spray gun so overspray is ahead of area being painted and new paint will wipe out overspray. Spray primer across fuselage, vertical and horizontal tail surfaces, and wing.

5. Color Matching

See aircraft logbooks for color codes.

6. Trim and Registration Numbers

Apply predominant color first over entire surface. Apply trim colors over base color after it dries. When top of fuselage is to be painted white with a dark color adjoining it, apply light color and feather into area to be painted with dark color. When light color dries, place masking tape and paper along separation line, and apply dark color.

Allow paint to dry several hours before removing masking tape. Remove tape by pulling slowly parallel to surface. This will reduce the possibility of peeling off finish with tape.

Apply registration numbers by painting or affixing self-adhering plastic figures. They must be solid color lines contrasting with background. Location and size of identification numbers vary, per aircraft size. Location and size are found in the Federal Aviation Regulations.

7. Paint System Compatibility

Before applying new paint, find what type finish was used previously. Refer to the Piper parts catalog for correct paint number and color.

Identify paint finishes by applying engine oil to a small surface area. Old nitrocellulose finishes will soften in a few minutes. Acrylics, urethanes, and epoxy finishes show no effects.

If not identified, wipe down a small area with rag wet with methyl ethyl ketone. MEK picks up pigments from acrylic finishes, but not from epoxy or cured urethane coatings. Wipe surface, do not rub. Heavy rubbing picks up epoxy and urethane pigments from coatings not fully cured.

The use of different types of paint, with several coatings, make repair of damaged and deteriorated areas difficult. Paint finishes are not always compatible. The following are general rules for compatibility and are not necessarily listed in order of importance.

A. Old-type zinc chromate primer may be overcoated with wash primers if in good condition. Acrylic lacquer finishes will not adhere to this material.

B. Modified zinc chromate primer will not adhere to bare metal. Never use it over a dried film of acrylic nitrocellulose lacquer.

C. Nitrocellulose coatings will adhere to acrylic finishes, but reverse is not true. Do not use acrylic nitrocellulose lacquers over old nitrocellulose finishes.

D. Acrylic nitrocellulose lacquers will not adhere to nitrocellulose and epoxy finishes and to bare metal. For best results, apply lacquers over fresh, successive coatings of wash primer and modified zinc chromate. They also adhere to freshly applied epoxy coatings (dried less than 6 hours).

E. Epoxy topcoats adhere to all paint systems in good condition. Use epoxy for general touch touchup, including touchup of defects in baked enamel coatings.

F. Old wash primer coats may be overcoated directly with epoxy finishes. Apply a new second coat of wash primer if an acrylic finish is to be applied.

G. Old acrylic finishes may be refinished with new acrylic provided old coating is thoroughly softened using acrylic nitrocellulose thinner before paint touchup.

H. Repair damage to epoxy finishes by using more epoxy. Neither lacquer finish will stick to epoxy surfaces. In some instances, air drying enamels may be used for touchup of epoxy coatings if edges of damaged areas are roughened with abrasive paper.

8. Common Paint Troubles

A. Poor Adhesion - Paint properly applied to correctly pretreated surfaces will adhere satisfactorily. When thoroughly dry, paint must not be easily removed. Poor adhesion can result from:

(1) Inadequate cleaning and pretreatment.
(2) Inadequate stirring of paint or primer.
(3) Coating at incorrect time intervals.
(4) Application under adverse conditions.
(5) Bad application.

B. Spray Dust - Spray dust caused by atomized particles drying before reaching surface being painted fail to flow as a continuous film. Usual causes are incorrect air pressure or distance gun is held from work.

C. Sags and Runs - Excess paint causes wet paint film to move by gravity and presents a sagging appearance. Incorrect viscosity, air pressure, and gun handling, or inadequate surface preparation are frequent causes.

D. Spray Mottle - Sometimes known as orange peel or pebble, is caused by incorrect paint viscosity, air pressure, spray gun setting, or the distance the gun is held from work.

E. Blushing is one of the most common troubles. It appears as clouding or blooming of paint film. It is more common with cellulose than synthetic materials. It may be caused by moisture in air supply line, adverse humidity, drafts, or sudden temperature changes.

9. Storage

A. Store paint, enamel, and other finishing material in dry storage away from direct sunlight and heat. Mark each container with a code for identification.

B. Storage facilities must comply to Occupational Safety and Health Act (OSHA) requirements regarding air circulation, lighting, and fire protection. Lock storage facilities to prevent children and unauthorized personnel entry.

C. Invert pigmented materials every inventory so pigments will not pack to can bottom. Properly dispose of empty containers.

D. Use older materials first. Useful life of some finishes is limited.

E. Storage area temperatures must be approximately 50-90°F. If finishes are stored in temperature extremes, allow them to return to room temperature before using.

10. Painting Facility

WARNING: DO NOT BREATHE PAINT FUMES. FUMES DEPLETE THE OXYGEN SUPPLY REQUIRED BY THE BODY.

A. Painting facilities must conform to local, state, and OSHA standards with respect to air circulation, exhaust emissions, lighting, and fire protection.

B. Provide sufficient air movement in painting area so there is only a slight finishing material odor. Exhaust fans must be belt-driven and located near floor level. Locate fan’s motor away from fumes.

C. All spraying area personnel must wear approved respiration safety equipment.

11. Aircraft Finish Care

See Aircraft Finish Care, 12-00-00.

WIRING

Airplanes equipped with the Garmin GFC 700 Automatic Flight Control System have additional EMI shielding on some harness assemblies. If repair is required, use the following procedures as appropriate.

EMI Shielding

(PIR-107873 A)

A. PRT-S Wraparound Heat Shrink Tubing

The follow procedure is for installing “PRT-S” wraparound, heat shrinkable, EMI shielded tubing.

(1) Cut to Size

(a) Select a tubing diameter appropriate for the intended repair per “Chart 1”.

NOTE: Selecting a PRT-S tubing size smaller than recommended may result in the material splitting during the heating process.

(b) Cut the tubing to length and also cut an appropriate length of drain wire.

NOTE: When PRT-S is removed from the original spool it may have a flat profile with creased edges. Do not be alarmed by this shape, it will disappear when the product shrinks.

(2) Wrap Drain Wire

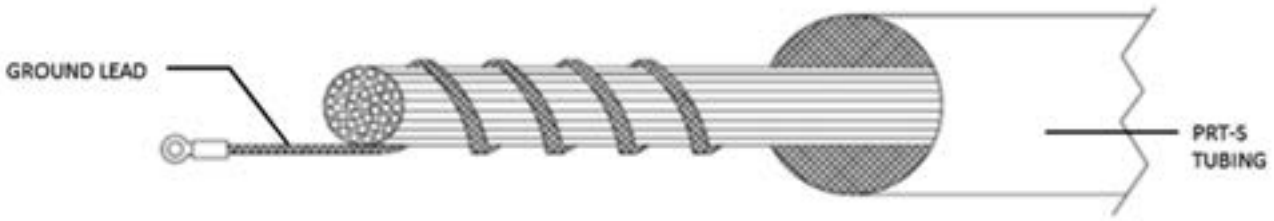
A ground lead can be added by simply wrapping a piece of woven braid around the primary wire bundle prior to installing the PRT-S tubing over the cable. The ground lead should be helically wound five revolutions around the outside of the primary wires. The helical winding insures that the bare lead will always be in contact with the shield due to the compressive contact of the shrink tubing. This winding will also create enough friction to keep the lead from pulling out if tugged on. See “Figure 1” on page 20972.

CHART 1
EMI SHIELDING SPECIFICATIONS

Harness	EMI Shielding
A/P Servo (Pitch/Roll)	SPC5092 or ZT14-04-005-3
A/P Servo (Pitch Trim/Yaw)	SPC5092 or ZT14-04-005-3
Magnetometer	SPC5092 or ZT14-04-005-1
OAT Probe	SPC5092 or ZT14-04-005-1
Radar	SPC5092 or ZT14-04-005-1

Wire Harness Size (Diameter Inches)	EMI Shielding Product Name / Manufacturer / P/N
up to 0.5	Wrap-A-Shield / Pro Power (aka SPC Tech) / SPC5092
0.125–0.187	PRT-S / Zippertubing / ZT14-04-005-1
0.25–0.37	PRT-S / Zippertubing / ZT14-04-005-3

(PIR-107872 B)



Wrapped Drain Wire
Figure 1



Wrapping PRT-S Around Harness
Figure 2

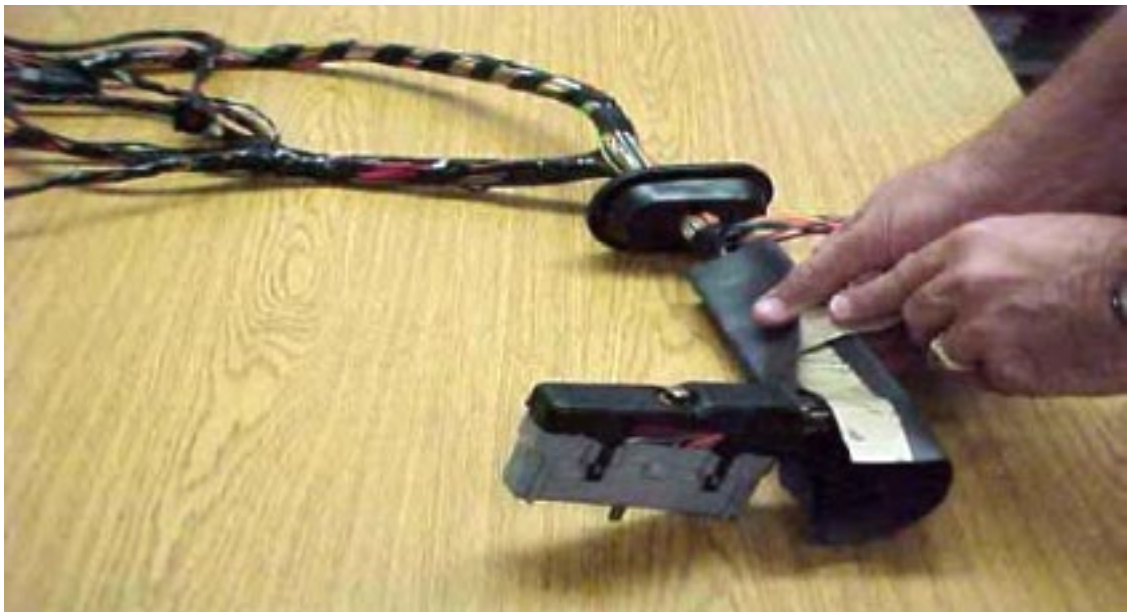
(3) Install the PRT-S Tubing

(a) Lay the tubing out so the metallic EMI shield material faces up, install the PRT-S tubing around the cable so the metallic shield cloth will be sandwiched between the wires and the heat shrinkable jacket material. Wrap the loose edge of the shield cloth around the wires and cover as much of the cable as possible. Peel off an inch or two of the adhesive's paper release liner on the shield material edge. Stick the adhesive edge to the wire bundle or cables. Wrap the entire assembly length in this manner. See "Figure 2".

NOTE: Wrap the shield around the cable as much as possible to ensure complete EMI shield coverage when the tubing is fully shrunk.

(b) Remove a short section of the paper release liner that covers the adhesive on the shrink tubing. Position the non-adhesive edge of the tubing over the exposed adhesive section and align so all the adhesive is covered but no more. Never over-wrap the tubing so much that you leave a loose flap of material at the overlap. Avoid contacting the adhesive with your fingers as much as possible. Remove short sections of the release liner until the entire length of tubing is sealed closed. Rub the overlap seam area firmly using your thumb to work out any wrinkles. Minor wrinkles will disappear when the tubing shrinks. See "Figure 3" on page 20973.

NOTE: Do not be concerned about the large size of the sealed tubing versus the wire size. It will always appear extremely large. You may shrink the tubing in place immediately after sealing; the closure adhesive system has no cure time.



Sealing / Closing PRT-S
Figure 3

(4) Heat Shrink

(a) PRT-S tubing begins to shrink at 90°C (194°F), however to achieve a timely and uniform installation it's desirable to use a heat gun temperature setting of 120-130°C (248-266°F). Always use a small reducer nozzle that will allow you to localize the stream. Never use a tubing reflector designed to spread the heat evenly or infrared type of heat gun.

(b) Begin shrinking the PRT-S tubing by directing the heat source at the overlap seam area near the center of the tubing. See "Figure 4" on page 20974. Apply heat to the overlap area only! The overlap area will tend to curl up into a "U" shape as it is heated. See "Figure 5" on page 20974. Continue applying heat in short bursts to this area until the surrounding tubing begins to pull the "U" shape back down flat. Do not be afraid to apply extra heat to the overlap area, as it will take extra dwell time for the inside layer of the overlap to shrink. Do not be alarmed if the outer layer of the overlap flap tends to lift off the adhesive and curl back slightly during initial heat exposure. With continued heating this lifted area will generally lay back down on its own, if not, lightly tap the lifted edge as necessary with a wooden stick. Shrink the overlap seam along its entire length before attempting to shrink the remaining tubing.

(c) After the overlap seam has fully shrunk begin applying heat to the remaining tubing. With small diameter tubing sizes you may find that most of the remaining tubing has already shrunk. Visually examine the entire circumference of the tubing and apply localized heat to any areas that show evidence of insufficient heating. While the tubing is hot you may work or form the PRT-S tubing and cable as necessary. Once properly positioned allow the finished assembly to cool to the point of being warm.



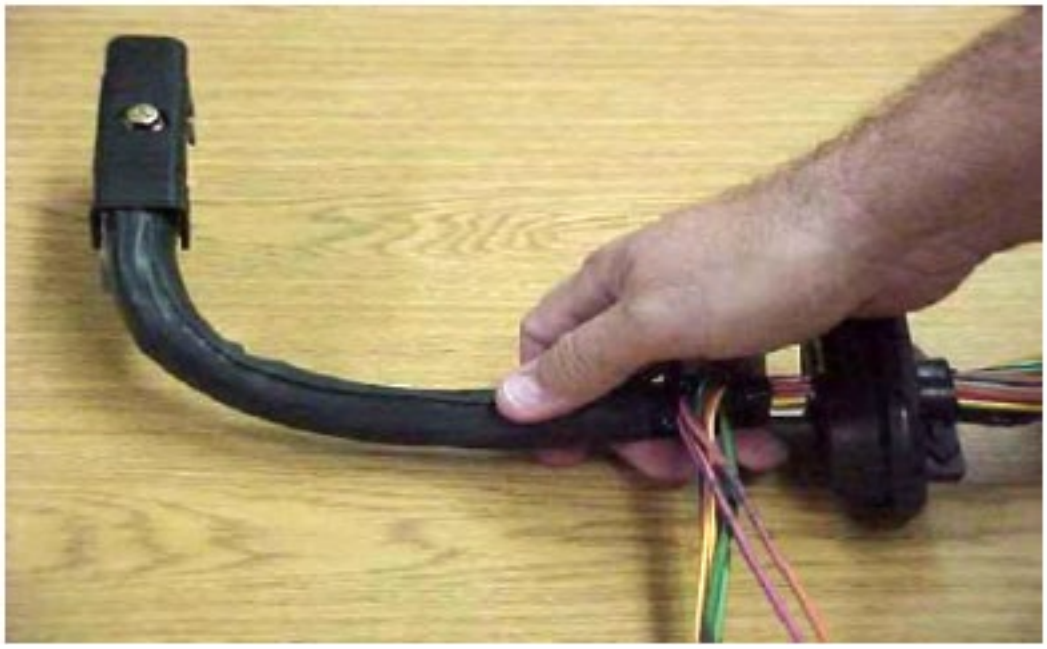
Heat Shrinking PRT-S
Figure 4



PRT-S Curling into "U" Shape
Figure 5

CAUTION: NEVER ATTEMPT TO REMOVE EXCESS ADHESIVE FROM THE OVERLAP AREA USING SOLVENT.

(d) The finished assembly may display a shiny, sticky area, parallel to the overlap seam. This is residual adhesive that was exposed as the overlap material pulled back during shrinking. The excess adhesive can be removed while the tubing is warm by rubbing your thumb along the overlap seam line. Rubbing the seam area will ensure good overlap contact and the excess adhesive will ball up ahead of your thumb. Remove excess adhesive as necessary and discard. See “Figure 6” and “Figure 7”.



Rubbing Adhesive Seam PRT-S
Figure 6



Residual Adhesive Ball PRT-S
Figure 7

B. SPC5092 Wrap-A-Shield Tubing

The following procedure is for installing “Wrap-A-Shield" tubing (SPC5092), a wraparound, EMI shielded tubing.

See “Figure 8”.

(1) Cut to Size

Cut the tubing to length and also cut an appropriate length of drain wire.

(2) Attach Drain Wire

(a) Locate the braid strap, and then solder the drain wire to the braid strap.

(b) Once the solder has cooled, wrap the drain wire around the primary wire bundle prior to installing the shield tubing over the cable.

NOTE: The ground lead should be helically wound four revolutions around the outside of the primary wires. This will form a strain relief to keep the lead from pulling out if tugged on.

(3) Install the Wrap-A-Shield Tubing

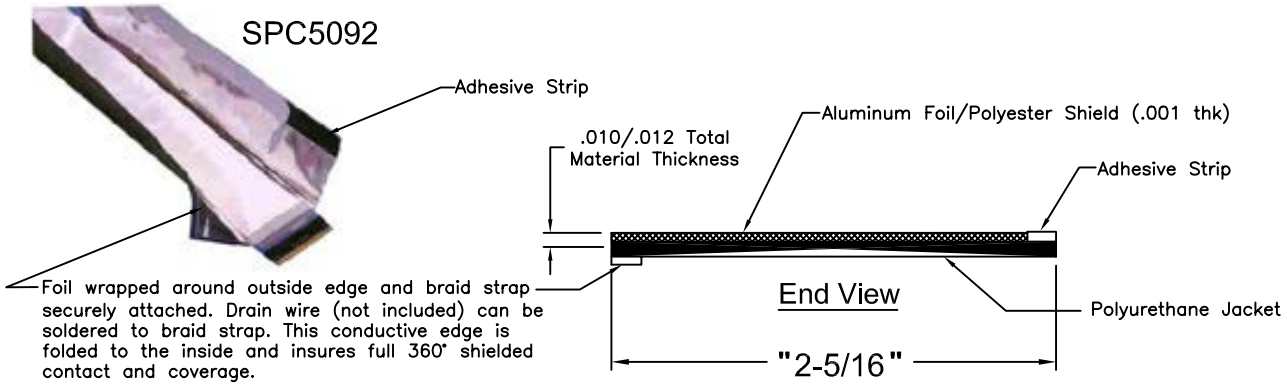
(a) Lay the tubing out so the metallic EMI shield material faces up, with the ground lead attached.

(b) Lay in your primary wire bundle, (don’t forget to wrap your ground lead as mentioned in Step (2)) starting from the braid strap edge begin to roll your shield and wire bundle towards the adhesive strip.

(c) Removing the backing paper from the adhesive strip, continue to roll your bundle, closing the tube as you roll over the exposed adhesive.

(d) Now secure the shield using MS3367-7-9 cable ties, starting one (1) inch from each end and at six (6) inch intervals.

(e) Finalize the assembly by rubbing your thumb along the overlapped seam to ensure good overlap contact with the adhesive.



SPC5092 Wrap-A-Shield
Figure 8

CHAPTER

21

ENVIRONMENTAL SYSTEMS

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CHAPTER 21

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE
21-List of Effective Pages	1	Oct 10/23
	2	Sep 30/18
21-Table of Contents	1	Oct 10/23
	2	Sep 30/18
	3	Sep 30/18
	4	Sep 30/18
21-00-00	1	Sep 30/18
	2	Sep 30/18
21-20-00	1	Sep 30/18
	2	Sep 30/18
	3	Oct 10/23
	4	Sep 30/18
	5	Sep 30/18
	6	Sep 30/18
	7	Sep 30/18
	8	Sep 30/18
	9	Sep 30/18
	10	Oct 10/23
21-40-00	1	Sep 30/18
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	28	Sep 30/18
	29	Sep 30/18
	30	Sep 30/18
	31	Sep 30/18

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	38	Sep 30/18			
	39	Sep 30/18			
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	42	Sep 30/18			
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GENERAL

This chapter deals with the operating, servicing, and inspecting procedures for the heating, air conditioning, and ventilating systems.

Description

There are six overhead fresh air vents which are supplied by a separate inlet in the dorsal fin. This system can be supplemented by an optional blower.

A Hartzell Engine Technologies (aka Kelly Aerospace, aka Janitrol) combustion heater Model B3040 (91E88-1), or B3500 (91E88-1(EL)), or Model I-3500 (91E88-1(IS)) is installed as standard equipment.

When the optional air conditioning system is not installed cabin ventilation and cooling is provided solely by fresh air through the overhead vent system.

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DISTRIBUTION

1. Cabin Vent System

See “Figure 1” on page 21202.

Overhead Vent Blower

The blower is mounted in the aft section of the fuselage and is connected to the overhead vent system. The blower draws air in from the dorsal fin and forces it through the ducting, whenever desired. (The three position blower switch in the overhead panel controls the two speed blower.)

(1) Removal

(a) Remove the access door from the aft wall of the baggage area.

(b) With the master switch off, disconnect the plug assemblies at the blower assembly.

(c) Remove the inlet and outlet hoses from the blower assembly by removing the clamps.

(d) Remove the screws, washers and nuts that secure the blower assembly to the hanger braces.

(e) Remove the screws and washers which secure the blower assembly to the retainer and hangers.

(f) Remove the blower assembly from the aircraft.

(2) Dismantling

(a) Remove the hose duct from the forward edge of the blower assembly by removing the nuts, washers and screws.

(b) Remove the cover from the blower assembly by removing the nuts, washers and screws.

(c) Remove the blower fan from the motor shaft by removing the set screw.

(d) For removal of the motor, proceed as follows:

1) Separate the plate from the motor cover by carefully drilling out the connecting rivets.

2) Cut the motor wires at the edge of the receptacle and plug and remove the wire ends from the blocks.

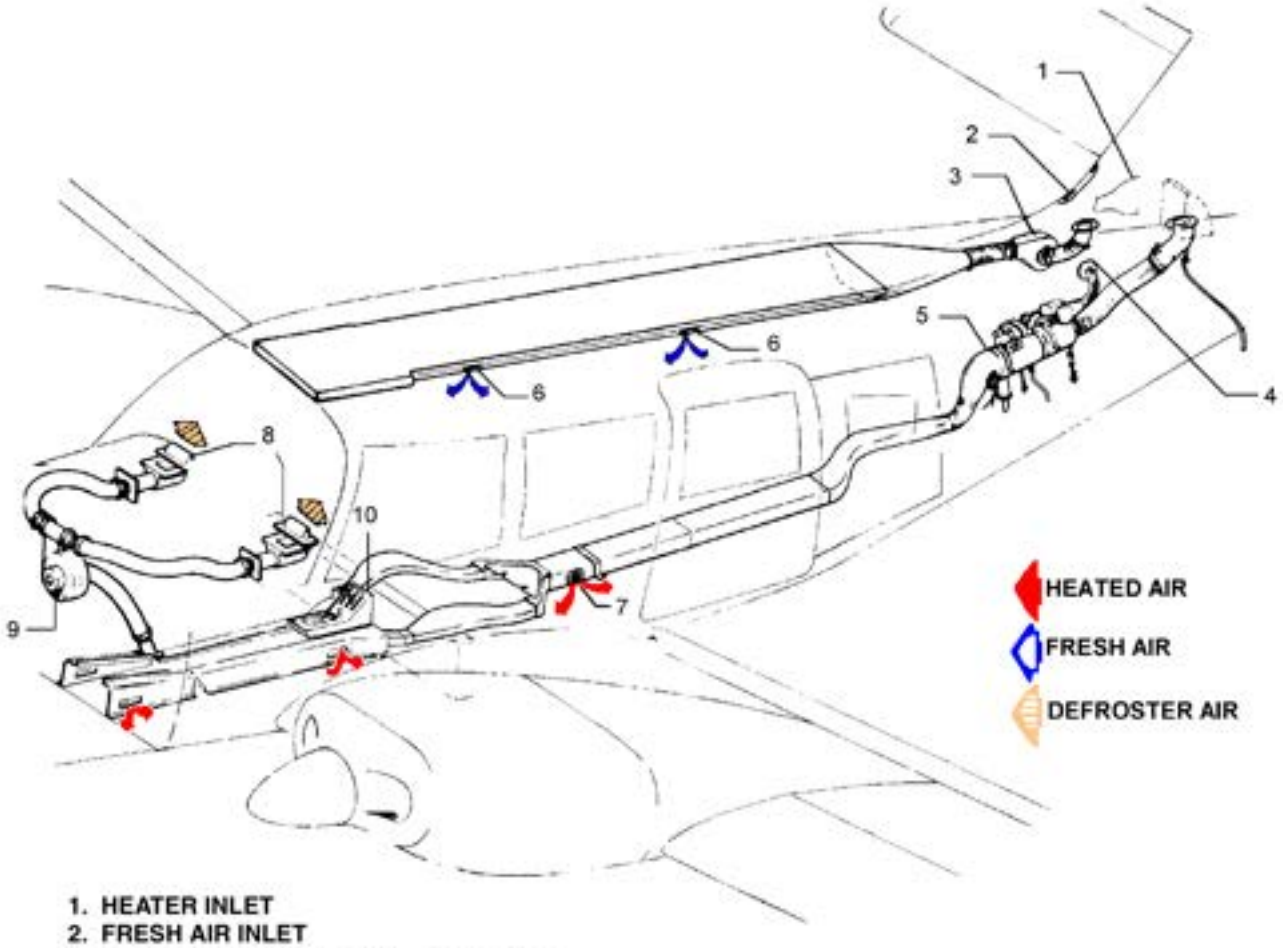
3) Remove the motor from the mounting plate by removing the nuts, washers and bolts.

CHART 1
BLOWER WIRE COLOR CODES

MOTOR - 24Vdc Dukes 1482-22-1		
	Pin Nos.	Wire Color
Ground	1	Green
Low Speed	2	Red
High Speed	2	Orange

NOTE: Pin number 1 is at the pointed side of the plug and receptacle.

C404
37069 78629 78630



1. HEATER INLET
2. FRESH AIR INLET
3. OVERHEAD VENT BLOWER - OPTIONAL
4. COMBUSTION AIR BLOWER INLET
5. COMBUSTION HEATER ASSEMBLY
6. FRESH AIR OUTLETS
7. HEATER OUTLETS
8. DEFROSTER OUTLETS
9. DEFROSTER BLOWER
10. HEATER AND DEFROSTER CONTROL ASSEMBLY

Cabin Heater, Defrosters, and Overhead Vent System
Figure 1

(3) Rebuilding

(a) Mount the motor on the plate and secure it with the bolts, washers and nuts. Be sure that the motor nuts are snug and the shaft spins freely.

(b) Position the cover over the motor plate with the motor wires protruding through the cover grommet.

(c) With the holes in the cover matching the holes in the motor plate, secure the two parts together with rivets.

(d) Apply PR-307 sealant to fill any opening left after the wires are brought through the grommet.

(e) Install the wires in the plug and receptacle.

(f) Position the blower fin on the motor shaft and secure with set screw.

(g) Secure the cover to the blower assembly with screws, washers and nuts.

(h) Position the hose duct on the blower assembly and secure it with screws, washers and nuts. The screws must be installed with their heads inside the duct.

(i) After cleaning the surfaces of all old sealant, use white rubber chalk PR-307 sealant to seal where the duct attaches to the blower assembly.

(4) Installation

(a) Position the blower assembly in the hangers and retainer and install the washers and screws.

(b) Install the nuts, washers and screws securing the blower assembly to the hanger braces.

(c) Seal all hose joints with 3M 390 Duct Tape; then install the inlet and outlet hoses securing them with the clamps.

(d) With the master switch off, connect the plug and receptacles at the blower (see “Chart 1” on page 21201).

(e) Check the blower for the proper operation.

(f) Install the access door to the aft wall of the baggage area and secure with the attaching hardware.

2. Avionics Cooling Blower (Optional)

A. Standard and Garmin G600 Installations

The avionics cooling blower is installed on the right side of the fuselage, behind the instrument panel, at F.S. 49.50. The cooling air is directed through ducts to installed avionics equipment. See “Figure 2” on page 21204.

NOTE: Actual equipment served will vary depending on optional equipment installed.

(1) Removal

(a) Remove ducts from top of blower. (Ducts are held in place by Ty-Raps.)

(b) Disconnect blower motor wires at connector.

(c) Remove screws, washers and nuts that attach the blower assembly to its mounting bracket.

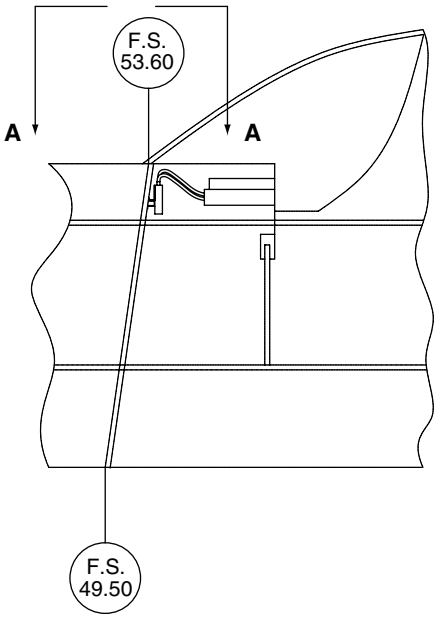
(2) Installation

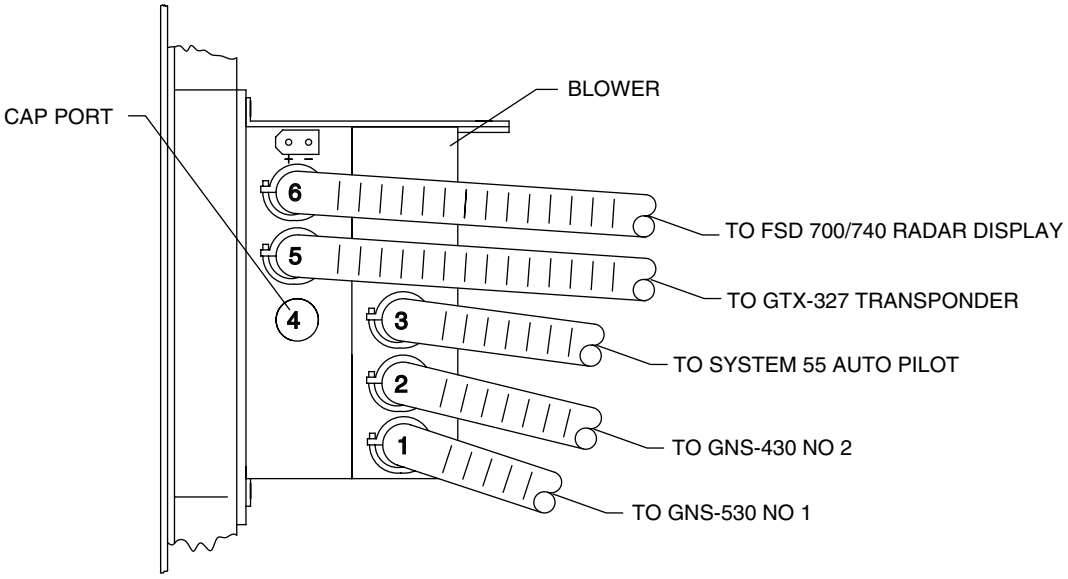
(a) Attach the blower assembly to the mounting bracket with the screws, washers and nuts.

(b) Reconnect the blower motor wires.

(c) Attach the ducts to the tops of the blower assembly. Ensure that the duct ends overlap the ports on top of the blower one-half inch minimum. Secure with Ty-Raps.

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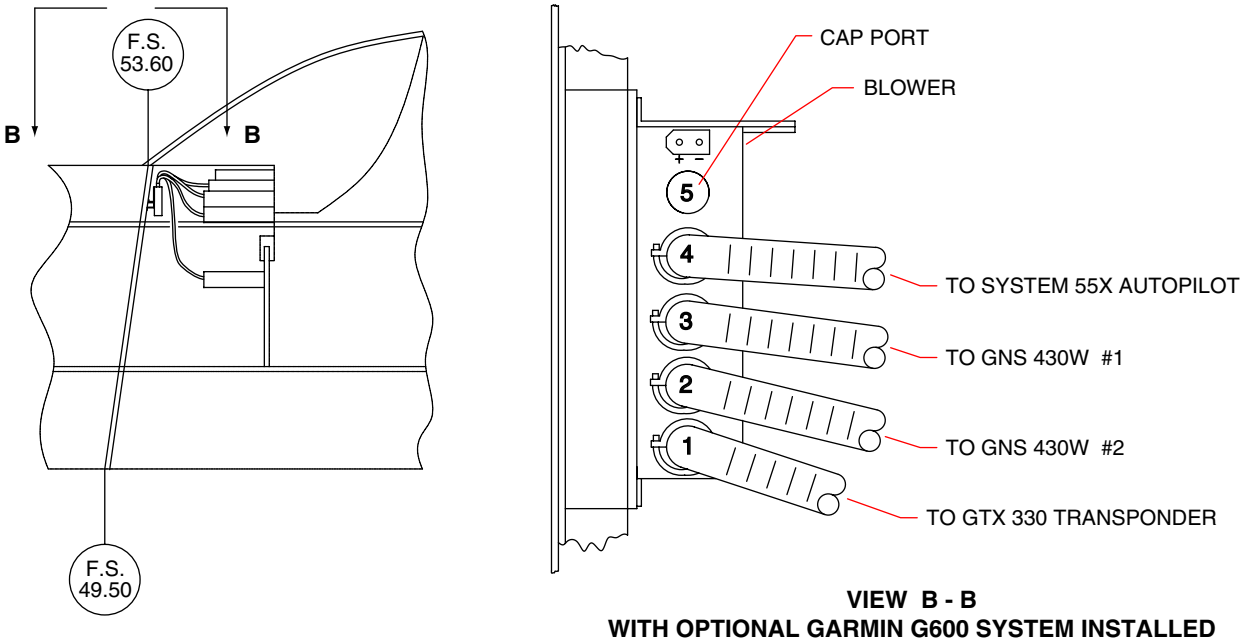


VIEW A-A

Effectivity
3449161, 3449178 and up

Avionics Cooling
Figure 2 (Sheet 1 of 2)

107350 H



VIEW B - B
WITH OPTIONAL GARMIN G600 SYSTEM INSTALLED

Avionics Cooling
Figure 2 (Sheet 2 of 2)

Effectivity
with Garmin G600

B. Garmin G1000 Installation

In S/N's 3449459, 3449467 and up, an avionics cooling blower is mounted to a bracket attached to the right fuselage side-panel just forward of (i.e. - behind) the instrument panel. In addition, three cooling fans are mounted in the avionics equipment racks attached to the instrument panel.

See "Figure 3" on page 21207.

NOTE: Actual equipment served will vary depending on optional equipment installed.

(1) Avionics Cooling Blower

(a) Removal

1) Remove ducts from top of blower. (Ducts are held in place by Ty-Raps.)
2) Disconnect blower motor wires at connector.
3) Remove screws, washers and nuts that attach the blower assembly to its mounting bracket.

(b) Installation

1) Attach the blower assembly to the mounting bracket with the screws, washers and nuts.
2) Reconnect the blower motor wires.
3) Attach the ducts to the tops of the blower assembly. Ensure that the duct ends overlap the ports on top of the blower one-half inch minimum. Secure with Ty-Raps.

(2) Rack-mounted Cooling Fans

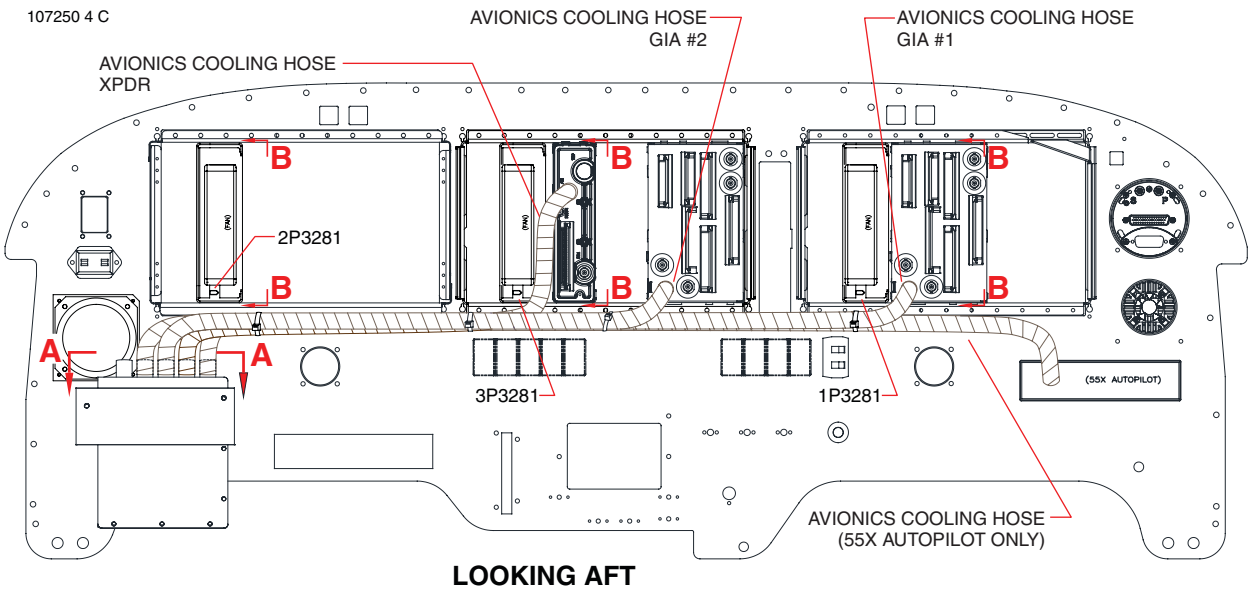
Rack-mounted cooling fans (Safe 328) are removed from the front (i.e., aft side) of the instrument panel. Accordingly, the associated PFD or MFD must be removed to allow access to the cooling fan. See 34-25-01.

(a) Removal

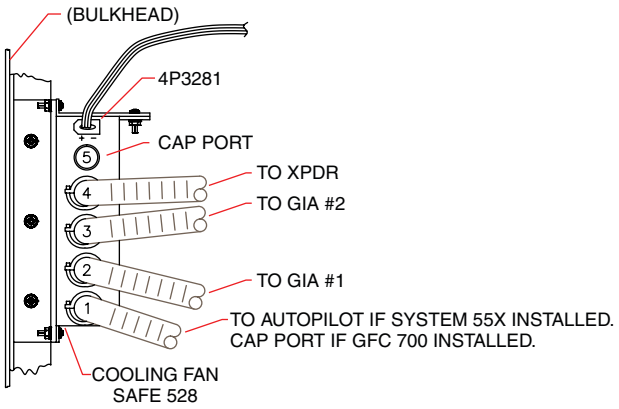
1) Disconnect fan wires at connector.
2) Remove screws (6 ea.) that secure fan rack to the avionics equipment mounting rack.
3) Remove fan rack and fan.
4) Remove screws (3 ea.) that secure fan to rack and remove fan.

(b) Installation

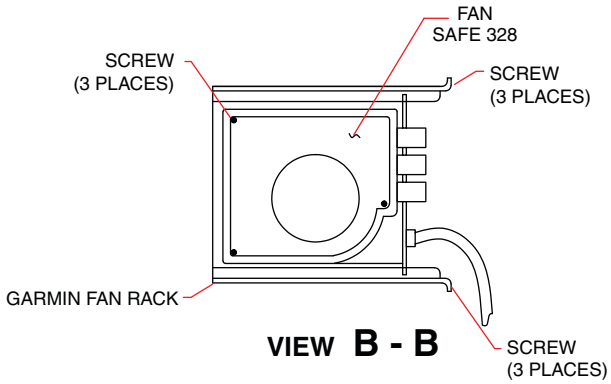
1) Insert fan into fan rack and secure with screws (3 ea.)
2) Insert fan rack into avionics equipment mounting rack and secure with screws (6 ea.).
3) Connect fan wires to fan.



LOOKING AFT



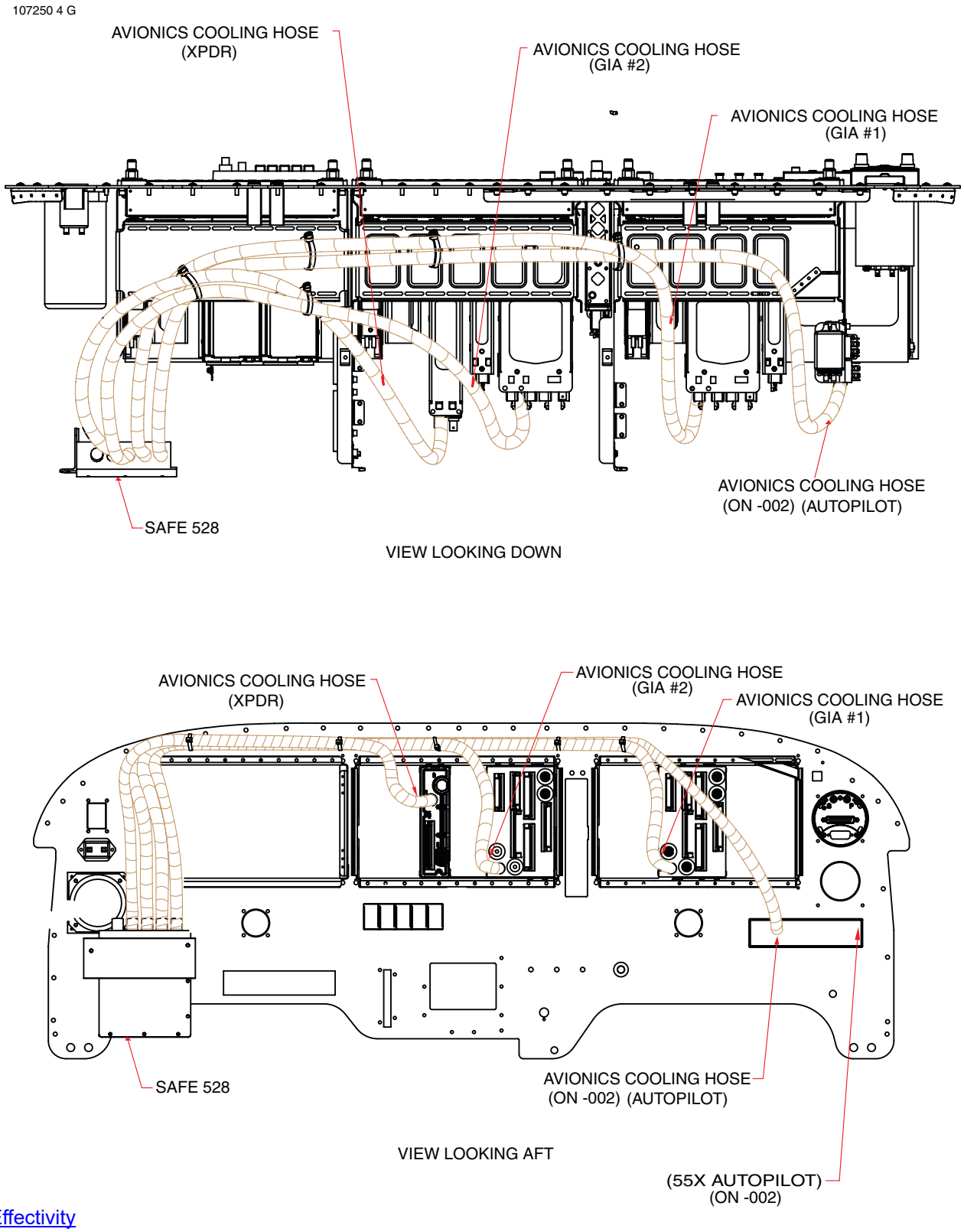
VIEW A - A
ROTATED 90° CLOCKWISE



VIEW B - B

Avionics Cooling
Figure 3 (Sheet 1 of 2)

Effectivity
with Garmin G1000 S/N's
3449459, 3449467 - 3449488



Effectivity
with Garmin G1000
S/N 3449489 and Up

Avionics Cooling
Figure 3 (Sheet 2 of 2)

C. Garmin G1000 NXi Installation

S/N's 3449509 and up.

See "Figure 4" on page 212010.

(1) Avionics Cooling Blower

(a) Removal

1) Remove ducts from top of blower. (Ducts are held in place by Ty-Raps.)

2) Disconnect blower motor wires at connector.

3) Remove screws, washers and nuts that attach the blower assembly to its mounting bracket.

(b) Installation

1) Attach the blower assembly to the mounting bracket with the screws, washers and nuts.

2) Reconnect the blower motor wires.

3) Attach the ducts to the tops of the blower assembly. Ensure that the duct ends overlap the ports on top of the blower one-half inch minimum. Secure with Ty-Raps.

(2) Rack-mounted Cooling Fans

Rack-mounted cooling fans (Lone Star Aviation Cyclone 21) are removed from the front (i.e., aft side) of the instrument panel. Accordingly, the associated PFD or MFD must be removed to allow access to the cooling fan. See 34-25-02.

(a) Removal

1) Disconnect fan wires at connector.

2) Remove screws (6 ea.) that secure fan to the avionics equipment mounting rack.

3) Remove fan rack and fan.

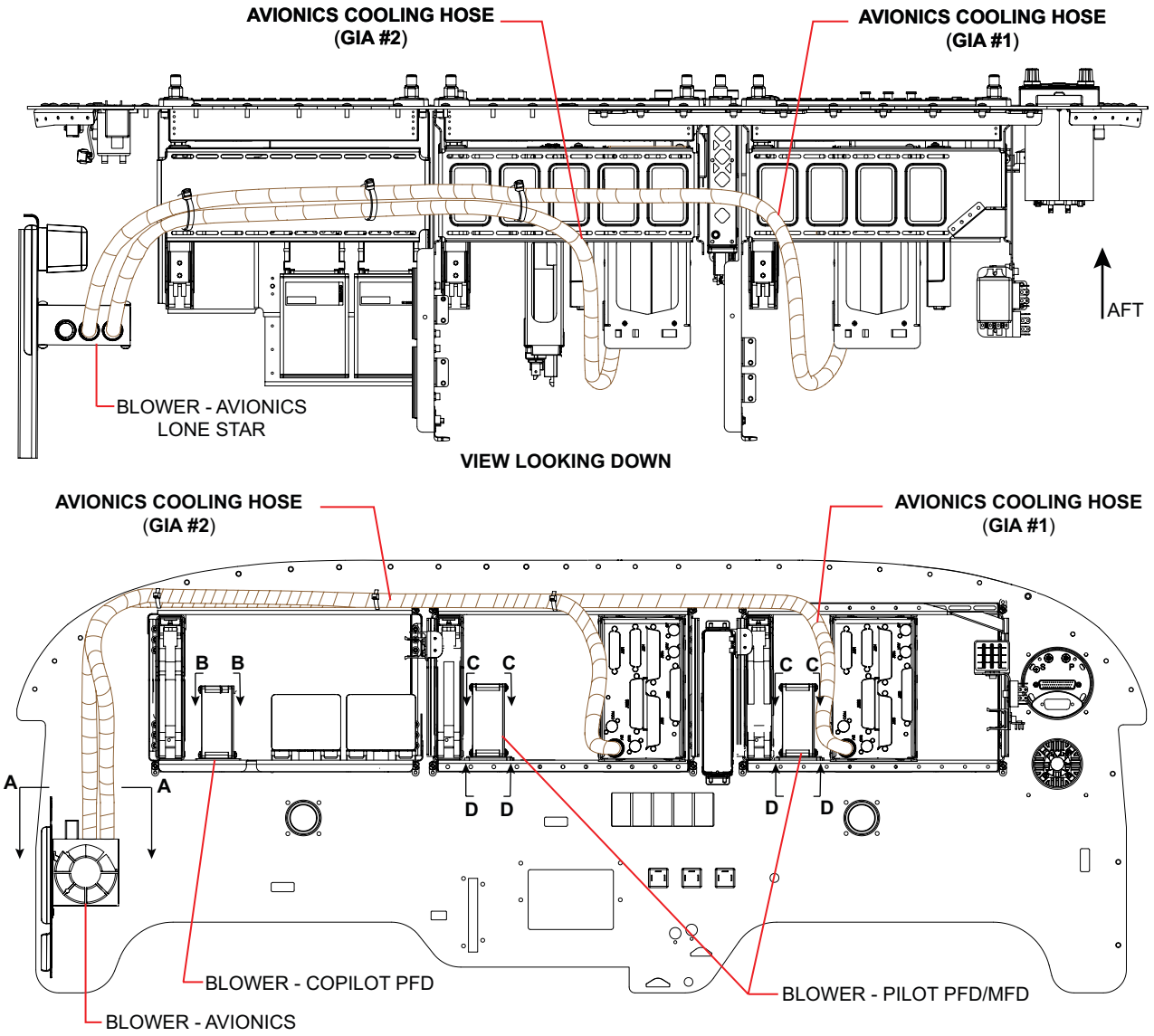
(b) Installation

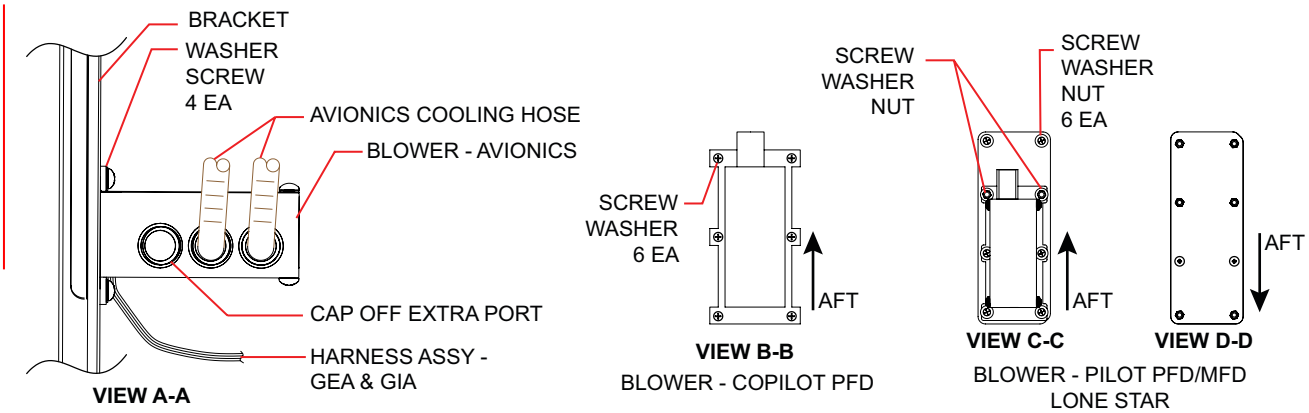
1) Insert fan rack into avionics equipment mounting rack

2) Secure with screws (6 ea.).

3) Connect fan wires to fan.

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Effectivity
with Garmin G1000 NXi

Avionics Cooling
Figure 4

HEATING

1. Hartzell Engine Technologies

   A. Description

   See “Figure 1” on page 21407.

   CAUTION: WHEN CABIN HEAT IS OPERATED, HEAT DUCT SURFACE BECOMES HOT. THIS COULD RESULT IN BURNS IF ARMS OR LEGS ARE PLACED TOO CLOSE TO HEAT DUCT OUTLETS OR SURFACE.

   A Hartzell Engine Technologies (aka Kelly Aerospace, aka Janitrol) combustion heater Model B3040 (91E88-1), or B3500 (91E88-1(EL)), or Model I-3500 (91E88-1(IS)) is installed as standard equipment.

   NOTE: Repetitive inspections associated with older model heaters can be eliminated by upgrading to an I-Series heater per Hartzell Service Bulletin No. A-101, latest revision.

   Heated air for the cabin and defroster operation is obtained from the combustion heater located in the tail section of the airplane. Fresh air is supplied to the heater from an intake located in the dorsal fin and routed through the heater and into the cabin through six adjustable outlets. Operation of the heater is controlled by a three-position switch located on the heater control console between the pilot’s and copilot’s seats and labeled FAN, OFF and HEATER. The FAN position will operate the ventilation blower on the heater and may be used for cabin ventilation or windshield defogging on the ground when heat is not desired. There is a defroster blower in the same distribution system to provide additional defrost capability when required. The defroster control switch must be in the ON position to energize the defroster blower.

   For cabin heat, the air intake lever located on the heater control console must be partially or fully open and the three-position switch set to HEATER. This will start the fuel flow and ignite the burner simultaneously. With instant starting and no need for priming, heat should be felt within a few seconds. There are two safety switches installed at the intake valve located aft of the heater unit which are activated by the intake valve and wired to prevent both fan and heater operation unless the air intake valve is moved off the closed position.

   Regulating the heater and airflow is accomplished by adjusting the levers on the heater control console. The right-hand lever regulates the air intake valve, while the left-hand lever regulates cabin temperature. Cabin temperature and air circulation can be varied to suit individual requirements by various combinations of lever settings.

   Heat may be supplied before starting the engines by turning on the master switch, opening the air intake valve, and placing the heater switch in the HEATER position.

   An overheat limit switch is located in the forward outboard end of the heater vent jacket and acts as a safety device to render the heater inoperative if a malfunction should occur. Operation of this switch results in illumination of the heater overheat warning light (red) in the annunciator panel.

   In Seneca IVs and Seneca Vs (circa 1997–2005), a red reset button on the switch can be reached through the bulkhead access panel into the aft fuselage.

   NOTE: Any of the above airplanes with a service replacement heater installed 2006 or later may function as described below.

   In Seneca Vs (circa 2006 and later), the overheat limit switch resets automatically, if activated.

   To prevent activation of the overheat limit switch upon normal heater shutdown during ground operation, turn the heater three-position switch to the FAN position for two minutes, while leaving the air intake lever in the open position, before turning the switch to the OFF position.

B. Basic Components

(1) Spark - Spray Ignition

See “Figure 2” on page 21408.

The controlled atomized spray from a specially designed spray nozzle, coupled with high voltage spark plug ignition, ensures instant firing and continuous burning under all flight conditions.

Heat is produced by burning a fuel-air mixture in the combustion chamber of the heater. Aviation gasoline is injected into the combustion chamber through the spray nozzle. The resulting cone-shaped fuel spray mixes with combustion air and is ignited by a spark from the spark plug. Electric current for ignition is supplied by an ignition unit which converts 28 volts to high voltage oscillating current to provide a continuous spark across the spark plug gap. A shielded, high voltage lead connects the ignition assembly to the spark plug. Combustion air enters the combustion chamber tangent to its surface and imparts a whirling or spinning action to the air. This produces a whirling flame that is stable and sustains combustion under the most adverse conditions because it is whirled around itself many times. Therefore, ignition is continuous and the combustion process is self-piloting. The burning gases travel the length of the combustion tube, flow around the inside of the inner tube, pass through crossover passages into an outer radiating area, then travel the length of this surface and out the exhaust.

Ventilating air passes through the heater between the jacket and combustion tube assembly outer surface and through an inner passage in the assembly. Consequently, ventilating air comes into contact with two or more heated cylindrical surfaces.

(2) Fuel Regulator and Shutoff Valve

See “Figure 11” on page 214026.

This unit provides preset, regulated fuel pressure as well as remote shutoff to the heater, regardless of fuel inlet pressure variations. It is set for 7.5 ± .5 psi. The shutoff valve is operated by a solenoid.

(3) Duct Switch

See “Figure 3” on page 21408.

This switch is installed in the ventilating manifold upstream from the heater to sense the ventilating air outlet temperature. To select the desired cabin temperature, the switch may be adjusted manually from a high of 250°F + 10° downward through a range of 146°F ± 6°. The switch has a differential of 15°F ± 5° at any given setting.

(4) Combustion Air Blower

This centrifugal type blower supplies combustion air to the combustion chamber of the heater.

(5) Ventilating Air Blower

This blower is attached to the inlet end of the heater assembly and provides a source of ventilating air through the heater. Ram air from the air intake is used during flight.

(6) Heater Hour Meter (Optional)

See “Figure 1” on page 21407.

A heater hour meter is available as an original equipment option; or, as a service kit (No. 764-094) from your Piper Distributor. Use of a heater hour meter permits heater maintenance and overhaul to be based on actual hours of operation.

(7) Operating Controls

See “Figure 1” on page 21202.

NOTE: The schematic diagram (“Figure 5” on page 214016) shows the heater circuit including the electrical wiring in the airplane.

The HEATER SWITCH is connected in the line that supplies electrical power to all heater equipment and controls. When this switch is in the OFF position, the entire heater system is inoperative. This switch has a FAN position which permits use of the ventilating air blower to circulate cool air through the system for summer ground operation. With the switch in FAN position, the heater is inoperative and only the ventilating air blower is energized.

C. Troubleshooting

See “Chart 1”.

CHART 1 (Sheet 1 of 3)
TROUBLESHOOTING HEATER

Trouble	Cause	Remedy
Heater fails to light.	Heater switch or circuit breaker off.	Turn on heater switch or close circuit breaker.
	Low voltage supply.	Apply external power supply.
		Attempt to start heater.
	Fuel cut off from tank.	Turn on heater switch.
	Regulator not operating properly.	Check for low pressure or replace regulator.
NOTE: When making the fuel pressure check, be sure fuel is flowing through the nozzle. The fuel regulator can be adjusted. Turn the adjusting screw clockwise to increase fuel pressure and counterclockwise to decrease it.		
	Restriction in fuel nozzle orifice.	Remove nozzle and clean or replace it.
	Fuel heater solenoid not operating.	Remove and check solenoid. Replace if faulty.
	Fuel lines clogged or broken.	Inspect all lines and connections. It may be necessary to disconnect lines at various points to determine where the restriction is located.
	Ignition vibrator inoperative.	Replace vibrator.
	Manual reset limit (overheat) switch open.	Press reset button firmly (overheat light will illuminate when heater switch is on) and recheck to determine reason for switch opening.

CHART 1 (Sheet 2 of 3)
TROUBLESHOOTING HEATER

Trouble	Cause	Remedy
Heater fails to light. (continued)	Combustion air pressure switch open. (Defective switch or low combustion air blower output.)	Check for low blower output due to low voltage and correct it. If switch is defective, replace it.
	Cycling switch open.	Replace if defective.
	Duct switch open.	Operate control to see if switch will come on. Replace switch if defective.
Ventilating air blower fails to run.	Heater switch “OFF”.	Turn heater switch ON.
	Broken or loose wiring to motor.	Check and repair wiring.
	Circuit breaker open.	Close circuit breaker.
	Worn motor brushes.	Replace motor brushes.
	Blower wheel jammed.	Remove and check the ventilating air blower wheel and realign if necessary.
	Motor burned out.	Remove blower assembly and replace motor.
	Defective radio-noise filter.	Replace filter.
Combustion air blower fails to run.	Faulty wiring to motor.	Inspect and replace faulty wiring.
	Poor ground connection.	Tighten ground screw.
	Worn motor brushes.	Replace motor brushes.
	Blower wheel jammed. (Usually indicated by hot motor housing.)	Overhaul the combustion air blower.
	Defective radio-noise filter.	Replace filter.
	Faulty or burned-out motor.	Remove combustion air motor for overhaul or replacement of motor.
Heater fires, but burns unsteadily.	Insufficient fuel supply.	Inspect fuel supply to heater, including shutoff valve, solenoid valve and fuel lines.
		Make necessary repairs.

CHART 1 (Sheet 3 of 3)
TROUBLESHOOTING HEATER

Trouble	Cause	Remedy
Heater fires, but burns unsteadily. (cont.)	Spark plug partially fouled.	Replace spark plug. See Caution.
CAUTION: DO NOT CREATE A SPARK GAP BY HOLDING THE LEAD TO THE HEATER JACKET. THIS CAN RESULT IN DAMAGE TO THE LEAD AND IGNITION UNIT AND THE OPERATOR MAY RECEIVE AN ELECTRICAL SHOCK.		
	Loose primary connection at ignition assembly.	Tighten the connection.
	Faulty vibrator.	Replace the vibrator.
	Combustion air blower speed fluctuates. (Can be caused by low voltage, loose blower wheel, worn brushes or motor.)	Remove and overhaul the combustion air blower assembly as required or correct low voltage condition.
	High voltage leak in lead between ignition assembly and spark plug.	Replace ignition assembly.
	Inoperative ignition assembly.	If vibrator is in good condition, replace ignition assembly only.
	Restriction in fuel nozzle orifice.	Clean or replace nozzle.
	Nozzle loose in retainer or improper spray angle.	Tighten or replace the nozzle as required.
Heater starts then goes out.	Lack of fuel at heater.	Check fuel supply through all components from the tank to the heater. Make necessary corrections.
	Inoperative or chattering combustion air pressure switch.	Adjust or replace switch.
	Inoperative overheat switch.	Replace switch.
	Inoperative cycling switch.	Adjust or replace the switch.
	Low voltage.	Attach external power.
Heater fails to shut off.	Fuel solenoid valve in heater stuck open.	Remove and replace solenoid assembly.
	Inoperative duct and cycling switch.	Check and repair.
	Defective heater switch.	Replace the heater switch.

D. Maintenance Service

Instructions contained in this section consist of periodic inspection, adjustments, and minor corrections required at normal designated intervals for the purpose of maintaining the heating system in peak operating condition. These inspections assume that a heating system includes accessory components mentioned in preceding paragraphs.

In airplanes with a Model I-3500 (91E88-1(IS) combustion heater installed, service and maintain per the appropriate Hartzell Engine Technologies component maintenance manual.

E. Inspection

(1) Preflight and/or Daily Inspection

(a) Inspect the ventilating air inlet, combustion air inlet, exhaust outlet and fuel drains for possible obstructions. Make sure that all of these openings are clear of any restrictions and that no damage has occurred to the exhaust, cold or hot fuel drains, water drain or fuel line drain.

(b) Place the HEATER SWITCH in the ON (or HEAT) position. The ventilating air blower and combustion air blower should operate. Operate both combustion and ventilating air blowers and check each for unusual current draw, noise or vibration.

NOTE: Proceed with the Operating Procedure, above, and verify heater system basic operation.

(2) 100 Hour Inspection

For heaters with 500 heater operating hours or twenty-four (24) months time-in-service since new, or overhauled with a new combustion tube assembly, each 100 hours or twenty-four (24) months, whichever occurs first, conduct the following inspection.

(a) Inspect ventilating air and combustion air inlets and exhaust outlet for restrictions and security at the airplane skin line.

(b) Inspect the drain lines to make sure they are free of obstructions. Run a wire through them if necessary to clear any obstructions.

(c) Check all fuel lines for security at joints and shrouds, making sure that no evidence of leaks exists. Also, check for security of attachment of fuel lines at the various attaching points in the airplane.

(d) Inspect electrical wiring at the heater terminal block and components for loose connections, possible chafing of insulation and security of attachment points.

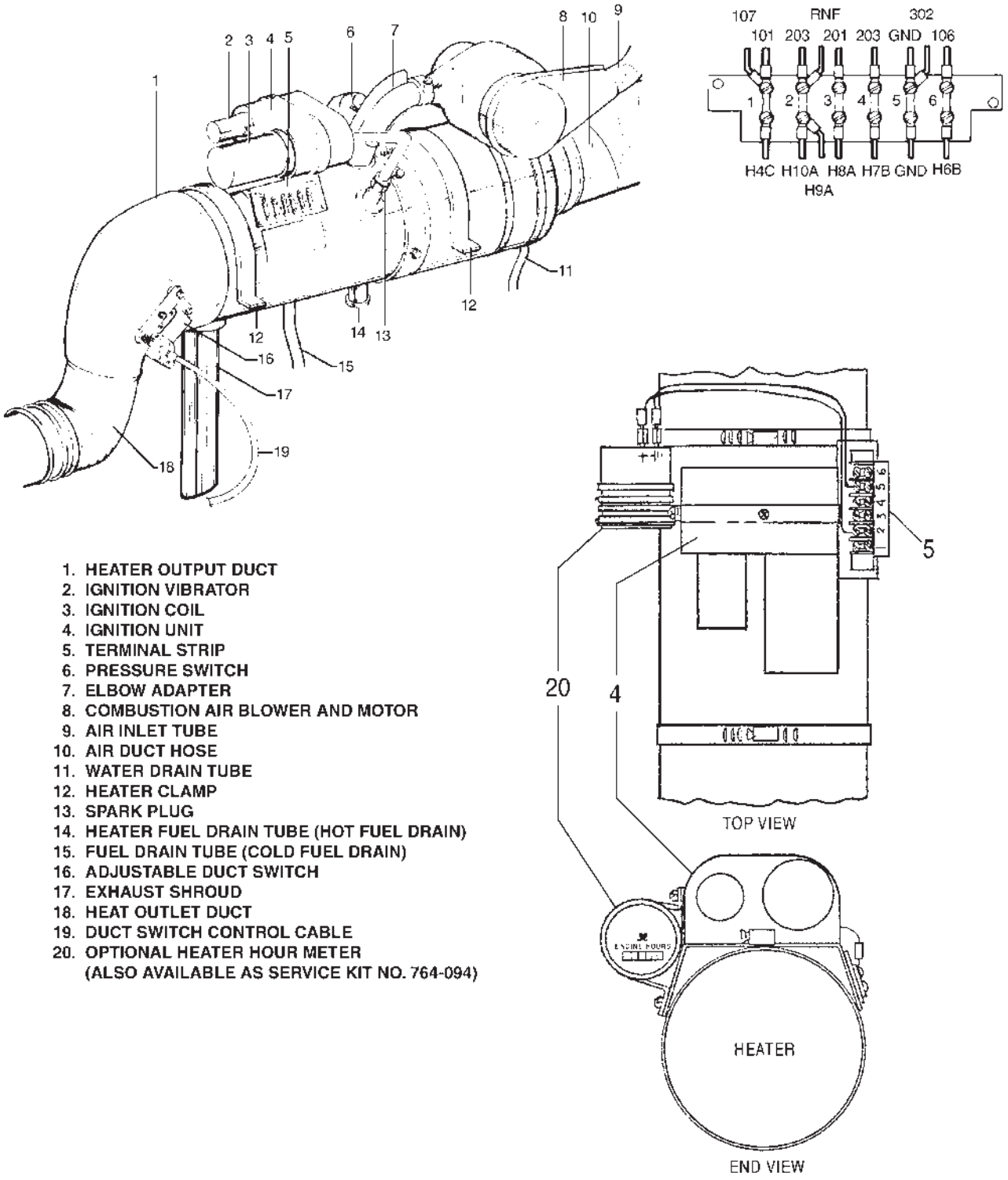
(e) Inspect the high voltage cable connection at the spark plug to make sure it is tight. Also, examine the cable sheath for any possible indications of arcing which would be evidenced by burning or discoloration of the sheath.

(f) Inspect the combustion air blower assembly for security of mounting and security of connecting tubing and wiring. Tighten any loose electrical terminals and air tube connections. Operate both the combustion and ventilating air blowers and check for unusual noise or vibrations.

(g) Check the condition and operation of the spark plug as described in Spark Plugs, below.

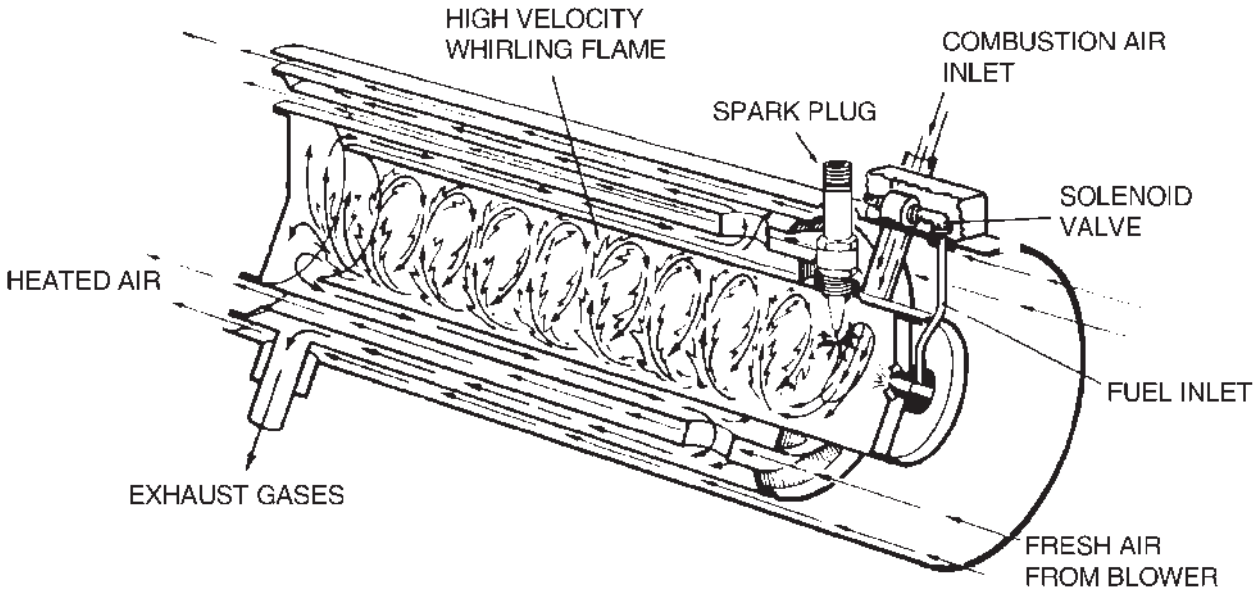
(h) Evaluate the condition of the combustion chamber by performing a “Pressure Decay Test” as described in Aircraft Heater Overhaul and Maintenance Manual (see Vendor Publications under Supplementary Publications in the Introduction).

(i) Following the 100 hour inspection, perform the “Preflight and/or Daily Inspection.”



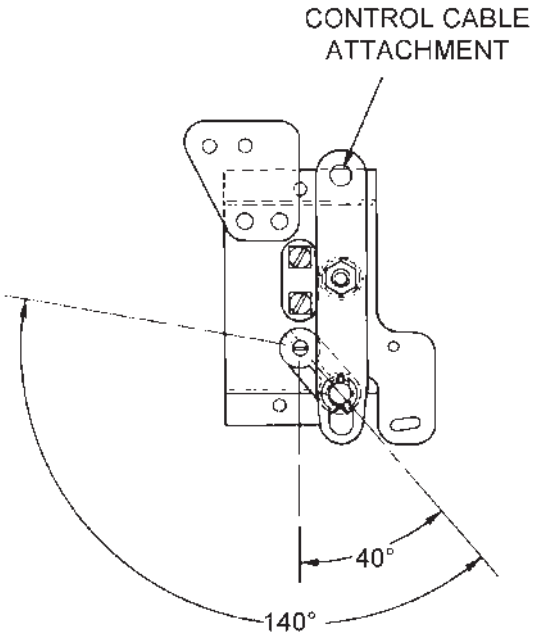
1. HEATER OUTPUT DUCT
2. IGNITION VIBRATOR
3. IGNITION COIL
4. IGNITION UNIT
5. TERMINAL STRIP
6. PRESSURE SWITCH
7. ELBOW ADAPTER
8. COMBUSTION AIR BLOWER AND MOTOR
9. AIR INLET TUBE
10. AIR DUCT HOSE
11. WATER DRAIN TUBE
12. HEATER CLAMP
13. SPARK PLUG
14. HEATER FUEL DRAIN TUBE (HOT FUEL DRAIN)
15. FUEL DRAIN TUBE (COLD FUEL DRAIN)
16. ADJUSTABLE DUCT SWITCH
17. EXHAUST SHROUD
18. HEAT OUTLET DUCT
19. DUCT SWITCH CONTROL CABLE
20. OPTIONAL HEATER HOUR METER
    (ALSO AVAILABLE AS SERVICE KIT NO. 764-094)

Heater and Combustion Air Blower Assembly
Figure 1



Diagramatic Cutaway of Heater to Show Whirling Flame Action
Figure 2

A360



Top View - Duct Switch
Figure 3

(3) Operating Procedure

(a) Place the master and heater switches in their ON position and place the air intake lever in the OPEN position. The ventilating air and combustion air blowers will operate and the heater will ignite.

NOTE: The blowers will not operate and the heater will not ignite with the air intake lever in the CLOSED position.

(b) Set the temperature control lever to the desired temperature setting. This controls the duct switch.

NOTE: If this control is set for ground operating comfort, it may be necessary to reposition it after being airborne, since ram air will increase the ventilating airflow and heater output.

(c) To stop the heater operation, turn the heater switch to the FAN position. The heater will shut off and the ventilating air blower will continue to operate. Allow the blower to operate for two minutes; this will cool down the heater before turning the heater switch off and closing the air intake valve. Turn off master switch.

(4) Operational Tests (PIR-PPS60055, Rev. K)

Connect a ground power unit (GPU) to the aircraft for ground tests. Complete all applicable ground tests prior to fight test.

(a) Test Preparation

1) Check all fittings and connections for security.
2) Check that all ducts, inlets and outlets are free from obstruction.
3) For the tests that require it, fabricate a fuel pump test fixture. (See “Figure 4” on page 214011.)

(b) Replacement Ground Test

Prior to first-time heater operation, the following test may be done after installation of a replacement fuel pump or replacement fuel regulator/shutoff valve.

1) Ensure there are at least five (5) gallons of fuel in the right fuel tank.
2) Install fabricated fuel pump test fixture after the regulator/shutoff valve. (See Test Preparation, above.)
3) Place the fuel selectors in the ON position.
4) Turn ON the GPU.
5) Place the AIR INTAKE lever in the OPEN position.
6) Put the HEATER switch on CABIN HEAT position.
7) Direct fuel into a suitable safe container using the bleed valve on the check fixture.
8) While observing the fuel discharge, verify that all air and preservatives are purged from the system.
9) Close the bleed valve on check fixture.
10) Change the HEATER switch to OFF position.
11) Place the AIR INTAKE lever in the CLOSED position.
12) Turn OFF the GPU.
13) Proceed to Ground Test 1.

(c) Ground Test 1

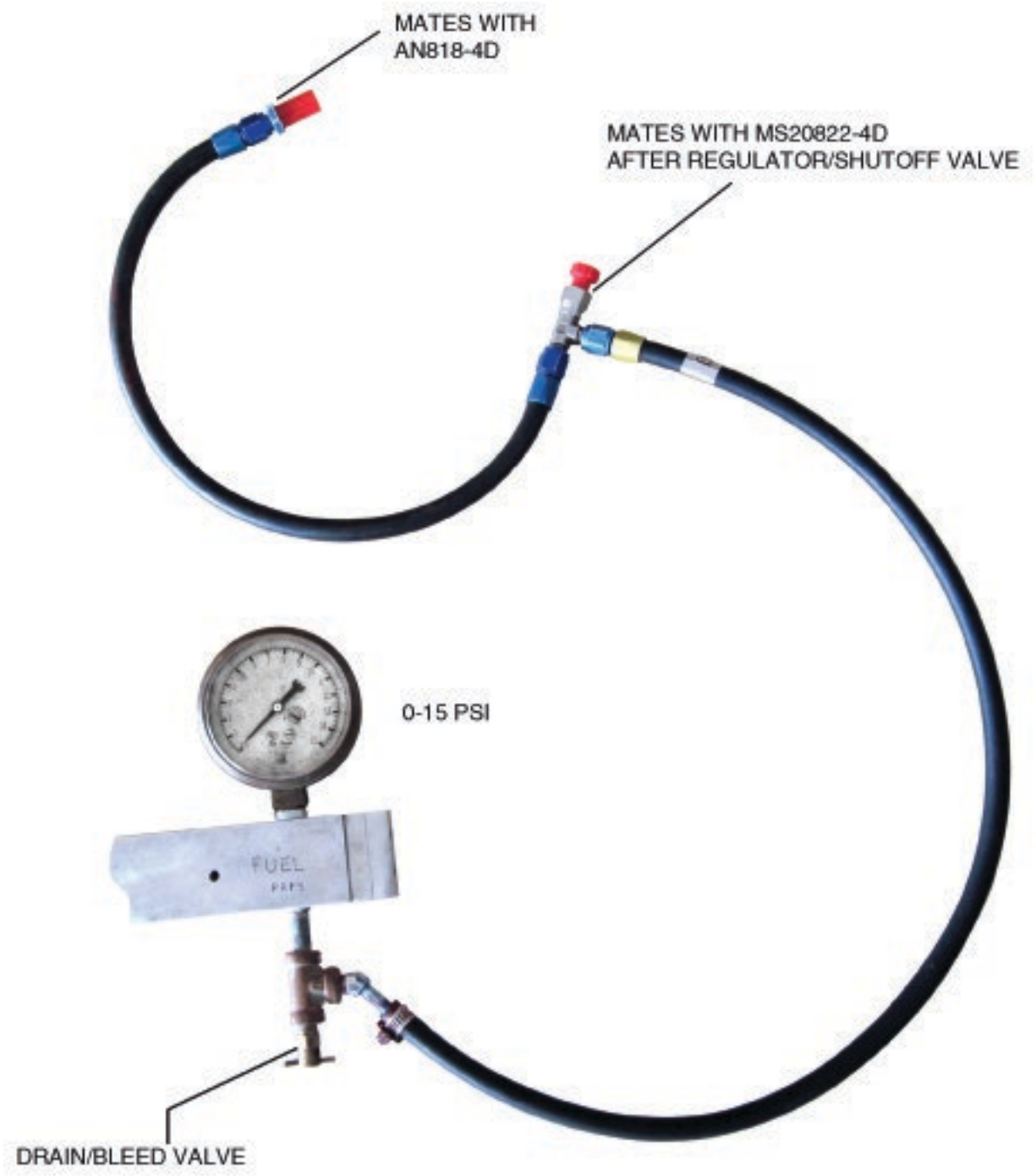
1) Install pressure test fixture after the regulator/shutoff valve. (See “Test Preparation” on page 21409.)
2) Turn ON GPU.
3) Place the AIR INTAKE lever in the OPEN position.
4) Move the TEMP control lever to the center of its travel limit.
5) Put the HEATER switch on CABIN HEAT position.
6) While heater is operating, the test fixture’s gauge should read within 7-8 psi.
7) Change HEATER switch to FAN.
8) The heater will stop operating; the ventilation blower will continue to operate.
9) Allow the blower to operate for a minimum of two minutes.
10) Change the HEATER switch to OFF position.
11) Move the AIR INTAKE lever to CLOSED position.
12) Turn off the GPU.
13) Remove test fixture and reconnect the fuel line.
14) Operate heater and observe heater fuel system to ensure there are no leaks.

(d) Ground Test 2

1) Verify that the GPU is disconnected.
2) Verify that the master switch is in the OFF position.
3) Disconnect wire specified below from the heater terminal strip to remove power from the fuel pump.
   a) Electro-mechanical equipped aircraft: H10A
   b) G1000-equipped aircraft: H43C16A
4) Connect and turn ON GPU.
5) Place the AIR INTAKE in the OPEN position.
6) Put the HEATER switch on CABIN HEAT position.
7) The blower shall operate. Air shall be flowing from the ventilating air outlets and the heater exhaust outlet. The heater shall not ignite.
8) Press the OVERHEAT indicator light.

   NOTE: Skip this step for G1000-equipped aircraft.

   a) The lamp shall illuminate when pressed.
   b) Press the test switch on the annunciator panel. The heater overheat lamp (in addition to all other annunciator lamps) shall illuminate.
9) Change the HEATER switch to OFF position.
10) Place the AIR INTAKE in the CLOSED position.
11) Turn OFF and disconnect GPU.
12) Verify that the master switch is in the OFF position.
13) Reconnect wire specified below to the heater terminal strip.
   a) Electro-mechanical equipped aircraft: H10A
   b) G1000-equipped aircraft: H43C16A



Combustion Heater Fuel Pump Test Gauge Fixture
Figure 4

(e) Ground Test 3

Perform the following test on every electro-mechanical aircraft. The following test may be performed during landing gear retraction testing (see 32-30-00, Landing Gear Retraction System Functional Test) provided that the relevant steps in “Test Preparation” on page 21409, and “Ground Test 2” on page 214010, are successfully completed.

NOTE: Ground Test 3 is not applicable for G1000-equipped aircraft. For these aircraft, proceed to Ground Test 4.

1) Place the aircraft on jacks.
2) Turn ON the GPU.
3) Place HEATER switch in FAN position.
4) The ventilation blower shall operate.
5) Place the landing gear handle in the UP position, the ventilation blower shall stop operating.
6) Place the landing gear in the DOWN position. The ventilation blower shall start operating.
7) Change HEATER switch to OFF position.
8) Turn OFF the GPU.
9) Remove aircraft from jacks.

(f) Ground Test 4

1) Turn ON the GPU.
2) Put the HEATER switch on CABIN HEAT position.
3) Move the TEMP lever to the far left of its travel limit.
   a) The heater shall ignite and continue to operate until the thermostat causes it to stop operating. Cycling in this manner shall continue until the HEATER switch is changed to OFF position.
   b) Allow the heater to cycle a minimum of two times.
4) Press the OVERHEAT indicator light.

   NOTE: Skip this step for G1000-equipped aircraft.

   a) The lamp shall illuminate when pressed.
   b) Press the test switch on the annunciator panel. The heater overheat lamp (in addition to all other annunciator lamps) shall illuminate.
5) Change the HEATER switch to FAN. Let it operate for two minutes. The heater shall stop and the blower shall continue to operate.
6) Move AIR INTAKE lever to CLOSED position. The blower shall stop operating.
7) Change the HEATER switch to CABIN HEAT position again. The heater shall not ignite.
8) Change the HEATER switch to OFF position.
9) Turn OFF the GPU.

(g) Ground Test 5

To be done only after all Ground Tests above have been completed.

1) Turn ON the GPU.
2) Put the HEATER switch on CABIN HEAT position.
3) Place the AIR INTAKE in the OPEN position.
4) Verify that there are no leaks at the connection points which were disconnected during preparation and earlier ground tests.
5) Change the HEATER switch to OFF position.
6) Place the AIR INTAKE in the CLOSED position.
7) Turn OFF the GPU.

(h) Flight Test

Perform this test during cruising flight.

1) Move the AIR INTAKE lever to OPEN position.
2) Put the HEATER switch on CABIN HEAT position.
3) Move the DEFROST lever to the ON position.
4) Verify that there is airflow from defrost and heater vents.
5) Return the DEFROST lever to OFF position.
6) Change HEATER switch to OFF position.
7) Wait 15 seconds and then move the AIR INTAKE lever to CLOSED.

F. Heater Electrical System Checks

(1) Electrical Checks

These tests are listed as an aid in isolating open circuited or inoperative components.

NOTE: The schematic wiring diagram (“Figure 5” on page 214016) shows, in addition to the heater circuitry, the aircraft control circuit. For the purposes of this manual, the circuitry shown in these illustrations will be utilized to describe voltage checks.

It must be assumed that power, which is furnished through the heater circuit breaker, is present at the HEATER SWITCH at all times. Always check the circuit breaker before performing voltage checks.

(2) Vent Blower Power Circuit Check

With the HEATER SWITCH in the FAN position, voltage (28-volts nominal) should be present at the following locations: (See “Figure 5” on page 214016.)

(a) Terminal No. 6 on the heater terminal strip if the air valve is open.

(b) From terminal No. 6 of the heater terminal strip through the radio noise filter to the ventilating air motor.

(c) Electrical ground circuit for the ventilating air motor is provided from terminal No. 5 of the heater terminal strip. Ventilating air motor is inoperative when the landing gear is up or air valve is closed.

(3) Heater Power Circuit Check

With the HEATER SWITCH in the HEAT position, voltage should be present at the following locations: (See “Figure 5” on page 214016.)

NOTE: Power for the ventilating air blower is the same as described above except that power is now supplied through the HEAT side of the HEATER SWITCH.

(a) Terminal No. 1 of the heater terminal strip if the air valve is open.

(b) From terminal No. 1 of the heater terminal strip through the radio noise filter to the combustion air motor and to terminal No. 1 of the overheat switch.

(c) From terminal No. 3 of the overheat switch through the combustion air pressure switch to terminal No. 2 of the heater terminal strip.

(d) From terminal No. 2 of the heater terminal strip to the ignition unit to the fuel regulator and shutoff valve and fuel pump through the adjustable duct switch to terminal No. 3 of the heater terminal strip.

(e) From terminal No. 3 of the heater terminal strip through the cycling switch to the fuel solenoid valve.

In the event that voltage is not present at one or more of the above listed points, the wiring must be traced back to the power source. If components are still inoperative after the wiring inspection, check the individual inoperative components for voltage and, if necessary, replace them.

(4) Removal of Heater

See “Figure 1” on page 21407.

(a) Ascertain that all heater controls are off.

(b) Remove the access panel to the aft section of the fuselage.

(c) Disconnect the heater outlet hose from the heater air distribution box by releasing the hose attachment clamp.

(d) Disconnect the duct switch control cable from the left side of the air distribution box.

(e) Note the hookup of the electrical leads to facilitate reinstallation. Disconnect the leads from the heater terminal block.

(f) Disconnect the fuel supply line at the heater by removing the cover of the fuel line connection shroud and disconnecting the line from the solenoid valve.

(g) Disconnect the fuel and water drains from the bottom of the heater and allow them to slide down.

(h) Disconnect the air inlet hose from the inlet end of the heater by releasing the hose attachment clamp.

(i) Disconnect the combustion air blower inlet hose from the blower assembly by removing the cotter key and clevis pin at the blower.

(j) Loosen the clamps from around the heater and remove the heater from the airplane. The exhaust shroud should remain in the airplane.

(k) With the heater removed, the necessary maintenance may be performed as required.

(5) Installation of Heater

See “Figure 1” on page 21407.

(a) Ascertain that all the heater components are on the heater. Position the exhaust tube shroud on the tube mounting flange located in the fuselage.

(b) Position the heater over its mounting brackets and ascertain that the exhaust tube extends into the exhaust shroud. Lower the heater to its mounting brackets. The exhaust tube should extend out the bottom of the fuselage.

(c) Move the heater slightly to obtain the best fit of the exhaust tube shroud and heater. Place the heater clamps around the heater and mounting bracket flanges and secure.

(d) Connect the combustion air blower inlet hose to the combustion air blower assembly on the heater and secure in place with the clevis pin and cotter key.

(e) Connect the air inlet hose to the inlet end of the heater and secure with clamp.

(f) Connect the fuel and water drain lines to the bottom of the heater.

(g) Connect the fuel supply line to the heater and cover over the fuel shroud and secure with two screws.

(h) Attach the duct switch control cable to the switch.

(i) Connect the electrical leads to the heater terminal block on the heater as shown in "Figure 1" on page 21407.

(j) Check the operation of the heater per previous instructions.

(k) Install the access panel to the aft section of the fuselage.

G. General Maintenance

Instructions in this paragraph pertain to maintenance of the basic heater and components while the heater is installed in the airplane. Instructions for removal of components are included provided the installation permits accessibility.

NOTE: No special service tools are required for normal periodic maintenance.

(1) Combustion Air Blower

(a) Removal:

1) Disconnect wire at quick-disconnect terminal.

2) Disconnect the inlet tubing from the inlet air adapter.

3) Loosen the clamps that hold the combustion air blower assembly in the support bracket and slide the motor out of the bracket.

(b) Replacing Motor Brushes.

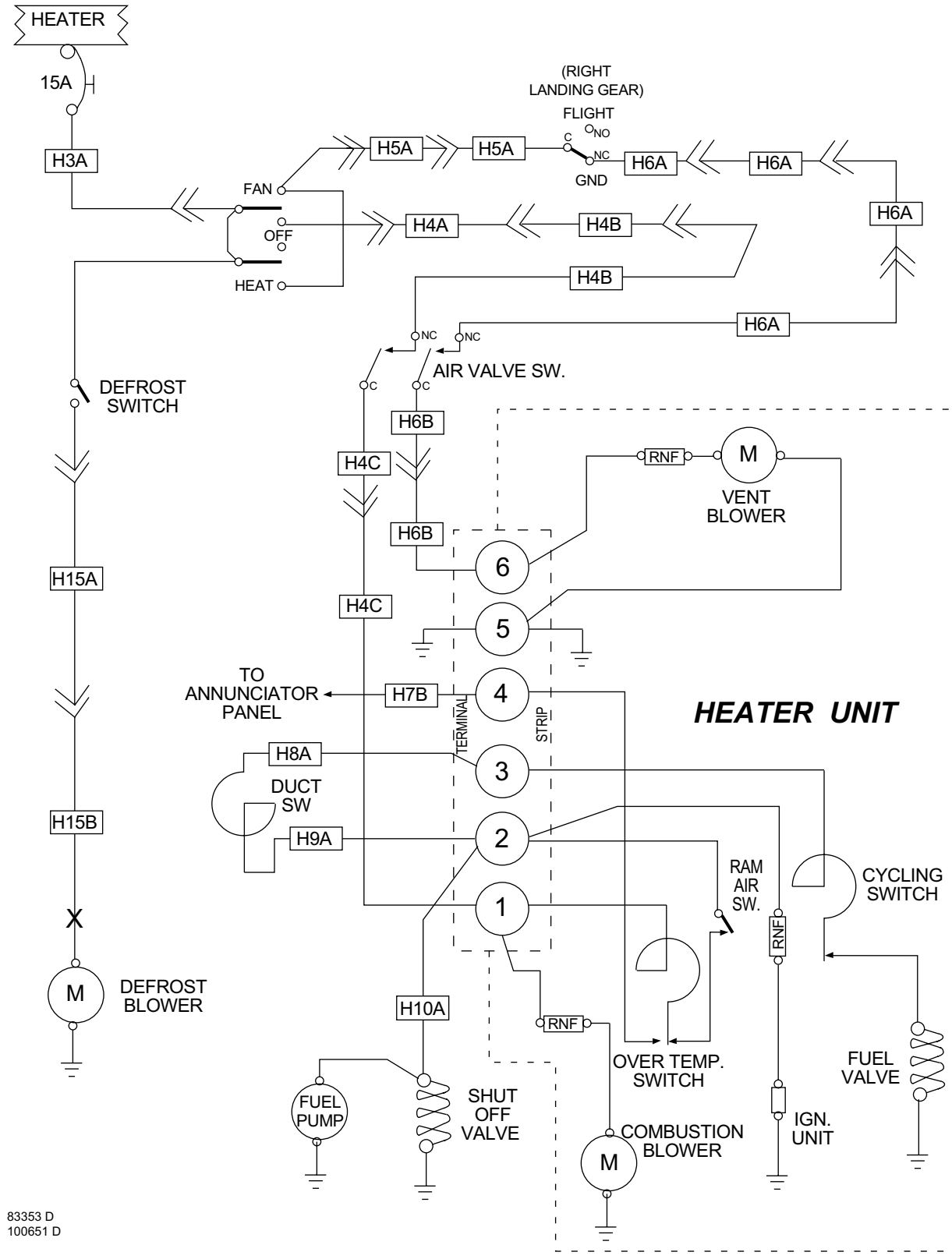
See "Figure 15" on page 214033.

1) Remove brush cap at one of brush locations. Note position of brush inside guide and carefully lift brush and brush spring out of guide. Be sure to hold brush so that it can be reinstalled in precisely the same position if no brush replacement is required.

2) Inspect the brush for wear. If brushes are worn to a length of .187 of an inch, they must be replaced.

3) Looking through the brush guide, inspect the commutator which should be smooth and medium brown to dark brown in color. Remove all dust from commutator with compressed air. If the commutator is grooved in the brush track, gouged, scored or shows signs of having burned spots, replace the complete motor assembly. If the commutator is in good condition, install new motor brushes and tighten brush caps into place. Make sure each brush is oriented so that the curved end fits the curvature of the commutator.

4) After installing new brushes, it is advisable to run in the brushes as follows: Connect the motor to a controlled voltage supply (rheostat). Operate the motor at approximately 1/2 its normal speed for the first hour; then gradually increase the speed until it is rotating at approximately normal speed. Continue the run in operation for at least two hours to properly seat the brushes before installing the blower in the aircraft.



Heater and Defroster Wiring Diagram
Figure 5

(c) Installation:

1) Prior to installing the combustion air blower, inspect all parts of the assembly for loose screws, loose nuts, and poor ground connection on the blower housing. Make sure the blower wheel is tight on the shaft and properly located in the housing. It should have just enough clearance to rotate at full speed without binding against the inlet housing. Blower performance is based upon this close-tolerance clearance. It is recommended that correct voltage be applied for this clearance check.

2) Install the blower inlet adapter in the same orientation as before removal.

3) Place combustion air blower assembly in position in the attaching clamp so air tubing can be connected and slide the tubing into position at the point where it was disconnected during removal. Do not tighten until after tightening the motor in the attaching strap.

4) Tighten blower motor mounting strap securely making certain air tubing is in proper alignment.

5) Secure air tubing by tightening the clamp or installing the sheet metal attaching screws.

6) Connect the wire lead at the quick-disconnect terminal.

7) Connect the ground lead securely to the mounting bracket.

8) Check motor operation. By disconnecting the wire at the No. 3 terminal on heater terminal strip, blower can be operated without fuel flow to the heater.

(2) Spark Plug

(a) Removal:

See “Figure 14” on page 214032.

1) Remove the necessary access panels on the rear of the fuselage to expose the spark plug area of the heater assembly.

NOTE: Ensure that heater electrical circuits are de-energized.

2) Unscrew and remove the high voltage lead connector at the spark plug. Exercise care to avoid fouling or damaging the connector.

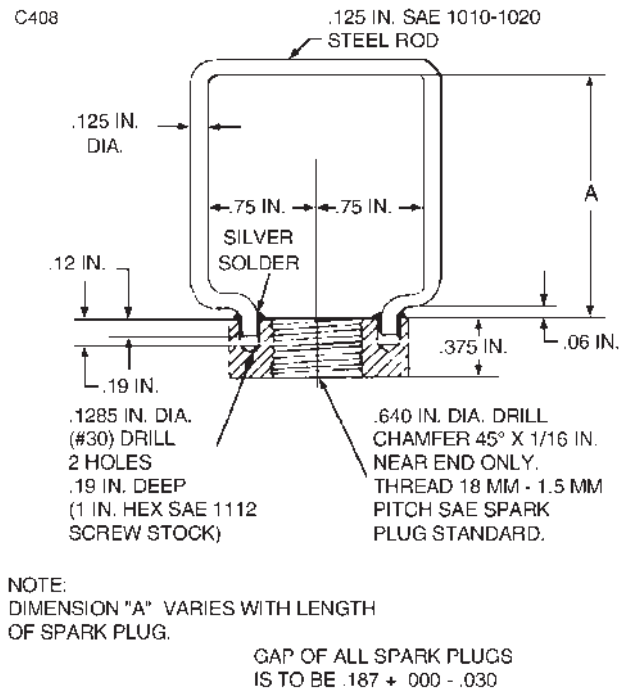
3) Remove the grommet.

4) Using a 7/8 inch deep hex socket, unscrew and remove spark plug. Make sure spark plug gasket is removed with spark plug. It will normally stick on spark plug threads, but if gasket should drop into ventilating air passages of heater, remove with a wire hook.

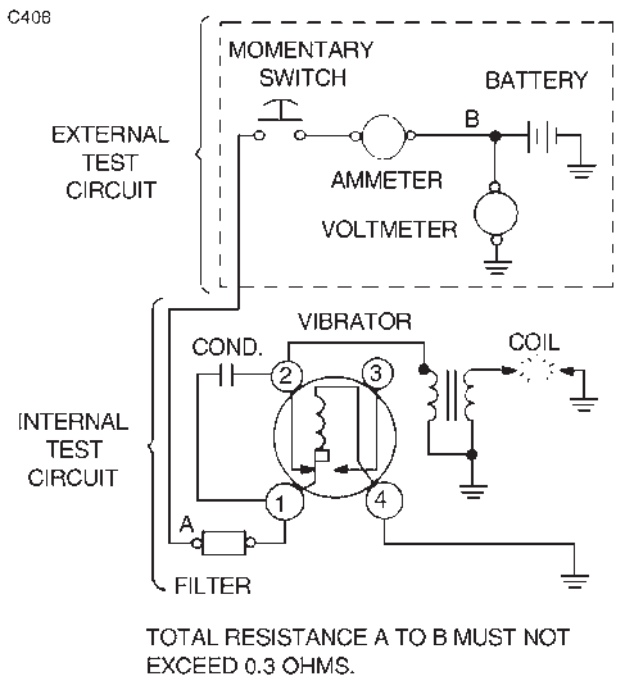
(b) Inspection and Servicing (Spark Plug):

If the spark plug appears to be in good condition, except for a mild coating of oxide on the porcelain and electrodes, it may be cleaned and reused. Cleaning is accomplished on a conventional airplane type spark plug cleaner, except that it will be necessary to use two or more adapters in order to raise the long extension of the plug far enough out of the cleaner nozzle opening to perform an effective job. Plug the ceramic insert cavity at the terminal end of the plug with a piece of paper or cloth to keep out any of the cleaning sand. Wipe this cavity out thoroughly with a cloth wet with carbon tetrachloride. If after cleaning the spark plug porcelain is white and the electrodes are not eroded, proceed to check the ground electrode in the heater and adjust the spark gap in accordance with Step 3 of this paragraph.

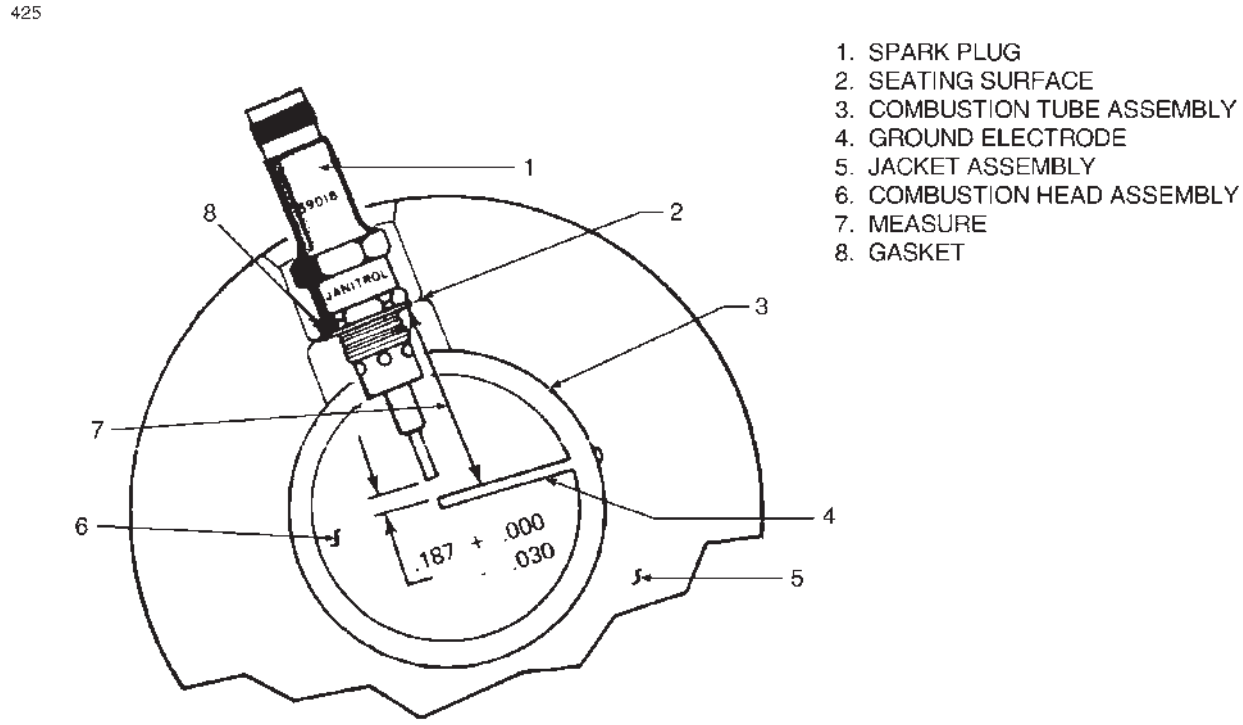
NOTE: If the spark plug fails to clean up properly and/or if the electrodes are badly eroded, it should be replaced.



Spark Plug Fixture
Figure 6



Wiring Test Setup
Figure 7



Spark Plug Gap Adjustment
Figure 8

(c) Spark Gap Check and Adjustment:

See “Figure 8” on page 214018.

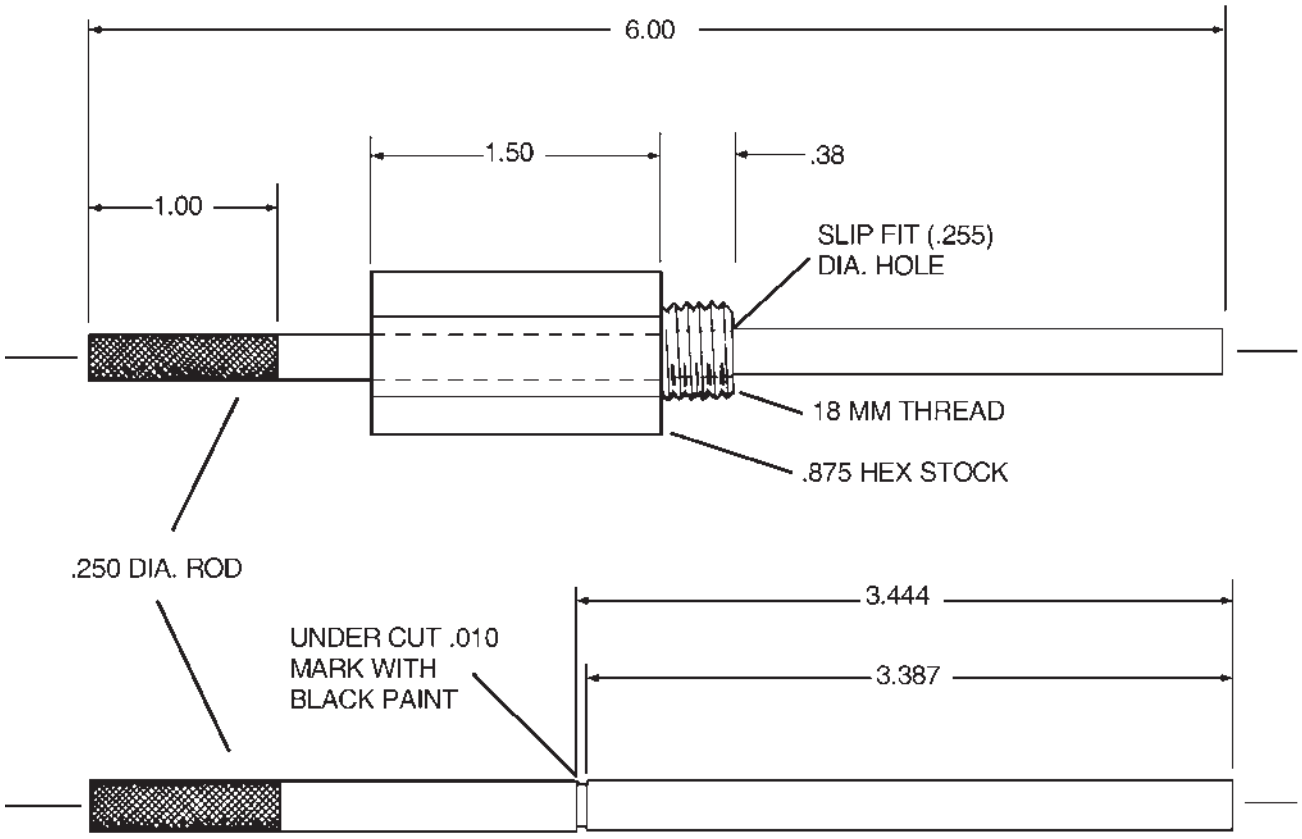
A spark gap of 0.156 to 0.188 inches must be maintained on the P/N 39D18 spark plug. This gap should be checked any time a plug is replaced or at the time of heater overhaul. A spark gap greater than that specified can shorten the life of the ignition assembly. There are several methods in which the spark gap of this heater may be checked. Method I is recommended when the heater is being overhauled and before the installation of the fuel nozzle. Methods II and III are suitable for checking the gap through the spark plug well when the heater is not disassembled.

1) Method I.
   a) Using a 5/32 inch drill (0.156) or a piece of 5/32 rod, reach through small opening in combustion head and find the ground electrode. (It is welded inside the head.)
   b) Move the drill along the side of the electrode on the spark plug side. (Movement should be from the outer edge towards the center.) The drill should just pass through the spark plug gap opening. Should the drill fail to pass through this opening, the gap is too narrow. If it passes through too freely, the gap is too wide. In either case, it will be necessary to bend the ground electrode in the direction required. This may be done by removing the spark plug and reaching through the opening.
   c) Recheck the gap after repositioning the ground electrode.
2) Method II.
   a) Measure the distance between the seating surface of the spark plug with a new gasket installed to the end of the plug electrode.
   b) Using a depth gauge, measure the distance between the ground electrode in the heater to the spark plug seating surface in the heater jacket and check this measurement against the measurement obtained in Step A. The difference should be between 0.156 to 0.188 of an inch.
   c) The ground electrode can be bent to obtain the required gap.
3) Method III.
   a) Fabricate the special tool from dimensions given in “Figure 9” on page 214020 for the spark plug gap adjustment tool.
   b) Install the threaded end of the tool into the spark plug hole.
   c) Slide the rod of the tool into the combustion head until it contacts the ground electrode.
   d) Check that the indicator ring on the rod lines up with the end of the tool. The ground electrode may be bent to obtain the required gap.

      NOTE: Inspect the ground electrode for erosion. If it is eroded to approximately half of its original 1/8 inch diameter, it should be replaced. This can be done as follows:

      1) Grind off the head of the rivet where it projects through the combustion head and remove the electrode.
      2) Install a new CRES rivet AN125452 which is 1.500 inches in length.
      3) Heliarc tack weld the rivet head to hold it in place.
      4) Check spark gap as noted in Methods I or II.

GAUGE FOR LOCATING GROUND ELECTRODE IN JANITROL HEATERS



MATERIAL CAN BE SAE TYPE 303, 321 OR ST. OR ALUMINUM - CASE HARDENED

Heater Plug Gap Adjustment Tool
Figure 9

(d) Installation:

Refer to “Figure 8” on page 214018.

1) If a new spark plug is being installed, be sure to adjust the spark gap. Do not bend the electrode on the spark plug.
2) Place a new spark plug gasket on the threads. If the gasket does not hold on the threads and would be likely to fall off during installation, place a small drop of Aviation Permatex or similar material on the gasket to stick it temporarily to the plug shell.
3) Screw the spark plug into the heater with a deep socket wrench. Tighten to a torque of 28 foot-pounds.
4) Install the grommet (See Figure 13, item 39) in the heater jacket opening.
5) Carefully insert the spring connector on the high voltage lead into the spark plug shell; press down gently and start the nut on the threads. Tighten the nut to 20 foot-pounds.
6) Operate the heater to check dependability and close all access openings.

(3) Ignition Unit

This unit converts 28-volt DC to high voltage, oscillating current capable of producing a continuous spark in the combustion chamber of the heater. This unit remains energized and produces a continuous spark during heater operation. It contains a condenser, resistor, radio noise filter and vibrator socket. It also has an externally mounted vibrator and ignition coil.

(a) Removal and Installation

1) Removal:

See “Figure 14” on page 214032.

NOTE: Make sure heater electrical circuits are de-energized.

a) Disconnect the primary wire from the primary terminal of the ignition assembly.
b) Carefully unscrew and disconnect the high voltage ignition cable at the spark plug. Exercise care to avoid fouling or damaging the connector.
c) Remove the four attaching screws and lift the ignition assembly off the heater jacket.

2) Installation:

See “Figure 14” on page 214032.

a) Place the ignition assembly in position on the heater jacket with the high voltage cable facing the spark plug end of the heater.
b) Install the four screws. Tighten the screws securely.
c) Carefully connect the high voltage lead to the spark plug.
d) Connect the primary lead to the primary terminal on the ignition unit and tighten the nut securely.
e) Check for proper heater operation.

(b) Inspection

Inspect components as directed in “Chart 2” on page 214023 and “Figure 10” on page 214023.

NOTE: Replace any component that fails to meet checks listed in “Chart 2” on page 214023.

(c) Testing

CAUTION: WHEN TESTING AN IGNITION UNIT, DO NOT USE A SCREWDRIVER AS A SUBSTITUTE FOR A SPARK PLUG AND SPARK PLUG FIXTURE.

The ignition unit does not require complete overhaul. The following test will indicate whether or not the unit is operational and whether the vibrator should be replaced before reinstallation in the aircraft. The following equipment is required to test the components:

1) A battery that will supply power at approximately 28 volts DC.
2) A voltmeter with a range of 0-30-volts.
3) A lead from the battery to the test fixture in which is included an ammeter with a range of 0-3 amperes and a normally open, momentary-closed switch. The total resistance of the lead including the ammeter and switch must not exceed 0.3 ohms.
4) A spark gap of 0.187 inch (plus 0, minus .030). A convenient means of arranging the correct spark gap is to install a spark plug, P/N 39D18, in a test fixture arranged to provide a ground electrode and a .187 inch spark gap. (See “Figure 6” on page 214018 for information on fabricating this fixture.)

   NOTE: Any one of several spark plugs may be used with the spark plug fixture detailed in “Figure 6” on page 214018. However, the “A “dimension in that sketch must be varied with the length of spark plug electrode to provide a gap of .187 inch for all spark plugs.

5) The high tension shielded ignition lead between the ignition unit and the spark plug is a part of the cover assembly.
6) Arrange the test equipment as shown in “Figure 7” on page 214018.

(d) Operational Test

1) Close the momentary switch and read the voltmeter and ammeter. Release the momentary switch immediately.
2) The amperage reading at 14-volts DC must be 1.50 + 0.25 amperes.
3) The amperage reading at 28-volts DC must not be more than 1.5 amperes.

(e) Vibrator

The vibrators should be replaced after 250 hours of operation. This schedule applies equally to vibrators installed in new units as well as new vibrators installed in ignition units that have been in service.

Removal and Installation

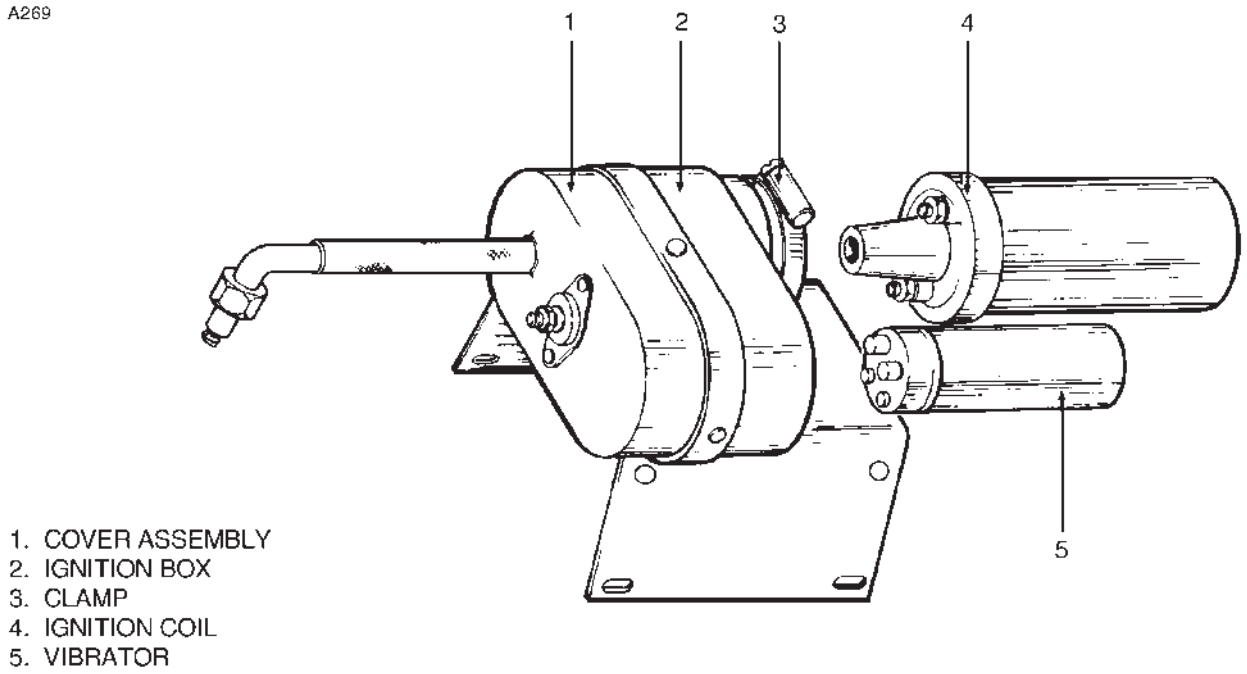
See “Figure 10” on page 214023.

1) Remove the clamp.
2) Remove the vibrator from the ignition unit; it may require a slight back-and-forth movement to remove it from the unit. A piece of masking or friction tape around the exposed portion of the vibrator will help to grip the vibrator for removal.
3) Install the new vibrator with the index mark aligned. The connector pins on the vibrator can be felt entering the pin sockets in the vibrator socket; then press the vibrator fully and firmly into position. Secure with the clamp.

   NOTE: If replacement of vibrator fails to correct operational failure, further disassembly and inspection may be required.

CHART 2
INSPECTION (IGNITION UNIT)

Index No.	Nomenclature	Inspection
1	Cover Assembly	Inspection for security of lead assembly to cover. Ignition cable, grommet, terminal and connector for carbon tracks, cracks or distortion. Repair or replace for any of above conditions.
2	Ignition Coil	Inspect for broken bakelite, carbon tracks, oil leaks, and dents in coil cover. Replace for any of the above conditions.



Ignition Unit Assembly
Figure 10

(4) Cycling Switch and Limit (Overheat) Switch

See “Figure 14” on page 214032.

(a) Removal:

1) If the limit switch is damaged or defective, disconnect the three electrical leads from the switch terminals. Be sure to mark the leads for proper reassembly. (The switch terminals are identified by numbers “1”, “2” and “3”.)

2) Remove the two attaching screws and lift the limit switch and spacers (gaskets) from the jacket opening.

3) If the cycling switch is damaged or defective, disconnect the electrical leads being sure to mark them for proper reassembly.

4) Remove the two screws and lift the cycling switch from the jacket opening.

NOTE: No attempt should be made to repair either of these switches. If they do not operate properly, they should be replaced.

(b) Installation:

See “Figure 14” on page 214032.

1) Install the limit switch and spacers (gaskets) by placing them in position in the heater jacket opening and installing two screws.

2) Tighten screws securely; then reconnect the electrical leads in accordance with markings made during disassembly.

Refer to wiring diagram, “Figure 5” on page 214016.

3) Install the cycling switch (See “Figure 14” on page 214032) by placing it in position in the heater jacket opening and securing it with the two screws. Tighten screws securely; then reconnect the electrical leads to their respective terminals as marked during disassembly. (See wiring diagram, “Figure 5” on page 214016.)

(5) Combustion Air Pressure Switch

See “Figure 14” on page 214032.

(a) Removal:

1) Disconnect electrical leads from the terminals of the combustion air pressure switch, being sure to mark them for proper reassembly. Disconnect the tube from the switch cap. Exercise caution not to exert excessive bending of the tube. (It is “tacked” to the combustion chamber inside the jacket.)

2) Unscrew and remove the combustion air pressure switch from the fitting on the combustion air inlet tube.

(b) Installation:

1) Install the combustion air pressure switch by rotating it on the threaded fitting of the combustion air inlet tube and tighten it securely. Exercise caution not to over-torque the switch as this could change the setting.

2) Connect electrical leads to their respective terminals in accordance with markings made during removal. If in doubt regarding proper connections, refer to the wiring diagram, “Figure 5” on page 214016. Connect the tube to the switch cap.

3) Check for proper heater operation.

(6) Fuel Regulator and Shutoff Valve

See “Figure 11” on page 214026.

The fuel regulator and shutoff valve is located below the floor panel between the main and rear spar on the right side of the cabin. The assembly is enclosed in a special fiberglass box with a removable access panel.

(a) 100 hour Inspection

Visually inspect the valve body for signs of fuel stains, paying careful attention to the diaphragm joint and the threaded mounting holes located in sides of the valve body (see Figure 10). Fuel leakage may appear as a greenish blue stain or residue in the area of the diaphragm joint or threaded mount hole. Use supplemental lighting if needed to facilitate visual inspection. Visual inspection must include all four sides of the regulator valve body. If signs of fuel leakage are found replace the valve using a new valve of appropriate part number with a manufacture date code of 02/02 or later. Record valve replacement in the logbook.

(b) Removal

1) Ascertain that the left fuel tanks are empty and the fuel selector controls are in the OFF position.
2) Gain access to the regulator and disconnect the electrical leads from regulator and shutoff valve.
3) Disconnect the fuel line from the outlet port and remove the regulator from the heater fuel pump. Cap all open fuel lines to prevent contamination.

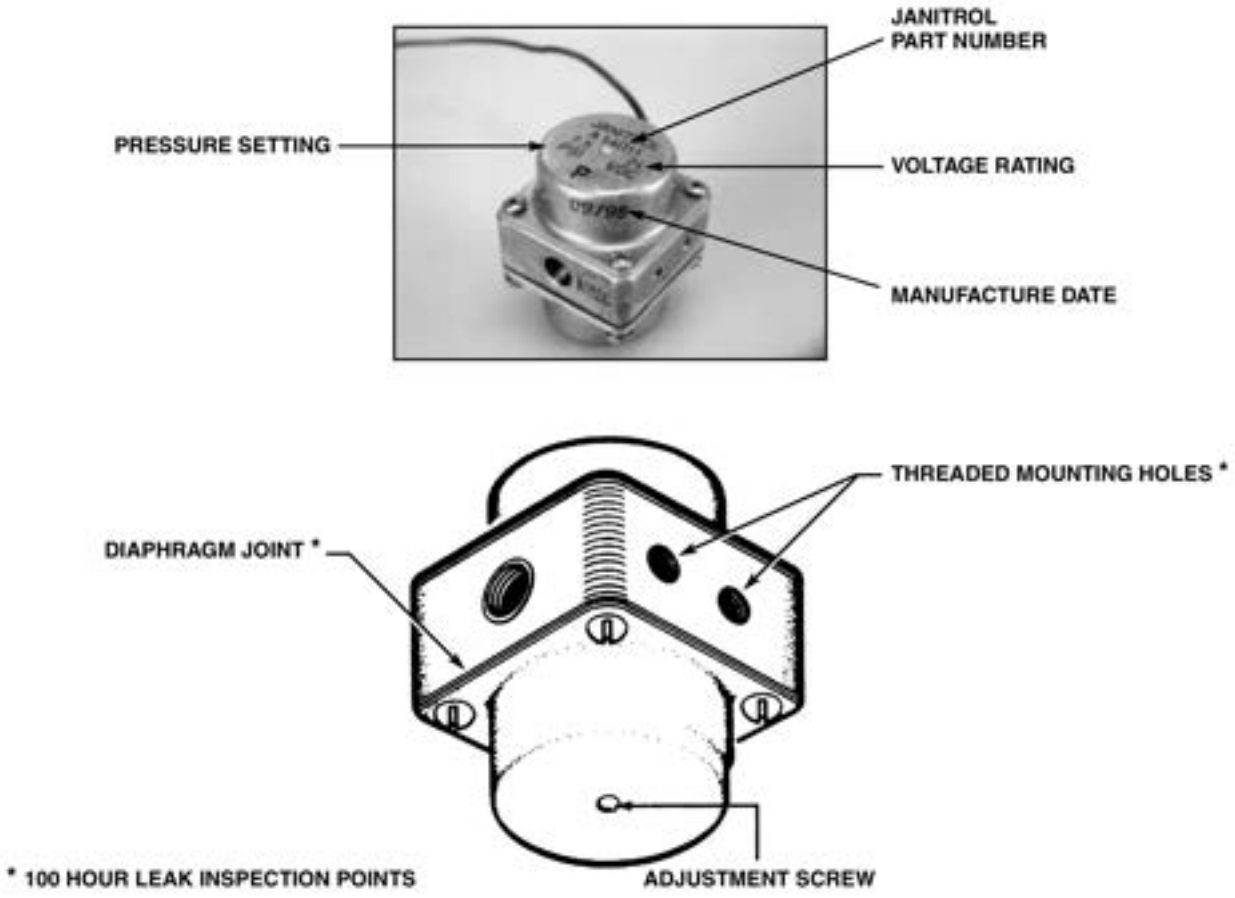
(c) Adjustment

The fuel regulator and shutoff valve used in this system is adjustable but not repairable. The following steps cover the proper adjustment of this unit:

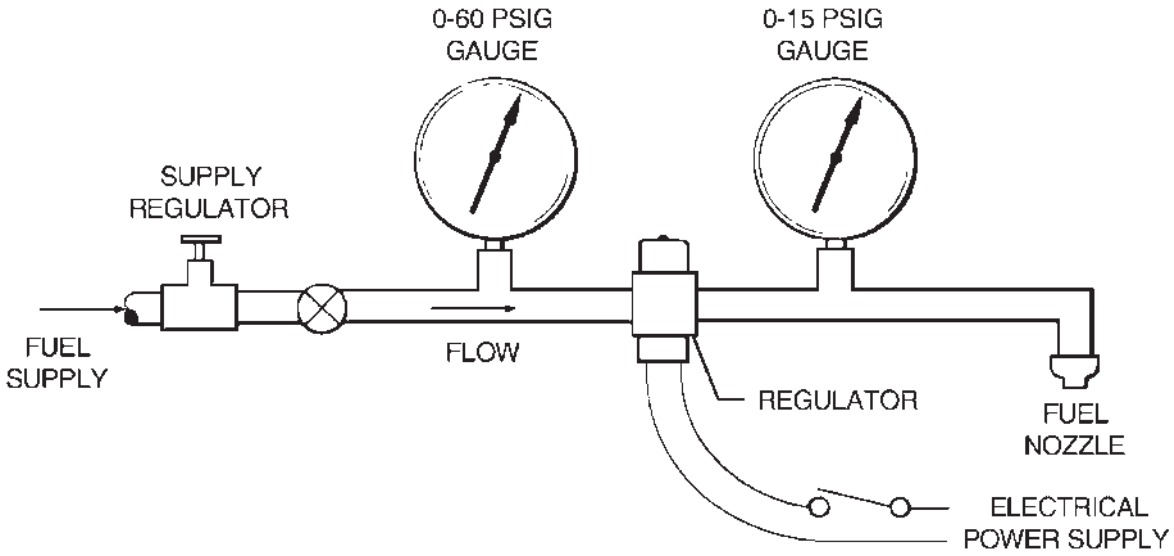
1) Install the regulator in a test stand similar to that shown in “Figure 12” on page 214027.
2) Install a 2.0 gph nozzle (Janitrol Part No. C08D09). Gasoline or Stoddard solvent can be used for testing.
3) Apply fluid pressure from fuel pump and energize the solenoid. Outlet pressure should be 7.0 ± .5 psi, if not, correct accordingly.
4) Using a screwdriver, break the seal over the adjustment screw and adjust the regulated outlet pressure to 7.0 ± .5 psi. (Turn clockwise to increase pressure or counterclockwise to decrease pressure.)
5) De-energize and energize the solenoid at least twice. The outlet pressure should be 6.5 to 7.5 psi with the solenoid energized. When the solenoid is de-energized, the pressure should drop to zero and the fuel flow from the nozzle should stop.
6) During the above test, observe for signs of external leakage. Any leakage is cause for rejection of the regulator. After satisfactory adjustment has been made, apply Glyptol around the threads of the adjustment screw and in the slot.

(d) Installation

1) Position the regulator between the fuel line and fuel pump. Ascertain that the inlet side of the regulator is towards the fuel pump.
2) Connect the regulator to the pump and the heater fuel line to the regulator outlet port.
3) Connect the electrical leads from the regulator.
4) Conduct Replacement Ground Test under Operational Tests, above.



Fuel Regulator and Shutoff Valve
Figure 11



Test Setup for Fuel Regulator and Shutoff Valve
Figure 12

(7) Heater Fuel Pump Maintenance

See “Figure 13” on page 214031.

Maintenance required is very limited, consisting of inspection and cleaning or replacing the filter element.

(a) In Seneca IVs and Seneca V S/N’s 3449001 through 3449413 as originally delivered:

1) Inspection

a) Remove and disassemble the pump as described below.

b) Inspect the filter element for condition and cleanliness. Clean or replace as required.

2) Removal

The heater fuel pump is located below the cabin floor panel between the main and rear spar on the right side of the cabin. It is enclosed in a fiberglass compartment which has a removable access cover.

a) Ascertain that the right fuel tanks are empty and both fuel selector controls are in the OFF position.

b) Remove the aft facing rear seats. Remove the center floor panel. Remove the access cover to fuel pump assembly.

c) Disconnect the electrical lead from the pump.

d) Drain fuel remaining in lines using the low-point drains located on the lower right fuselage.

e) Disconnect the fuel line from the inlet end of the pump and the regulator from the outlet end. Cap all open fuel lines to prevent contamination.

f) Remove the bolts which secure the pump to its mounting bracket.

3) Disassembly

a) Remove the pump as described above.

b) Remove the three screws securing the cover plate and remove the cover plate. Discard the cover plate gasket.

c) Remove filter element.

4) Assembly

a) Replace filter element.

b) Install a new cover plate gasket and install the cover plate. Secure with three screws.

c) Torque screws 18–20 in.-lbs.

5) Installation

a) Position the fuel pump to its mounting bracket and secure in place with bolts.

b) Connect the regulator to the pump outlet and the fuel line to the pump inlet.

NOTE: Be careful not to rotate the tube assembly on reinstallation.

c) Connect the electrical lead to the pump.

d) Conduct Replacement Ground Test under Operational Tests, above.

e) Replace the access cover to fuel pump assembly. Replace the center floor panel. Replace the aft facing rear seats.

(b) In Seneca V S/N's 3449414 and up (and in earlier Seneca Vs and all Seneca IVs with service replacement Heater Fuel Pump Kit P/N 461-771 installed):

1) Fuel Filter Replacement.

a) Ascertain that the right fuel tanks are empty and both fuel selector controls are in the OFF position.

b) Access the fuel pump as described under Removal, below.

c) Disconnect the electrical lead from the pump.

d) Drain fuel remaining in lines using the low-point drains located on the lower right fuselage.

e) Disconnect the fuel line from the fuel filter at the inlet end of the pump. Cap all open fuel lines to prevent contamination.

f) Unscrew the used fuel filter from the pump. Install the new fuel filter on the pump. Torque filter per "Figure 13" on page 214031.

g) Connect the fuel line to the fuel filter at the inlet end of the pump.

NOTE: Be careful not to rotate the tube assembly on reinstallation.

h) Connect the electrical lead to the pump.

i) Conduct Replacement Ground Test under Operational Tests, above.

j) Replace the access cover to fuel pump assembly. Replace the center floor panel. Replace the aft facing rear seats.

2) Removal

The heater fuel pump is located below the cabin floor panel between the main and rear spar on the right side of the cabin. It is enclosed in a fiberglass compartment which has a removable access cover.

a) Ascertain that the right fuel tanks are empty and both fuel selector controls are in the OFF position.

b) Access the fuel pump as follows: remove the aft facing rear seats; remove the center floor panel; and remove the access cover to fuel pump assembly.

c) Disconnect the electrical lead from the pump.

d) Drain fuel remaining in lines using the low-point drains located on the lower right fuselage.

e) Disconnect the fuel line from the fuel filter at the inlet end of the pump and the regulator from the outlet end. Cap all open fuel lines to prevent contamination.

f) Remove the bolts which secure the pump to its mounting bracket.

3) Installation

a) Position the fuel pump to its mounting bracket and secure in place with bolts.

b) Connect the regulator to the pump outlet and the fuel line to the pump inlet.

NOTE: Be careful not to rotate the tube assembly on reinstallation.

c) Connect the electrical lead to the pump.

d) Conduct Replacement Ground Test under Operational Tests, above.

e) Replace the access cover to fuel pump assembly. Replace the center floor panel. Replace the aft facing rear seats.

(8) Duct Switch

See “Figure 1” on page 21407.

(a) Removal:

1) Disconnect the electrical leads from the terminals on the exposed face of the switch and mark to facilitate installation.

2) Remove the two attaching screws and washers from the duct switch bracket.

3) Carefully lift out the switch and gasket (if gasket is used).

(b) Cleaning and Inspection:

Brush off any dust or lint from the switch operating mechanism (exposed inside the duct) and wipe the external surfaces with a clean cloth.

(c) Installation:

1) Insert the switch carefully with gasket (if used) into the ventilating duct opening and secure with the two attaching screws and washers.

2) Connect the two electrical leads to their respective terminals on the face of the switch as marked during removal.

3) Operate the heater with the duct switch set above ambient temperature to check operation.

H. Inspection of Fuel Nozzle Orifice

See “Figure 14” on page 214032.

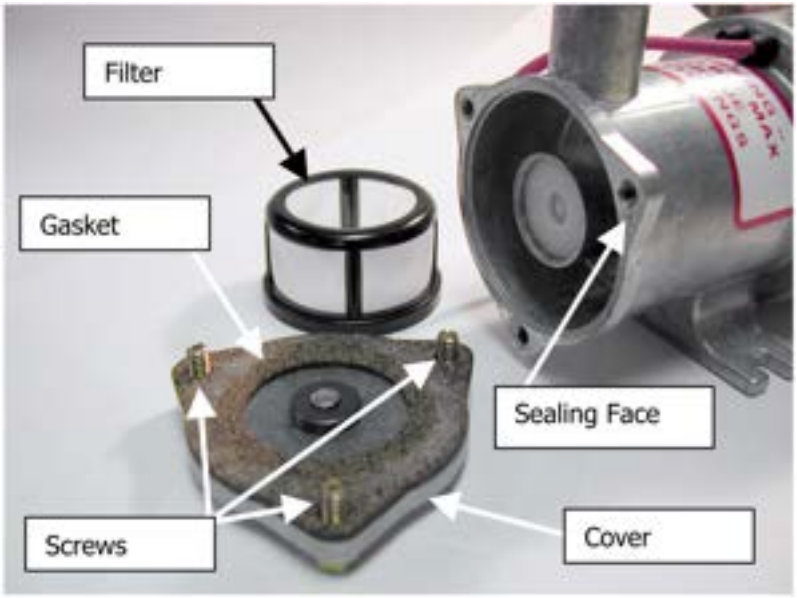
(1) Loosen the four screws and rotate the blower and motor housing to disengage the ventilating air blower from the end of the heater jacket. It is not necessary to disconnect the electrical connections to remove the nozzle.

(2) Remove the fuel shroud cover by removing the screws. Remove solenoid and elbow.

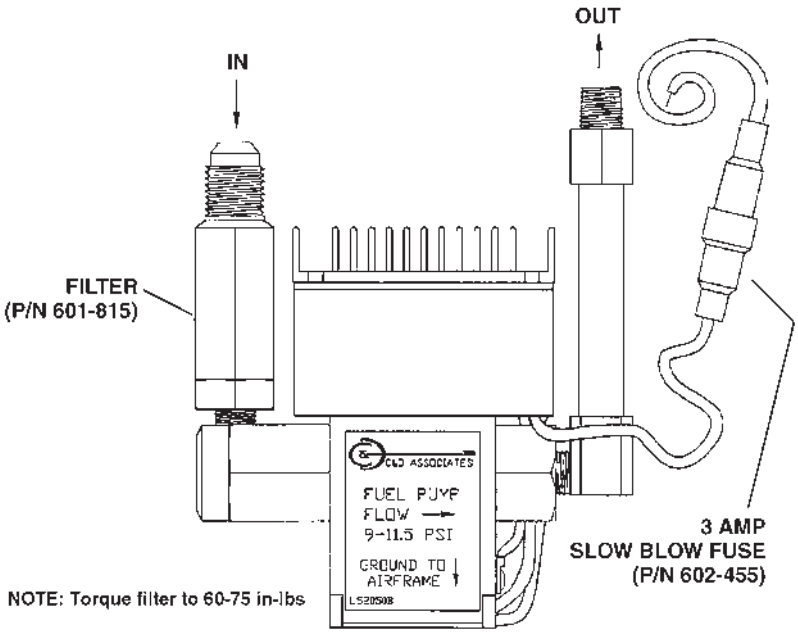
(3) Reach inside the inlet end of the jacket assembly with a 3/4 inch deep socket to remove the nut, washer and gasket and lower fuel shroud box.Remove the two screws and carefully withdraw the nozzle holder and valve assembly from the combustion head assembly.

(4) Carefully unscrew and remove the spray nozzle from the nozzle holder. Remove the gasket.

(5) After cleaning the nozzle, reinstall the parts removed in essentially the reverse order from removal. Be sure to hold the fuel-tube fitting when tightening the nut to avoid damage to the fuel tube.



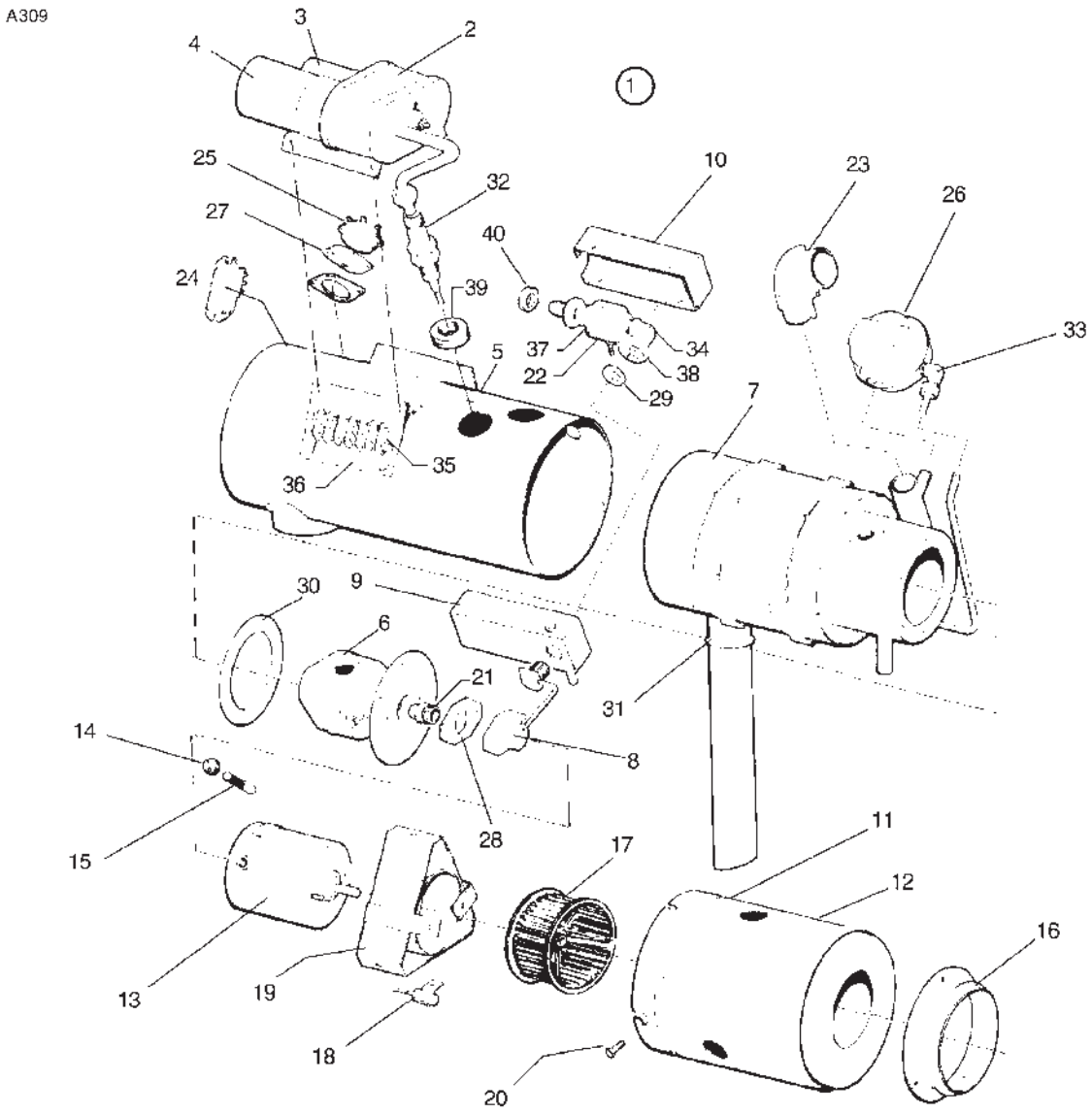
Heater Fuel Pump
Figure 13 (Sheet 1 of 2)



HEATER FUEL PUMP
(P/N 461-773 installed as part of Kit P/N 461-771)

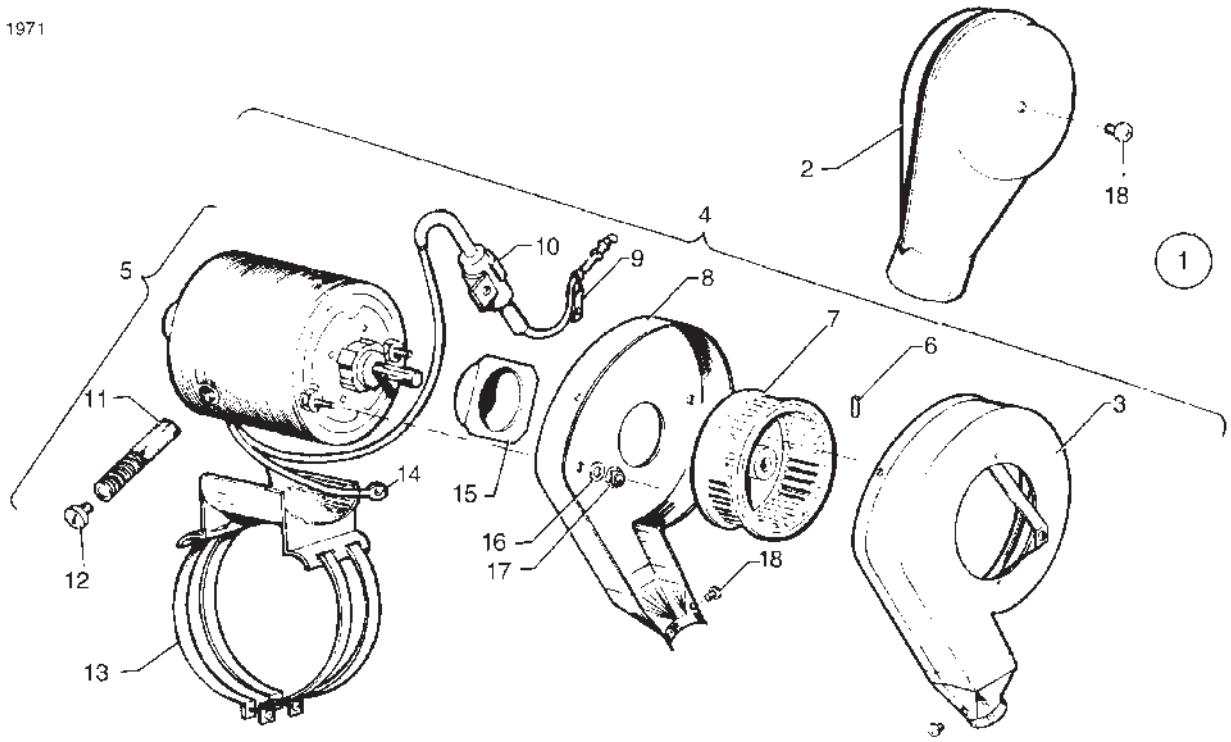
Heater Fuel Pump
Figure 13 (Sheet 2 of 2)

Effectivity
3449414 and up; and
3449001 thru 3449413
and Seneca IVs
with Kit No. 461-771 installed



1. HEATER ASSEMBLY
2. IGNITION ASSEMBLY
3. VIBRATOR - IGNITION
4. COIL - IGNITION
5. JACKET ASSEMBLY
6. HEAD ASSEMBLY - COMBUSTION
7. TUBE ASSEMBLY - COMBUSTION
8. FUEL FEED AND NOZZLE HOLDER ASSEMBLY
9. BOX ASSEMBLY - FUEL SHROUD, LOWER
10. BOX ASSEMBLY - FUEL SHROUD, UPPER
11. BLOWER ASSEMBLY - VENT AIR
12. HOUSING - BLOWER
13. MOTOR ASSEMBLY - VENT AIR BLOWER
14. CAP - BRUSH ASSEMBLY
15. BRUSH ASSEMBLY - MOTOR
16. ADAPTER
17. FAN - VENT AIR BLOWER
18. CAPACITOR ASSEMBLY
19. BRACKET ASSEMBLY - MOTOR
20. FASTENER
21. NOZZLE - FUEL
22. SOLENOID ASSEMBLY - FUEL
23. ADAPTER - ELBOW STRIP
24. SWITCH - CYCLING
25. SWITCH - LIMIT
26. SWITCH - PRESSURE
27. GASKET - LIMIT SWITCH
28. GASKET
29. GASKET
30. GASKET
31. GASKET - ASBESTOS
32. PLUG - SPARK
33. ELBOW
34. ELBOW
35. STRIP - TERMINAL
36. INSULATOR - TERMINAL STRIP
37. NIPPLE
38. NUT
39. GROMMET
40. GROMMET

Exploded View of Heater Assembly
Figure 14



1. COMBUSTION AIR BLOWER AND MOTOR ASSEMBLY
2. ADAPTER ASSEMBLY - BLOWER INLET
3. HOUSING - BLOWER OUTER HALF
4. COMBUSTION AIR BLOWER AND MOTOR
5. MOTOR ASSEMBLY - COMBUSTION AIR BLOWER
6. SET SCREW - BLOWER FAN
7. FAN - COMBUSTION AIR BLOWER
8. HOUSING - BLOWER INNER HALF
9. STRAP - CABLE
10. CAPACITOR
11. BRUSH ASSEMBLY
12. CAP - BRUSH ASSEMBLY
13. MOUNT - COMBUSTION AIR BLOWER SUPPORT
14. ELECTRICAL LEAD
15. SPACER
16. WASHER
17. LOCKNUT - AN345-10
18. SCREWS

Exploded View - Combustion Air Blower and Motor Assembly
Figure 15

I. Heater System - Operational Test

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

(1) Check all fittings and connections for condition and security of mounting, and all ducts for freedom of obstructions.

(2) Disconnect wire (H 10A) from the heater terminal No. 2; this will remove electrical power to the fuel valve and pump so heater will not ignite.

(3) Turn the master switch and “HEATER” switch on and open the air intake valve. Both blowers (combustion air and ventilating air) should operate. Check at heater exhaust and ventilating air outlets to ensure airflow.

(4) Momentarily insert a wedge under the leaf of the main gear squat switch. The ventilation blower should stop operating.

(5) Turn off heater switch and remove wedge at squat switch.

(6) To ensure that the heater fuel line is free of airlock, cautiously loosen the fuel connection at the heater. This will bleed the line between the heater and fuel source. Then tighten the fuel line connection.

(7) Reconnect the wire (H 10A) to the heater terminal No. 2.

(8) Place the air intake lever in the “OPEN” position and the temperature control lever in the center of its travel.

(9) Install a 0 to 10 psi pressure gauge in the outlet line of the fuel regulator by installing a “T” fitting in the OUTLET opening of the regulator.

(10) Turn on the master switch; then press the press-to-test in the annunciator panel — red overheat light should illuminate with other warning lights.

(11) Turn on heater switch. The heater should ignite and continue to operate until the thermostat turns it off. Cycling in this manner should continue until the heater switch is turned off.

(12) With the heater in operation, check the pressure gauge. The gauge should read from 6.5 to 7.5 psi; if the heater is running and the pressure indicated is more or less than required, adjust the regulator accordingly. If the required pressure cannot be reached after a couple turns of the regulator’s adjustment screw, troubleshoot the fuel pump.

(13) Place the heater switch in the “FAN” position. The heater should turn off and the ventilation blower should continue to operate. Allow this blower to operate for a full two minutes; then place the air intake lever in the closed position. The blower should turn off.

NOTE: This procedure should be followed after every shutdown to cool off the burn chamber.

(14) With the air intake closed, turn on the heater switch; the heater should not ignite and neither fan should operate. Turn off the heater switch and master switch.

(15) Remove the pressure gauge and “T” fitting from the regulator.

J. Overhaul Instructions

The heater assembly shall be overhauled after 1500 hours of heater time or when the “Pressure Decay Test” requirements cannot be met. See latest Aircraft Overhaul and Maintenance Manual as described under Supplementary Publications, Vendor Publications, Heater, in the Introduction.

K. Test Procedure

(1) General Information

A test of all components should have been made after overhaul to ensure proper operation. Some shops may not have complete testing facilities for measuring airflows, pressure drops, and other factors which would be accomplished in a laboratory-type test. If such a test cannot be made, install the heater and check operation on the ground and in the air to determine if operation is normal. In shops where complete test equipment is available and a complete functional test can be performed, the test routine described in subsequent paragraphs should be made.

(2) Equipment Required

See “Figure 16” on page 214036.

(a) An improvised stand to hold the heater during test. The heater should be located far enough away from any combustible material or atmosphere to avoid hazard. A location should be chosen where exhaust can be dispelled. Do not add an excessive extension to the heater exhaust.

(b) A source of fuel capable of being regulated at seven psi.

(c) The combustion air blower to be used with the heater should be used for the test.

(d) A 12-volt current supply which may be a dc generator with a rheostat, ammeter, and voltmeter in the line to control and indicate the current draw and voltage output.

(e) Two water manometers (zero to 5.0 inch water column) for measuring the pressure in the ventilating air duct and in the combustion air stream.

(f) A piece of duct to be attached to the downstream end of the heater. It should have a minimum length of 24 inches and the same diameter as the heater being tested. A 2.25 inch diameter orifice should be centrally located at the outlet end. An aperture should be provided for the thermometer and duct switch and a static tap should be attached as shown in “Figure 16” on page 214036.

(g) A thermometer with 500°F scale.

(h) A fuel-pressure gauge.

(i) A controlled source of compressed air for final leakage test.

(3) Operational Test (On Test Bench)

See “Figure 16” on page 214036 and “Figure 17” on page 214037.

(a) Connect the heater to the test setup as shown in “Figure 16” on page 214036. Make sure the combustion air blower is mounted securely and that the heater is clamped to its supporting stand.

(b) Insert the duct switch in the sheet metal extension tube at the location shown in “Figure 16” on page 214036.

(c) Connect components and heater as outlined in the wiring connection diagram, “Figure 17” on page 214037. The power supply switch should be open.

(d) Connect the power source to the heater.

(e) Disconnect wire lead from terminal No. 3 on the heater side of heater terminal strip to prevent the heater from lighting and close the power source switch to check operation of blowers. The combustion air blower and ventilating air blower should operate at full speed with no blower wheel interference. If either blower fails to run, locate and correct the trouble before proceeding with the test.

(f) Connect a voltmeter from open side of combustion air pressure switch terminal to ground to determine if the switch is closed, which would be indicated by a full voltage reading on the meter. If a full voltage reading is not obtained, the combustion air supply is either inadequate or the switch is defective or improperly adjusted. Make necessary corrections.

(g) Observe the manometer connected to the ventilating air pressure tap, which should show a reading of 1.1 inches of water (minimum) at rated voltage.

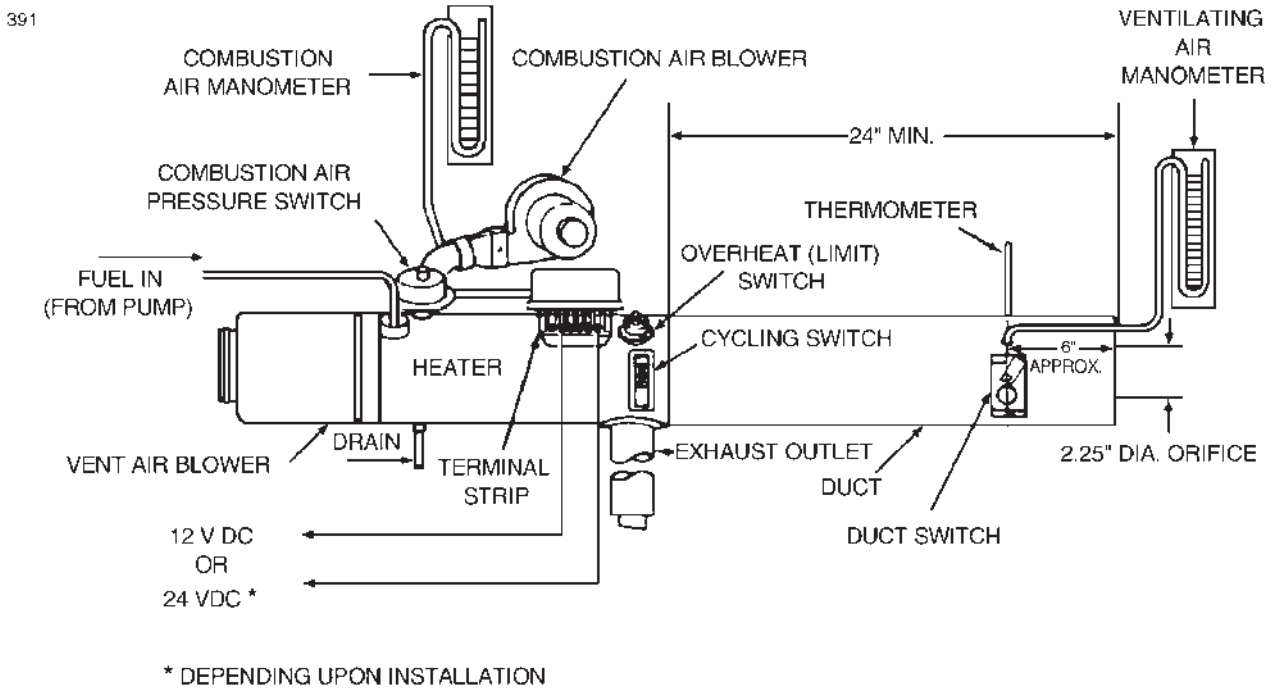
(h) Observe the manometer connected to the combustion air tube tap, which should show a reading of 1.5 inches of water (minimum) at rated voltage.

(i) Open power supply switch and reconnect terminal lead disconnected in preceding Step (e).

(j) Close the power supply switch and turn on the fuel supply. The heater should light within five seconds (may require slightly longer for air to be purged from fuel lines on the first trial).

(k) Observe operation of duct switch, which should control heater operation according to the switch setting.

(l) If the duct switch fails to control the temperature according to the setting, place the control lever in high “H” position and notice the control variation. A high reading of 250°F ± 10° should be obtained (reading will vary in different applications).



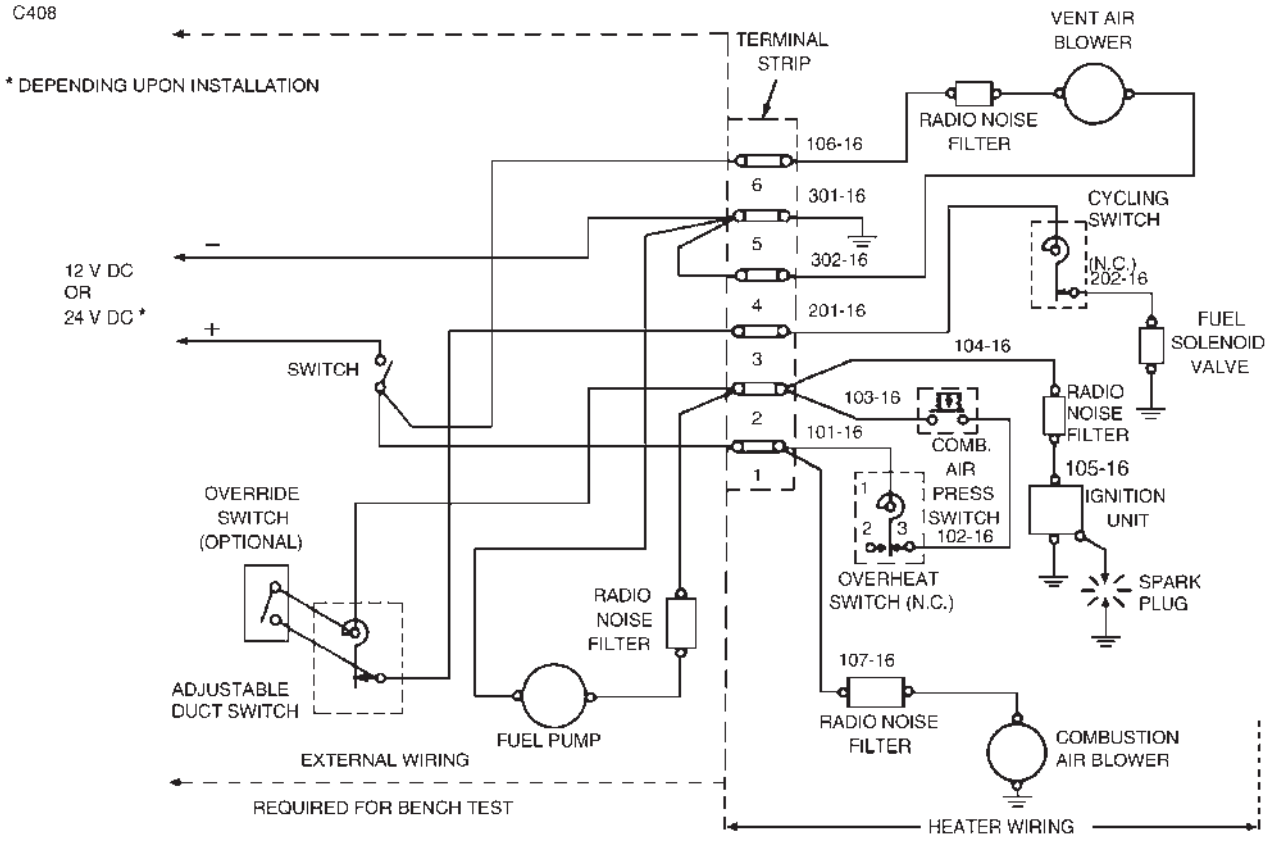
Suggested Setup of Heater Operation Test
Figure 16

(m) Connect a jumper across the terminals of the duct switch to make it inoperative and observe action of the cycling switch. The cycling switch should cycle to control the outlet air temperature at approximately 250°F (nominal). This is a function of ambient temperature and airflow conditions. If operation is within a range of 190°F to 290°F, the switch is operating normally. If the switch is out of range, it can be reset in the same manner as described for the duct switch, except that no control lever or indicator stop are used. If adjustment fails to restore proper temperature range, replace the switch.

(n) With duct switch still jumped, place a jumper across the cycling switch terminals to check operation of the overheat switch. Block the ventilation air outlet and notice if the overheat switch shuts off the heater. It should open at between 300°F and 402°F. (This is also a function of ambient temperature and airflow.) After the switch shuts off, remove ventilating air restriction; remove jumpers from cycling and duct switches and press firmly on the overheat switch reset button until it “clicks.” The heater should light and operate.

(o) Shut down the heater and check all components visually to make sure no damage has occurred to any of them.

(p) Remove heater and other components from the test setup and install it in the airplane.



Wiring Connection for Heater Operation Test
Figure 17

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COOLING

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

1. Fresh Air (Standard)

   See 21-20-00.

   When the optional air conditioning system is not installed cabin ventilation and cooling is provided solely by fresh air through the overhead vent system.

2. Air Conditioning (Optional)

   A. Description and Operation

      See “Figure 1” on page 21502.

      WARNING: REFRIGERANT R12 IS USED IN THE SENECA IV AND REFRIGERANT R134A IS USED IN THE SENECA V. IT IS IMPORTANT THAT ANYONE SERVICING THE AIR CONDITIONING SYSTEM BE FAMILIAR WITH THE REFRIGERANT, LUBRICANT, AND COMPONENTS USED IN THAT PARTICULAR SYSTEM.

      NOTE: Operate air conditioning system at least once a month to keep system lubricated and prevent sticking valves.

      Besides the specific plumbing, the air conditioning installation, from a servicing standpoint, actually involves five significant component installations; the compressor, condenser, condenser scoop, evaporator-dehydrator, and controls installations.

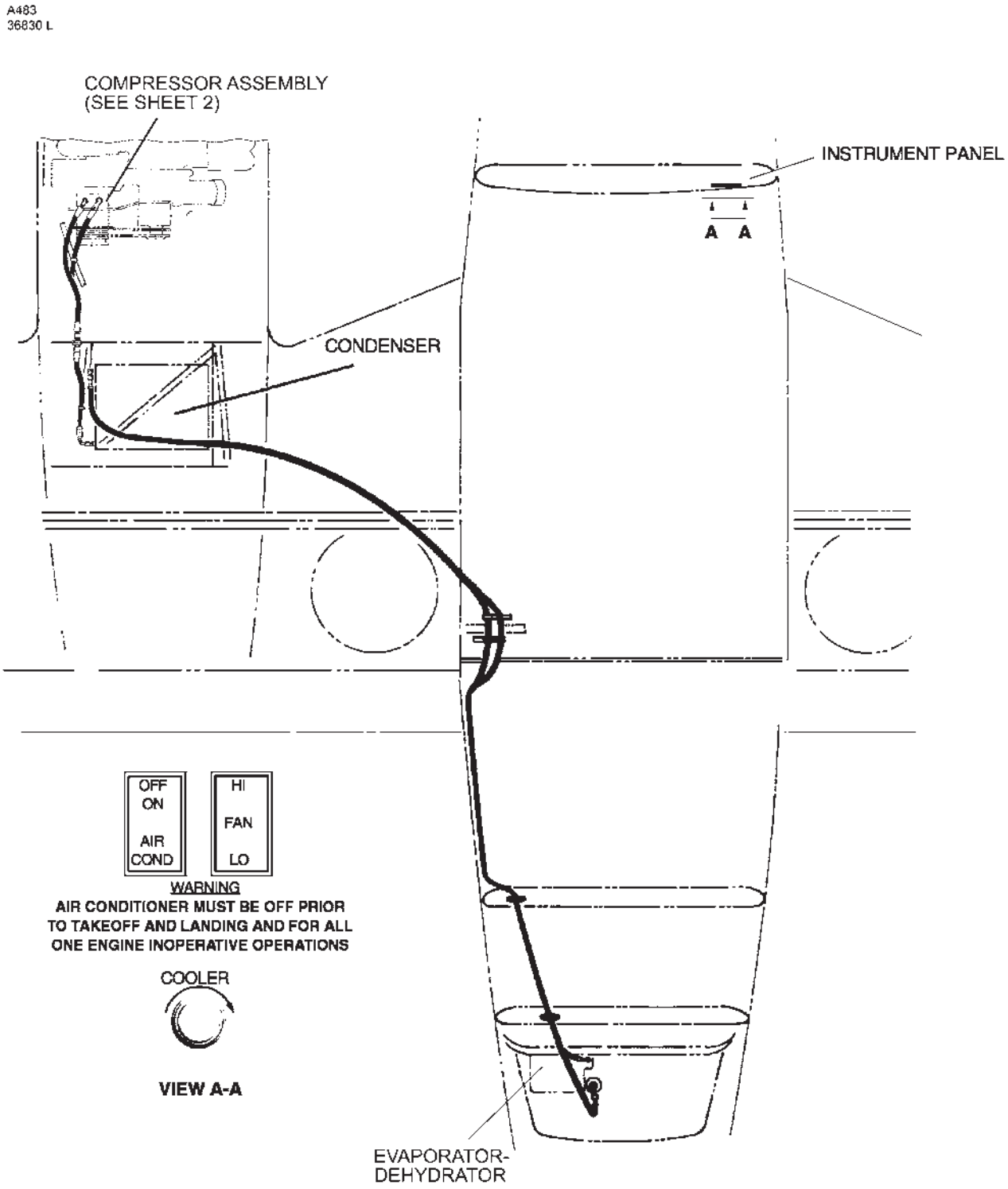
      The compressor, a piston type unit, is mounted to the left, rear side of the left engine. A V-belt connection from a gear box extension off the accessory case, drives the compressor through an electromagnetic clutch. The compressor is supported and made adjustable by front and rear brackets also mounted to the accessory case. Access to the unit and lines can be made by removing the upper cowl of the left engine.

      The condenser is mounted on the left nacelle to its air scoop support structure and is accessible upon removal of the upper nacelle hatch cover. A scoop is also located at the same location, and is designed to be electrically open. The design is such that the condenser can be cooled without increasing drag. A set of louvers further back on the nacelle allows air flow out of the nacelle.

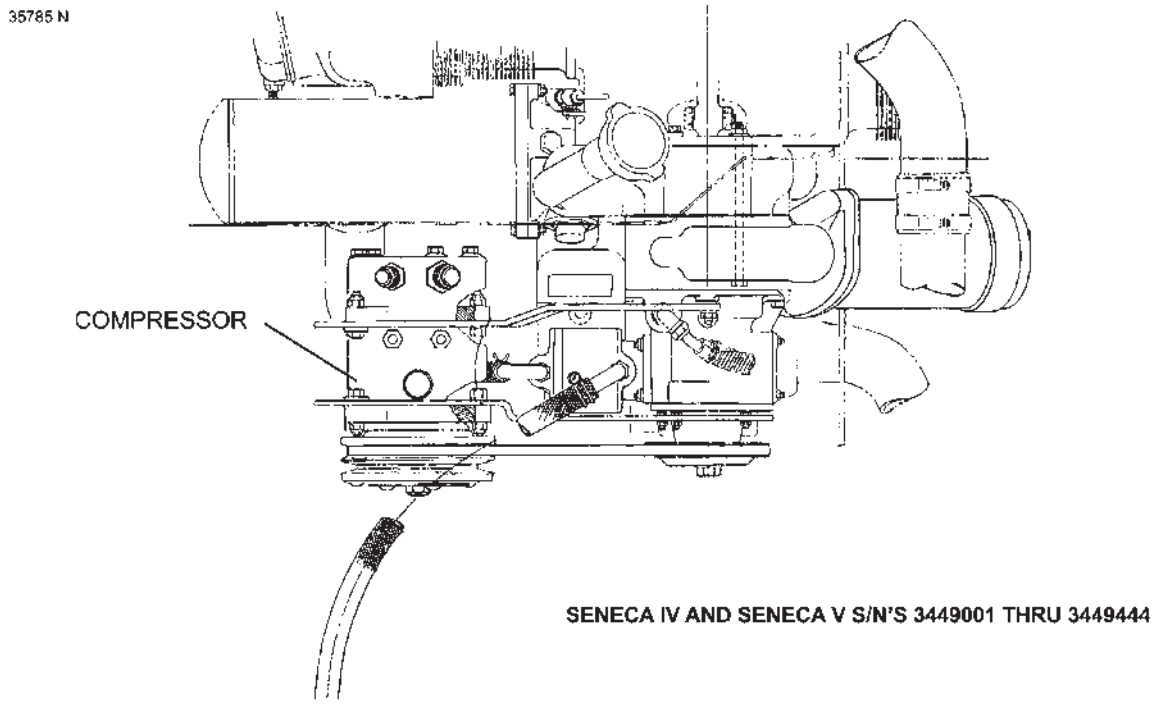
      The evaporator-dehydrator installation is incorporated in the rear of the fuselage just aft of the baggage compartment. Access to the unit can be made by removing the false bulkhead in the rear of the baggage area.

      Controls for the system are located on the right side of the instrument panel and consist of a HIGH-OFF-LOW fan switch, an ON-OFF air conditioning switch, and thermostat control.

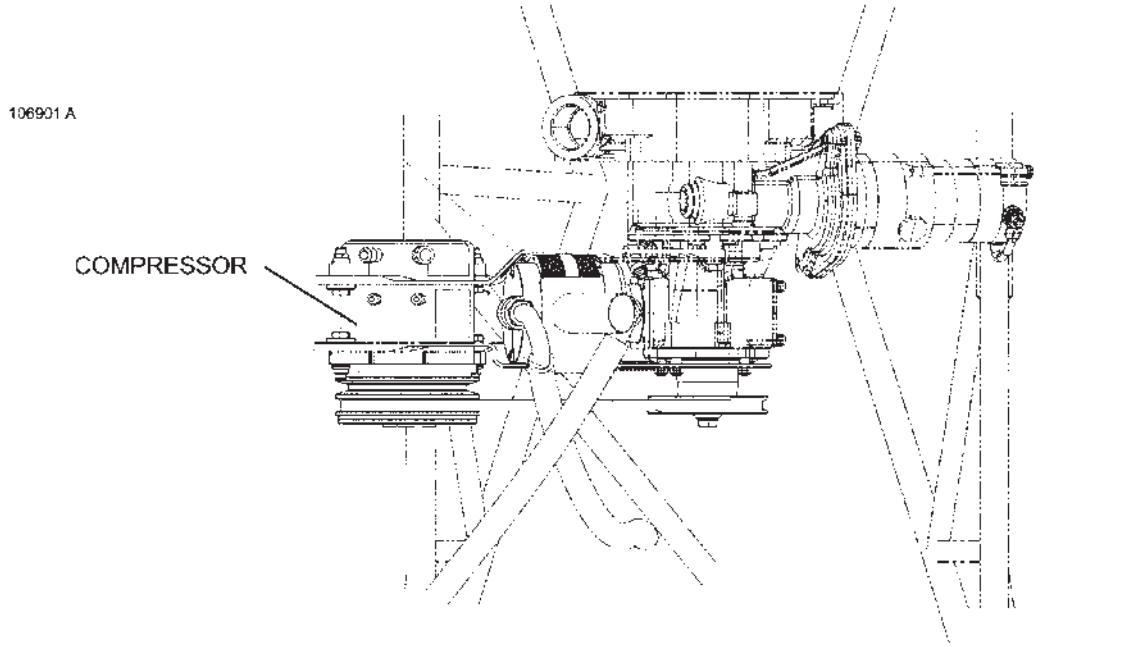
      The air conditioning system is also an independent unit which filters, dehumidifies and cools the cabin air by recirculating it through the evaporator-dehydrator. To operate the system, the fan switch must first be placed in the HIGH or LOW position to provide power to the air condition switch. This also provides a second way of using the system in that the fan can be used only to recirculate the air. With the fan and air condition switches on, the clutch on the compressor is engaged, the scoop opened, and the circulating fan turned on. Temperature is then controlled by a radial thermostat control on the copilot’s side of the instrument panel.



Air Conditioning Installation
Figure 1 (Sheet 1 of 2)



SENECA IV AND SENECA V S/N'S 3449001 THRU 3449444



SENECA V S/N'S 3449445 AND UP;
AND 3449001 THRU 3449444 WITH KIT 88502-002 INSTALLED

Air Conditioning Installation
Figure 1 (Sheet 2 of 2)

In the Seneca IV, with the system in operation, refrigerant is pulled into the compressor as a vapor. The vapor is compressed and routed to the condenser where it is cooled and turned into a liquid. The liquid refrigerant then passes to the receiver-dehydrator where fluid is filtered and any moisture removed. At this point the liquid is regulated at a steady flow by an expansion valve. This thermally controlled metering valve governs the flow of the liquid refrigerant into the evaporator where the liquid changes to a gas and absorbs the heat from the air passing over the coils. From the evaporator the vaporized liquid returns to the compressor to restart the cycle. A pressure switch is also incorporated in the system to automatically control the condenser maximum head pressure by temporarily declutching the -compressor in the event the pressure becomes excessively high.

NOTE: The air conditioning system should be operated at least once a month to prevent sticking valves and keep the system lubricated.

The air conditioning system in the Seneca V uses HFC-134a. Refrigerant enters the compressor as a vapor. The compressor pressurizes the heat laden vapor until the vapor temperature becomes warmer than the outside air temperature. The compressor then pumps the vapor to the condenser where the refrigerant is cooled and changes to liquid. The liquid now passes to the receiver/dehydrator. The receiver/dehydrator filter removes moisture and ensures a steady flow of liquid refrigerant (which is visible in the receiver/dehydrator's sight glass) into the evaporator through the expansion valve. The expansion valve is a temperature controlled metering valve which regulates the flow of liquid refrigerant to evaporator. The evaporator enables the liquid refrigerant to absorb heat from the outside air passing over coils, converting it back to a vapor. From the evaporator, heat laden refrigerant in a vapor state returns to compressor, and the cycle repeats.

NOTE: "HFC-134a" means air conditioning refrigerant which is generically identified and documented as 1,1,1,2-Tetrafluoroethane and/or CAS# 811-97-2. Brand names and commercial designations which meet these requirements include (but are not limited to):

HFC134a, HFA134a, R134a, Dymel® 134a, Fluorocarbon 134a, Forane® 134a, Genetron® 134a, Halocarbon 134a, KLEA® 134a, Norfluane, Referon® 134a, and SUVA™ 134a.

B. Troubleshooting

See "Chart 1" on page 21505.

CHART 1 (Sheet 1 of 5)
TROUBLESHOOTING AIR CONDITIONING SYSTEM

Trouble	Cause	Remedy
High discharge pressure.	Refrigerant overcharge.	Purge excess refrigerant.
	Air in system.	Check for leaks. Bleed charge from system. Evacuate and recharge system.
	Overheated condenser due to blocked air passage.	Clean bugs and dirt from condenser fins. Straighten bent fins.
	Flooded evaporator indicated by heavy frosting on suction line and compressor suction service valve.	Check capillary bulb is securely clamped to suction line. If capillary bulb is OK, replace expansion valve.
	Restriction in liquid line from condenser.	Check for kinked hoses and clogged filter.
Low discharge pressure.	Refrigerant undercharge. Sight glass shows bubbles or foam.	Add refrigerant until bubbles disappear. Check system for leaks.
	Damaged compressor valves or dirt under valves.	Replace compressor.
	Damaged compressor. Worn or broken piston or piston rings.	Replace compressor.
Low suction pressure accompanied by icing of evaporator.	Low air supply through evaporator.	Repair blower or blower motor. Clean stoppage in air ducts.
	Very dirty evaporator fins and coils.	Clean and flush with water.
Low suction pressure. (Evaporator not cold enough.) Suction gauge reads vacuum indicating evaporator lacks refrigerant. clogged. Inoperative expansion	Refrigerant undercharge. Moisture freezing in expansion valve. Valve shows frost.	Add refrigerant. Install new dryer. Evacuate and recharge.
	Expansion valve inlet screen	Remove screen. Clean with solvent and replace.
	valve. Valve stuck closed or capillary bulb has lost charge.	Warm capillary by holding in hand. If suction pressure does not change replace expansion valve.
	Restriction in liquid line. Restriction will show frost.	Locate restriction and repair.

CHART 1 (Sheet 2 of 5)
TROUBLESHOOTING AIR CONDITIONING SYSTEM

Trouble	Cause	Remedy
High suction pressure.	Capillary bulb clamp loose on suction line. Suction line shows frost.	Clean contact surfaces of suction line and cap bulb. Tighten clamp.
	Expansion valve not closing. Evaporator flooded. Suction line frosted to compressor.	Replace expansion valve.
	Compressor drive belt slipping.	Adjust belt tension.
	Magnetic clutch slipping.	Check electrical circuit for proper voltage to clutch coil. Clean oily clutch surfaces.
	Leaking or broken compressor.	Replace compressor valves.
Condenser door will not close when air conditioner switch is in OFF position.	Faulty K-2 relay.	Replace relay.
System does not cool.	**If electrical:**	
	Blown fuse in control head.	Replace fuse.
	Open circuit breaker.	Set circuit breaker.
	Broken or disconnected electrical wire.	Check all terminals for loose connections. Check wiring for hidden breaks.
	Broken or disconnected ground wire.	Check ground wire is not loose, broken, or disconnected.
	Clutch coil burned out or disconnected.	Verify voltage to clutch. Replace if inoperative.
	Thermostat sensing element defective.	Check thermostat and cabin comfort control panel.
	Blower motor disconnected or burned out motor.	Verify voltage to blower. Repair or replace if inoperative.

CHART 1 (Sheet 3 of 5)
TROUBLESHOOTING AIR CONDITIONING SYSTEM

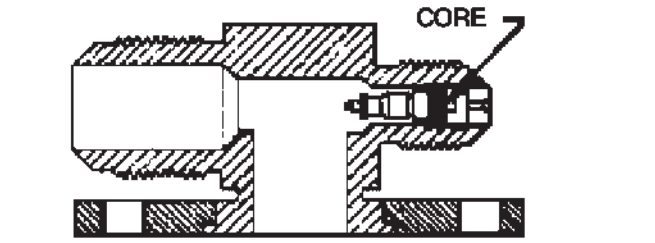
Trouble	Cause	Remedy
System does not cool. (cont.)	**If mechanical:**	
	Loose or broken drive belt.	Replace drive belts and tighten to specifications.
	Compressor partially or completely frozen.	Remove compressor. Service or replace.
	Expansion valve stuck in open position.	Replace expansion valve.
	If refrigeration:	
	Broken refrigerant line.	Examine all lines for evidence of breakage by external stress or rubbing wear.
	Leak in system.	Evacuate system, apply static charge, leak test system, and repair leak as necessary.
	Compressor shaft seal leaking.	Replace compressor.
	Clogged screen or screens in receiver dehydrator or expansion valve; plugged hose or coil.	Repair as necessary.
System cooling inadequate.	**If electrical:**	
	Blower motor operation sluggish.	Remove blower motor for service or replacement.
	If mechanical:	
	Compressor clutch slipping.	Remove clutch assembly for service or replacement. Check clutch airgap and coil.
	Obstructed blower passage.	Examine passage for obstruction. Correct as necessary.
	Insufficient air circulation over condenser coils; fins clogged with dirt or bugs.	Clean condenser coils.
	Clogged evaporator filter.	Clean with solvent.

CHART 1 (Sheet 4 of 5)
TROUBLESHOOTING AIR CONDITIONING SYSTEM

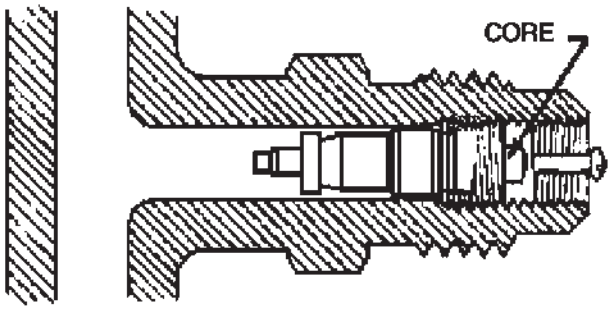
Trouble	Cause	Remedy
System cooling inadequate. (cont.)	**If refrigeration:**	
	System refrigerant low.	Recharge system until bubbles disappear in receiver dehydrator and gauge readings stabilize to specifications.
	Clogged screen in expansion valve.	Purge system, replace expansion valve.
	Expansion valve thermal bulb has no charge.	Purge system, replace expansion valve.
	Clogged receiver dehydrator screen.	Purge system, replace receiver dehydrator.
	Excessive moisture in system.	Purge system, replace receiver dehydrator.
	Air in system.	Purge, evacuate, and charge system. (Replace receiver dehydrator.)
Excessively noisy system.	**If electrical:**	
	Defective winding or connection in compressor clutch coil.	Replace or repair as necessary.
	If mechanical:	
	Loose or worn drive belts, crankshaft pulley, or idler pulley or bearing.	Tighten or replace as required.
	Engine components such as: alternator, water pump, valves, timing or mounts.	Check.
	Compressor mounting bolts or brackets - broken or loose.	Check, repair, replace.
	Compressor oil level low.	Fill with proper amount of specified oil.
	Compressor failure.	Check shaft turning smoothness. Remove compressor for service or replacement.
	Magnetic clutch failure.	Check airgap, clutch pulley, front plate, coil, and bearing. Adjust, repair, or replace, as required.

CHART 1 (Sheet 5 of 5)
TROUBLESHOOTING AIR CONDITIONING SYSTEM

Trouble	Cause	Remedy
Excessively noisy system. (cont.)	**If refrigeration:**	
	Excessive system charge.	Remove excess refrigerant until high pressure gauge drops within specifications.
	Low system charge.	Check system for leaks. Recharge system.
	Excessive moisture in system.	Replace dehydrator, purge, evacuate, and recharge system.

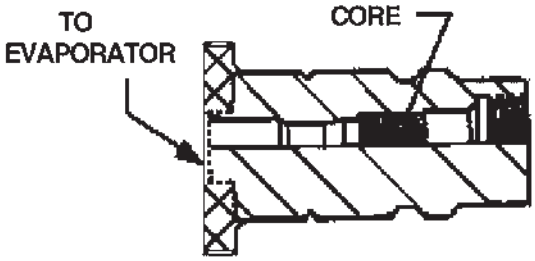


COMPRESSOR VALVE (SCHRADER CHARGING)

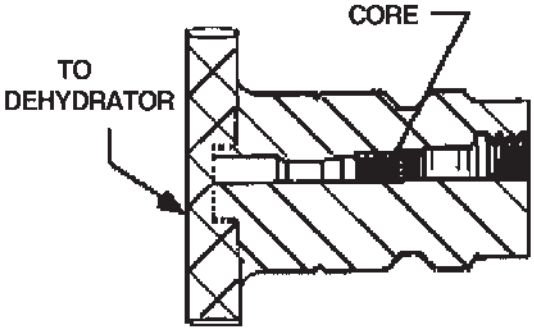


INLINE SERVICE VALVE (SCHRADER)

SENECA IV



LOW SIDE (PRESSURE) VALVE



HIGH SIDE (PRESSURE) VALVE

SENECA V

Service Valves
Figure 2

3. Servicing Cooling System

NOTE: The maximum refrigerant capacity is 2.25 pounds. The total refrigerant capacity required is determined separately for each system and is the amount that will result in bubble-free operation at the sight gauge, as specified in the post charging operational check.

A. Definitions

(1) High Side. The “high side” consists of all lines and components between the compressor outlet and the expansion valves. It includes the condenser and receiverdryer sight gage.

(2) Low Side. The “low side” consists of all lines and components between the expansion valves and the compressor inlet. It includes the evaporators.

B. Service Valves

i.e. - ports, See “Figure 1” on page 21502, “Figure 2” on page 215010, and “Figure 14” on page 215039.

(1) Description

The aircraft is equipped with inline service valves mounted in the suction and discharge lines of the evaporator assembly located behind the cabin rear closeout panel. These valves are installed to Test, Bleed (Discharge), Evacuate and Charge the air conditioning system. All normal air conditioning service should be performed at the manifold assembly mounted valves.

On the Seneca IV, these valves are the two-position screw-on type Schrader valves. On the Seneca V, these valves are the quick disconnect type Schrader valves.

The valve in the short line between the receiver and the expansion valve is the high side service port. The other valve, located nearby, is the low side service port.

NOTE: On Seneca IV models, service valves are also located on the compressor. However, use of these valves in servicing is not recommended.

NOTE: If a Schrader service valve is not serviceable, the core assembly must be replaced.

(2) Replacement

CAUTION: WHENEVER THE AIR CONDITIONING REFRIGERANT LINES OR SYSTEM ARE OPENED FOR ANY REASON, THE LINES AND FITTINGS SHOULD BE CAPPED AND SEALED IMMEDIATELY TO PREVENT DIRT AND OTHER CONTAMINANTS FROM ENTERING THE SYSTEM. (DO NOT PUT A PLUG INTO THE HOSES OR FITTINGS.)

The Seneca IV models have service valves on the compressor. These valves are sealed with a gasket placed in the valve port boss. Lubricate the gasket with refrigerant oil of the type used in the compressor, place the valves with the tube fitting facing aft and secure with .312 bolts. torque to 15-23 inch-pounds.

CHART 2
TEMPERATURE VS. PRESSURE

Refrigerant R12		Refrigerant HFC134a	
Evaporator Pressure Gauge Reading psi	Evaporator Temperature °F	Evaporator Pressure Gauge Reading psi	Evaporator Temperature °F.
0	-21	-5	-27
2.4	-15	0	-15
4.5	-10	2	-9
10.1	2	4	-4
11.2	4	6	0
12.3	6	8	4
13.4	8	10	7
14.6	10	12	11
15.8	12	14	14
17.1	14	16	17
18.3	16	18	20
19.7	18	20	22
21	20	22	25
22.4	22	24	28
23.1	23	26	30
23.8	24	28	33
24.6	25	30	35
25.3	26	32	37
26.1	27	34	39
26.8	28	36	41
27.6	29	38	43
28.4	30	40	45
29.2	31	42	47
30	32	44	49
30.9	33	46	51
31.7	34	48	53
32.5	35	50	54
33.4	36	55	58
34.3	37	60	62
35.1	38	65	66
36	39	70	69
36.9	40		
37.9	41		
38.8	42		
39.7	43		
41.7	45		
43.6	47		
45.6	49		
48.7	52		
49.8	53		
55.4	57		
60	62		
64.9	66		

C. Malfunction Detection

NOTE: If the cooling system has leaked refrigerant or is discharged, the compressor oil level must be checked.

Detection of system malfunction largely depends on the mechanic's ability to interpret gauge pressure readings into system problems. A system operating normally will have a low side gauge pressure reading that will correspond with the temperature of the refrigerant evaporating, allowing for a few degrees temperature rise due to loss in tube walls and fins. The high side will have a gauge pressure that will correspond with the temperature of the refrigerant condensing in the condenser, allowing for a few degrees temperature drop due to loss in tube walls and fins.

Any deviation from that which is normal indicates a malfunction within the system due to faulty control device, obstruction, defective part, or improper installation.

Detection of system malfunction is made easier with knowledge of the relationship between temperature and pressure of the refrigerant. See "Chart 2" on page 215012 for specific values.

NOTE: Gauge readings are about one inch mercury or 1/2 psi higher than chart reads for each 1000 feet elevation above sea level.

Actual air temperature of air passing over the evaporator coils will be several degrees warmer allowing for a temperature rise caused by the loss in the fins and tubing of the evaporator.

The importance of a seasonal check-up of the air conditioning system should be brought to the attention of the customer whenever possible. A thorough check of the system performed in a methodical manner will reveal trouble the customer is often not aware of. Locating and repairing the trouble early will usually result in savings to the customer both in time and additional troubles that too often result from neglect.

A performance test of the system is the only positive way in which the complete system can be checked for efficient operation. The air conditioning system should be given this test before work begins on the system whenever possible. However, if the system is completely inoperative, repairs must be performed before the system can be properly tested. The test can uncover further work that must be performed before the system is brought to its full operating efficiency. The performance test should always be performed after repair work has been done and before the airplane is released to the customer. The serviceman performing this test carefully will ensure that the repairs have been properly performed and that the system will operate satisfactorily.

The performance test, when properly performed includes a thorough examination of the outside of the system as well as the inside. Many related parts are overlooked because it is felt they are of no bearing on the operating efficiency of the unit. For this reason, a thorough visual inspection of the complete system should be performed, followed by an operating inspection of the system.

D. Special Servicing Procedures

The efficiency of this system depends upon the pressure-temperature relationship of pure refrigerant. As long as the system contains only pure refrigerant plus a specified amount of compressor oil (which is mixed with the refrigerant), it is considered to be chemically stable. Foreign materials within the system will affect the chemical stability, contaminate the system, and decrease its efficiency.

(1) Refrigerant Safety Precautions

WARNING: REFRIGERANT R12 IS USED IN THE SENECA IV AND REFRIGERANT R134A IS USED IN THE SENECA V. IT IS IMPORTANT THAT ANYONE SERVICING THE AIR CONDITIONING SYSTEM BE FAMILIAR WITH THE REFRIGERANT, LUBRICANT, AND COMPONENTS USED IN THAT PARTICULAR SYSTEM.

WARNING: AIR CONDITIONING REFRIGERANT (BOTH R12 AND R134A) IS ODORLESS AND COLORLESS IN EITHER ITS LIQUID OR GASEOUS STATE. BOTH R12 AND R134A, USED FOR CHARGING REFRIGERATION SYSTEMS, ARE IN A PRESSURIZED CONTAINER IN LIQUID FORM. BOTH REFRIGERANTS ARE INERT AT ROOM TEMPERATURE. THE PRESSURE AND REFRIGERATION EFFECTS OF RELEASE AND EVAPORATION OF THE PRESSURIZED LIQUID IS DANGEROUS.

WARNING: WEAR SUITABLE EYE PROTECTION WHEN HANDLING REFRIGERANTS. THE EYE WILL FREEZE IF CONTACTED BY ESCAPING LIQUID REFRIGERANT.

IF LIQUID REFRIGERANT CONTACTS THE EYE, THE FOLLOWING ACTIONS SHOULD BE TAKEN:

(A) DO NOT RUB THE EYE.

(B) SPLASH LARGE QUANTITIES OF COOL WATER INTO THE EYE TO RAISE THE TEMPERATURE.

(C) TAPE ON AN EYE PATCH TO AVOID THE POSSIBILITY OF DIRT ENTERING THE EYE.

(D) RUSH TO A PHYSICIAN OR HOSPITAL FOR IMMEDIATE PROFESSIONAL AID.

(E) DO NOT ATTEMPT TO TREAT IT YOURSELF.

WARNING: WEAR SUITABLE CLOTHING AND GLOVES. IF LIQUID REFRIGERANT STRIKES THE SKIN, FROSTBITE CAN OCCUR. IF EXPOSED, TREAT WITH WARM (NOT HOT) WATER AND SEEK MEDICAL ATTENTION.

(2) System Servicing Precautions

WARNING: DISCHARGE SYSTEMS SLOWLY TO PREVENT THE ESCAPE OF LIQUID REFRIGERANT AND THE LOSS OF THE LUBRICATING OIL.

NOTE: The term “Discharge,” as used throughout this section, in no sense implies or suggests discharging refrigerant to atmosphere. In all cases when discharging, an environmentally approved refrigerant recovery station is to be used.

WARNING: DO NOT LEAVE SYSTEMS OPEN TO THE ATMOSPHERE WHEN DISCHARGED. MOISTURE AND OTHER CONTAMINATION MAY ENTER AND DAMAGE OPEN SYSTEMS.

NOTE: When HFC-134a comes in contact with moisture it absorbs it into the system, which will lead to system failure.

WARNING: USE ONLY APPROVED REFRIGERATION OIL IN THE COMPRESSOR:

MINERAL OIL FOR SENECA IV; POLYALKYLENE GLYCOL (PAG) OR POLYOL ESTER (POE) FOR SENECA V.

IF ANY DOUBT EXISTS ABOUT THE CLEANLINESS OF THE COMPRESSOR OIL, REPLACE IT WITH NEW OIL.

WARNING: NEVER INTRODUCE ANYTHING BUT PURE R12 REFRIGERANT (SENECA IV) OR R134A REFRIGERANT (SENECA V) INTO THE SYSTEM.

WARNING: KEEP REFRIGERANT OIL CONTAINERS TIGHTLY SEALED AND CLEAN TO PREVENT ABSORPTION OF MOISTURE OR OTHER CONTAMINATION.

WARNING: NEVER REUSE OIL REMOVED FROM THE SYSTEM -- DISCARD IT.

CAUTION: WHEN LOCTITE REFRIGERANT SEALANT HAS BEEN USED ON A JOINT IT MUST BE HEATED TO 400°F PRIOR TO DISASSEMBLY. LOCTITE MUST BE USED TO SEAL ANY PIPE THREADS IN THE SYSTEM LINES.

CAUTION: REPLACE THE RECEIVER-DEHYDRATOR ASSEMBLY ON ANY SYSTEM WHICH HAS BEEN OPERATING WITH A LEAK ALLOWING AIR TO ENTER THE SYSTEM. IF A RECEIVER-DEHYDRATOR IS LEFT OPEN TO THE ATMOSPHERE IT SHOULD BE REPLACED DUE TO THE LOSS OF EFFECTIVENESS OF THE DRYING COMPOUND IT CONTAINS.

CAUTION: A NEW RECEIVER-DEHYDRATOR SHOULD BE OPENED AND CONNECTED TO THE SYSTEM ONLY WHEN READY TO CHARGE THE SYSTEM WITH REFRIGERANT.

CAUTION: RECOMMENDED TORQUE VALUES MUST BE USED ON ALL FLARE FITTING AND O-RING JOINTS. SEE CHART 3.

CAUTION: IF AIR CONDITIONING REFRIGERANT LINES OR SYSTEM IS OPENED, LINES AND FITTINGS MUST BE CAPPED AND SEALED IMMEDIATELY TO PREVENT DIRT AND OTHER CONTAMINANTS FROM ENTERING THE SYSTEM. (DO NOT PUT A PLUG INTO THE HOSES OR FITTINGS.)

CAUTION: UNITED STATES ENVIRONMENTAL REGULATIONS PROHIBIT THE RELEASE OF REFRIGERANT INTO THE ATMOSPHERE. SPECIAL EQUIPMENT IS REQUIRED WHEN TESTING, DISCHARGING, OR CHARGING THE SYSTEM.

CAUTION: UNITED STATES ENVIRONMENTAL REGULATIONS REQUIRE THAT AIR CONDITIONING SYSTEM REPAIRS BE ACCOMPLISHED BY A QUALIFIED SHOP WITH APPROPRIATELY TRAINED PERSONNEL.

CHART 3
RECOMMENDED TORQUE SPECIFICATIONS

ALUMINUM TUBING

Metal Tube O.D.	Thread and Fitting Size	Ft./Lb.
1/4	7/16	5-7
3/8	5/8	11-13
1/2	3/4	15-20
5/8	7/8	21-27
3/4	1-1/16	28-33

FLARE CONNECTIONS			O-RING CONNECTIONS		
Tube OD	Thread size	Ft./Lb.	Tube OD	Thread size	Ft./Lb.
3/8	5/8	18-20	3/8	5/8	11-13
1/2	3/4	36-39	1/2	3/4	15-20
5/8	7/8	52-57	5/8	7/8	21-27

E. Servicing the System with a Charging Stand

(PIR-PPS-50003-3, Rev. H)

CAUTION: MINERAL OIL AND PAG ARE NOT COMPATIBLE. USE A SEPARATE MANIFOLD TEST SET AND / OR TEST/CHARGING STAND AND RECOVERY SYSTEM FOR EACH REFRIGERANT TYPE.

CAUTION: USE RECOVERY UNIT SPECIFICALLY DESIGNED FOR THE TYPE OF REFRIGERANT USED IN THE AIRCRAFT SYSTEM. UNINTENDED EFFECTS MAY OCCUR IF REFRIGERANTS ARE COMBINED.

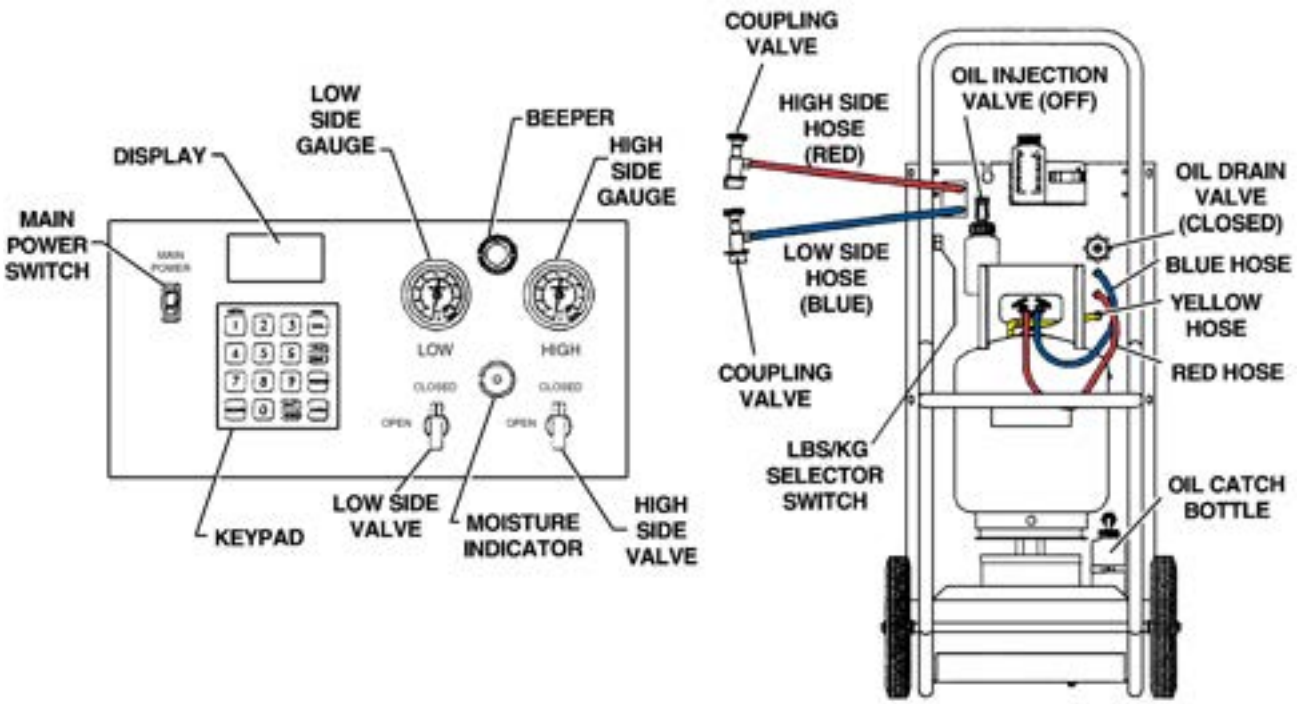
(1) Discharging (Bleeding/Purging) the System (with a Robinair 34700 or similar charging stand/ recovery station) (See “Figure 3” and “Figure 6” on page 215025). (Required only if system contains refrigerant.)

CAUTION: REFRIGERANT CAN CAUSE FREEZING OF SKIN. BE PARTICULARLY CAREFUL NOT TO ALLOW CONTACT WITH THE EYES.

CAUTION: DO NOT ALLOW REFRIGERANT TO ESCAPE TOO RAPIDLY, AS EXCESSIVE OIL MAY BE CARRIED OUT OF SYSTEM. WHEN HISSING STOPS, SYSTEM IS EMPTY AND VALVE SHOULD BE CLOSED IF NO FURTHER WORK IS PLANNED.

CAUTION: APPLIES TO ROBINAIR 34700 OR SIMILAR CHARGING/RECOVERY STATION. SEE OPERATOR’S MANUAL OF STATION BEING USED FOR DETAILED INSTRUCTIONS FOR DISCHARGING SYSTEM.

CAUTION: UNITED STATES ENVIRONMENTAL REGULATIONS PROHIBIT THE RELEASE OF REFRIGERANT INTO THE ATMOSPHERE. SPECIAL EQUIPMENT IS REQUIRED WHEN TESTING, DISCHARGING, OR CHARGING THE SYSTEM.



Robinair 34700 Charging Stand
Figure 3

NOTE: The term “Discharge,” as used throughout this section, in no sense implies or suggests discharging refrigerant to atmosphere. In all cases when discharging, an environmentally approved refrigerant recovery station is to be used.

(a) Gain access to service valves by removing rear closeout panel in cabin.

(b) Remove protective caps from service valves.

(c) Connect high side (red) hose to air conditioner high side service valve. On systems equipped with quick disconnect connections, open coupler valve.

(d) Connect low side (blue) hose to air conditioner low side service valve. On systems equipped with quick disconnect connections, open coupler valve.

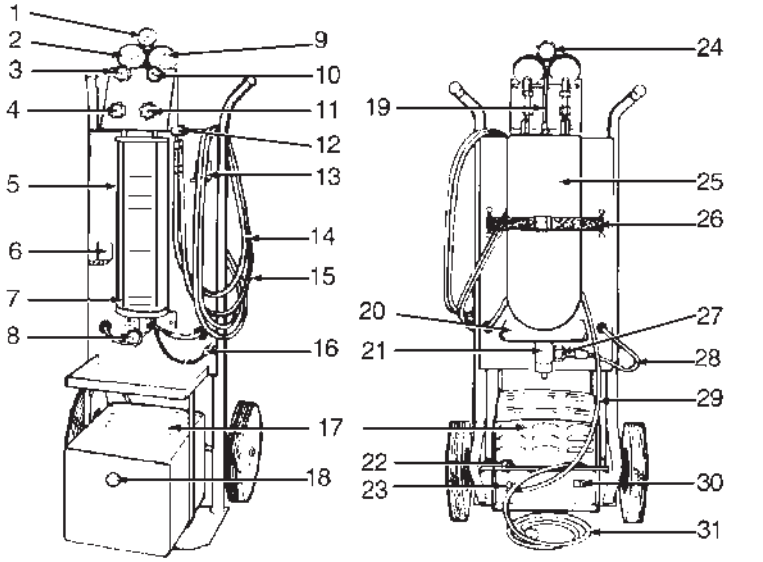
(e) Check the low side gauge (GAUGE 1) and high side gauge (GAUGE 2) to determine that there is pressure in the system. If there is no pressure, there is no refrigerant in the system to recover.

(f) Check that the oil drain valve is closed.

(g) Open both the low side and high side valves on control panel.

(h) Open the red GAS (vapor) valve and the blue LIQUID valve on the charging station’s refrigerant tank.

1. CYLINDER PRESSURE GAUGE
2. COMPOUND GAUGE
3. VALVE, LOW PRESSURE CONTROL
4. VALVE, VACUUM CONTROL
5. CHARGING CYLINDER
6. BRACKET
7. SIGHT GLASS
8. CYLINDER BASE VALVE
9. HIGH PRESSURE GAUGE
10. VALVE, HIGH PRESSURE CONTROL
11. VALVE, REFRIG. CONTROL
12. CHARGING LINE HOSE HOLDER
13. BRACKET
14. LOW PRESSURE CHARGING LINE
15. HIGH PRESSURE CHARGING LINE
16. HEATING ELEMENT PLUG
17. VACUUM PUMP
18. OIL FILL LOCATION
19. NECK ASSEMBLY
20. REFRIGERANT DRUM SUPPORT
21. REFRIGERANT DRUM VALVE
22. VACUUM PUMP VALVE
23. VACUUM PUMP EXHAUST PORT
24. TOP CYLINDER VALVE
25. REFRIGERANT DRUM



26. WEB STRAP
27. REFRIGERANT DRUM REDUCER
28. CHARGING CYLINDER HOSE
29. VACUUM PUMP INTAKE HOSE
30. VACUUM PUMP SWITCH
31. VACUUM PUMP POWER CORD

Kent Moore J23500 Charging Stand
Figure 4

(i) Slowly open the oil drain valve to see if system oil separator contains oil. If it does, let oil drain into the oil drain bottle (located at the bottom of the rear side of the charging station) until separator is empty.

(j) Close the oil drain valve. Dispose of collected oil in an environmentally accepted manner. Return collection bottle to its place on the charging stand.

(k) Plug unit into a proper voltage outlet. Turn MAIN POWER switch ON.

(l) Press the RECOVER key on charging station keypad.

(m) To assure complete recovery of refrigerant:

1) Wait 5 minutes. Observe pressure gauges for a rise above zero.

2) If a rise occurs, press the HOLD/CONT key.

3) Repeat as necessary until system maintains pressure for two minutes.

(n) Slowly open oil drain valve. Drain oil into the oil catch bottle. When all recovered oil has been completely drained, close oil drain valve.

NOTE: Drain oil separator after each job. Display will indicate OIL (OUNCES) or OIL (GRAMS) as a reminder.

(o) Measure the amount of oil in the catch bottle. The same amount of new oil must be added to the system before charging the system.

(p) To enter diagnostic mode, simultaneously press the SHIFT/RESET and ENTER keys. To display amount of refrigerant recovered by the unit, press the 3 key. The panel display will read the amount of recovered refrigerant in pounds or kilograms.

(q) Simultaneously press the SHIFT/RESET and ENTER keys to clear internal counter. Press SHIFT/RESET to return to the main menu.

(2) Leak Detection

See “Figure 2” on page 215010 and “Figure 3” on page 215017 .

When using a charging stand, a leak may be located as follows:

(a) Ensure that aircraft and/or ground power is OFF.

(b) Close all valves on the charging stand.

(c) Remove the protective caps from the high and low side service ports on the evaporator.

(d) Connect the blue and red hoses to the service ports as shown in “Figure 5” on page 215021.

(e) Proceed following the instructions in either paragraph (f) or (g), below.

(f) Using gaseous dry nitrogen:

1) Remove the pressure switch located on the evaporator assembly.

2) Connect a regulated (0-300 psig) gaseous dry nitrogen source to the pressure switch port on the evaporator assembly.

3) Slowly pressurize the system to 200 psig max. with nitrogen and turn off the nitrogen source.

4) Monitor pressure on the charging station gauge for 20 minutes. A leak free system will maintain the 200 psig pressure for 20 minutes.

5) If there is no pressure drop for 20 minutes, slowly release nitrogen pressure and disconnect the nitrogen source from the evaporator assembly. Re-install the pressure switch, lubricating the threads with Retro-fix CCI ESTER-25065 oil (P/N 197-511). Proceed to step (i).

6) If there is a pressure drop, find leak(s) by applying a soap solution to all connections.

7) Tighten/re-tighten fittings as necessary to stop leak(s). If leaks are due to damaged or worn components, proceed with refrigerant recovery/system discharge, perform repairs or component replacement and repeat leak detection procedure.

(g) Using HFC-134a refrigerant:

1) Ensure that there is at least one pound of refrigerant in the charging cylinder.

2) Open the high pressure control valve and the refrigerant control valve on the charging stand. Allow one pound of refrigerant to enter the system.

3) Close the high pressure control valve and the refrigerant control valve.

CAUTION: IT IS RECOMMENDED THAT A THICK SOLUTION OF SOAP AND WATER BE USED TO CHECK FOR LEAKS INSTEAD OF THE PROPANE LEAK DETECTOR THAT IS PROVIDED WITH SOME BRANDS OF CHARGING STANDS.

4) Locate leak(s) using an electronic leak detector designed to detect HFC134a refrigerant. Or, use soap and water in a thick solution.

5) If no leaks are found, proceed to step (h).

6) Tighten/re-tighten fittings as necessary to stop leak(s). If leaks are due to damaged or worn components, proceed with refrigerant recovery/system discharge, perform repairs or component replacement and repeat leak detection procedure.

(h) Recover remaining refrigerant from system using the Robinair 34700 (or other approved) charging station (See “Servicing the System with a Charging Stand” on page 215017 (1) Discharging the System). Any quantity of oil recovered from aircraft must be measured and an equal amount of new oil (i.e. - PAG with HFC134a) must be added to system before recharging.

(i) Evacuate the system.

CHART 4
SYSTEM VACUUM

	System Vacuum	Boiling Point of Water °F / °C
COMPOUND GAUGE READING IN INCHES OF MERCURY VACUUM	27.99	101 / 37.78
	28.89	80 / 26.67
	29.40	60 / 15.56
	29.71	40 / 4.44
	29.82	20 / -6.67
	29.88	0 / -17.78

(3) Evacuating the System

NOTE: Perform a Leak Detection check, above, before evacuating the system.

If the system has been operated in a discharged condition or anytime the system has been open to atmospheric pressure, the receiver-dehydrator must be replaced and the system evacuated to remove any trapped air and moisture which has entered it. A vacuum pump capable of pulling 29 inches of mercury or better should be used. As the pressure in the air conditioning system is lowered, the boiling temperature of the water (moisture) that may be present is also lowered. This then forces any moisture, in the form of water vapor, out of the system. Chart 4 demonstrates the effectiveness of moisture removal under a given vacuum.

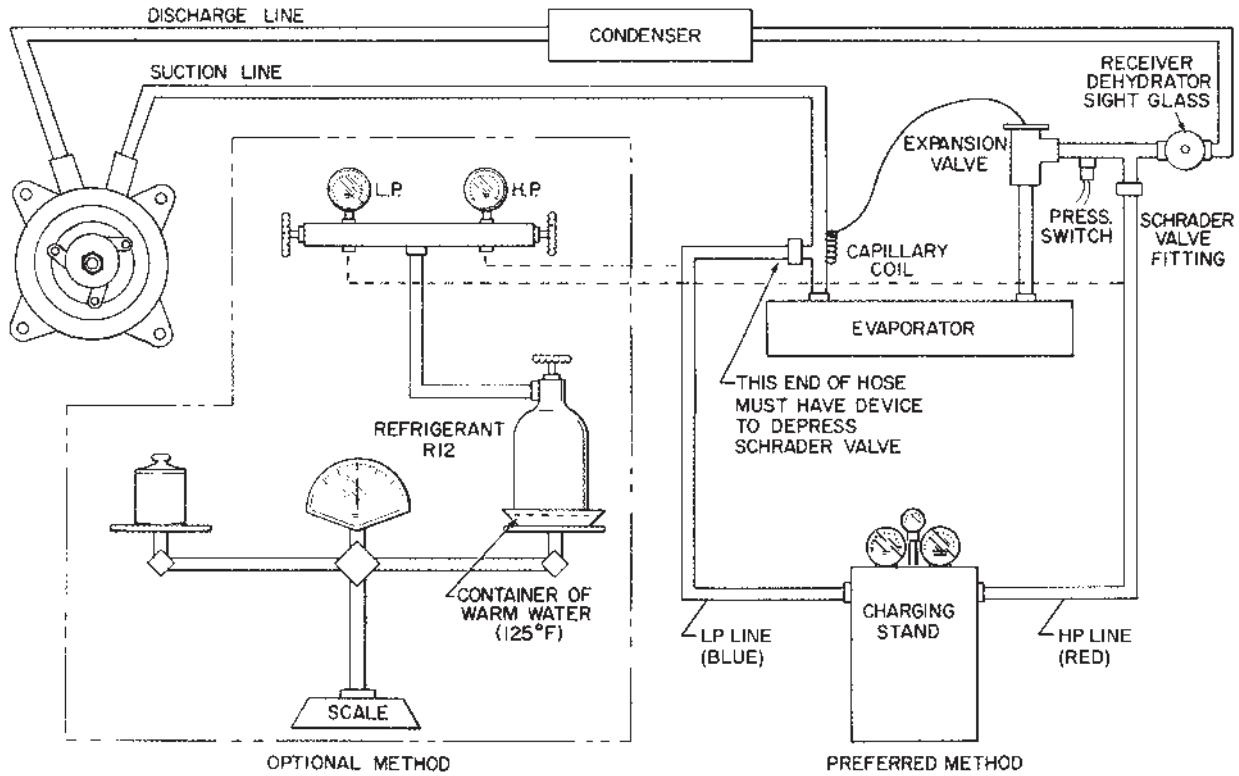
NOTE: Compound gauge reading will be approximately one inch lower, numerically, for each 1000 feet elevation above sea level.

(a) Using a Kent Moore J23500 or similar charging stand:

See “Figure 4” on page 215018.

1) Ensure that aircraft and/or ground power is OFF.
2) Close all valves on the charging stand.
3) Remove closeout panel at the rear of the cabin to gain access to the service valves.
4) Remove protective caps from the high and low side service ports on the evaporator.
5) Remove the protective cap from the vacuum pump outlet.

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Charging Hookup
Figure 5

6) Connect the blue and red hoses to the service ports.
7) Start the vacuum pump.
8) Open the valve on the vacuum pump. Open the low pressure control valve and the vacuum control valve on the charging stand.
9) After five minutes of pump operation, the high pressure gauge should indicate slightly below zero.
    a) If it doesn't, stop the pump and eliminate the blockage in the system replacing the faulty component, then repeat steps (1) - 9)).
    b) If it does, open the high pressure control valve on the charging stand and continue to evacuate the system.
10) Operate the vacuum pump for fifteen minutes, or until the compound gauge indicates 24 to 26 in. Hg. whichever occurs first.
11) Close the low pressure control valve and the high pressure control valve on the charging stand. Stop the vacuum pump and observe the compound gauge. If the gauge rises at a rate faster than 1 in. Hg. in 5 minutes, there is a leak in the system. Locate and fix the leak. Repeat the evacuation steps above.
12) Open the low pressure control valve and the high pressure control valve on the charging stand. Continue pumping and hold the system pressure below 26 in. Hg. for a minimum of 30 minutes. All the pumping time specified above may be included in the 30 minutes provided that no leaks or blockages are noted, and provided that the system is not opened by removal or disconnection of components.
13) Close the low pressure control valve, the high pressure control valve and the vacuum control valve. Stop the vacuum pump and perform the charging procedure immediately.

(b) Using a Robinair 34700 or similar charging/recovery stand: (See "Figure 3" on page 215017.)
1) Ensure that aircraft and/or ground power is OFF.
2) Close all valves on charging stand.
3) Remove closeout panel at rear of cabin to gain access to service valves.
4) Remove protective caps from the high and low side service ports on the evaporator.
5) Connect the blue and red hoses to the service ports (See "Figure 6" on page 215025), on systems equipped with quick disconnect connections, open coupler valves.
6) Open blue (low side) valve (1) on unit's control panel.
7) Open both the red GAS (vapor) valve and the blue LIQUID valve on the tank (See "Figure 6" on page 215025).
8) Program the length of evacuation time.
    a) Press the VACUUM key on control panel key pad.
    b) Display will show unit is in VACUUM mode.
    c) See operator's manual for further detail.
9) Enter the required time in minutes and seconds (30:00 minutes minimum) by pressing appropriate keys and then ENTER on keypad. The display will show selected time in minutes and seconds. Example: one hour and fifteen minutes (1:15) would be entered as 7500. The display will show 75:00. Thirty minutes is entered as 3000. The display will show 30:00.
10) To start the vacuum pump press the VACUUM key on keypad again.
11) Vacuum sequence will continue for the programmed time. Digital display will then show CPL, indicating that the evacuation is completed.

12) If, after 5 minutes of pump operation, the RED gauge does not indicate a little below zero:
    a) Stop the pump by pressing the “1” key or the SHFT/RESET key.
    b) Eliminate blockage in the system by replacing faulty parts.
    c) Repeat steps (1) - 12)).
13) If, after 5 minutes of pump operation, the RED gauge indicates a little below zero, open red (high side) valve (2), and continue evacuation.
    a) System vacuum (i.e. - low side gauge (GAUGE 1)) should attain 24 to 26 inches of mercury (in. Hg.) in 10 to 15 minutes.
    b) Allow pump to hold a vacuum of 26 in. Hg. (or below) for a minimum of 15 minutes.
    c) Failure to achieve or hold a vacuum of 26 in. Hg. (or below) in either (a) or (b), above, indicates a leak in the system. Locate leak as described in Leak Detection, above.
    d) Repair leak. Repeat steps (1) - 13)).
14) With the low side (1) and high side (2) valves OPEN, continue pumping, holding system below 26 in. Hg. for a minimum of 30 minutes.

    NOTE: All specified pumping times may be included in the 0:30 minutes, provided no blockage or leaks are noted, and provided the system is not opened by disconnecting or removing components.

15) When panel display reads CPL (complete), close both the low side valve (1) and the high side (2) valves.
16) Perform charging procedure immediately, see Charging the System, below.

(4) Charging the System

NOTE: Always evacuate the system (see above), before charging.

(a) Using a Kent Moore J23500 or similar charging stand.

See “Figure 4” on page 215018 and “Figure 5” on page 215021.

1) Open the valve at the base of the charging cylinder and fill the charging cylinder with sufficient refrigerant to charge the system. If refrigerant stops filling the cylinder, open the bleed valve at the top of the charging cylinder to relieve head pressure and allow refrigerant to continue filling the charging cylinder.
2) Close the bleed valve and the valve at the base of the charging cylinder.
3) Turn the charging cylinder sight glass to match the pressure reading on the charging cylinder pressure gauge. Keep the sight glass in this position during the remainder of the charging operation.
4) Connect the heating element plug to a 110 volt power outlet.
5) With the low pressure control valve (3) closed, open the refrigerant control valve (11) and the high pressure control valve (10).
6) Allow the correct amount of refrigerant to enter the high side of the system.
7) Close the high pressure control valve (10) and the refrigerant control valve (11).
8) Disconnect the hoses from the airplane’s system.

(b) Using a Robinair 34700 charging station or equivalent.

See “Figure 3” on page 215017 and “Figure 6” on page 215025.

CAUTION: THE FOLLOWING PROCEDURE APPLIES TO ROBINAIR 34700 OR SIMILAR CHARGING STATION. SEE OPERATOR’S MANUAL OF CHARGING STATION BEING USED, FOR DETAILED INSTRUCTIONS FOR CHARGING SYSTEM.

1) Check that main power switch and/or ground power is OFF.
2) Check that the LBS/KG selector switch on back of unit is in desired measurement mode. Be sure to turn OFF the main power switch before changing the measurement mode.

   NOTE: You may enter the amount of refrigerant to be charged when the unit is turned ON. The unit will store the amount in memory until it is turned off.

3) Remove protective caps from the high and low side service ports on the evaporator.
4) Connect the blue and red hoses to the service ports (See “Figure 6” on page 215025), on systems equipped with quick disconnect connections, open coupler valves.

   CAUTION: DO NOT PLACE ANY WEIGHT, INCLUDING HANDS AND/OR FEET, ON REFRIGERANT TANK OR SCALE DURING CHARGING PROCESS. ANY WEIGHT DISTURBANCE WILL CAUSE AN INDIRECT TRANSFER OF REFRIGERANT.

   CAUTION: ADD REFRIGERANT THROUGH THE LOW PRESSURE SIDE ONLY.

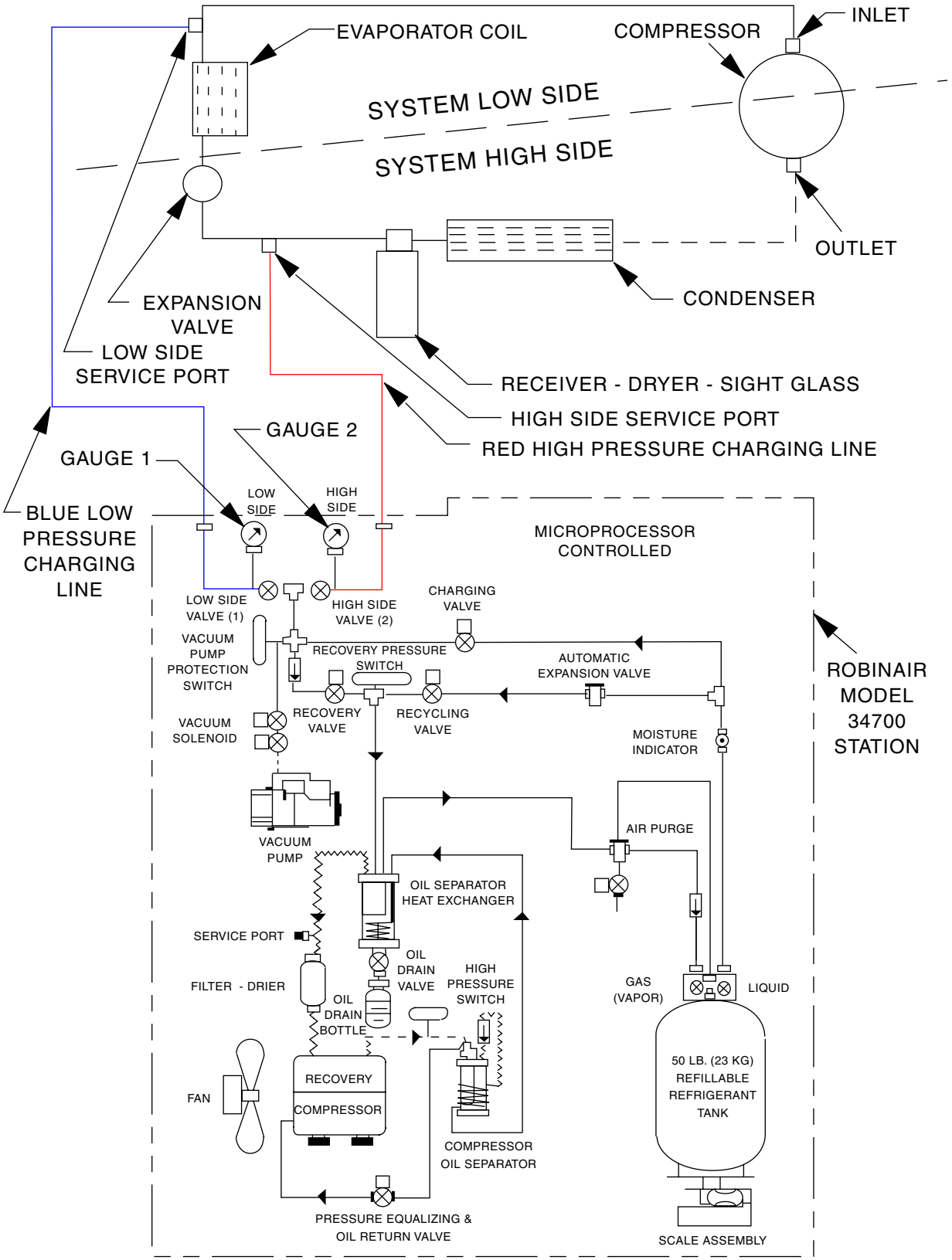
5) Open the low side (blue) valve on the unit’s control panel.
6) If the messages PROGRAM and CHARGE do not display, press the CHG key to enter the PROGRAM mode.

   NOTE: The amount of refrigerant required must be determined for each airplane. It is the amount that will result in bubble-free operation at the system sight gauge. The PA-34-220T requires approximately 2.25 LBS.

7) Enter amount of refrigerant required to charge the system by pressing the appropriate number keys and ENTER on keypad.

   NOTE: You may enter the amount of refrigerant to be charged when the unit is turned ON. The unit will store the amount in memory until it is turned off.

8) To begin charging process, press CHG key on keypad.
   a) The digital display will read AUTOMATIC and show the amount of refrigerant programmed for the charge.
   b) As the solenoid opens, it will make an audible sound.
   c) The display will count down to zero, and display message CPL, when charging is complete.
9) Close low side (blue) valve. Check that the high (red) valve is also closed. Also close coupler valves.
10) Perform Post Charging Operational Check, See “Post Charging Operational Check” on page 215026.



Robinair 34700 Charging Station Hose Hookup
Figure 6

(5) Post Charging Operational Check

See “Figure 3” on page 215017, “Figure 4” on page 215018, and “Figure 6” on page 215025 - for numbers in parentheses see “Figure 4” on page 215018; for numbers in brackets see “Figure 6” on page 215025.

(a) Ensure that the low pressure control valve (3) [1] and the high pressure control valve (10) [2] are closed. Hook up the charging stand to the system as shown in “Figure 5” on page 215021 and “Figure 6” on page 215025.

(b) With Robinair 34700 style stands equipped with quick disconnect couplings only, ensure coupler valves are open.

CAUTION: ASCERTAIN THAT THE AREA AROUND THE AIRPLANE IS CLEAR AND THAT A QUALIFIED PERSON IS AT THE CONTROLS OF THE AIRPLANE.

CAUTION: ENSURE THE AIRPLANE IS HEADED INTO THE WIND.

(c) Activate the system and operate the engine at 1,000 rpm for 2 minutes. Then operate the engine at 2,000 rpm for 2 minutes.

(d) Check the sight gauge on the receiver-dehydrator during the engine operation at 1,000 and 2,000 rpm. Any indication of bubbles passing the sight gauge indicates that additional refrigerant is required.

(e) If additional refrigerant is required, add it slowly through the refrigerant control valve (11) (Kent Moore J23500 style stands only) and the low pressure control valve (3) [1] until the sight glass remains free of bubbles. Regulate the flow of refrigerant with the low pressure control valve. Do not allow the compound gauge (2) or the low side gauge [1] to exceed a reading of 40 psi.

(f) With the engine operating at 1,000-1,500 rpm, the low and high side gauges should indicate as shown in “Chart 5”.

(g) With the charge properly established, stop the engine and, with Robinair 34700 style stands equipped with quick disconnect couplings only, close the coupler valves.

(h) Close the low pressure control valve (3) [1] and, with Kent Moore J23500 style stands only, the refrigerant control valve (11).

(i) Remove the charging stand. Replace all protective caps and covers.

CHART 5
AMBIENT TEMPERATURE INDICATION

GAUGE (PRESSURE)	AMBIENT TEMPERATURE	GAUGE INDICATION
Low Side	All	10 to 35 psig
High Side	Up thru 75° F	125 psig minimum to 175 psig maximum
High Side	Over 75° F	150 psig minimum to 300 psig maximum

F. Servicing the System with a Manifold Set

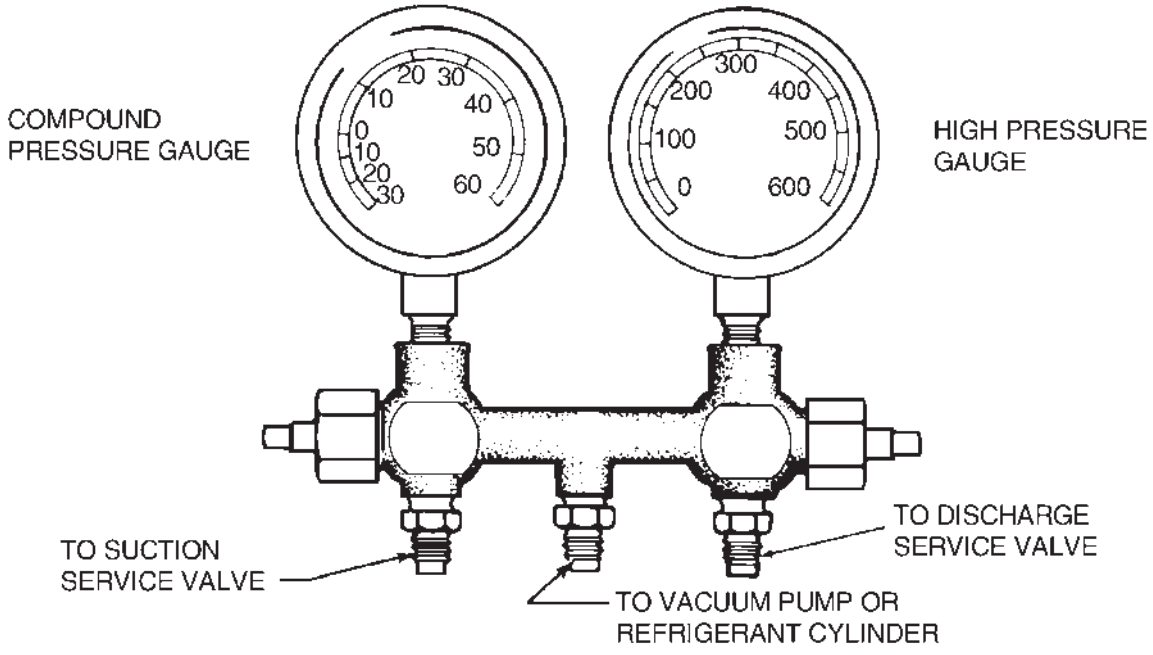
(1) Test Gauge and Manifold Set

The proper testing and diagnosis of the air conditioning system requires that a manifold gauge set (or a charging stand) be attached to the system. This set consists of two gauges mounted to a manifold. One gauge is a high pressure gauge used in the discharge side of the system. The other is a low pressure gauge used in the suction side of the system. The manifold is a device having fittings for both gauges and connection hoses with provisions for controlling the flow of refrigerant through the manifold. (See “Figure 7” and “Figure 8” on page 215028.)

The center port of the manifold set is used for charging or evacuation procedures, or any other service that may be necessary.

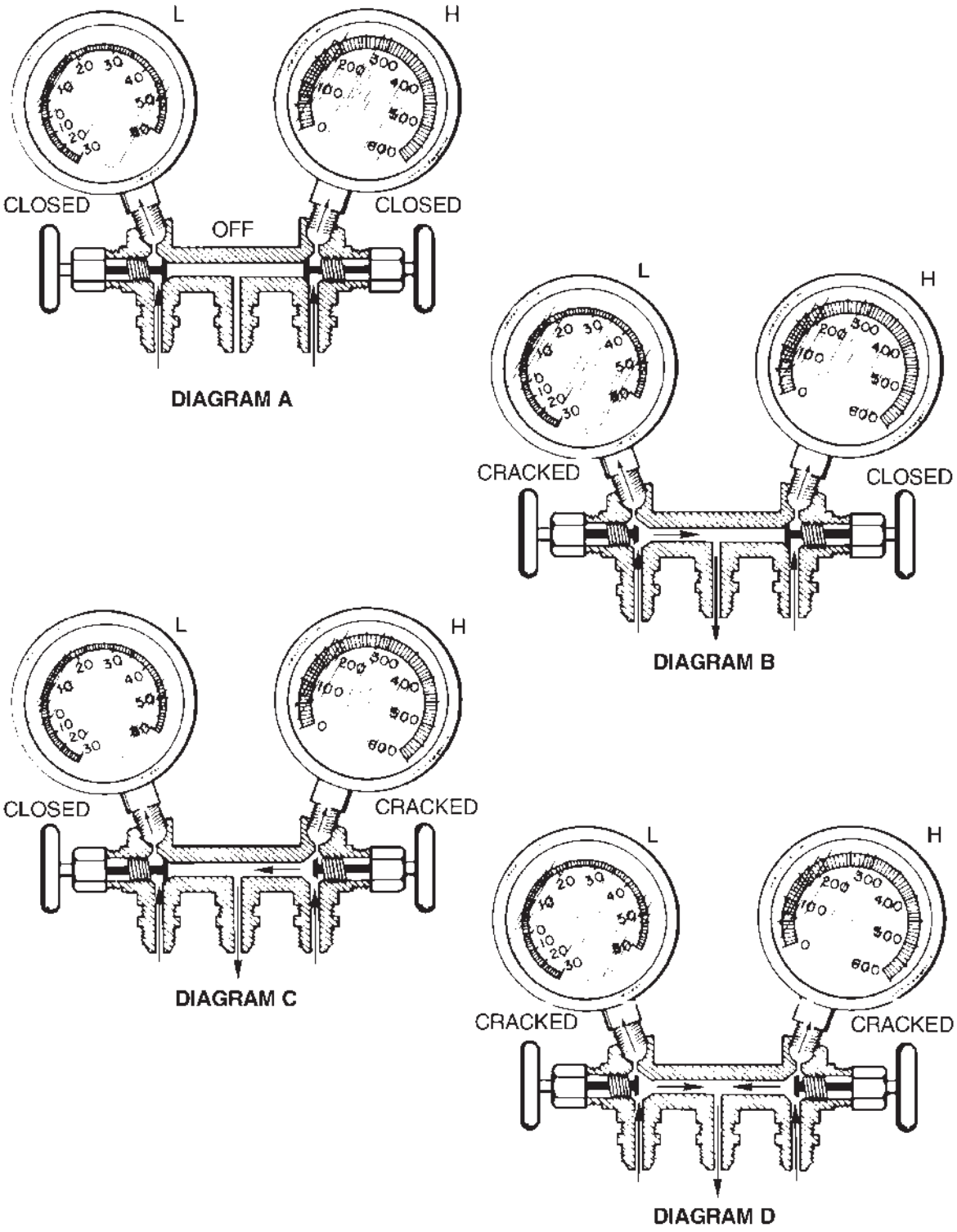
Both the high or low side of the manifold have hand shut-off valves. When the hand valve is turned all the way in, in a clockwise direction, the manifold is closed. The pressures on that side of the system will, however, be recorded on the gauge above the hose.

Cracking the hand valve, in the counterclockwise direction, opens the systems to the middle service port of the manifold set. This is desirable only when it is necessary to let refrigerant out of or into the system. (See “Figure 7” and “Figure 8” on page 215028.)



Test Gauge and Manifold Set
Figure 7

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Manifold Set Operation
Figure 8

(2) Leak Detection

See “Figure 9”.

(a) Close both the low side and high side valves on manifold hand set.

(b) Connect manifold hand set middle port (yellow) hose to a regulated (0-300 psig) gaseous dry nitrogen source or a container of HFC-134a refrigerant.

NOTE: Do Not use HFC-134a refrigerant in a Seneca IV.

(c) Open nitrogen source or refrigerant container service valve.

(d) Open the manifold hand set high side valve until a pressure of 50 psig is reached on low side gauge. Close high side valve.

(e) Locate leak(s) using soap and water in a thick solution; or, if using HFC-134a, an electronic leak detector designed to detect HFC134a refrigerant can also be used.

(f) Tighten/re-tighten fittings as necessary to stop leak(s). If leaks are due to damaged or worn components, proceed with refrigerant recovery/system discharge, perform repairs or component replacement and repeat leak detection procedure.

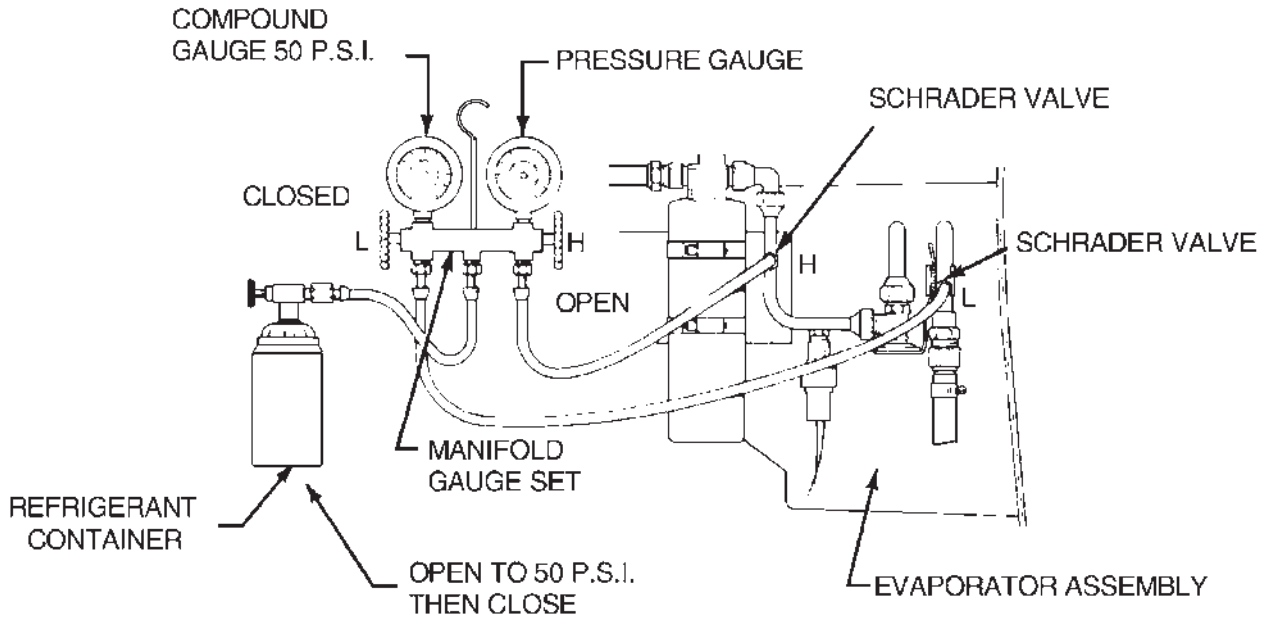
(g) Check that both the high side and low side valves on the manifold hand set are closed.

(h) Close service valve on nitrogen source or refrigerant container. Disconnect yellow manifold hand set center hose from nitrogen source or refrigerant container.

(i) On systems equipped with quick disconnect connections, close coupler valves. Disconnect manifold hand set red and blue hoses from airplane service ports. Remove manifold hand set.

(j) If HFC-134a refrigerant was used, recover remaining refrigerant from system using the Robinair 34700 (or other approved) charging station (see Servicing the System, Charging, above). Any quantity of oil recovered from aircraft must be measured and an equal amount of new oil (i.e. - PAG with HFC134a) must be added to system before recharging.

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Leak Test Hookup
Figure 9

(k) When refrigerant recovery is complete, on systems equipped with quick disconnect connections, close coupler valves. Disconnect charging/test station from service ports.

(l) Evacuate the system, see below or Servicing the System, Evacuating, above.

(m) Immediately charge the system, see below or Servicing the System, Charging, above.

(3) Evacuating the System

See “Figure 10”.

NOTE: Perform a Leak Detection check, above, before evacuating the system.

If the system has been operated in a discharged condition or anytime the system has been open to atmospheric pressure, the receiver-dehydrator must be replaced and the system evacuated to remove any trapped air and moisture which has entered it. A vacuum pump capable of pulling 29 inches of mercury or better should be used. As the pressure in the air conditioning system is lowered, the boiling temperature of the water (moisture) that may be present is also lowered. This then forces any moisture, in the form of water vapor, out of the system. Chart 4 demonstrates the effectiveness of moisture removal under a given vacuum.

(a) Ascertain that all system pressure is released.

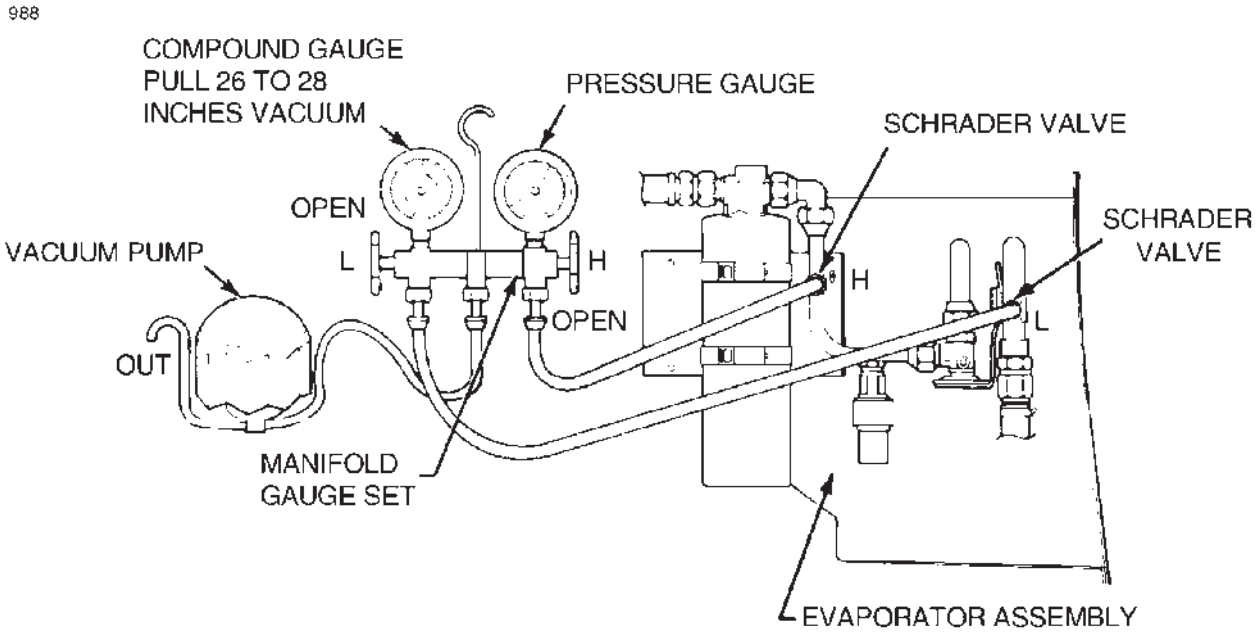
(b) Connect the manifold set hoses to the service ports and vacuum pump as shown in “Figure 10”.

(c) Close the high side (pressure) and low side (suction) hand valves on the manifold set.

(d) Start the vacuum pump.

(e) Open the low side manifold set hand valve. The low side gauge should show a vacuum.

(f) After five minutes of pump operation the high side gauge should indicate slightly below zero. If it does not, stop the pump and eliminate the blockage in the system by replacing the faulty component, then repeat the previous evacuation steps.



Evacuation Hookup
Figure 10

(g) Operate the vacuum pump for fifteen minutes or until the low side gauge indicates 24 to 26 in. Hg. whichever occurs first.

(h) Close the low side hand valve, stop the vacuum pump and observe the low side gauge. If the gauge rises at a rate faster than 1 in. Hg. in 5 minutes, there is a leak in the system. Locate and repair the leak, then repeat the previous evacuation steps.

(i) With both the low and high side valves open, continue the pumping and hold the system below 26 in. Hg. for a minimum of 30 minutes. All the previous pumping time may be included in the 30 minutes provided that no leaks or blockages are noted, and provided that the system is not opened by removal or disconnection of components.

(j) Close the low and high side hand valves, stop the vacuum pump and perform the charging procedure immediately.

(4) Charging the System Using the Airplane Compressor

This method is the least desirable due to the requirement of operating the airplane's engine to run the compressor.

CAUTION: ASCERTAIN THAT THE AREA AROUND THE AIRPLANE IS CLEAR AND THAT A QUALIFIED PERSON IS AT THE CONTROLS OF THE AIRPLANE.

(a) Keep the system under the vacuum established during the evacuating procedure with both hand valves in the closed position.

(b) Attach a container of the appropriate refrigerant to the manifold set and open the container service valve.

(c) Loosen the center hose at the manifold set until a hiss can be heard. Allow the gas to escape for 2 to 3 seconds, then tighten the connection.

(d) Open the high side manifold set hand valve, observe the low side gauge, then close the high side hand valve. The low side gauge should immediately change from an indication of a vacuum to an indication of pressure. If it does not, the system is blocked, and the blockage must be corrected before proceeding.

(e) Start the engine and operate it at 1000 rpm.

(f) Adjust the airplane air conditioning controls for maximum cooling, high blower speed.

(g) Keep the refrigerant cylinder in an upright position. A slug of liquid refrigerant entering the system would damage the compressor.

(h) Open the low side manifold set hand valve and allow two pounds of refrigerant in the gas state to enter the system.

(i) Close the low side manifold set hand valve.

(j) Proceed with the Post Charging Operational Check, below.

(5) Post Charging Operational Check

CAUTION: ASCERTAIN THAT THE AREA AROUND THE AIRPLANE IS CLEAR AND THAT A QUALIFIED PERSON IS AT THE CONTROLS OF THE AIRPLANE.

NOTE: Head the airplane into the wind during these checks.

(a) With the manifold set installed, and both hand valves closed, actuate the system and operate the engine at 1,000 rpm for 2 minutes, then operate the system at 2,000 rpm for 2 minutes.

(b) Check the system sight gauge (on the receiver-dehydrator) during operation at 1,000 and 2,000 rpm. Any indication of bubbles passing through the sight gauge indicates that additional refrigerant is required.

(c) Add additional refrigerant slowly through the low side manifold set hand valve until the sight glass remains free of bubbles.

(d) Close the low side hand valve and refrigerant container valve.

CHART 6
AMBIENT TEMPERATURE INDICATION

GAUGE	AMBIENT TEMPERATURE	GAUGE INDICATION
Low Pressure	All	10 to 35 psig
High Pressure	Up thru 75°F	125 psig min to 175 psig max
High Pressure	Over 75°F	150 psig min to 275 psig max

(e) With the engine operating at 1,000 to 1,500 rpm, the gauges should indicate as shown in "Chart 6".

(f) Once the charge is properly established, stop the engine, close the refrigerant container service valve. Remove the manifold set and replace all protective caps and covers.

(6) Adding Partial Charge to System

The system can be topped off with refrigerant by the following method:

(a) Remove the closeout panel at the rear of the cabin.

(b) Connect a charging hose to a refrigerant cylinder and also to the low pressure Schrader valve fitting on the manifold assembly.

(c) Purge the charging hose by allowing a small amount of refrigerant gas to escape at the Schrader valve fitting.

CAUTION: ASCERTAIN THAT THE AREA AROUND THE AIRPLANE IS CLEAR AND THAT A QUALIFIED PERSON IS AT THE CONTROLS OF THE AIRPLANE.

NOTE: Head the airplane into the wind during this procedure.

(d) Start the engine, operate at 1000 rpm and turn the air conditioner on maximum cool.

(e) Remove the plastic plug (if installed) from the sight glass in top of the receiver-dehydrator.

(f) With a low refrigerant charge in the system, bubbles will be seen passing through the sight glass when the system is operating.

(g) Open the valve on the refrigerant cylinder.

(h) Allow refrigerant to flow into the system until the bubbles disappear from the sight glass.

(i) Close the refrigerant valve and check to see that the sight glass remains clear during system operation.

(j) When the sight glass stays clear of bubbles, add an additional pound of refrigerant to the system. (Engine should be operating at 1,000 rpm.)

NOTE: This is done with OAT at 70°F, or higher, with the air conditioner operating.

(k) Shut off the air conditioner and engine. Remove the charging hose from the Schrader valve with care due to refrigerant remaining in the line.

(l) Reinstall closeout panel.

G. Compressor

See “Figure 13” on page 215037.

The engine driven compressor is mounted on the rear of the left engine. A V-belt connected to the accessory drive adapter pulley drives the compressor through a magnetic clutch.

(1) Servicing

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

CAUTION: AN IMPORTANT FACTOR IN AIR CONDITIONING SERVICING IS CLEANLINESS. TAKE CARE TO PREVENT DIRT OR FOREIGN MATERIAL FROM ENTERING THE SYSTEM. ALL HOSE AND TUBING ENDS SHOULD BE CAPPED IMMEDIATELY. ANY LUBRICATION REQUIRED IN THE ASSEMBLY OF THE COMPONENTS SHOULD BE REFRIGERANT OIL OF THE TYPE USED IN THE COMPRESSOR.

Servicing the compressor in the field is not recommended. Compressor service should be done by a qualified shop which has the special equipment and trained personnel required to properly service the unit.

Field maintenance of the Sanden compressor is limited to removal and installation, checking the compressor oil level, and replacement of worn drive belts. Contact Sanden International, for special tools and instructions for detailed compressor maintenance.

(2) Removal

(a) Be sure the air conditioning circuit protector is in the off position.

(b) Remove engine cowling.

(c) Disconnect the electrical leads to the magnetic clutch on the compressor.

(d) Discharge the air conditioning system to an appropriate environmentally approved refrigerant recovery station. See Servicing the System, Discharging, above.

(e) Remove the suction and discharge lines from the connections on the compressor.

NOTE: All open lines should be capped immediately to prevent dirt and moisture from entering the system.

(f) Loosen compressor mounting bolts (4) to release belt tension and remove belt from pulley. (Do not force belt over pulleys.)

(g) Support compressor and remove bolts, washers, and nuts securing compressor to forward and rear mounting brackets.

(h) Remove compressor.

CHART 7
SANDEN COMPRESSOR OIL LEVEL VS. MOUNTING ANGLE

Mounting Angle	0°	10°	20°	30°	40°	50°	60°	90°
Oil Level (in notches)	3-5	5-7	6-8	7-9	8-10	8-10	9-11	9-11

(3) Installation

NOTE: If installing a new compressor, drain new compressor of all OEM oils by removing oil plug and draining oil reservoir. Turn compressor on end so that oil may drain out through suction and discharge ports. Rotate compressor pistons several times to make sure residual oils are drained. Fill compressor with 3.4 fl. oz. of Piper compressor oil (197-511). Ensure that the seat and O-ring is clean and wet with oil. Reinstall the oil filler plug, first checking that the sealing O-ring is not twisted. Torque the plug from 6 to 9 foot-pounds (0.8 to 1.2 kg-m). Do not overtighten the plug to stop a leak. If plug leaks, remove it, and install a new O-ring.

(a) Align compressor mounting lugs with forward and rear mounting brackets and install bolts, washers, and nuts hand tight.

(b) Place drive belt over compressor clutch and accessory drive adapter pulleys. Ensuring drive belt is properly seated in both pulleys, rotate compressor in mounting brackets slots to obtain a drive belt tension of 45 to 50 lbs. Torque the four (4) mounting bolts 240 to 260 in.-lbs.

(c) Check oil level in compressor as described in Checking Compressor Oil, below.

(d) Connect discharge and suction lines to their respective fittings on the compressor.

(e) Evacuate the system per Servicing the System, Evacuating, above.

(f) Charge the system per Servicing the System, Charging, above.

(g) Install engine cowling.

(4) Checking Compressor Oil Level

Whenever a system component has been replaced or there is an obvious leak, use the following procedure to check the compressor oil level (after making necessary repairs):

WARNING: ENSURE THE AREA AROUND THE AIRCRAFT IS CLEAN AND FREE OF LOOSE OBJECTS, BEFORE OPERATING THE ENGINE AND AIR CONDITIONER ON THE GROUND.

(a) Run compressor for 10 minutes at engine idle rpm.

(b) Recover all refrigerant from the system. Be careful not to lose oil.

(c) Determine the compressor mounting angle by positioning the angle gauge (Sanden P/N 32448) or a propeller protractor across the flat surfaces of the two front mounting ears.

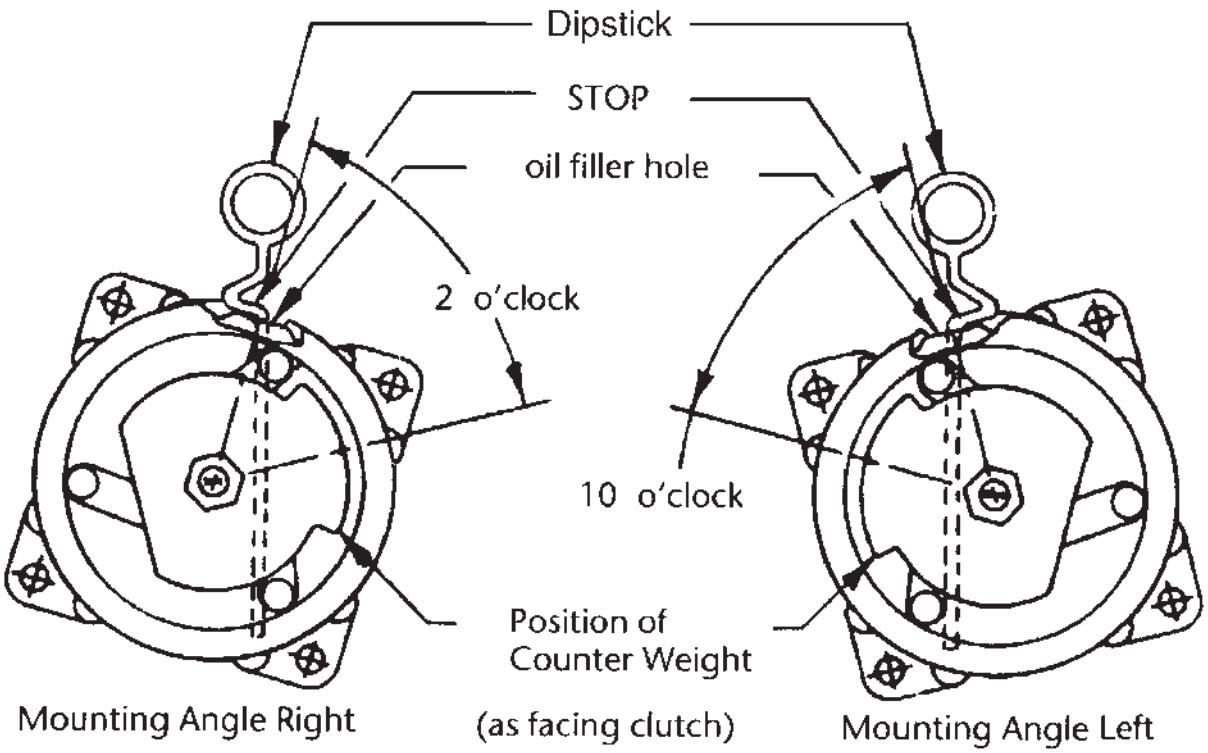
(d) Center the bubble and read the mounting angle to the closest degree.

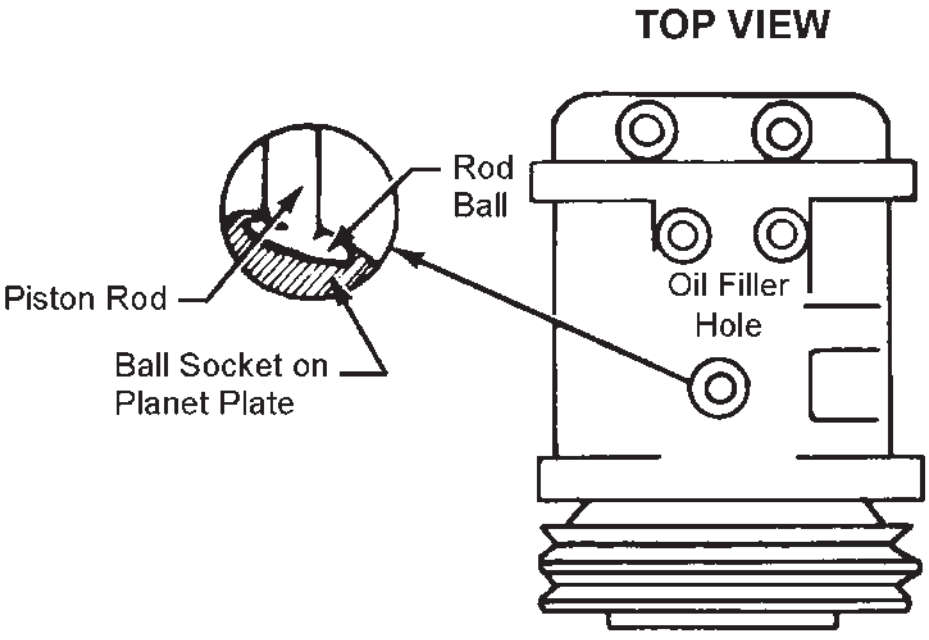
CAUTION: DO NOT REMOVE THE OIL FILLER PLUG WITH PRESSURE IN THE SYSTEM.

(e) Remove the oil filler plug.

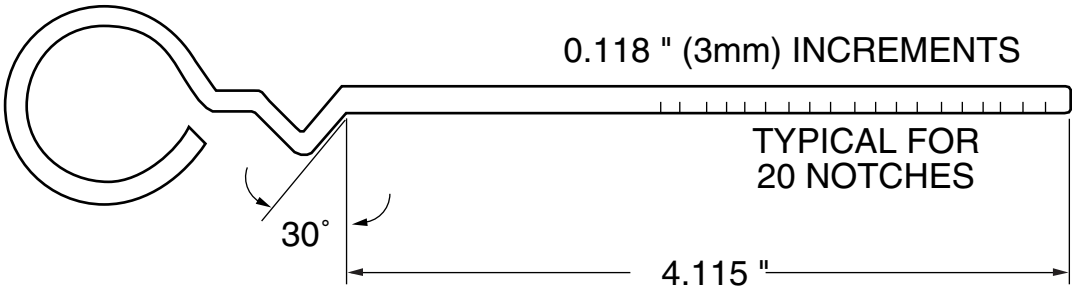
(f) Look through the oil filler plug hole and rotate the clutch front plate to position the internal parts as shown in “Figure 11” on page 215035. Center the parts as they are moving to the rear of the compressor (discharge stroke).

NOTE: This step is necessary to clear the dipstick of internal parts and to allow its insertion to full depth.





Checking Compressor Oil Level
Figure 11



Fabricated Dipstick
Figure 12

(g) Insert the dipstick (“Figure 12”) to its stop position (See “Figure 11” on page 215035). The stop is the angle near the top of the dipstick.

1) The point of the angle must be to the left if the mounting angle is to the right, or to the right if the mounting angle is to the left.

2) The bottom surface of the angle must be flush with the surface of the oil filler hole.

(h) Remove the dipstick and count the increments of oil.

(i) Use Chart 7 to determine the correct oil level for the mounting angle of the compressor.

(j) If the increments read on the dipstick do not match the table, add or subtract the appropriate oil to the mid-range value - i.e. - if the angle is 20°, the desired oil level is 7.

(k) Install the oil filler plug, first checking that the sealing O-ring is not twisted. Ensure that the seat and O-ring are clean.

(l) Torque the plug from 6 to 9 foot-pounds (0.8 to 1.2 kg-m). Do not overtighten the plug to stop a leak. If plug leaks, remove it, and install a new O-ring.

(m) Evacuate and charge the system as described above.

(5) Adjusting Drive Belt Tension

(a) Remove left engine cowling.

(b) Loosen compressor mounting bolts (4) rotate compressor in mounting brackets slots to obtain a drive belt tension of 45 to 50 lbs. Torque the four (4) mounting bolts 240 to 260 in.-lbs.

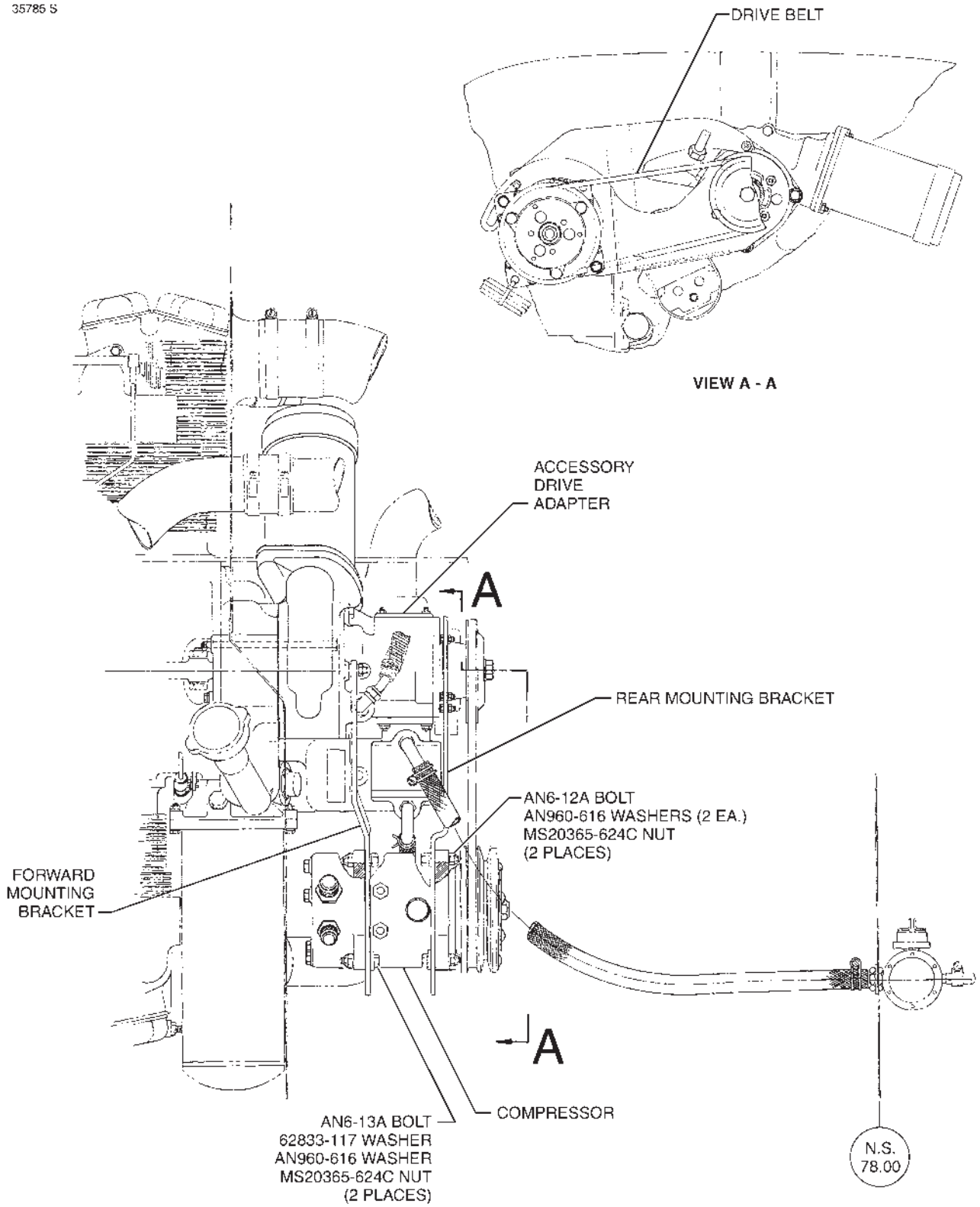
WARNING: ENSURE THE AREA AROUND THE AIRCRAFT IS CLEAN AND FREE OF LOOSE OBJECTS, BEFORE OPERATING THE ENGINE AND AIR CONDITIONER ON THE GROUND.

(c) Run the engine for 20 minutes at 1900 rpm with the compressor engaged.

(d) Shut down engine and recheck belt tension. Tension should hold at 45 to 50 lbs.

(e) Check tension every 100 hours or annual inspection, whichever comes first.

(f) Install engine cowling.



Compressor Installation
Figure 13

H. Refrigerant Lines and Routing

CAUTION: DISCHARGE SYSTEM COMPLETELY BEFORE HOSE COUPLINGS ARE UNCOUPLED. (SEE SERVICING THE SYSTEM, DISCHARGING, ABOVE.)

NOTE: United States environmental regulations prohibit the release of refrigerant into the atmosphere. Special equipment is required when discharging or recharging system.

Handle refrigerant lines carefully. Refrigerant lines are flexible high pressure hoses. Hoses in power plant area are routed for maximum protection from heat and abrasion. They couple at firewall to hoses routed through the two inboard, external hat sections on bottom of fuselage, up through floor to condenser and evaporator in tail cone. Discharge is in the right hand hat section. The suction is in the left hand hat section.

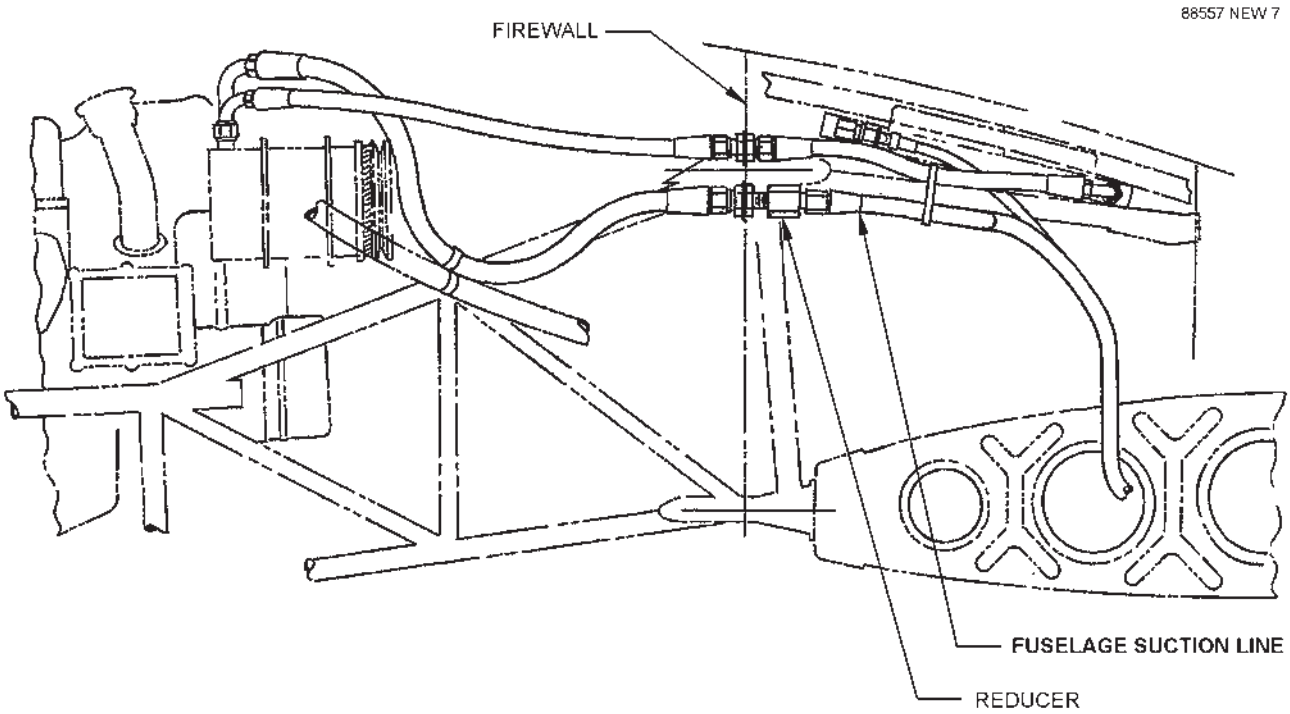
Repairs

In Seneca IVs only:

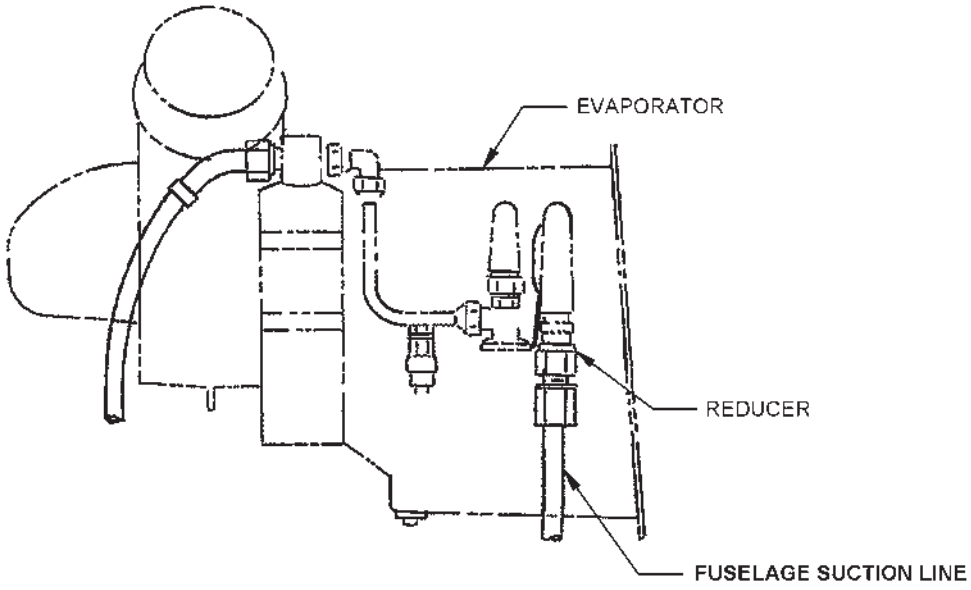
(a) If the fuselage suction line is damaged or leaking at the forward hose fitting attached at the firewall (see “Figure 14” on page 215039); order and install Piper Service Kit No. 88557-001.

(b) If the fuselage suction line is damaged or leaking at the aft hose fitting attached to the evaporator (see “Figure 14” on page 215039); order and install Piper Service Kit No. 88557-002.

(c) If the fuselage suction line is damaged or leaking from a middle section of hose at least eight (8) inches from the nearest end fitting; order and install Piper Service Kit No. 88563-001.



LOOKING INBOARD AT LEFT NACELLE



LOOKING OUTBOARD FROM CENTERLINE

Suction Line
Figure 14

I. Receiver-Dehydrator

(1) Removal

CAUTION: IF RECEIVER-DEHYDRATOR IS NOT SERVICEABLE, IT MUST BE REPLACED. RECEIVER-DEHYDRATOR MUST BE REPLACED WHEN SYSTEM HAS OPERATED WITHOUT A CHARGE OR HAS BEEN LEFT OPEN.

The unit is mounted on inboard side of evaporator assembly housing.

(a) Discharge system of all refrigerant. See “System Servicing Precautions” on page 215015 and “Evacuating the System” on page 215021.

(b) Uncouple refrigerant lines at receiver-dehydrator. (Follow all WARNINGS and CAUTIONS under “Special Servicing Procedures” on page 215014.)

(c) Remove clamp attaching unit to evaporator housing.

(2) Installation

CAUTION: A NEW RECEIVER-DEHYDRATOR SHOULD BE OPENED AND CONNECTED TO THE SYSTEM ONLY WHEN READY TO CHARGE THE SYSTEM WITH REFRIGERANT.

On systems utilizing HFC 134a refrigerant, use only receiver-dehydrators marked with a GREEN arrow.

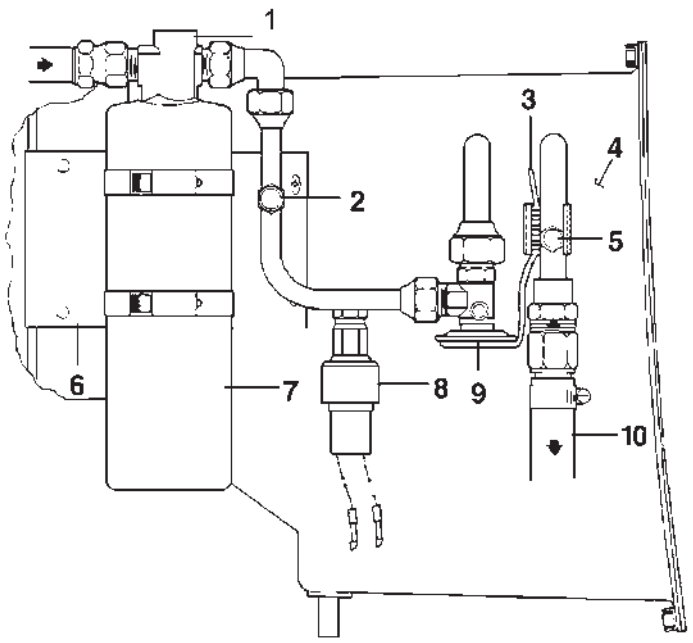
(a) Slip mounting bracket around receiver and put it in place on evaporator housing with tube fitting on top. Align fittings to proper line before securing mounting bracket.

(b) Replace O-rings on HFC-134a systems.

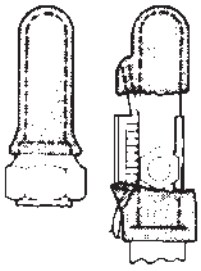
(c) Tighten fittings to torque listed in “Chart 3” on page 215016.

(d) Evacuate and charge system per “System Servicing Precautions” on page 215015 and “Evacuating the System” on page 215021. and “Charging the System” on page 215023.

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WRAP TAPE AROUND THERMOSTAT CAPILLARY LEAVING SERVICE PORT ACCESSIBLE.

1. SIGHT GLASS
2. SERVICE VALVE (SCHRADER) (HI)
3. CAPILLARY COIL
4. HOUSING ASSEMBLY
5. SERVICE VALVE (SCHRADER) (LOW)
6. RECEIVER DEHYDRATOR CLAMP
7. RECEIVER DEHYDRATOR
8. PRESSURE RELIEF SWITCH (RANCO)
9. EXPANSION VALVE
10. OUTLET HOSE

Components Installation
Figure 15

J. Condenser

The condenser is located in the left nacelle aft of the firewall, between stations 78.00 and 98.00.

(1) Removal

(a) Remove the hatch cover assembly.

CAUTION: CAP THE OPEN LINES TO PREVENT MOISTURE AND DUST FROM CONTAMINATING THE SYSTEM.

(b) With the system completely discharged, disconnect the suction and discharge hoses at the condenser fitting. See “Special Servicing Procedures” on page 215014.

(c) Remove the screws which hold the condenser to the mounting brackets.

(d) Remove the condenser from the nacelle, being careful not to bend the fins of the core or damage connecting tube.

(2) Installation

(a) Place the condenser in the left nacelle with the line connections on the outboard side.

(b) Attach the condenser to the mounting brackets.

(c) Seal and couple the hose fittings. On Seneca IV, apply a small amount of Loctite refrigerant sealant to the flare only to ensure leak free connections. On Seneca V, replace o-rings (o-rings must be suitable for use with HFC134a).

(d) With the condenser secured, proceed to evacuate and recharge the system per previous instructions.

(e) When the system is completely charged, check it for any leaks.

(f) Replace and secure hatch cover assembly.

K. Condenser Air Scoop Rigging

See “Figure 16” on page 215043.

The condenser scoop assembly in the left nacelle, is operated by an electric motor through a torque tube and pushrod. Travel of the scoop is limited by switches, activated by a rotating cam plate. The scoop should be rigged to the proper specifications for the most efficient cooling results. Proceed as follows:

(1) Open scoop door to ground open position. Hand rotate torque tube assembly to ground open position; lobe straight up. Insert bolt to connect scoop door and rod assembly.

(2) Move door to closed position. With door in closed position, lobe will be straight down. If snug fit is not obtained adjust rod end until a snug fit is obtained.

(3) Complete installation of attachment hardware at rod to door.

(4) The flight open position, is obtained by adjusting the micro switch and not by a mechanical adjustment.

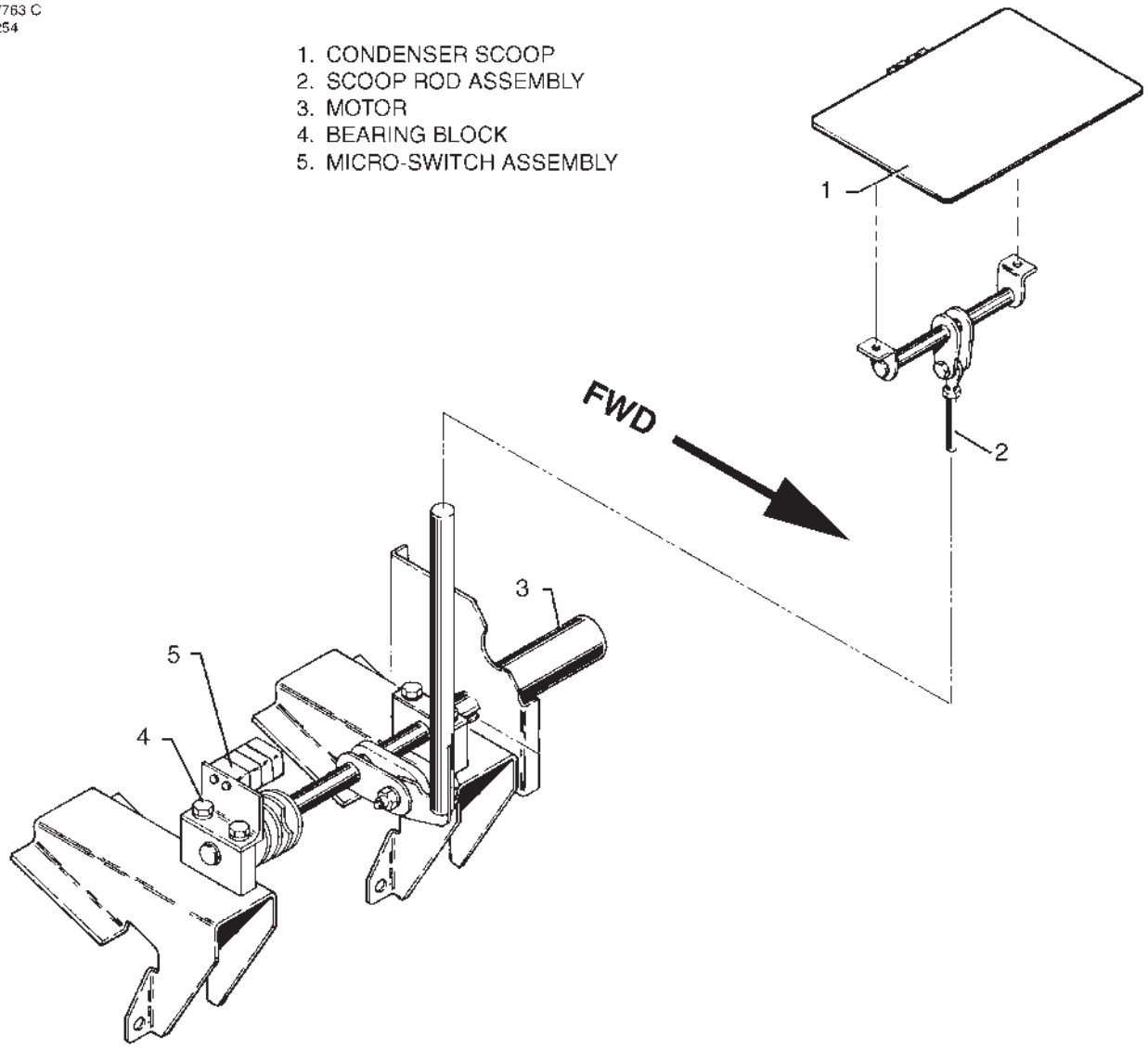
(5) Move door to flight position until rod assembly contacts torque tube. Back rod assembly away from torque tube .06 to .08 inches and position microswitch until audible “click” is heard (no continuity, C to NC wires) and secure microswitch in place.

(6) Move door to closed position and verify snug fit. Adjust the two micro switches relative to the cam, so that both micro switches are relax. (No continuity, connector pin 2 to pin 1 and 4.)Only during approximately 5° angular travel at the closed position of the crank. Tighten the two screws to maintain this adjustment.

(7) Move door to ground open position and verify ground open position dimension. Position micro switch until audible “click” is heard (no continuity, C to NC wires) and secure in place.

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1. CONDENSER SCOOP
2. SCOOP ROD ASSEMBLY
3. MOTOR
4. BEARING BLOCK
5. MICRO-SWITCH ASSEMBLY



Condenser Air Scoop Installation
Figure 16

(8) Place aircraft on jacks, activate circuit breaker marked “Air Condition,” cycle scoop door to verify each position.

NOTE: With reference to “Figure 17” on page 215045, the door should be open 0.8 + 0, -0.10 inch for the INFLIGHT condition and 3.50 minimum inch for the GROUND position. Do not bend actuating pushrod.

(9) Over ride the squat switch on the right main gear by depressing micro switch lever, when inflight position is obtained ensure that the motor stops before rod assembly locks against torque tube.

(10) Reinstall hatch cover assembly and remove aircraft from jacks.

L. Expansion Valve

(1) Removal

See “Figure 18” on page 215046.

The expansion valve is in evaporator assembly between reciver dehydrator and evaporater inlet. The capillary coil is attached to evaporator outlet line.

NOTE: If expansion valve is not serviceable, it must be replaced with a new part.

(a) Remove access panels, and discharge system. See “System Servicing Precautions” on page 215015 and “Evacuating the System” on page 215021.

(b) Remove capillary coil from outlet line. (Do not kink capillary tube.)

(c) Uncouple all related tube fittings. See “Special Servicing Procedures” on page 215014.

(2) Installation

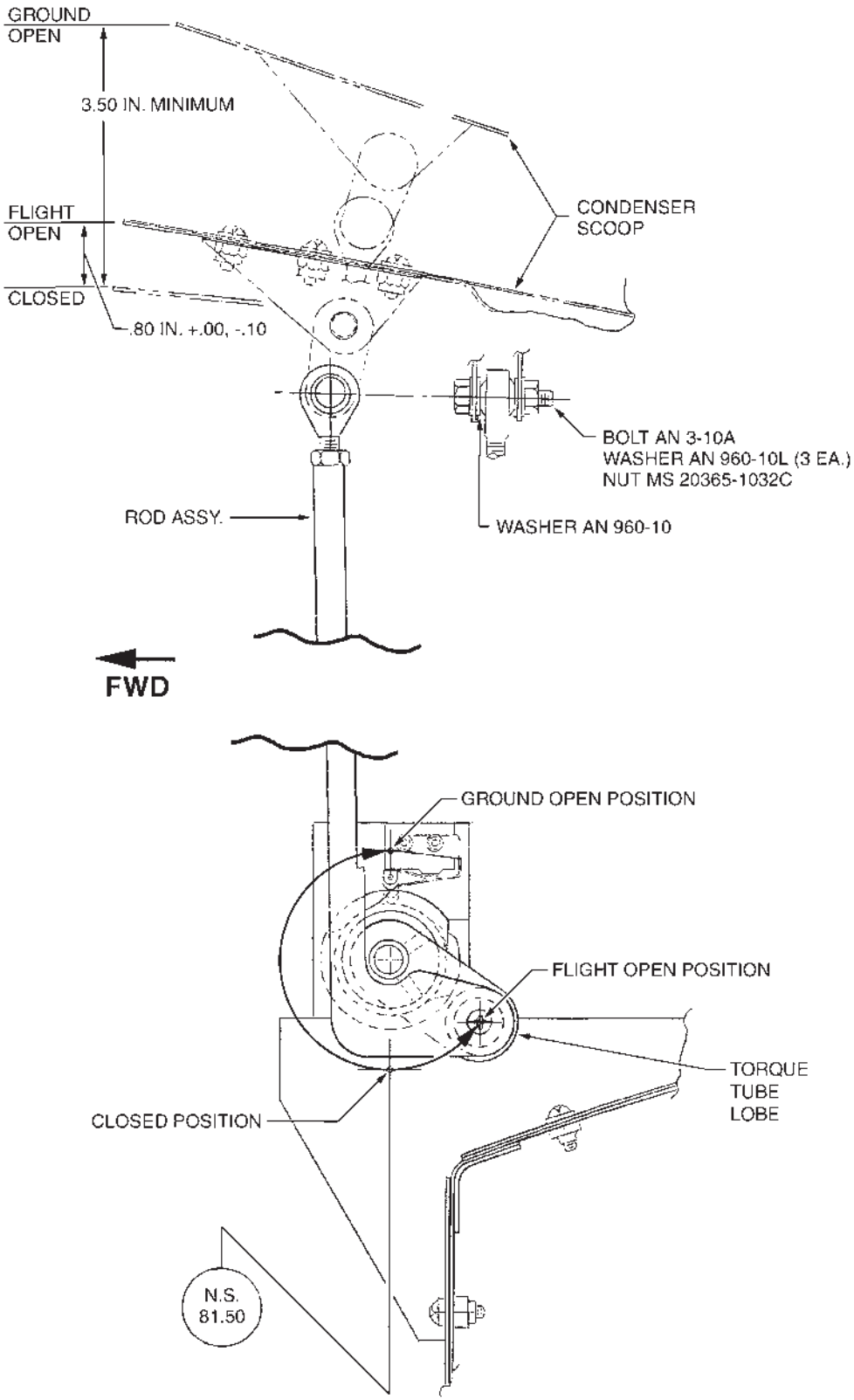
(a) Install expansion valve in inlet line of evaporator core. Apply appropriate lubricant on O-rings and replace O-rings on fittings, torque fittings per “Chart 3” on page 215016.

(b) Secure capillary coil to evaporator outlet line.

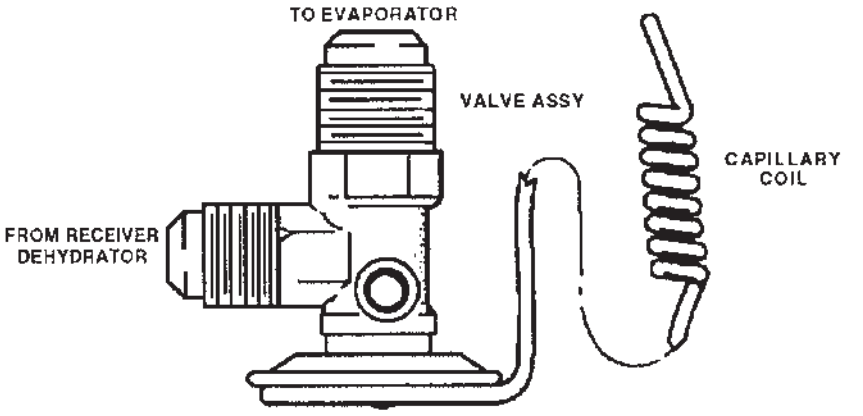
(c) Evacuate and charge system per “System Servicing Precautions” on page 215015 and “Evacuating the System” on page 215021.

(d) Check for leaks. See “Leak Detection” on page 215019.

(e) Replace access panels.



Condenser Air Scoop Rigging
Figure 17



Expansion Valve
Figure 18

M. Evaporator

(1) Removal

CAUTION: DISCHARGE THE SYSTEM BEFORE DISASSEMBLING ANY COMPONENTS FOR SERVICE.

CAUTION: UNITED STATES ENVIRONMENTAL REGULATIONS PROHIBIT THE RELEASE OF REFRIGERANT INTO THE ATMOSPHERE. SPECIAL EQUIPMENT IS REQUIRED WHEN DISCHARGING OR RECHARGING SYSTEM.

Evaporator assembly consists of evaporator core, receiver-dehydrator, expansion valve, circulating fan, pressure switch, necessary housing, and plumbing. The housing is made of thermoplastic material and the condensed moisture is dumped overboard through a hose clamped to fitting on bottom of evaporator housing.

Evaporator assembly is behind cabin rear closeout panel, attached to mounting panel with twelve (12) screws, washers, and a bracket securing the back to mounting panel.

(a) Remove air conditioning filter cover, filter, and rear access panels.

(b) Uncouple the liquid line from inlet side of receiver-dehydrator and suction line from evaporator core outlet (see Special Servicing Procedures above).

(c) Disconnect related electrical wires.

(d) Remove flexible air duct from housing outlet and remove drain hose from housing.

(e) Remove temperature probe from evaporator housing.

(f) Remove screws attaching support bracket and evaporator housing to mounting panel.

(g) Remove assembly through access hole in bulkhead.

(2) Installation

(a) Cement gasket in place on flanges of evaporator housing and attach large end of mounting gasket to back of housing.

(b) Install housing through access hole with air duct outlet on top and mate mounting flanges to surface of mounting panel and insert screws (Do not tighten at this time).

(c) Line mounting bracket with mating holes in mounting panel, insert screws and tighten. Tighten screws in flange and check that the gasket is in place, flange seal must be air tight.

(d) Couple suction and discharge lines to their proper fittings (apply Loctite refrigerant sealant to tube flares only).

(e) Evacuate and charge system per Servicing the System, Evacuating and Discharging, above.

(f) Check for leaks (see Leak Detection, above). If no leaks are detected, seal, and install access panel on evaporator housing.

(g) Couple flexible air duct and drain tube.

(h) Make and check electrical connections (see 91-21-50, Figure 1).

(i) Check blower operation and refrigerant systems.

(j) Install rear closeout panel.

N. Pressure Relief Switch

CAUTION: BEFORE RELIEF SWITCH REMOVAL, AIR CONDITIONING SYSTEM MUST BE DISCHARGED. (SEE DISCHARGING.)

CAUTION: UNITED STATES ENVIRONMENTAL REGULATIONS PROHIBIT THE RELEASE OF REFRIGERANT INTO THE ATMOSPHERE. SPECIAL EQUIPMENT IS REQUIRED WHEN DISCHARGING OR RECHARGING SYSTEM.

(1) Remove electrical connections from switch.

(2) Remove switch assembly from service port on steel line.

(3) Apply sealant sparingly to flare. When O-ring is present in HFC-134a systems, lube O-ring with PAG oil.

(4) Install new switch.

(5) Charge system per Servicing the System, Charging, above.

O. Electrical Installation

The wiring harness is connected to switches in the climate control center on the right side of the instrument panel. The harnesses cross the instrument panel to the left side where two wires are taken off for the compressor clutch. The harness then passes aft along the left side of fuselage connecting to the blower motor, the pressure relief switch, and the condenser actuating motor. Two fuses behind the air conditioning system control panel and a 10 amp circuit breaker mounted in circuit breaker panel protect the complete air conditioning electrical system.

(1) Adjustment of Throttle Switch

The throttle switch is mounted forward and below the throttle arm. The switch must be adjusted to actuate at the last quarter inch of full open throttle travel. Position the switch so that the throttle arm contacts the center of the switch actuator button.

(2) Fuse Replacement

Locate the fuse to be replaced behind the air conditioning system control panel.

(a) Open the fuse holder by applying a slight pushing, counterclockwise twisting, pressure.

(b) Remove blown fuse and insert a new 5 amp fuse.

(c) Close the fuse holder by applying a slight pushing and clockwise twisting pressure.

(3) Electrical Schematic

See 91-21-50, Figure 1.

CHAPTER

22

AUTOFLIGHT

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CHAPTER 22

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CHAPTER 22 - AUTOFLIGHT

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AUTOPILOT

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

1. General

An Autopilot/Flight Director (AP/FD) or Automatic Flight Control System (AFCS) system is installed as standard equipment in these airplanes.

A. Seneca IV - S/N's 3447001 thru 3447029 (1995–1996), and
Seneca V - S/N's 3449001 thru 3449151 (1997–1999).

A Bendix/King Autopilot/Flight Director (AP/FD), manufactured by then Allied Signal (now Honeywell), was installed in these airplanes. Maintenance information for those systems is not included in this manual. Follow the service literature published by the AP/FD equipment manufacturer. This includes mechanical service such as: adjusting bridle cable tension, servo removal and installation, servo clutch adjustments, etc.

Honeywell (King/Allied Signal) technical support, parts support, and service publications can be obtained as described under Supplementary Publications, Vendor Publications, in the Introduction.

B. Seneca V - S/N's 3449152 thru 3449491, less 3449459 (2000 thru 2014)

The S-TEC System 55 / 55X is installed in these airplanes. Maintenance information for this system is provided below.

C. Seneca V - S/N's 3449459, 3449492 and up (2014 and up)

The Garmin GFC 700 Automatic Flight Control System (AFCS) is fully integrated within the G1000/G1000 NXi system which installed in these airplanes. Maintenance information for this system is provided below.

2. S-TEC System 55 / 55X (Seneca V only.)

A. S-TEC System 55 (2000)

This system is installed as standard equipment in S/N's 3449152 thru 3449195, less 3449161.

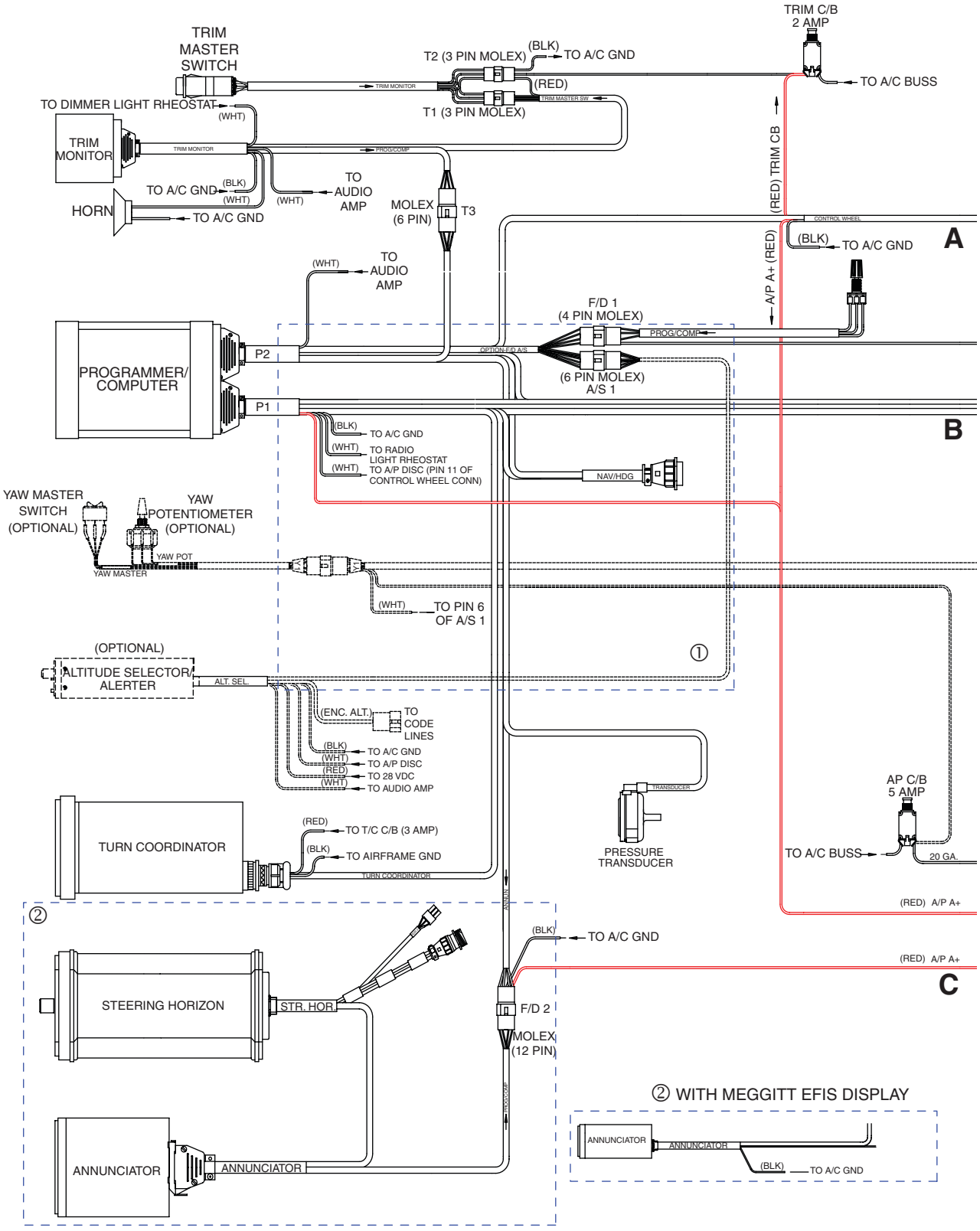
B. S-TEC System 55X (2001 thru 2014)

This system is installed as standard equipment in S/N's 3449161 and 3449196 thru 3449491, less 3449459.

C. Description

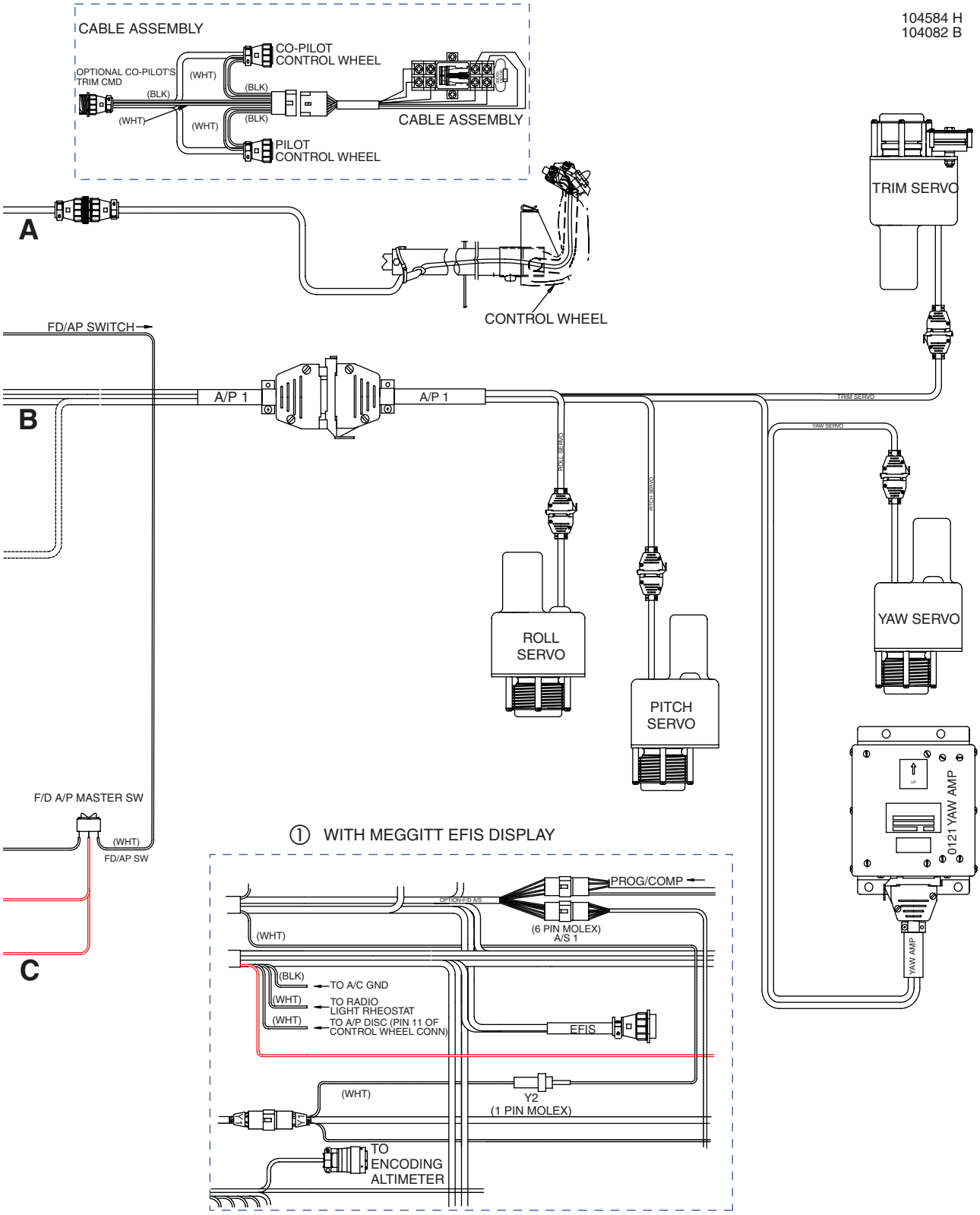
See "Figure 1" on page 22102.

The Pitch Computer receives select input signals from the Altitude Pressure Transducer, Accelerometer, Glideslope Deviation Indicator and Altitude Selector/Alerter (if installed). It then computes pitch servo commands for vertical speed, altitude hold and glideslope intercept and tracking. Sensing for trim annunciation or automatic stabilator trim is provided by the pitch servo. Drive for the stabilator trim servo is provided by the pitch computer.



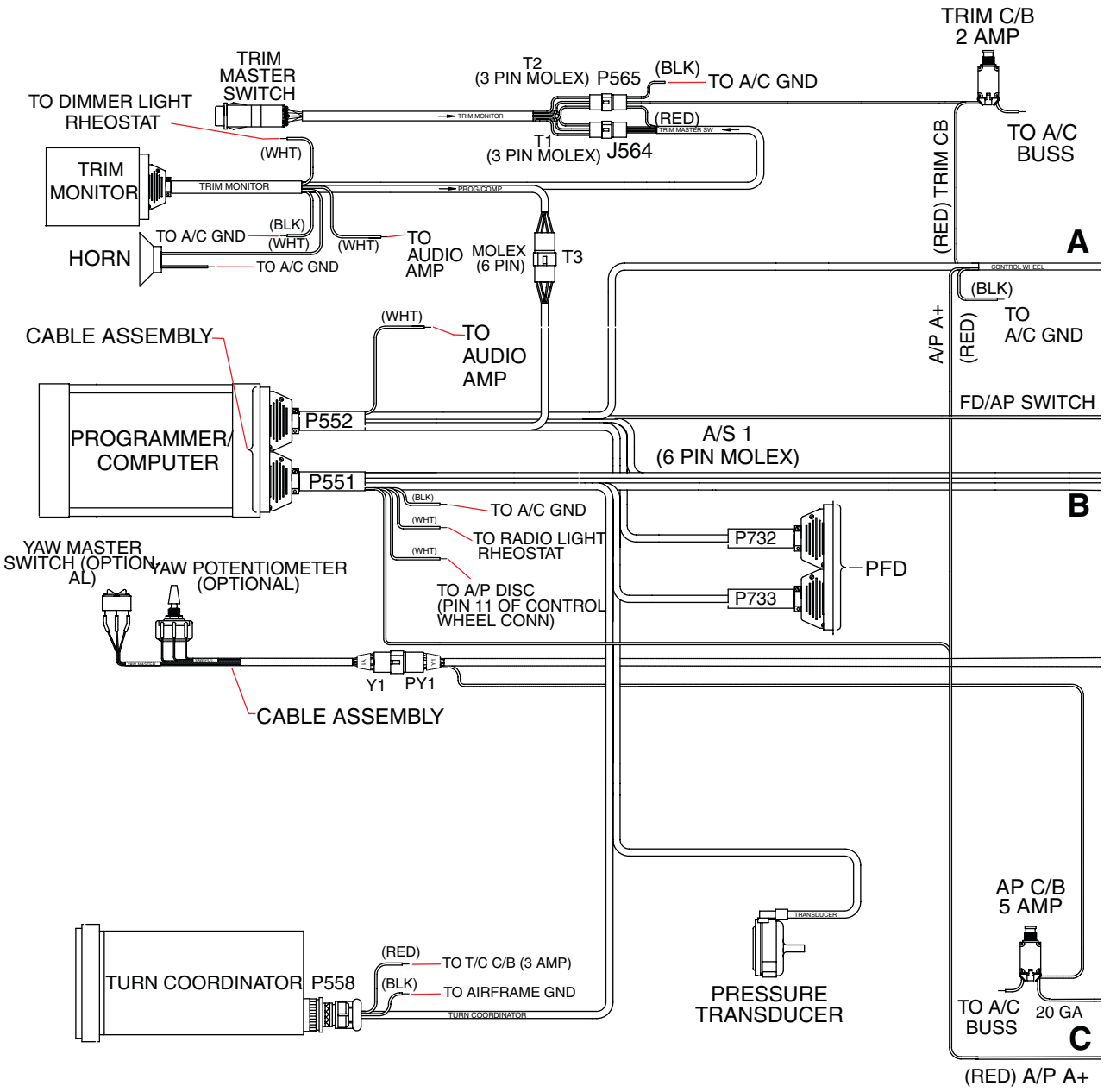
Effectivity
3449152 and up

System 55 / 55X Autopilot Installation
Figure 1 (Sheet 1 of 9)



System 55 / 55X Autopilot Installation
Figure 1 (Sheet 2 of 9)

Effectivity
3449152 and up

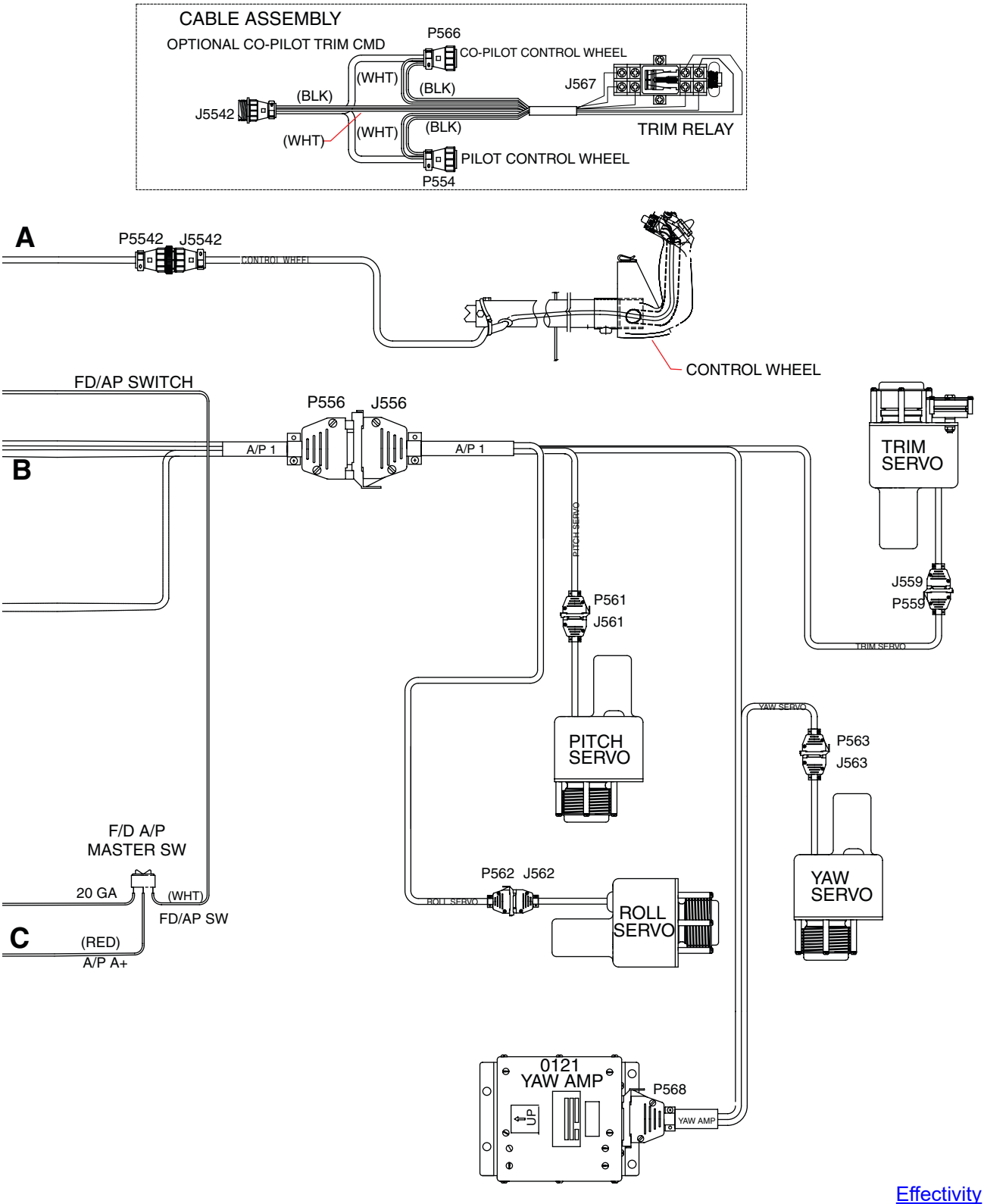


Effectivity
Seneca V
With Avidyne Entegra

System 55 / 55X Autopilot Installation
Figure 1 (Sheet 3 of 9)

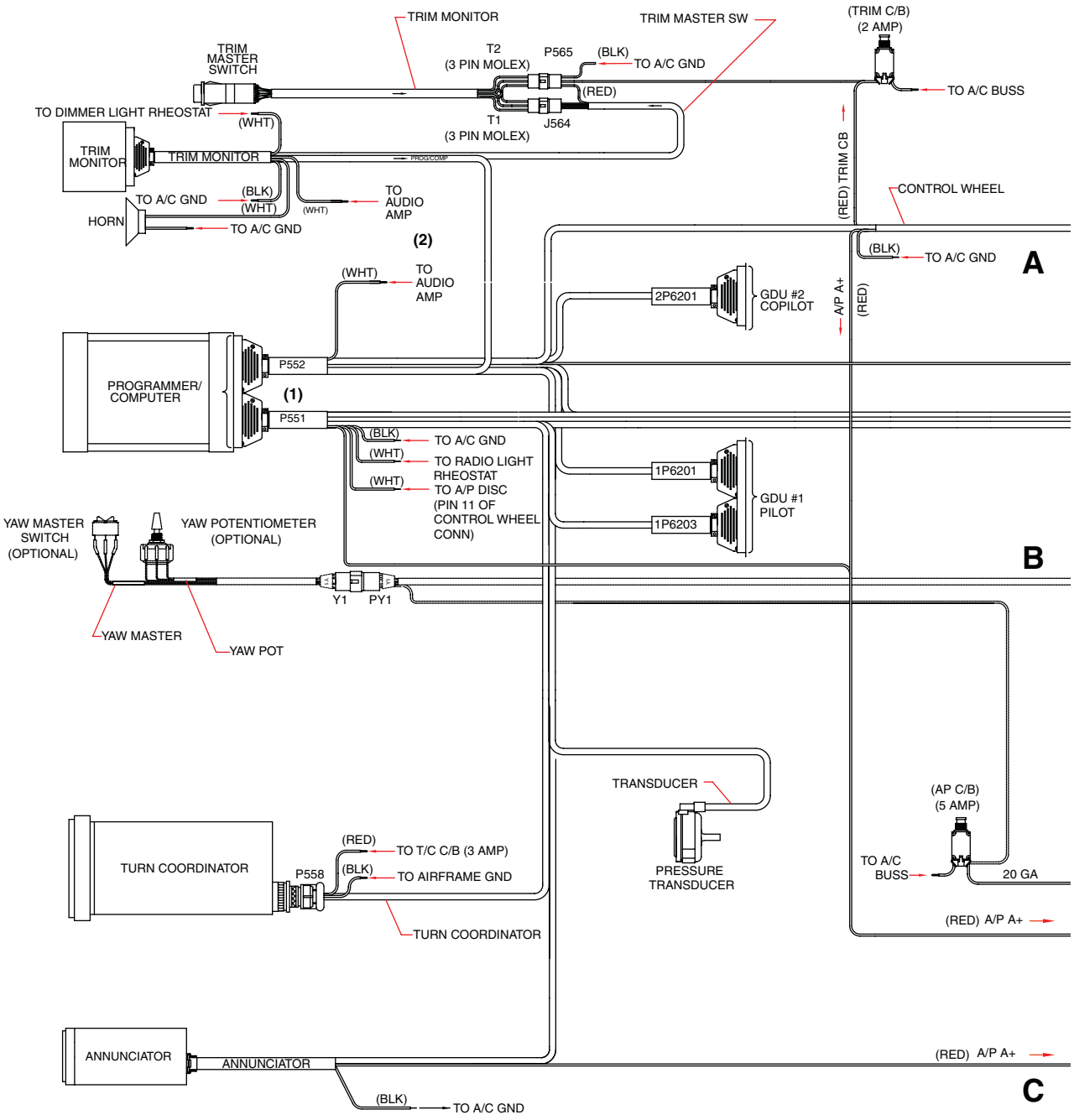
Effectivity
Seneca V
with S-TEC Autopilot

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System 55 / 55X Autopilot Installation
Figure 1 (Sheet 4 of 9)

Effectivity
Seneca V
With Avidyne Entegra

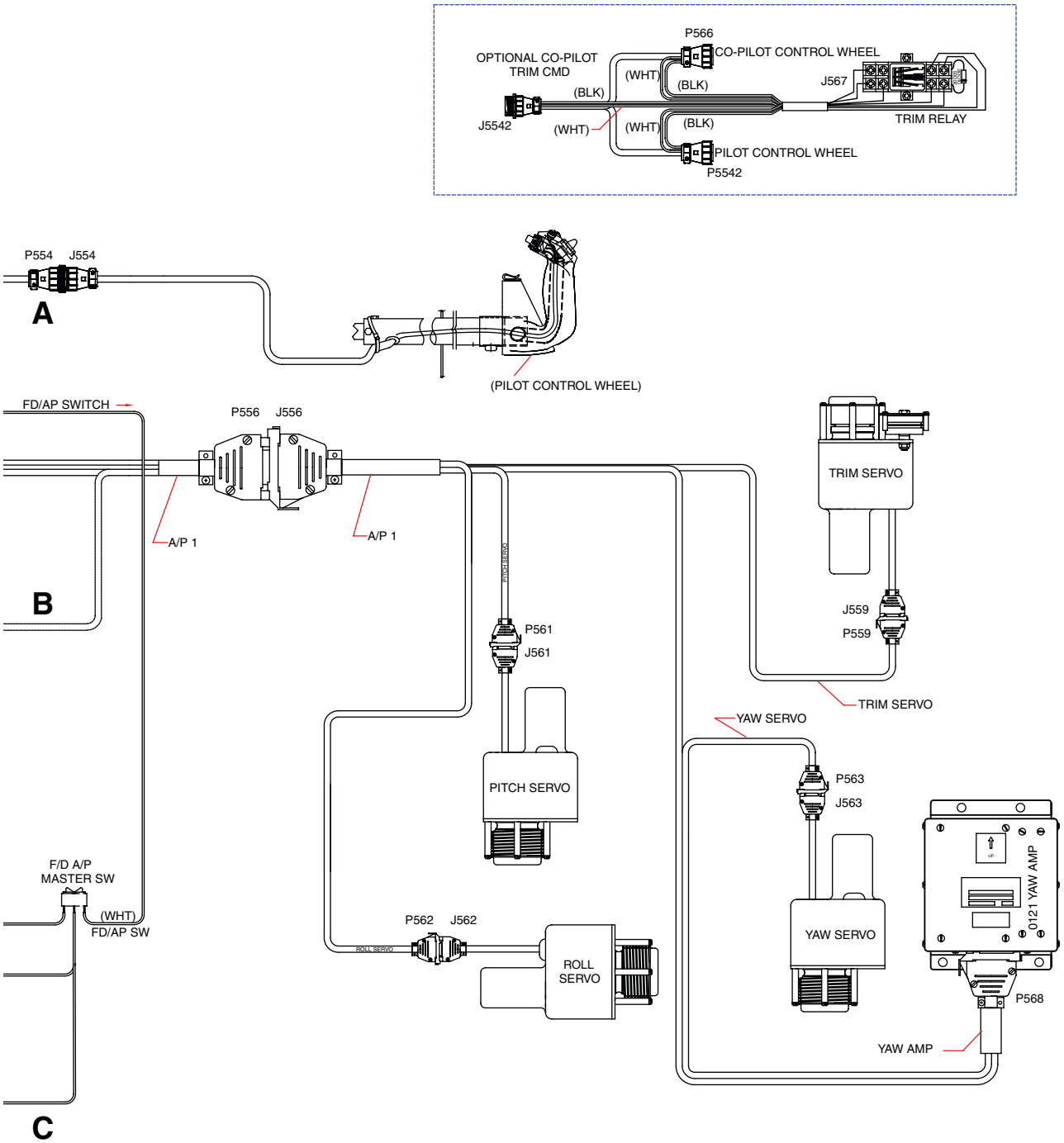


(1) CAUTION: The 50 pin connectors of both cable assemblies are identical. Ensure that connector P551 plugs into P1 and connector P552 plugs into P2.

(2) The autopilot and trim monitor audio outputs are terminated into a resistor network which is shown in 91-23-50, Figure 3, Sheet 2.

Effectivity
Seneca V
With Garmin G600

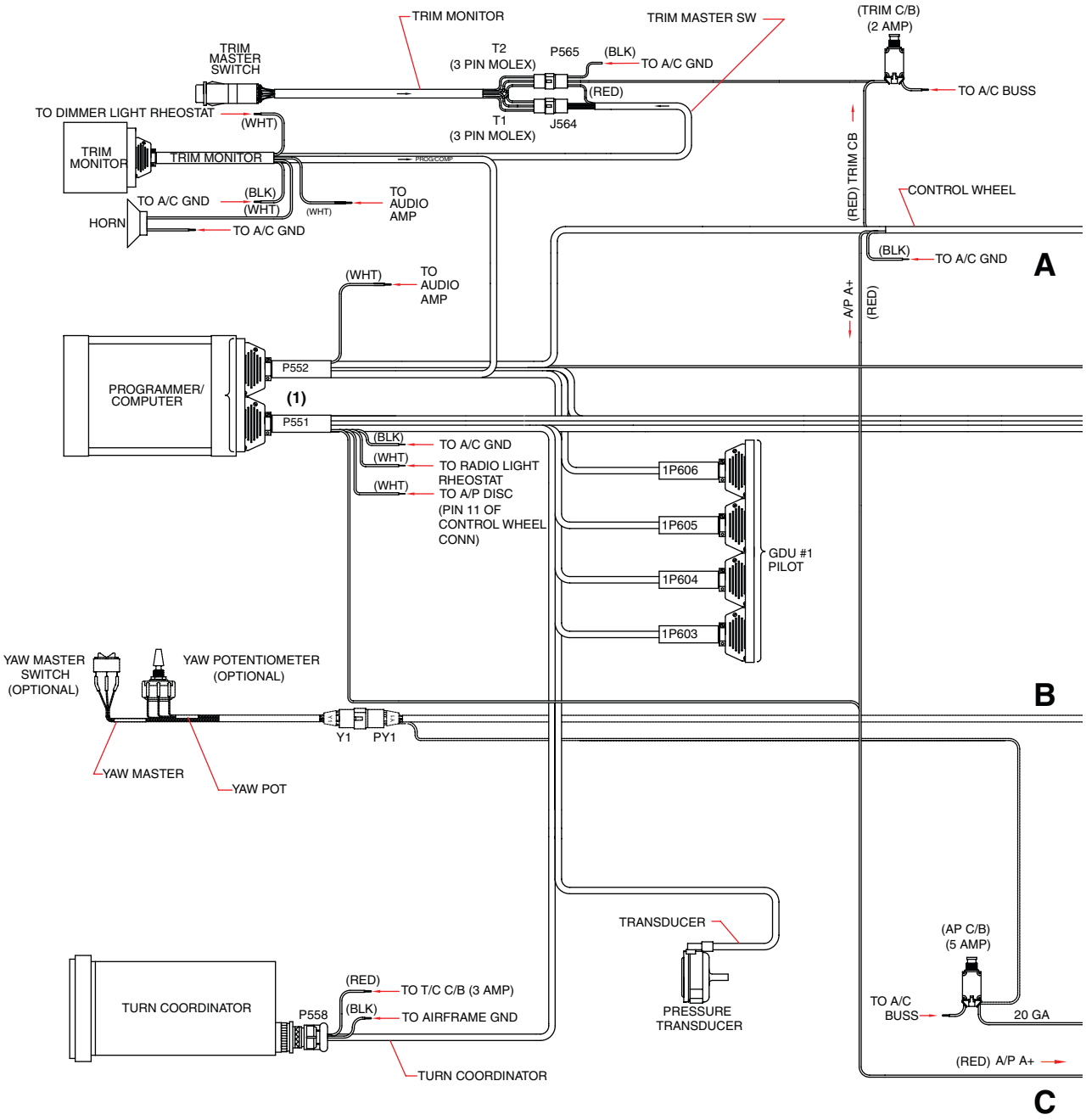
System 55 / 55X Autopilot Installation
Figure 1 (Sheet 5 of 9)



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System 55 / 55X Autopilot Installation
Figure 1 (Sheet 6 of 9)

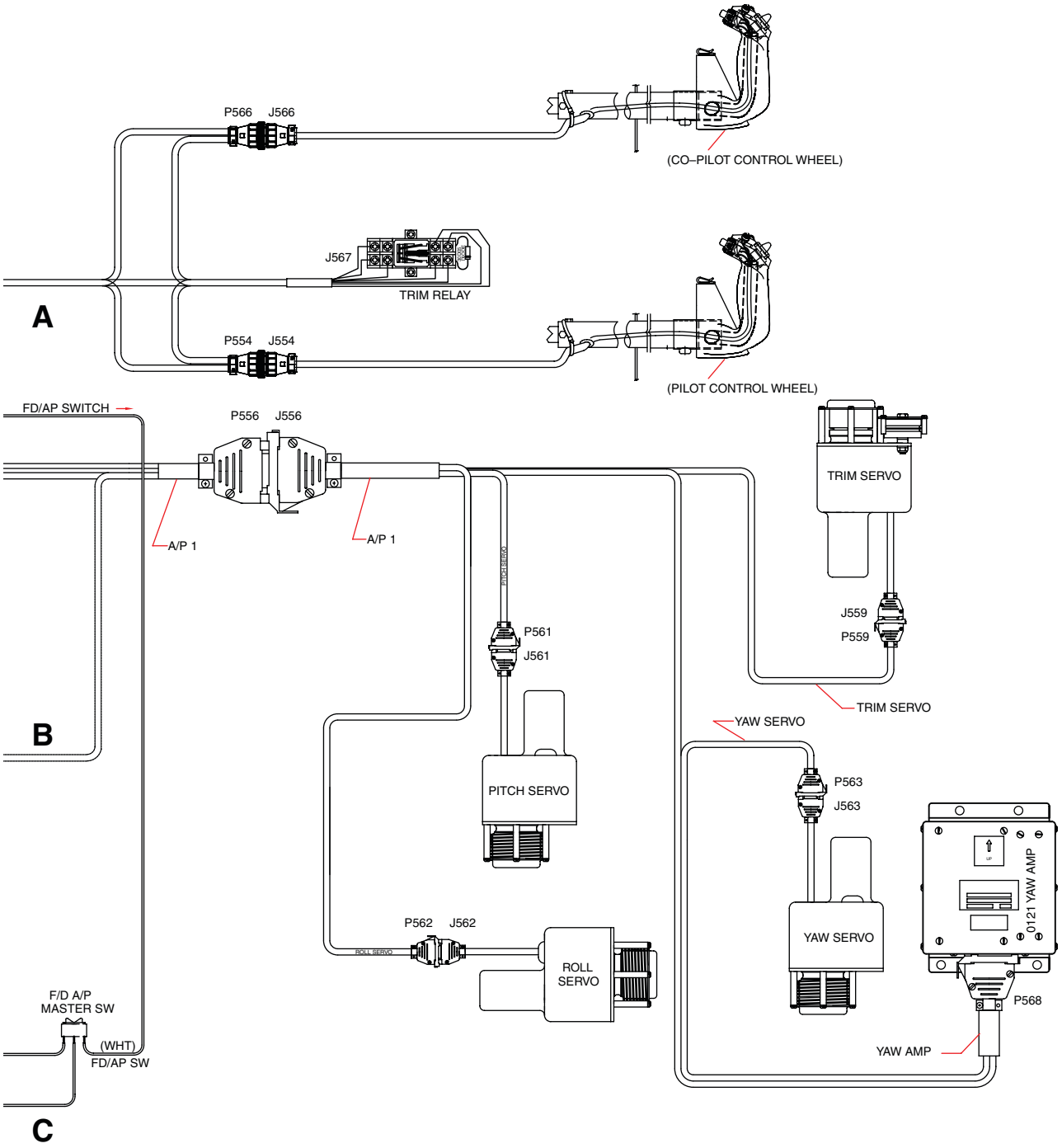
Effectivity
Seneca V
With Garmin G600



(1) CAUTION: The 50 pin connectors of both cable assemblies are identical. Ensure that connector P551 plugs into P1 and connector P552 plugs into P2.

Effectivity
Seneca V
With Garmin G1000

System 55 / 55X Autopilot Installation
Figure 1 (Sheet 7 of 9)

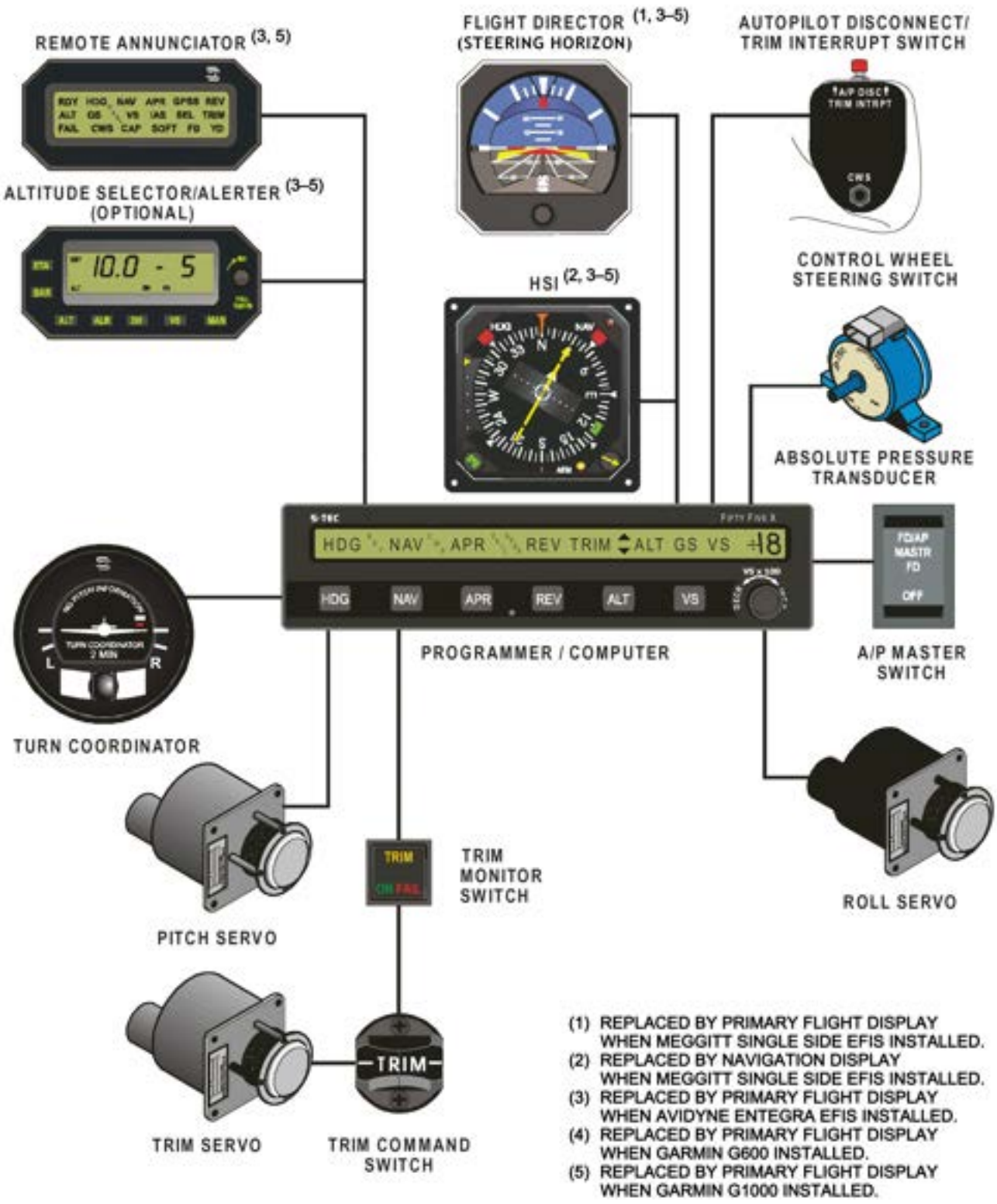


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System 55 / 55X Autopilot Installation
Figure 1 (Sheet 8 of 9)

Effectivity
Seneca V
With Garmin G1000

Effectivity
Seneca V
with S-TEC Autopilot



Effectivity
3449152 and up

System 55 / 55X Autopilot Installation
Figure 1 (Sheet 9 of 9)

Effectivity
Seneca V
with S-TEC Autopilot

A typical S-TEC System 55/55X Autopilot installation includes the following:

(1) Panel Mounted:

Programmer/Computer, Turn Coordinator, Annunciator, D.G. or HSI, and Steering Horizon.

(2) Remote Mounted:

Roll Servo, Pitch Servo, Trim Servo, Trim Monitor, A/P Disconnect switch, and Altitude (Pressure) Transducer.

Servo installations use aluminum brackets to secure the servos to the airframe. Attachment to the airplane's primary flight control and trim systems is accomplished with bridle cables and extension attachments.

D. Troubleshooting

System functionality can be determined using functional checks described in the AFM Supplement and autopilot Pilot's Operating Handbook. More detailed troubleshooting should be accomplished by authorized S-TEC Dealers, holding the appropriate FAA certification, with required test equipment and service data.

E. GPSS (System 55X only.)

The Global Positioning System Steering (GPSS) is a function of the 55X autopilot only. In the GPSS mode, the converter receives ground speed and bank angle digital signals that are calculated and converted to a commanded turn rate. The turn rate is then scaled and converted to a DC heading error signal that is compatible with S-TEC autopilots. The end result is an autopilot that can be directly coupled to the roll steering commands produced by the GPS Navigator, eliminating the need for the pilot to make any further adjustments to the HSI course arrow or the DG's heading bug.

F. System Operation

Operation of the autopilot and other systems is described in the FAA-approved Airplane Flight Manual Supplement (AFMS) - see airplane Pilot's Operating Handbook (POH), Section 9. Specialized controls, annunciation, operation and interpretation are covered in this supplement and in the S-TEC Autopilot POH that supplements the approved AFMS.

G. Maintenance

Except as provided in 5-20-00, servicing and/or maintenance of the autopilot system is On-Condition.

NOTE: Servicing of S-TEC System 55/55X Autopilot installations is best accomplished by approved S-TEC dealers holding the appropriate FAA-certification. Locations of and access to the components installed are described and depicted individually below. Removal and replacement of components is generally indicated by functional checks provided in the AFM Supplement, S-TEC Autopilot POH and/or below.

H. Post-Maintenance Operational Checkout

(1) Standard, Meggitt Single Side EFIS, and Avidyne Entegra

Complete the following checkout procedure after any maintenance to the system is performed.

NOTE: The Systems 55/55X incorporate a SELF-TEST that requires a 100% pass rate before the autopilot can be engaged.

NOTE: For airplanes equipped with Meggitt Single Side EFIS (see 34-21-00), references below to the flight director and HSI are to those functions in the Primary Flight Display (PFD) and Navigation Display (ND), respectively.

NOTE: For airplanes equipped with Avidyne Entegra (see 34-22-00), references below to the remote annunciator, flight director and HSI are to those functions in the Primary Flight Display (PFD).

(a) Apply aircraft power.

(b) Avionics Master Switch — ON

(c) Autopilot Master Switch — Set to FD / AP

NOTE: Observe that all segments of the Programmer / Computer display and annunciators illuminate for five (5) seconds during test. Satisfactory completion of the SELF-TEST is indicated when the Ready (RDY) annunciator remains on at the end of the five (5) second self-test. Should a fault be detected, the FAIL annunciator will remain on at the conclusion of the self-test and the autopilot will not operate.

(d) Trim Master (ON / OFF) Switch — ON

(e) HDG and VS switches — PRESS / RELEASE

Ensure that HDG and VS illuminate on the Fifty Five X annunciator.

(f) VS Knob — ROTATE CW

Pitch control (i.e. - the control yoke) should move slowly out (pilot may have to assist a heavy yoke).

(g) VS Knob — ROTATE CCW

Pitch control should move slowly in.

(h) A/P DISC Trim Interrupt Switch (on control yoke) — PRESS

Verify the autopilot disconnects.

(i) HDG Mode — ENGAGE

(j) DG or HSI HDG bug — MOVE LT / RT

Roll control should follow the HDG bug.

NOTE: If HSI equipped, center the course arrow under the lubber line and push the NAV button. Move the course arrow on the HSI left then right. Roll control should follow the course arrow. Channel a valid VOR signal and move course arrow just enough to deflect the left / right needle one (1) or two (2) dots. Roll control should follow the Course Deviation Indicator (CDI) left / right needle during the test. (This test is only valid if the left / right needle is centered with the course arrow under the lubber line.)

NOTE: If DG equipped, center the HDG bug under the lubber line. Channel a valid VOR signal. Move the OBS to cause left / right CDI needle deflection. The roll control should follow the left / right needle movement.

(k) REV Mode button — PUSH

Roll control should respond opposite to the course arrow and CDI left / right needle inputs.

(l) Altitude Hold (ALT) button — PUSH

Slowly pull out (nose up) on the pitch control (i.e. - control yoke). Autotrim should run nose down with TRIM flashing on the remote annunciator and the autopilot computer / programmer after approximately 3 seconds. Slowly move control yoke forward (nose down). After 3 seconds, autotrim should move nose up with TRIM flashing on the remote annunciator and the autopilot computer / programmer after approximately 3 seconds.

(m) Trim Master (ON / OFF) Switch — OFF

(n) Manual Electric Trim Test:

1) Trim Master (ON / OFF) switch — ON

a) Move each segment of the Manual Electric Trim Command Switch FWD and AFT.
Trim should not run.

b) Move both segments of the Trim Command switch FWD.
Trim should run nose down.

c) Move both segments of the Trim Command switch AFT.
Trim should run nose up.

2) Re-trim aircraft for takeoff and check controls for freedom of movement. Be sure the autopilot and trim servos are dis-engaged.

(o) Flight Director Test:

1) Autopilot Master Switch — SELECT FD

Note the roll, pitch and trim servos are disengaged. The steering bar should be in view on the attitude indicator.

2) HDG Mode — ENGAGE

MOVE HDG bug 45 degrees left. The roll steering bar should slowly indicate a left steering command. Repeat the same test for the right side.

3) VS Mode — ENGAGE

SELECT 1500 FPM rate of climb. Note the pitch steering bar moves slowly up. Repeat the same test for the down direction.

4) Autopilot Master Switch — SELECT FD / AP

The servos should re-engage.

5) Trim Master ON / OFF Switch — ON

6) Manual Electric Trim Command Switch — MOVE FWD or AFT

The autopilot should disconnect.

NOTE: The Manual Electric Trim Command Switch will disconnect the autopilot only if there is a Pitch Mode engaged.

(p) Post-Installation Ground Checks for the Yaw Damper System, if installed.

1) With aircraft levelled laterally, and trim pot on instrument panel approximately centered, adjust the pot in the yaw damper amplifier (located under plug in top of amplifier) to provide a null (no servo action) at the yaw servo.

2) Align the cap on the yaw trim pot knob (on instrument panel) so that the white line is centered vertically.

3) Check operation of yaw trim pot by turning trim pot both left and right, and note rudder pedal movement in the same direction (i.e., a counterclockwise or left turn of the trim pot causes left rudder pedal depression, etc.).

(2) Garmin G600

(PIR-PPS55028, Rev. F.)

(a) Autopilot Power-Up Test

1) Apply electrical power to Avionics.
2) Select AP/FD master switch to AP/FD. YD master switch OFF.
3) Verify RDY displayed on AP controller within 3 minutes.
4) Pull AP T&B circuit breaker.
5) Verify AP controller display is blank.
6) Select AP master switch OFF. Re-set all circuit breakers.

(b) Autopilot Engagement Test

1) Apply electrical power to Avionics.
2) Select FD/AP master switch to FD/AP.
3) Wait until GDU #1 has a valid heading, attitude, altitude, and GPS location (from the GNS 430W navigator).
4) Press the HDG and ALT buttons on the AP control panel.
5) Verify the pitch and roll clutches engage and resist movement of the control wheels.
6) Press the AP DISC switch on the pilot's control wheel.
7) Verify the autopilot disengages, accompanied by an annunciation on the AP modes panel and an aural alert.
8) Verify that the control wheels are free in pitch and roll axes.
9) If copilot electric trim installed, re-engage AP and repeat steps 6) – 8) using copilot's control wheel.
10) Select FD/AP master switch to "FD".
11) Select heading and altitude modes.
12) Verify that HDG and ALT modes are displayed on the ST-145 remote annunciator and that the command bars are displayed on both GDUs.
13) Verify that the AP remains disengaged (no resistance on control wheels).

(c) FD/AP Operational Test

1) Aircraft level.
2) Apply electrical power to Avionics.
3) Both GDUs operating in normal mode.
4) Select FD/AP master switch to FD/AP.
5) Push the HDG knob to synchronize the heading bug to the current aircraft heading.
6) Select HDG and ALT (or VS) modes on the AP controller.
7) Verify the flight director is displayed on both GDUs.

NOTE: The command bars on both GDUs should be aligned with the aircraft reference and the control wheels should be centered and stationary. There may be some roll motion in the yokes if the aircraft is not level.

8) Turn the HDG knob counterclockwise.
9) Verify that the command bars on both GDUs and both control wheels rotate counterclockwise.
10) Turn the HDG knob clockwise.
11) Verify that the command bars on both GDUs and both control wheels rotate clockwise.
12) Push the HDG knob to synchronize the heading bug to the current aircraft heading. Wait until the flight directors align with the aircraft reference.

13) Select VS mode on AP computer. On the AP controller, rotate the VS control clockwise.
14) Verify that the command bars on both GDUs command a pitch up attitude and both control wheels move backward. (In some aircraft, the down spring may require manual assistance to get aft control stick movement).
15) On the AP controller, rotate the VS control counterclockwise.
16) Verify that the command bars on both GDUs command a pitch down attitude and both control wheels move forward.
17) Disengage AP.

(d) Autopilot Clutch Torque and Cable Tension

1) Electrical power applied to Avionics.
2) Select FD/AP master switch to FD/AP.
3) Engage the Autopilot by selecting HDG and ALT modes on the AP computer.
4) Manually overpower the autopilot clutches in pitch and roll. If the Autopilot clutches cannot be overpowered, check the servo clutch torque settings.
5) Press the pilot's electric trim switch for nose up trim. Autopilot will disconnect.
6) While the trim is running, restrain the aircraft pitch trim wheel and verify that the trim clutch can be overpowered. If it cannot be overpowered, check the trim servo clutch torque setting.
7) Actuate and hold the manual electric trim switch in the nose up direction.
8) Verify that the trim wheel moves smoothly throughout the entire trim range during manual electric trim operation.
9) Actuate and hold the manual electric trim switch in the nose down direction.
10) Verify that the trim wheel moves smoothly throughout the entire trim range during manual electric trim operation. If the trim wheel hesitates, this may indicate that the pitch trim clutch is slipping and proper clutch setting and cable tension should be verified. If both clutch setting and cable tension are within tolerance, check the aircraft pitch trim system for excessive friction.

(e) Electric Trim and CWS

1) Electric power applied to Avionics.
2) Trim master switch selected ON.
3) Perform the following steps using the pilot's control yoke.
4) Actuate the electric trim switch in the UP and DOWN directions.
5) Verify the following:
   a) TRIM annunciation is displayed on AP controller with arrow in correct direction.
   b) Trim clutch engages.
   c) Pitch trim wheel and trim tab drive in the selected direction.
6) Activate each half of the electric trim switch.
7) Verify the trim clutch does not engage, the trim wheel does not move, and trim alarm activates.
8) Press the AP DISC switch and hold while actuating the electric trim switch.
9) Verify trim does not run and the trim wheel rotates freely when moved manually.
10) Release the AP DISC and electric trim.
11) If copilot trim is installed, repeat steps 4) – 10) using the copilot's electric trim switch.
12) Activate the pilot's electric trim switch. Remove power from the Pitch Trim servo by pulling the Pitch Trim circuit breaker.

13) Verify trim does not run and the trim wheel rotates freely when moved manually.
14) Release the trim switch and restore power to the Pitch Trim Servo.
15) Engage the AP by selecting the HDG and ALT modes.
16) Press and hold the pilot's CWS button.
17) Verify the following:
    a) CWS is displayed on AP remote annunciator.
    b) While the CWS button is pressed, there is no residual force on the control stick in the pitch and roll axis.
    c) When the CWS button is released, the servos re-engage and maintain the roll attitude and vertical speed.
18) If copilot trim is installed, repeat steps 16) – 17) using the copilot's CWS button.
19) With the Autopilot still engaged and the pilot's CWS button pressed, move the control wheel to its aft limit. Release the CWS button and apply continuous forward pressure, slowly moving the control wheel.
20) Verify the trim wheel begins moving in a trim up direction.
21) Grip the control wheel and press the pilot's CWS button.
22) Verify trim motion stops.
23) Move the control wheel to the forward limit and release the CWS button. Slowly move the control wheel aft.
24) Verify the trim wheel begins to trim nose down.
25) Relieve pressure on the wheel.
26) Verify the trim motion stops and the trim wheel is free to turn.
27) If copilot trim installed, perform the following:
    a) Activate copilot's electric trim switch in the UP/DN direction, then activate the Pilot's electric switch in the opposite direction.
    b) Verify the Pilot's trim switch takes priority.

(f) Flight Director Offset

1) Electrical power applied to Avionics.
2) AP/FD switch selected to FD/AP.
3) Center the heading bug on GDU #1 by pressing the HDG knob on the PFD.
4) Activate the autopilot in heading and altitude hold mode.
5) On GDU #1, observe the position of the flight director bar in relation to the aircraft reference symbol.
6) On GDU #1, if the flight director command bars are aligned with the aircraft reference symbol in the pitch axis, no further steps are required.
7) If the flight director command bars are offset from the aircraft reference symbol in the pitch axis, take note of the number of degrees of offset for adjustment.
8) Ensure an Installer Unlock Card is inserted in GDU #1.
9) Start GDU #1 in configuration mode and navigate to the FLIGHT DIRECTOR page in the FCS page group.
10) Change PITCH V Ref as needed. Note that the V/DEG ratio for the flight director is displayed on the same page. For example, if the PITCH V/DEG is 0.217, changing the PITCH V ref by 0.217 would move the flight director by one degree.
11) The resulting FD Pitch can be read at the bottom of the screen. Adjust the PITCH V ref until the FD Pitch is 0.00 ± 0.25°.

12) Restart GDU #1 in normal mode and verify that the flight director is aligned with the aircraft reference symbol.

13) Repeat steps 8) – 12) on GDU #2.

(g) Altitude Preselector Test

NOTE: Only perform test on GDU1.

1) Power up the GDU 620 in normal mode and wait until valid air data is displayed.

2) On the GDU 620, set the selected altitude to match the displayed field elevation plus approximately 1000 feet.

3) Engage the autopilot in HDG mode. Press ALT and VS simultaneously to select altitude select mode.

4) Verify that ALT and VS are displayed on the autopilot.

NOTE: If ALT is not displayed, the autopilot is not receiving the RS-485 altitude preselector data from the GDU 620.

5) Disengage the autopilot.

(h) Yaw Damper

1) Ensure aircraft is level in the lateral (roll) axis.

2) Adjust yaw damper trim control on instrument panel to center of rotation (rotation centered between mechanical stops).

3) Align the cap on the yaw damper trim control on instrument panel so that the white line is centered vertically.

4) Connect Yaw Damper Alignment box between cable harness and yaw amplifier. Connect a voltmeter to the YD alignment box.

5) Apply electrical power to Avionics.

6) Select AP/FD Master switch ON.

7) Select Yaw Damper Master switch ON.

8) Select “POT” on yaw damper alignment box.

9) Adjust the Yaw potentiometer on the YD alignment box for a null voltage.

10) Select “SERVO” on the YD alignment box.

11) Verify a null voltage from YD alignment box.

12) If voltage not null, remove silver cap from YD amplifier and adjust the centering potentiometer in the YD amplifier to obtain a null voltage from the YD alignment box. Allow a few seconds after each adjustment for the voltage to settle.

13) Turn Yaw potentiometer on the YD alignment box full counterclockwise.

14) After 10 seconds, note voltage at YD alignment box.

15) Turn Yaw potentiometer on the YD alignment box full clockwise.

16) After 10 seconds, note voltage at YD alignment box.

17) Verify that voltages noted at full rotation are within ±0.3V (ignoring polarity).

18) Remove YD alignment box and re-connect YD amplifier cable.

19) Turn YD trim control counterclockwise.

20) Verify left rudder pedal depression.

21) Turn YD trim control clockwise.

22) Verify right rudder pedal depression.

23) Switch OFF YD master switch.

(i) NAV Receiver Interface Test

1) Electrical power applied to Avionics, GDU #1 running in configuration mode.
2) Select AP/FD using AP/FD master switch.
3) Engage the autopilot in HDG and ALT modes.
4) Select the Autopilot Test page from the FCS page group.
5) Set the lateral deviation to 30% R.
6) Set the lateral flag to VALID.
7) Select NAV mode on the autopilot computer.
8) Verify the following:
   a) NAV mode activates
   b) Flight director indicates right bank command
   c) Control wheels and ailerons apply turn to the right
9) Set the lateral flag to INVALID.
10) Verify NAV mode flashes and FAIL is indicated on AP mode display.
11) Set the lateral flag back to VALID.
12) Set the lateral deviation to 30% L.
13) Verify the following:
   a) Flight director indicates left bank command
   b) Control wheels and ailerons apply turn to the left
14) Set the lateral flag to INVALID.
15) Verify NAV mode flashes and FAIL is indicated on AP mode display.
16) Set the lateral flag back to VALID.
17) Set the lateral deviation to 0%. Wait for controls to center.
18) Select HDG and ALT modes on the AP controller.
19) Set the ILS/GPS APRCH to ACTIVE.
20) Set the vertical deviation to 30% UP.
21) Set the vertical flag to VALID.
22) Press the APR button on the AP computer.
23) Verify that the GS mode is shown as armed (ALT mode remains displayed).
24) Reduce vertical deviation to 5%.
25) Verify GS mode is displayed as active (ALT mode no longer displayed).
26) Set the vertical deviation to 30% UP.
27) Verify the following:
   a) Autopilot moves the control yoke or stick backward.
   b) Flight director indicates a pitch up command.
28) Set the vertical flag to INVALID.
29) Verify GS mode flashes and FAIL is indicated on AP mode display.
30) Set the vertical flag back to VALID.
31) Set the vertical deviation to 30% DOWN.
32) Verify the following:
   a) Autopilot moves the control yoke or stick forward.
   b) Flight director indicates a pitch down command.
33) Set the vertical flag to INVALID.

34) Verify GS mode flashes and FAIL is indicated on AP mode display.
35) Disengage the autopilot.

(j) GPS Roll Steering Test

1) Electrical power applied to Avionics, GDU #1 running in configuration mode.
2) AP/FD master switch selected to “AP/FD”.
3) Select the Autopilot Test page.
4) Set the GPS ANNUNCIATE discrete output to ACTIVE.
5) Select GPSS mode by pushing NAV button on AP controller twice.
6) Verify GPSS autopilot mode is annunciated.
7) Set the BANK ANGLE to 10°R.
8) Verify that the autopilot rolls the control yoke or stick to the right.
9) Verify the flight director indicates a right turn command.
10) Set the bank angle to 0°.
11) Verify that the autopilot rolls the control yoke or stick level.
12) Verify the flight director indicates a wings level command.
13) Set the bank angle to 10°L.
14) Verify that the autopilot rolls the control yoke or stick to the left.
15) Verify the flight director indicates a left turn command.
16) Disconnect the autopilot.

(3) Garmin G1000

(PIR-PPS55032, Rev. D.)

(a) Autopilot Power-Up Test

1) Select the AVION MASTR switch to ON.
2) Select AP master switch to “FD/AP”. YD master switch OFF.
3) Verify RDY is displayed on AP controller and the PFDs AFCS status box within 3 minutes.
4) Pull AUTOPILOT circuit breaker.
5) Verify AP controller display is blank.
6) Select AP master switch OFF. Reset all circuit breakers.

(b) Autopilot Engagement Test

1) Select AP master switch to “FD/AP”.
2) Press the HDG and ALT buttons on the AP computer.
3) Verify HDG and ALT are displayed on AP computer and the PFDs AFCS status box.
4) Verify the pitch and roll clutches engage and resist movement of the control wheels.
5) Press the AP DISC switch on the pilot’s control wheel.
6) Verify the autopilot disengages, accompanied by an annunciation (flashing RDY) on the AP computer and an aural alert.
7) Verify that the control wheels have free motion in the roll and pitch axes.
8) Re-engage the autopilot and repeat steps 2–7 using co-pilot’s control wheel.
9) Select AP master switch to “FD”.
10) Select the heading and altitude modes.
11) Verify that HDG and ALT are displayed on the AP computer.
12) Verify that FD, HDG, and ALT are displayed on the PFDs AFCS status box, and that the command bars are displayed on the PFDs.
13) Verify that the AP remains disengaged (the control wheels have free motion).

(c) FD/AP Operational Test

NOTE: The command bars on the PFDs should be aligned with the aircraft reference and the control wheels should be centered and stationary. There may be some roll motion of the yokes if the aircraft is not level.

1) Ensure that the aircraft is level.
2) Select AP master switch to “FD/AP”.
3) Push the HDG knob on the PFD to synchronize the heading bug to the current aircraft heading.
4) Press the HDG and ALT buttons on the AP computer.
5) Verify that HDG and ALT are displayed on the AP computer and the PFDs AFCS status box, and that the command bars are displayed on the PFDs.
6) Turn the HDG knob on the PFD counterclockwise.
7) Verify the command bars on the PFDs and the control wheels rotate counterclockwise.
8) Turn the HDG knob on the PFD clockwise.
9) Verify the command bars on the PFDs and the control wheels rotate clockwise.
10) Push the HDG knob to synchronize the heading bug to the current aircraft heading. Wait until the flight director command bars align with the aircraft symbol.
11) Press the VS button on the AP computer.
12) Turn the VS knob on the AP computer clockwise.

13) Verify the command bars on the PFDs pitch up and the control wheels move aft. (In some aircraft, the down spring may require manual assistance to get control stick movement.)
14) Turn the VS knob on the AP computer counterclockwise.
15) Verify the command bars on the PFDs pitch down and the control wheels move forward.
16) Disengage the AP.

(d) Autopilot Clutch Torque and Cable Tension Test

1) Select AP master switch to “FD/AP” and TRIM switch “ON”.
2) Press the HDG and ALT buttons on the AP computer.
3) Verify that HDG and ALT are displayed on the AP computer and the PFDs AFCS status box, and that the command bars are displayed on the PFDs.
4) Verify the autopilot clutches can be manually overpowered in the pitch and roll axes.

NOTE: If servo clutches cannot be overpowered, check the servo clutch torque settings. Refer to the appropriate autopilot installation in 22-10-00 for torque settings.

5) Press the HDG and ALT buttons on the AP computer.
6) Verify that HDG and ALT are displayed on the AP computer and the PFDs AFCS status box, and that the command bars are displayed on the PFDs.
7) Actuate and hold PITCH TRIM switch in either the NOSE UP or NOSE DOWN direction.
8) Verify the autopilot disconnects, RDY is displayed on the autopilot computer flashing for 5 seconds and then remains, and TRIM is displayed on the PFD flashing while the trim is running.
9) While trim is running, restrain the aircraft pitch trim wheel and verify that the trim clutch can be overpowered.

NOTE: If trim servo clutch cannot be overpowered, check the trim servo clutch torque setting. Refer to the appropriate autopilot installation in 22-10-00 for torque settings.

10) Actuate and hold PITCH TRIM switch in the NOSE UP direction.
11) Verify the trim wheel moves smoothly throughout the entire trim range during manual electric trim operation, and TRIM is displayed on the PFD flashing while the trim is running.
12) Actuate and hold PITCH TRIM switch in the NOSE DOWN direction.
13) Verify the trim wheel moves smoothly throughout the entire trim range during manual electric trim operation, and TRIM is displayed on the PFD flashing while the trim is running.

(e) Electric Trim and CWS

Verify the AFCS system buttons and switches are operating correctly by performing the following steps (for both the pilot and copilot control wheels where applicable).

1) Select AP master switch to “FD/AP” and TRIM switch “ON”.
2) Actuate both sections of the PITCH TRIM switch to activate Manual Electric Pitch Trim (MEPT). Verify the trim clutch engages the trim wheel and trim tab drive in the requested direction, and TRIM is displayed on the autopilot computer flashing while the trim is running. Check operation in both the up and down direction.
3) Activate Co-Pilot’s MEPT switch in the UP/DN direction. Then activate the Pilot’s MEPT switch in the opposite direction. Verify the Pilot’s MEPT switch takes priority.

4) Activate each half of the MEPT switch. Verify the trim clutch does not engage, the trim wheel does not move.
5) Press the AP DISC TRIM INTER switch and hold while actuating the manual electric trim switch. Verify trim does not run and the trim wheel rotates freely when moved manually. Release the switches.
6) Activate the MEPT switch. Remove power from the Pitch Trim servo by pulling the Pitch Trim circuit breaker. Verify trim does not run and the trim wheel rotates freely when moved manually. Release the switches.
7) Restore power to the Pitch Trim Servo.
8) Press the HDG and ALT buttons on the AP computer.
9) Verify that HDG and ALT are displayed on the AP computer and the PFDs AFCS status box, and that the command bars are displayed on the PFDs.
10) Press and hold the CWS switch and verify the control wheel moves freely when moved manually and CWS is displayed on the AP computer.
11) Release the CWS switch and press the AP DISC TRIM INTER switch on the pilots control wheel. Verify the autopilot disengages and RDY is displayed on the autopilot computer flashing for 5 seconds, accompanied by an aural alert. Verify the control wheels move freely in pitch and roll axes.

(f) Yaw Damper

1) Remove power from the aircraft.
2) Ensure the aircraft is level in the lateral (roll) axis.
3) Adjust yaw damper trim control on instrument panel to center of rotation (rotation centered between mechanical stops).
4) Align the cap on the yaw damper trim control on instrument panel such that the white line is centered vertically.
5) Connect Yaw Damper Alignment box between cable harness and yaw amplifier.
6) Connect a voltmeter to the Yaw Damper alignment box.
7) Apply power to aircraft, and select AVION MASTR switch to ON.
8) Select AP master switch to “FD/AP”.
9) Select YAW master switch to “YAW ON”.
10) Select “POT” on yaw damper alignment box.
11) Adjust the Yaw potentiometer on the YD alignment box for a null voltage.
12) Select “SERVO” on the YD alignment box.
13) Verify a null voltage from YD alignment box.
14) If voltage not null, remove silver cap from YD amplifier and adjust the centering potentiometer in the YD amplifier to obtain a null voltage from the YD alignment box. Allow a few seconds after each adjustment for the voltage to settle.
15) Turn Yaw potentiometer on the YD alignment box full counterclockwise.
16) After 10 seconds, note voltage at YD alignment box.
17) Turn Yaw potentiometer on the YD alignment box full clockwise.
18) After 10 seconds, note voltage at YD alignment box.
19) Verify that voltages noted at full rotation are within ±0.3V, ignoring polarity.
20) Remove power from aircraft.
21) Remove YD alignment box and reconnect YD amplifier cable.
22) Apply power to aircraft, and select AVION MASTR switch to ON.
23) Turn YD trim control counter clockwise.

24) Verify left rudder pedal depression.

25) Turn YD trim control clockwise.

26) Verify right rudder depression.

27) Switch Yaw master switch to “YAW OFF”.

(g) Nav Receiver Interface Test

Perform the following test using ramp test equipment. Operate the equipment according to the test equipment manufacturer’s instructions.

NOTE: The PFD HSI does not show a course deviation bar unless a valid VHF NAV frequency is tuned.

1) Select AP Master Switch to “FD/AP”.

2) Simulate a VOR signal on radial 360°. Tune the NAV 1 and NAV 2 receivers to the simulation frequency.

3) Set the HSI on PFD1 to VOR1 by pressing the CDI soft key until VOR1 is selected. Set the HSI on PFD2 to VOR2 by pressing the CDI soft key until VOR2 is selected. Rotate CRS1 and CRS2 knobs to set VOR1 and VOR2 course pointers to 360°. (CDI Sync must be “OFF” on MFD Aux Group, SYSTEM SETUP page)

4) Verify full scale deflection of VOR1 and VOR2 CDI by varying the selected course at least 10° left and right. Reset course pointers to 360°.

5) Engage the autopilot and press the NAV key on the autopilot computer. Using the CRS1 knob alter course by 10° to the right. Verify the flight director and aircraft controls respond by flying to the VOR course. Repeat to the left.

6) Set CRS1 and CRS2 course pointers to 360°.

7) Simulate a Localizer/Glideslope signal. Tune this signal on NAV 1 and NAV 2 receiver. Set the PFD1 HSI to LOC1 and PFD2 HSI to LOC2 by pressing CDI soft key until LOC1 and LOC2 is selected. Use the test equipment to center the deviation bars (localizer and glideslope) on PFD1 and PFD2.

8) Press the APR key on the autopilot computer. Verify that the LOC and GS annunciations are displayed on PFD1 and PFD2. Apply right/left and up/down localizer/glideslope signals using the test equipment. Verify that the Flight Director and flight controls respond appropriately.

I. Panel-Mounted Components

The flight director, HSI, autopilot programmer/computer, altitude selector/alerter (if installed), remote annunciator, and turn coordinator are either face-mounted or rack-mounted in the instrument panel. See 39-10-00 for removal and installation instructions.

J. Component Locator

See “Figure 2”.

K. Trim Monitor

See “Figure 2”.

The trim monitor is mounted to a bracket under the instrument panel on the left side.

(1) Removal
  (a) Disconnect autopilot harness.
  (b) Remove screws (4) holding trim monitor to mounting bracket and remove trim monitor.

(2) Installation
  (a) Place trim monitor in position on mounting bracket and secure with screws (4).
  (b) Connect autopilot harness.
  (c) Perform Post-Maintenance Operational Checkout, above.

L. Pressure Transducer

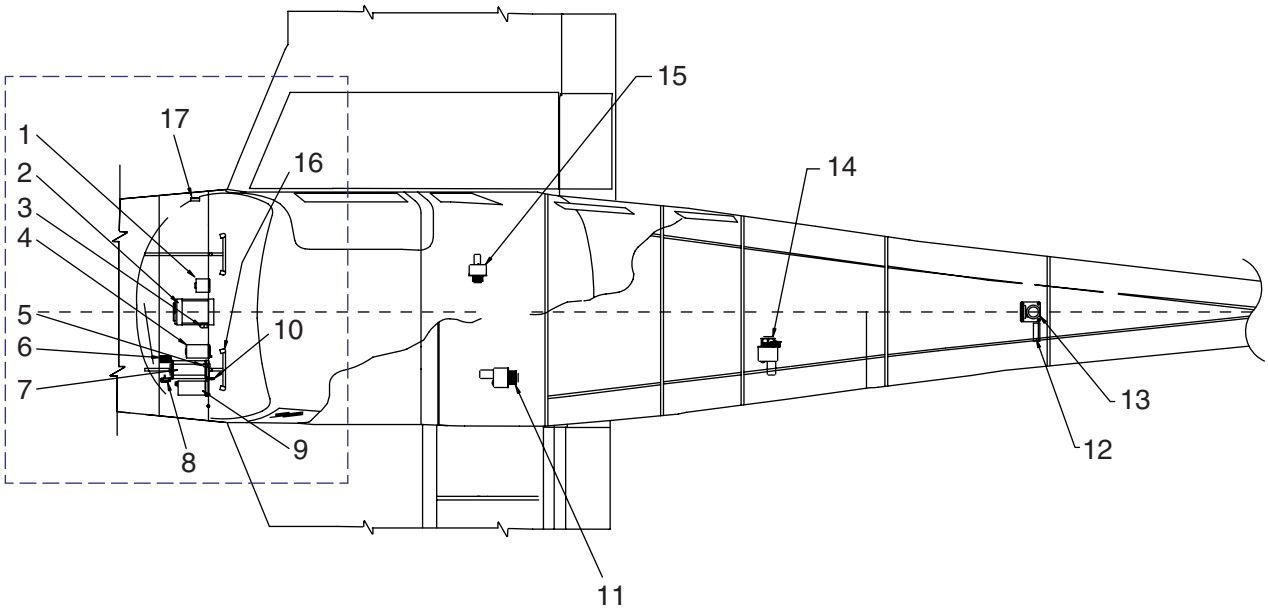
See “Figure 2”.

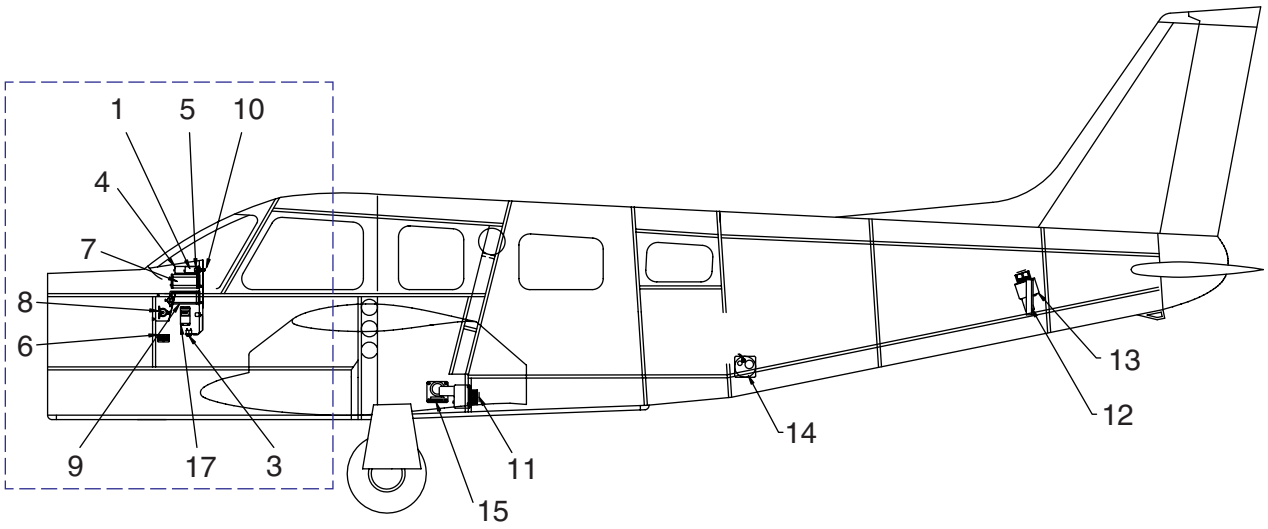
The pressure transducer is located on the aft upper left side of the bulkhead at F.S. 191.00.

(1) Removal
  (a) Remove the Ty-Rap and disconnect the transducer from the static-system by removing the flexible hose.
  (b) Disconnect the autopilot harness.
  (c) Remove screws and washers (2 ea.) and remove transducer.

(2) Installation
  (a) Place transducer in position. Secure transducer to instrument panel with screws and washers (2 ea.)
  (b) Connect the transducer to the static system by sliding the flexible hose over the hose barb. Then position and tighten Ty-Rap.
  (c) Connect the autopilot harness.
  (d) Perform Post-Maintenance Operational Checkout, above.

104584 L
104082 B



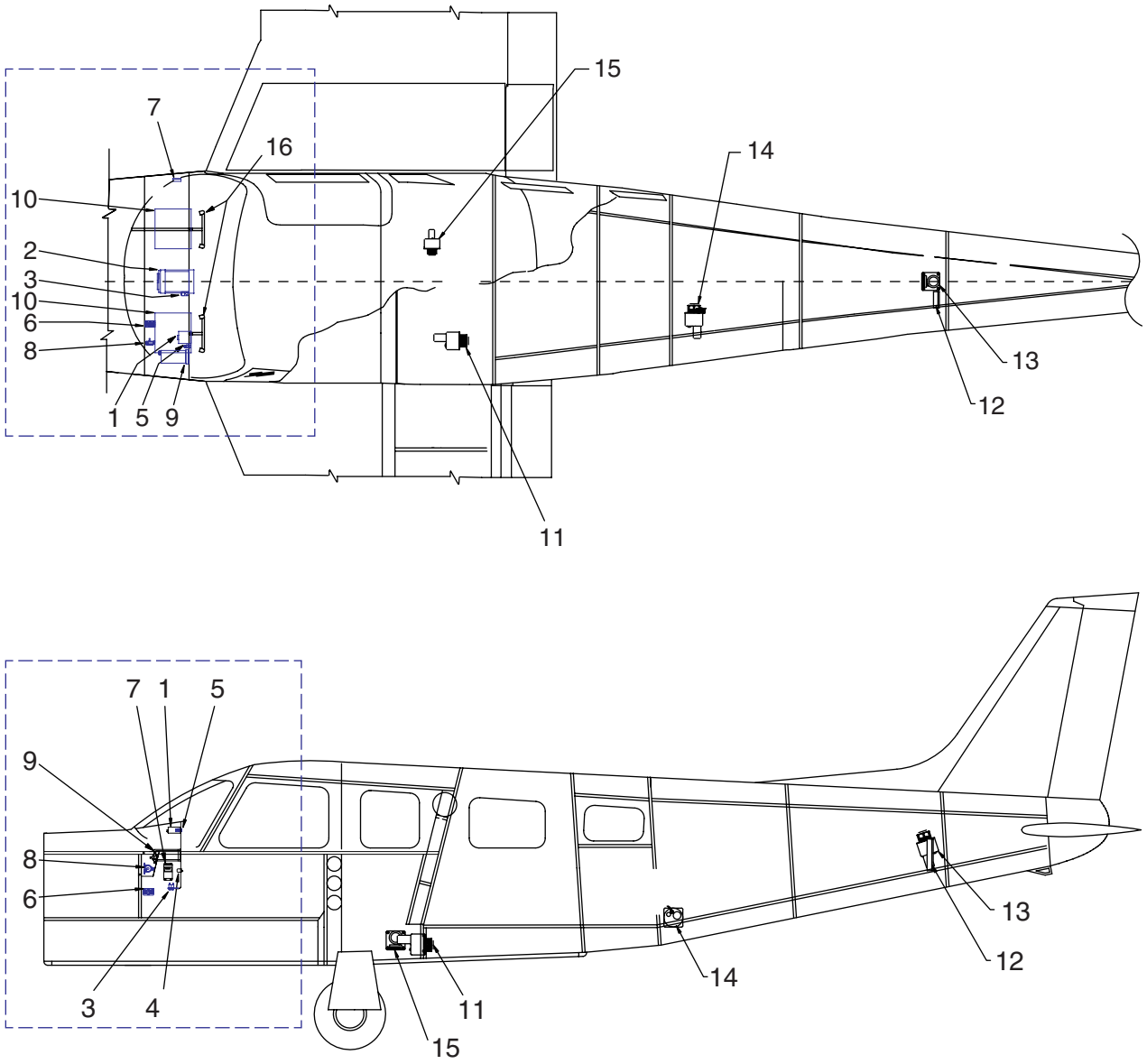


1. ANNUNCIATOR
2. PROGRAMMER/COMPUTER
3. HORN
4. ALTITUDE SELECTOR
5. TRIM/MASTER ANNUNCIATOR SWITCH
6. TRIM RELAY
7. STEERING HORIZON
8. TRANSDUCER
9. TURN COORDINATOR
10. FD PARALLEX POTENTIOMETER
11. ROLL SERVO
12. YAW AMPLIFIER
13. YAW SERVO
14. TRIM SERVO
15. PITCH SERVO
16. CONTROL WHEEL ASSEMBLY
17. TRIM MONITOR

System 55 / 55X Component Locator
Figure 2 (Sheet 1 of 3)

Effectivity
3449152 and up

107356 C



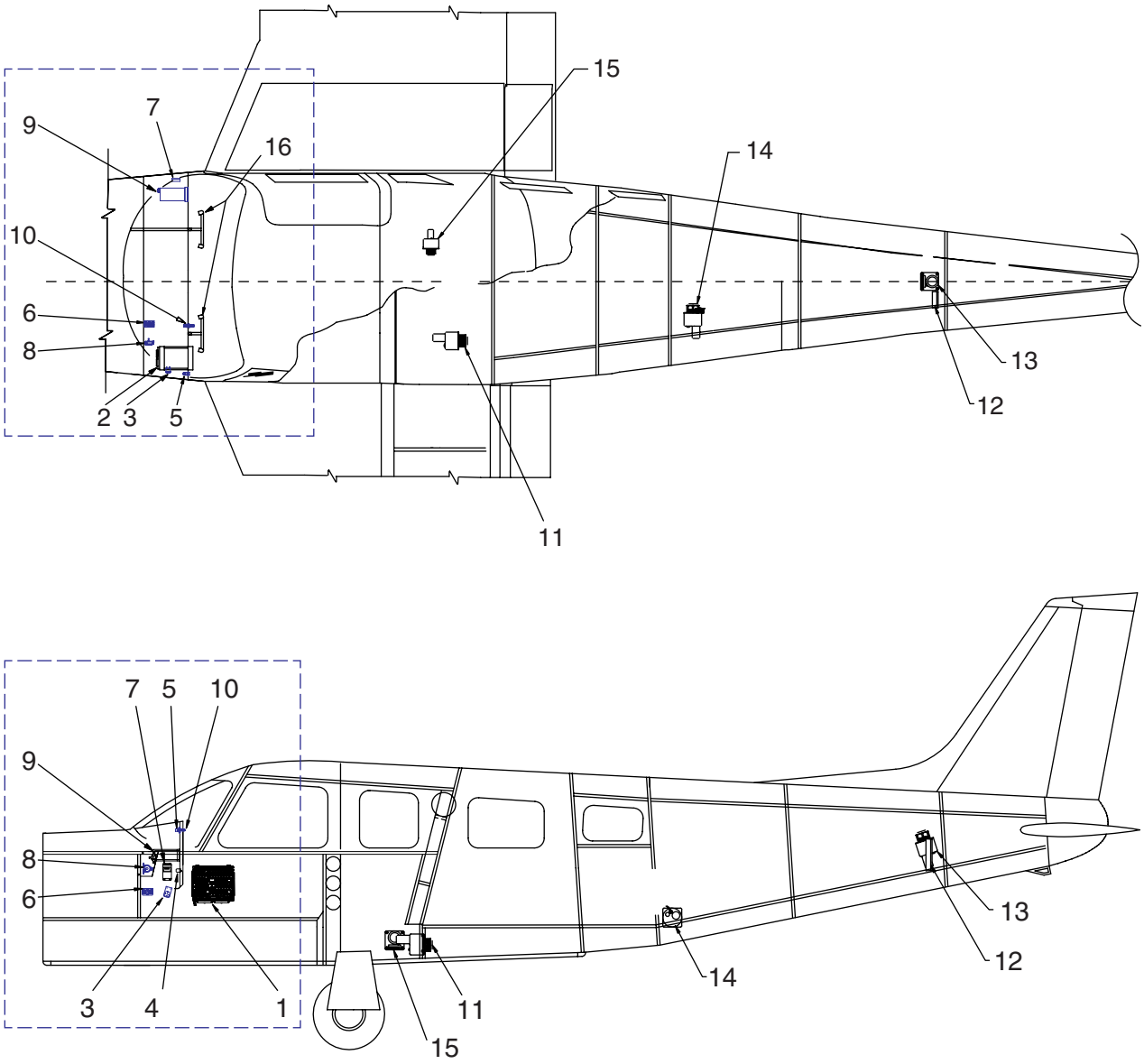
1. ANNUNCIATOR
2. PROGRAMMER/COMPUTER
3. HORN
4. FD/AP AND YAW DAMP SWITCHES
5. TRIM/MASTER ANNUNCIATOR SWITCH
6. TRIM RELAY
7. TRIM MONITOR
8. TRANSDUCER
9. TURN COORDINATOR
10. GDU's #1 AND #2
11. ROLL SERVO
12. YAW DAMPER AMPLIFIER
13. YAW SERVO
14. TRIM SERVO
15. PITCH SERVO
16. CONTROL WHEEL ASSEMBLY

Effectivity
Seneca V
With Garmin G600

System 55 / 55X Component Locator
Figure 2 (Sheet 2 of 3)

Effectivity
Seneca V
with S-TEC Autopilot

107253 1



1. CIRCUIT BREAKERS
2. PROGRAMMER/COMPUTER
3. HORN
4. FD/AP AND YAW DAMP SWITCHES
5. TRIM/MASTER ANNUNCIATOR SWITCH
6. TRIM RELAY
7. TRIM MONITOR
8. TRANSDUCER
9. TURN COORDINATOR
10. YAW POTENTIOMETER
11. ROLL SERVO
12. YAW DAMPER AMPLIFIER
13. YAW SERVO
14. TRIM SERVO
15. PITCH SERVO
16. CONTROL WHEEL ASSEMBLY

System 55 / 55X Component Locator
Figure 2 (Sheet 3 of 3)

Effectivity
Seneca V
With Garmin G1000

M. Roll Servo

See “Figure 2” on page 221025. and “Figure 3”.

The roll servo is mounted underneath the left rear seat. A bridle cable and clamps attach the servo capstan to the aileron balance cable.

(1) Removal

(a) Remove the rear seats.

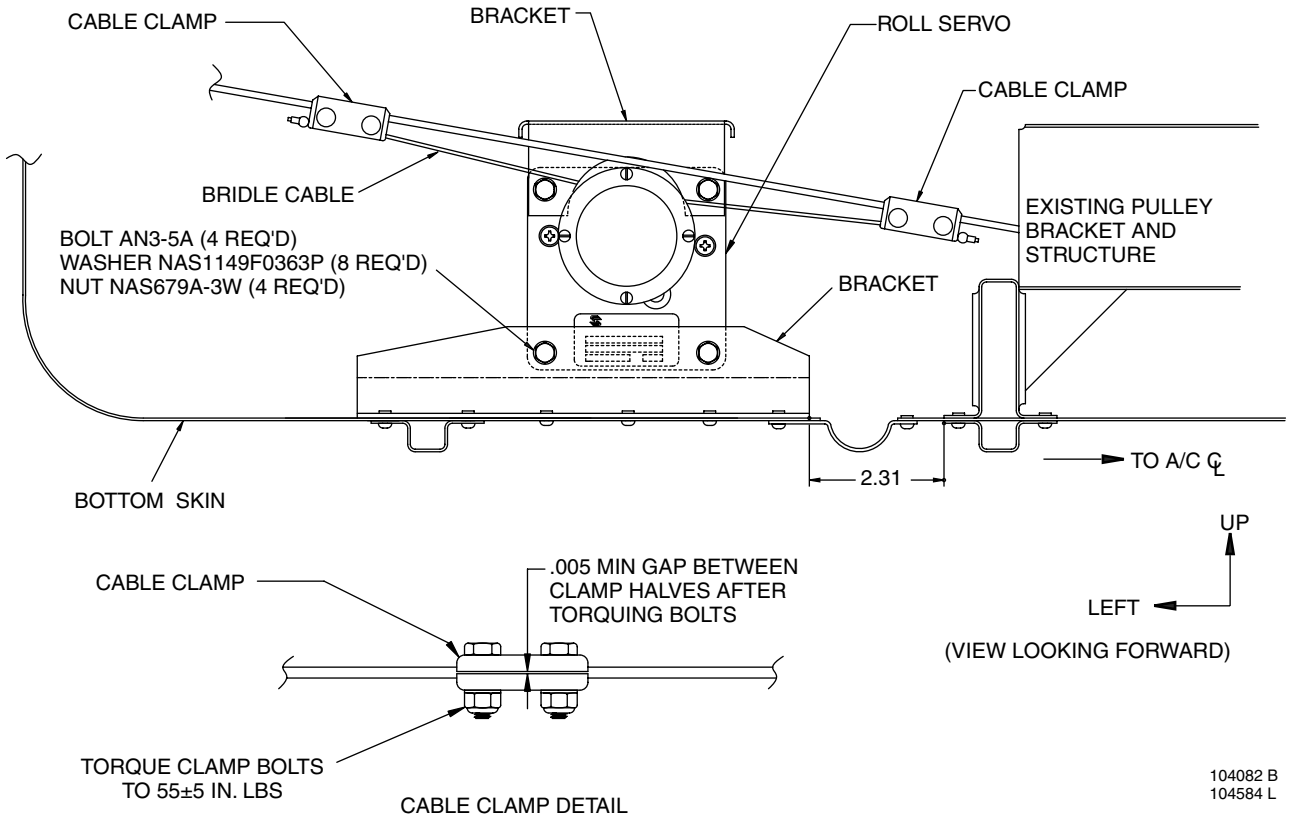
(b) Remove adjacent carpet.

(c) Remove screws securing floor panel and remove panel.

(d) Disconnect autopilot harness.

(e) Remove nuts and bolts (2 ea.) securing each cable clamp (2) and remove cable clamps from aileron balance cable and autopilot bridle cable.

(f) Remove nuts and bolts (4 ea.) and washers (8 ea.) securing roll servo to mounting bracket and remove roll servo with attached bridle cable.



Effectivity
3449152 and up

Roll Servo Installation
Figure 3

Effectivity
Seneca V
with S-TEC Autopilot

(2) Installation

(a) Rig ailerons per Aileron Control Rigging and Adjustment, 27-10-00.

(b) Place the control column tee bar in full forward position and secure by use of a suitable tool or by placing weights on the aft side of the stabilator, if stabilator cables have been previously tensioned.

(c) Lock the ailerons in neutral (i.e. - aligned with flaps) position using a suitable contour fixture at the inboard ends of the ailerons and the outboard ends of the flaps. Verify control wheels are centered and secure in that position.

(d) Remove screws (4) and remove capstan cover and cable guards from servo.

(e) Adjust roll servo clutch torque per Servo Clutch Torque Adjustment, below.

(f) Wrap autopilot bridle cable, align capstan, and tighten center-ball setscrew as shown in “Figure 4” on page 221029.

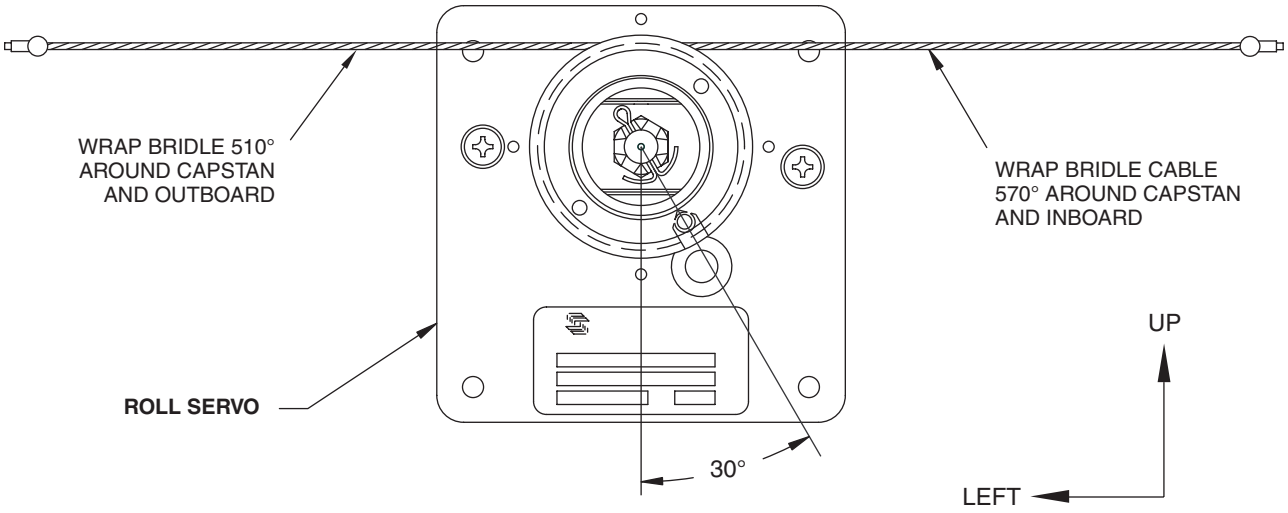
(g) Replace cable guards and capstan cover, secure with screws (4).

(h) Position servo as shown in Figure 3 and install and secure nuts (4 ea.), washers (8 ea.), and bolts (4 ea.) holding servo to mounting bracket.

(i) Position cable clamps (2) as shown in “Figure 3” on page 221028 and tighten nuts and bolts (2 ea.). Adjust cable clamps in or out along the aileron balance cable to obtain a bridle cable tension of 15 + 10, -2 lbs (System 55X) or 15 ± 2 lbs (System 55). Torque cable clamp bolts to 55 ± 5 in. lbs.

(j) Remove the locking fixtures at the inboard ends of the ailerons. Aileron neutral (i.e. - aligned with flaps) position should be maintained with the control wheels in neutral. A droop of 1/8 inch is allowable.

(k) Remove the control wheel/tee bar locks. Check to ensure that the left aileron up and right aileron down stops are contacted simultaneously and vice versa. Adjust stops as required.



(CAPSTAN COVER AND CABLE GUARDS REMOVED FOR CLARITY)

Roll Servo Capstan Wrapping
Figure 4

Effectivity
3449152 and up

(l) Rotate the left (pilot's) control wheel in each direction until the bellcranks contact the stops. The sprocket stops on the tee bar shall not be contacted until additional "override" movement (cushion) of the wheel occurs. A "cushion" on 0.030 to 0.040 inches is to be maintained as measured between the sprocket pin and adjustable control wheel stop bolts.

(m) Place the ailerons in the neutral (aligned with the flaps) position. For each aileron, from the neutral position, check that the "up" travel and the "down" travel are within the limits as specified in 27-00-00, Chart 2.

1) Center bubble of a protractor over surface of aileron at neutral position. Note reading.

2) Move aileron full up and down. Check degree of travel in each direction. Degree of travel on protractor is determined by taking the difference between protractor reading at neutral and up, and neutral and down. Bubble must be centered at each reading.

When measuring "down" travel from the neutral position, a light "up" pressure shall be maintained at the center of the aft edge of the aileron. When measuring "up" travel from the neutral position, a light "down" pressure shall be maintained at the center of the aft edge of the aileron (at the "up" position only), just sufficient to remove the slack between the bellcrank and the aileron.

(n) If steps (j) thru (m), above, reveal the aileron controls out of rig, repeat steps (a) thru (i).

(o) Connect autopilot harness.

(p) Check aileron controls for free and correct movement.

(q) Perform Post-Maintenance Operational Checkout, above.

(r) Replace floor panel and secure with screws.

(s) Replace carpeting.

(t) Replace the rear seats.

N. Trim Servo

See “Figure 2” on page 221025 and “Figure 5” on page 221032.

The trim servo is located just left of the aircraft centerline, just aft of the bulkhead at F.S. 187.80 and is accessed through the aft fuselage access door. The left stabilator trim cable wraps around the servo idler pulley and servo capstan.

(1) Removal

(a) Remove aft fuselage access door.

(b) Tie a pull rope to the left stabilator trim cable forward of the servo and tie-off the pull rope to structure aft.

NOTE: This may be easier to accomplish by removing the rear seats, carpet and floor panel.

(c) Tie a pull rope to the left stabilator trim cable aft of the turnbuckle in the rear fuselage aft of the trim servo and tie-off the pull rope to structure forward.

NOTE: The pull ropes apply tension to the trim cables to prevent the cables from unwrapping from the trim wheel drum or the trim barrel, and to prevent the cables from fouling at any of the pulleys.

(d) Slack-off the turnbuckle in the left stabilator trim cable segment aft of the trim servo sufficient to relieve tension on the left stabilator trim cable as it wraps around the trim servo idler pulley and capstan.

(e) Disconnect the autopilot harness.

(f) Remove the capstan cover and cable guards (4) by removing the retaining screws (4).

(g) Remove the bolt, nut, and washer securing the idler pulley to the trim servo baseplate and mounting bracket and remove the idler pulley components.

NOTE: The idler pulley breaks down into the following components upon removal of the bolt, above: mounting plate/cable guard assembly, idler pulley, and two washers.

(h) Remove the remaining bolts, nuts, and washers (3 ea.) securing the trim servo to its mounting bracket and remove the trim servo.

(2) Installation

(a) Adjust trim servo clutch torque per Servo Clutch Torque Adjustment, below.

(b) With the capstan cover and cable guards removed, position the trim servo as shown in Figure 5. Secure with bolts, nuts, and washers (3 ea. - i.e. - aft two and forward left).

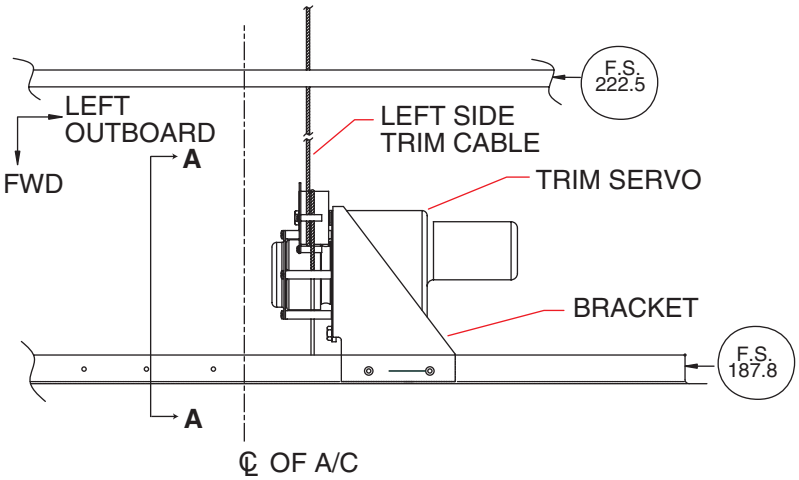
(c) Assemble the idler pulley cable guards (3) to the mounting plate with screws (1 ea.) and star washers (2 ea.). Place the center bolt through the mounting plate/cable guard assembly and slide a washer over the threaded end and up against the mounting plate. Set the mounting plate/cable guard/bolt assembly aside.

(d) Drape the slack left stabilator trim cable over the servo capstan.

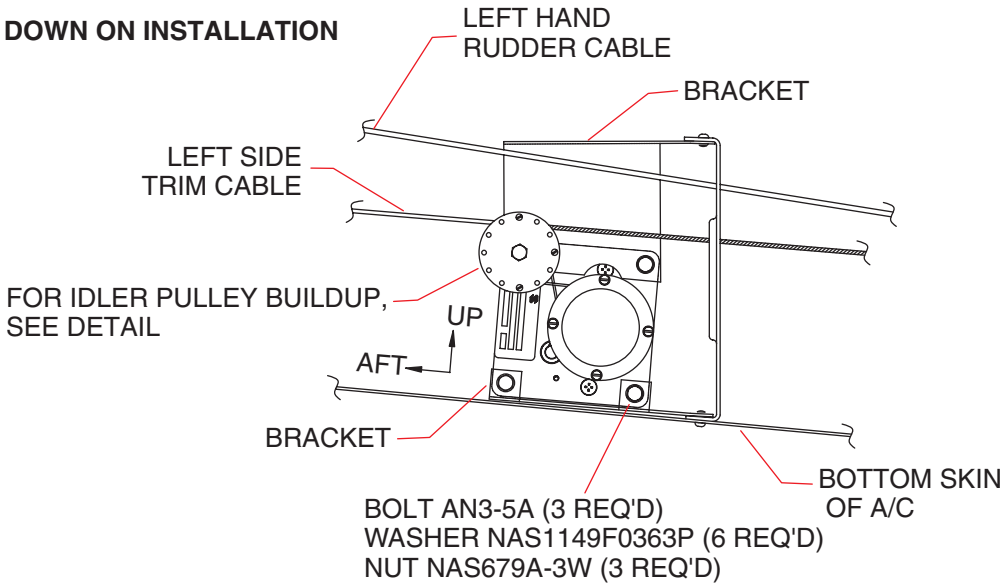
CAUTION: IN STEPS (e) THRU (i), BELOW, USAGE OF LEFT AND RIGHT IS RELATIVE TO THE VIEW OF THE TECHNICIAN IN THE CABIN BAGGAGE AREA LOOKING AFT, EXCEPT WHERE AIRPLANE COMPONENT PARTS ARE SPECIFICALLY NAMED.

(e) Place thumb and forefinger on top of the capstan over the trim cable in its groove. Pressing the trim cable into its groove, slide thumb and forefinger down around opposite sides of the servo capstan and pull the trim cable slack towards you and to your left.

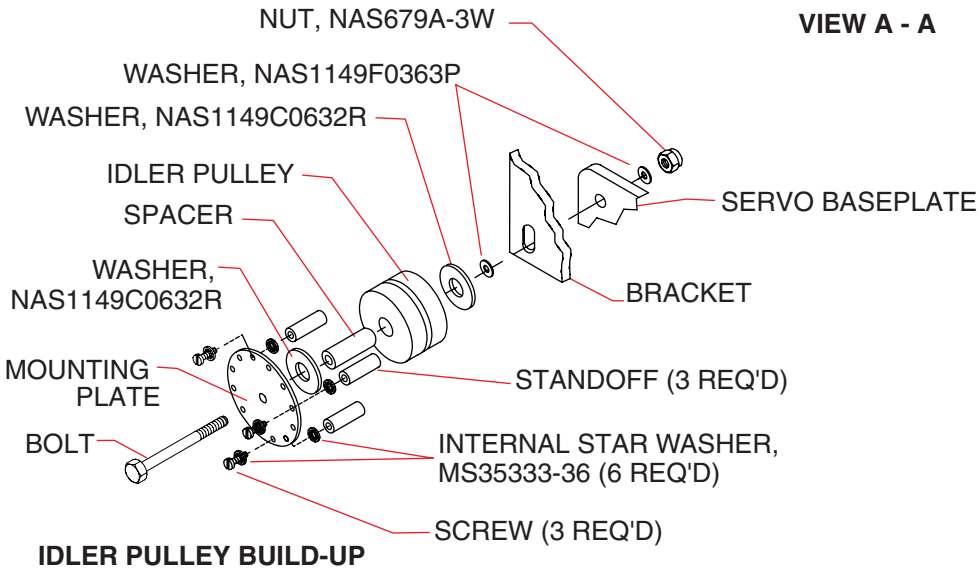
(f) Holding the trim cable in that position, install the capstan cover and cable guards as shown in “Figure 5” on page 221032.



VIEW LOOKING DOWN ON INSTALLATION



VIEW A - A



IDLER PULLEY BUILD-UP

104082 B
104584 L

Effectivity
3449152 and up

Trim Servo Installation
Figure 5

Effectivity
Seneca V
with S-TEC Autopilot

(g) Hold the idler pulley aft of the trim servo and to the right of the airplane's left stabilator trim cable. Move the idler pulley left to the left stabilator trim cable and capture the trim cable in the bottom cable groove on the left of the idler pulley.

(h) Keeping the trim cable in the bottom groove, slide the idler pulley forward along the trim cable, left of the servo capstan, and bring it approximately to its installed position (See "Figure 5" on page 221032). At this point, the aft portion of the trim cable should be routed left, around the front, and to the right of the idler pulley and to the left, around the rear, and to the right of the capstan.

(i) Holding the idler pulley in this position, reach down and pull the forward portion of the trim cable over the idler pulley and seat it in the top cable groove on the idler pulley. The left trim cable should now be routed as shown in "Figure 5" on page 221032.

(j) Place the spacer inside the idler pulley.

(k) Position the mounting plate/cable guard/bolt assembly as shown in "Figure 5" on page 221032 and slide the bolt through the spacer inside the idler pulley.

(l) Place a washer over the bolt end and put the bolt through the trim servo mounting bracket and baseplate. Secure with a nut and washer, taking care to ensure that the cable guards are positioned, and the left stabilator trim cable is routed, as shown in "Figure 5" on page 221032.

(m) Take up the slack in the left trim cable with the turnbuckle and remove the pull ropes.

(n) Rig stabilator trim per Stabilator Trim Rigging and Adjustment, 27-30-00.

(o) Connect the autopilot harness.

(p) Perform Post-Maintenance Operational Checkout, above.

(q) Check elevator trim controls for free and correct movement.

(r) Reinstall aft fuselage access door. If removed, reinstall floor panel, carpet, and rear seats.

O. Pitch Servo

See “Figure 2” on page 221025 and “Figure 6” on page 221035 .

The pitch servo is located under the right rear seat. A bridle cable and clamps attach the servo capstan to the right (upper) stabilator cable.

(1) Removal

(a) Remove rear seats.

(b) Remove carpet and floor panel.

(c) Disconnect autopilot harness.

(d) Remove nuts and bolts (2 ea.) securing each cable clamp (2) and remove cable clamps from right (upper) stabilator cable and autopilot bridle cable.

(e) Remove nuts and bolts (4 ea.) and washers (8 ea.), securing pitch servo to mounting bracket and remove pitch servo with attached bridle cable.

(2) Installation

(a) Rig stabilator controls per Stabilator Controls Rigging and Adjustment, 27-30-00.

(b) Remove screws (4) and remove capstan cover and cable guards from servo.

(c) Adjust pitch servo clutch torque per Servo Clutch Torque Adjustment, below.

(d) Wrap autopilot bridle cable, align capstan, and tighten center-ball setscrew as shown in “Figure 6” on page 221035 .

(e) Replace cable guards and capstan cover, secure with screws (4).

(f) Position pitch servo as shown in “Figure 6” on page 221035 and secure with bolts, nuts, and washers (4 ea.).

(g) Position cable clamps (2) as shown in “Figure 6” on page 221035 and tighten nuts and bolts (2 ea.). Adjust cable clamps in or out along the stabilator cables to obtain a bridle cable tension of 15 ± 2 lbs (System 55) or 15 + 10, -2 lbs (System 55X). Torque cable clamp bolts to 55 ± 5 in. lbs.

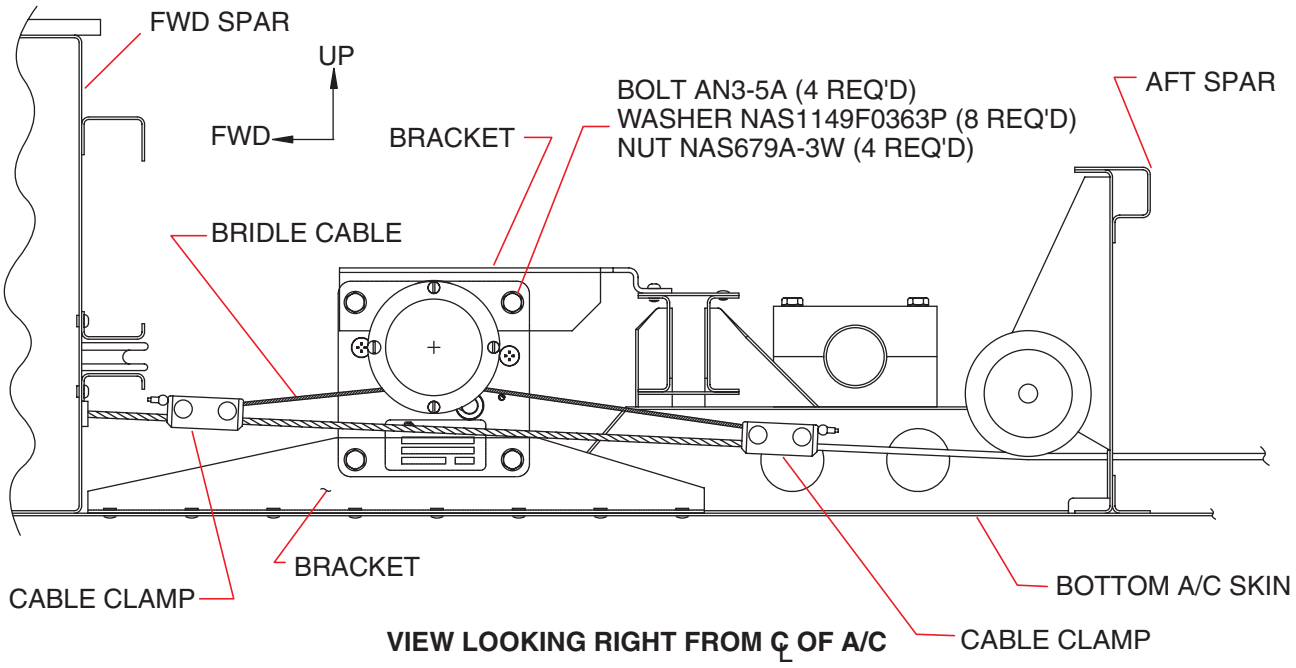
(h) Connect autopilot harness.

(i) Perform Post-Maintenance Operational Checkout, above.

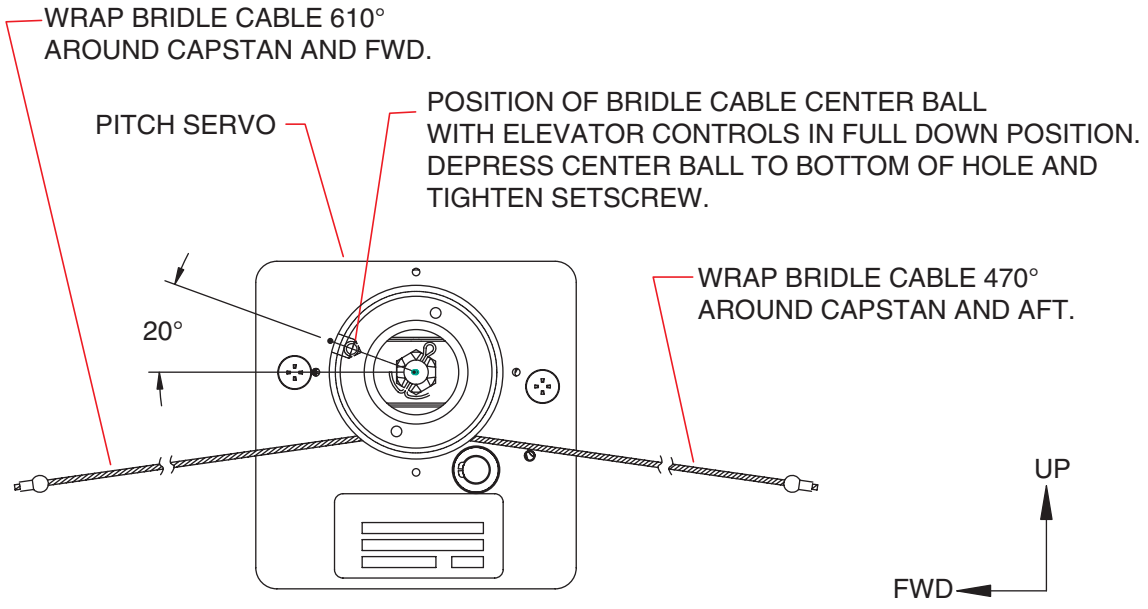
(j) Check stabilator controls for free and correct movement.

(k) Reinstall floor panel and carpet.

(l) Reinstall rear seats.



VIEW LOOKING RIGHT FROM $\mathbb{C}_L$ OF A/C



BRIDLE CABLE WRAPPING DETAIL
(CAPSTAN COVER REMOVED)

101817 G
104082 B
104584 L
107253

Pitch Servo Installation
Figure 6

Effectivity
3449152 and up

Effectivity
Seneca V
with S-TEC Autopilot

P. Servo Clutch Torque Adjustment

See “Figure 7” on page 221037 and “Chart 1”.

(1) Remove servo per instructions under specific servo, above.
(2) Place servo in a holding fixture (i.e. - vice) with capstan up.
(3) Remove capstan cover, cable guards, and cable.
(4) Check capstan torque by attaching the capstan adjusting tool (special tool - see parts catalog) to the capstan and using a currently calibrated torque wrench as shown in “Figure 7” on page 221037.
  (a) Acceptable torque is specified in “Chart 1”.
  (b) If adjustment is required, proceed as follows.
(5) Remove cotter pin from end of servo shaft and remove castle nut, shim washers, and tension washers.
(6) Replace tension washers as required (see “Chart 1”).
(7) Replace shim washers and castle nut.
(8) Tension castle nut so that capstan torque is as specified in “Chart 1”.

Q. Flap Compensator

A flap compensator potentiometer (pot) is mounted underneath the right rear seat. The pot is mounted on a bracket attached to the forward side of the pitch servo mounting bracket. The arm of the pot is linked to the flap torque tube by a cable assembly.

Adjustment

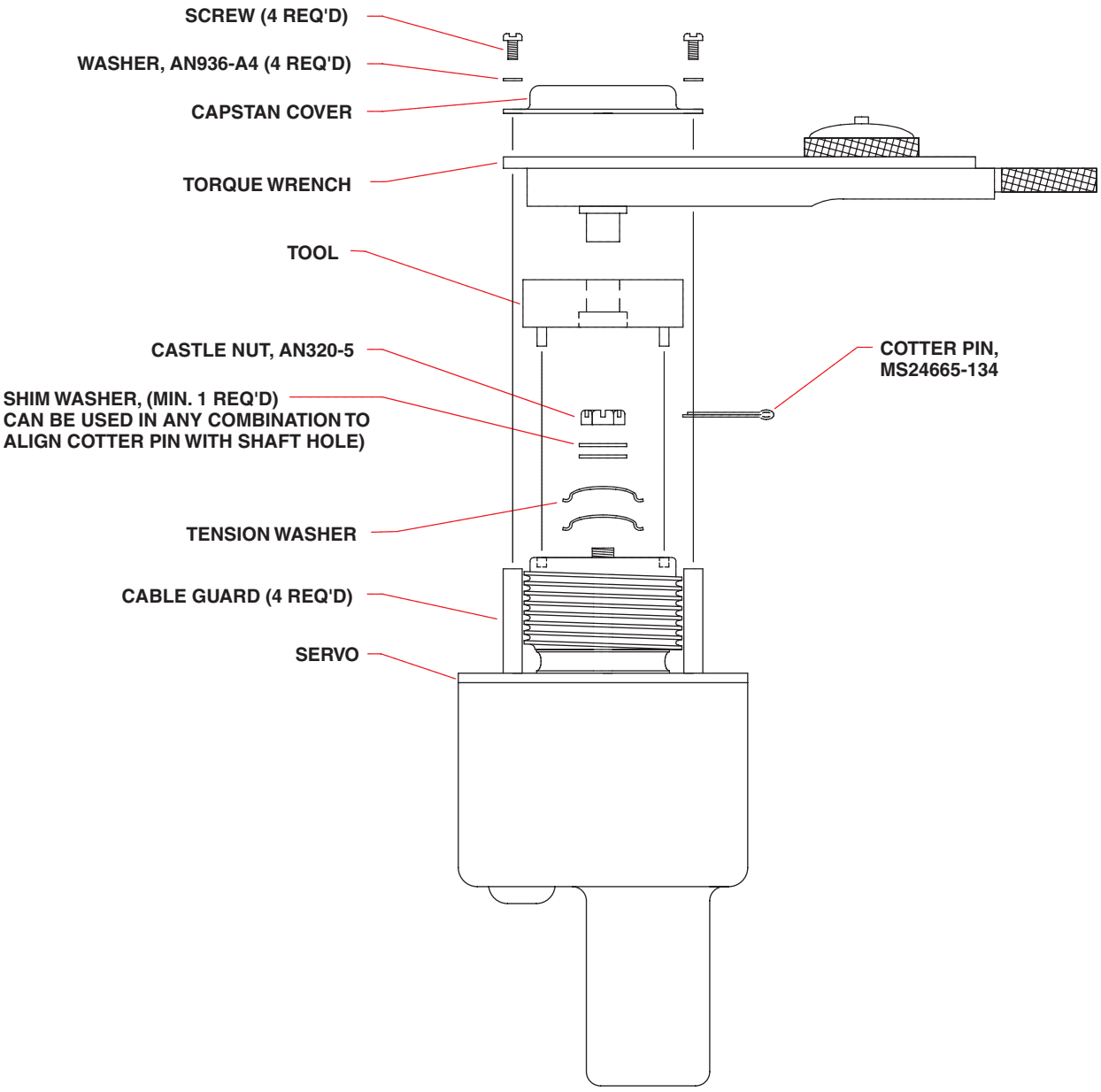
(a) With flaps in the full up position and set screw loose, turn A/P master switch ON.
(b) Connect a digital voltmeter (3 1/2 digit) between airframe ground and center terminal (wiper) of potentiometer (pot).
(c) Turn pot shaft clockwise to stop. Voltmeter should read 5.00 vdc.
(d) Turn pot shaft slowly counter-clockwise until voltage just starts to decrease from 5.00 volts.
(e) Tighten set screw and recheck wiper voltage for 4.95 to 5.00 vdc.

CHART 1
SERVO CLUTCH TORQUE ADJUSTMENT

SERVO	TORQUE (In. Lbs.)	CLUTCH WASHERS	
		System 55	System 55X
Roll	40 ± 3 (1)	0.040 (3 ea.)	0.032 (1 ea.) 0.040 (2 ea.)
Pitch	45 ± 3 (1)	0.032 (3 ea.) 0.040 (1 ea.)	0.040 (3 ea.)
Trim	20 ± 3 (1)	0.040 (2 ea.)	0.040 (2 ea.)
Yaw	50 ± 3 (1)	0.032 (3 ea.) 0.040 (1 ea.)	0.040 (3 ea.)

(1) ± 2 for System 55 installations.

104082 B
104584 L



Servo Clutch Torque Adjustment
Figure 7

Effectivity
3449152 and up

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3. Garmin GFC 700 Automatic Flight Control System (AFCS) (Seneca V only.)

S/N's 3449459, 3449492 and up (2014 and up)

A. Description

See "Figure 8" on page 221048 and also see 34-25-01 (G1000) or 34-25-02 (G1000 NXi).

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY.

CAUTION: THIS SECTION PROVIDES INFORMATION SUFFICIENT ONLY TO LOCATE, REMOVE, AND INSTALL COMPONENTS. DETAILED TROUBLESHOOTING OR REPAIR IS REQUIRED, IT SHOULD BE PERFORMED BY A QUALIFIED AVIONICS REPAIR SHOP USING THE APPROPRIATE MAINTENANCE MANUALS.

The GFC 700 Automatic Flight Control System (AFCS) provides a three-axis fail-safe digital flight control system with Flight Director, Autopilot, Pitch Auto-Trim and Yaw Damper (optional) functions. The GFC 700 functionality is distributed across several units of the G1000 / G1000 NXi avionics systems and is designed for precision approach to Category 1 minimums. The controls are located on the MFD bezel. The mode logic and Flight Director computations are performed within the GIA 63W (G1000) or GIA 64 (G1000 NXi) Integrated Avionics units. The autopilot computations and monitoring are performed within the GSA 8X Servo Actuators with GSM 86 Servo Gearboxes. Dual Control Display Units and Integrated Avionics units provide redundancy.

B. Troubleshooting

(1) See "Ground Checks" on page 221039, below, and "Return to Service" on page 221049.

(2) See the following under Integrated Avionics System - Garmin G1000 in 34-25-01 (G1000) and 34-25-02 (G1000 NXi):

(a) Troubleshooting

(b) LRU Test Procedures

C. Ground Checks

(PIR-107867, Rev. C/34G34A020 C.)

(1) Setup

(a) Ensure G1000 / G1000 NXi software is loaded, properly configured, and checked out per 34-25-01 (G1000) and 34-25-02 (G1000 NXi).

(b) Level the airplane per 8-20-00.

(c) Apply external power.

(2) References

(a) Garmin G1000 Cockpit Reference Guide for Piper PA-34-220T Seneca V, 190-01405-00 and Garmin G1000 NXi Cockpit Reference Guide for Piper PA-34-220T Seneca V, 190-02478-00.

(b) Appropriate Pilot's Operating Handbook (POH) serialized to the airplane.

(c) Avionics/Electrical System Installation schematics found in Chapter 91.

NOTE: In the sections that follow, an Autopilot disconnect should be accompanied by an aural alert (two-second tone) unless otherwise specified.

(3) Pre-Flight Test (PFT)

(a) Set the AVION MASTR switch to OFF. After 30 seconds select the AVION MASTR switch to ON. Verify the GFC 700 begins an automatic pre-flight test after AHRS and ADC parameters become valid.

(b) Verify that a white 'PFT' annunciation is displayed on PFD1 and PFD2. A momentarily red 'AFCS' annunciation displayed before PFT starts is acceptable.

(c) Upon successful completion of the test, an aural alert will sound and the annunciation will clear. If the 'PFT' annunciation turns red, the test has failed. See "Troubleshooting" on page 221039.

(4) AFCS Switch Checks

Verify that the AFCS system buttons and switches are operating correctly by performing the following steps (for both the pilot and copilot control yokes where applicable).

(a) Actuate both sections of the PITCH TRIM (TRIM UP/TRIM DN) switch to activate Manual Electric Pitch Trim (MEPT). Verify the trim clutch engages and the trim wheel and trim tab drive in the requested direction. Verify TRIM is annunciated on both PFDs while trim is in motion. Check operation in both the up and down direction.

(b) Activate Co-Pilot's MEPT switch in the UP/DN direction. Then activate the Pilot's MEPT switch in the opposite direction. Verify the Pilot's MEPT switch takes priority.

(c) Activate each half of the MEPT switch. Verify the trim clutch does not engage, the trim wheel does not move, and PTRM annunciation on the PFD after approximately 3 seconds.

(d) Press the AP DISC TRIM INTER switch and hold while actuating the manual electric trim switch. Verify trim does not run and the trim wheel rotates freely when moved manually. Release the switches.

(e) Activate the MEPT switch. Remove power from the Pitch Trim servo by pulling the Pitch Trim circuit breaker. Verify trim does not run and the trim wheel rotates freely when moved manually. Release the switches.

(f) Pull AUTOPILOT circuit breaker. Reset the PITCH TRIM and AUTOPILOT circuit breakers.

(g) Engage the autopilot by pressing the AP key on the MFD. Verify the pitch and roll clutches engage and resist movement of the control wheel. Press and hold the CWS switch and verify the control wheel moves freely when moved manually.

(h) Release the CWS switch and press the AP DISC TRIM INTER switch on the pilots control wheel. Verify the autopilot disengages with a flashing amber 'AP' annunciation on PFD1 and PFD2, accompanied by an aural alert. Verify that the control wheel is free in pitch and roll axes.

(i) Press the TO/GA button on the power lever. Verify 'TO' is annunciated on PFD1 and PFD2 for both PITCH and ROLL modes and the command bars should be at ten (10) degrees nose up and wings-level.

(j) Press the Flight Director (FD) key on the MFD to deactivate the TO mode. Press the AP key to engage the autopilot. Press the CWS button for a minimum of 5 seconds and release; verifying there is no residual force on the control stick for the pitch and roll axis while the CWS is pressed. Servos reengage after CWS is released.

(k) Disengage the autopilot by pressing the AP DISC TRIM INTER switch on the co-pilot's control wheel. Engage VS mode by pressing the VS key on the MFD. Verify PFD1 and PFD2 display 'VS' in green and indicates a pitch reference of '0 FPM'.

NOTE: The ALT Select knob must be activated for the FLC key to function properly.

(l) Press the FLC key on the MFD and verify that 'FLC' is annunciated on PFD1 and PFD2 in green.

(m) Press the ALT key on the MFD and verify that the 'ALT' annunciation is displayed in green on PFD1 and PFD2 with an altitude reference equal to the aircraft altitude (within the nearest 20 feet).

(n) Press the FD key and verify that the mode annunciations and command bars are removed from the display.

(5) Autopilot Disconnect Checks – Normal Mode

Engage the autopilot by pressing the AP key on MFD prior to each of the following checks and reset the circuit breaker (if required) prior to the subsequent check. For each check, verify a five-second flashing amber AP annunciation, a three-second disconnect aural alert, and FD bars remaining on display.

(a) Press the MEPT switch on the pilot's control yoke.

(b) Press and hold the left section of the MEPT switch on the pilot's control yoke. Verify the trim wheel rotates freely when moved manually.

(c) Press and hold the right section of the MEPT switch on the pilot's control yoke. Verify the trim wheel rotates freely when moved manually.

(d) Press the MEPT switch on the co-pilot's control yoke.

(e) Press and hold the left section of the MEPT switch on the co-pilot's control yoke. Verify the trim wheel rotates freely when moved manually.

(f) Press and hold the right section of the MEPT switch on the co-pilot's control yoke. Verify the trim wheel rotates freely when moved manually.

(g) Press TO/GA button on the throttle handle. (Press the FD key to remove the TO mode annunciation from the display).

(h) Press AP button on the MFD.

(6) Autopilot and Yaw Damper Disconnect Checks – Normal Mode

Engage the autopilot by pressing the AP key on the MFD prior to each of the following checks and reset the circuit breaker (if required) prior to the subsequent check. For each check, verify five-second flashing amber AP and YD (if installed) annunciations, a three-second disconnect aural alert, and FD bars remaining on display.

(a) Press the A/P DISC TRIM INTER switch on the pilot's control yoke.

(b) Press the A/P DISC TRIM INTER switch on the co-pilot's control yoke.

(c) Pull the AUTOPILOT circuit breaker. After check is complete, pull the PITCH TRIM circuit breaker and then reset both breakers.

(7) Autopilot and Yaw Damper Disconnect Checks – Abnormal Mode

Engage the autopilot by pressing the AP key on AFCS mode controller prior to each of the following checks and reset the circuit breaker (if required) prior to the subsequent check. For each check, verify a continuous flashing red/white AP and a five second yellow YD (if installed) annunciation, a red boxed AFCS annunciation (unless specified otherwise), and a continuous disconnect aural alert (until silenced by the A/P DISC TRIM INTER switch).

NOTE: After resetting each circuit breaker, allow time for the AHRS or GPS to initialize prior to conducting the next text.

(a) Pull the INTEG AV 1 circuit breaker. Verify FD bars are removed from PFDs. Various CAS messages may appear due to GIA 1 powering down.

(b) Pull the INTEG AV 2 circuit breaker. Verify FD bars remain on PFDs. Various CAS messages may appear due to GIA 2 powering down.

(c) Pull the AHRS 1 circuit breaker. Verify FD bars are removed. Verify red boxed AFCS annunciation does not appear. If dual AHRS/ADC and YD are installed, verify YD is not flashing until A/P DISC TRIM INTER switch is activated.

For G1000 only:

(d) Pull the PFD 1 circuit breaker. Verify red boxed AFCS annunciation does not appear.

(8) Yaw Damper Disconnect Checks – Manual “Normal” Method

(a) Engage the autopilot by pressing the AP key on MFD.

(b) Press the YD button.

(c) Verify a five-second yellow YD annunciation, and FD bars remaining on display.

(9) Autopilot Clutch Overpower Check

The GFC 700 uses electronic torque limiting as well as mechanical slip clutches to limit the maximum servo effort. When the system is on the ground, the electronic torque limiting is removed, allowing manual checks of the slip-clutch settings.

(a) Engage the Autopilot by pressing the AP key on the MFD.

(b) Manually overpower the autopilot clutches in pitch and roll. If any Autopilot clutch cannot be overpowered, verify the correct servo clutch is installed. See “Chart 2” and “Figure 10” on page 221050. If the correct clutch is installed and cannot be overpowered, replace the clutch.

(c) Actuate and hold PITCH TRIM switch in either the NOSE UP or NOSE DOWN direction to disconnect the autopilot. While the trim is running, restrain the aircraft pitch trim wheel and verify that the trim clutch can be overpowered. If it cannot be overpowered, verify the correct trim servo clutch is installed. See “Chart 2” and “Figure 10” on page 221050. If the correct clutch is installed and cannot be overpowered, replace the clutch.

(d) Engage the autopilot by pressing the AP key on the MFD. Actuate and hold the manual electric trim switch in either the up or down direction to disconnect the autopilot. Verify that the trim wheel moves smoothly in both directions throughout the entire trim range during manual electric trim operation. If the trim wheel hesitates, this may indicate that the pitch trim clutch is slipping and proper clutch setting and cable tension should be verified. See “Troubleshooting” on page 221039, “Trim Servo” on page 221060, and “Chart 2”, below. If clutch is as specified and cable tension is within tolerance, check the aircraft pitch trim system for excessive friction. See Rigging and Adjustment under Elevator Trim Controls in 27-30-00. Replace clutch as required.

CHART 2
GSM 86 SERVO CLUTCH TORQUE SETTINGS

Servo	Clutch Part Number (Piper/Garmin)	Torque (In.-Lbs.)
Pitch	PS50193-70 / 011-02147-12	72
Pitch Trim	PS50193-69 / 011-02147-05	24
Roll	PS50193-57 / 011-02147-07	28
Yaw	PS50193-59 / 011-02147-03	90

(PIR-107850 A.)

(10) Manual Electric Pitch Trim Speed Check

(a) Run the MANUAL ELECTRIC PITCH TRIM in one direction until it runs against the mechanical stop.

(b) Run the trim in the opposite direction and, using a stop watch or equivalent device, time the trim speed to the opposite mechanical stop. Verify that the elapsed time for full travel measures 30 +/- 5 seconds.

(11) Autopilot Operation Checks

(a) Engage the Autopilot by pressing the AP key on the MFD. Push the HDG knob to synchronize the heading bug to the current aircraft heading. Select HDG mode by pressing the HDG key on the MFD. Verify the command bars are level and the control wheel is stationary. (There may be some roll motion in the yoke if the aircraft not perfectly level.)

(b) Turn the HDG knob to the left and right and check that the command bars move in the correct direction and the control wheel follows the command bars. Push the HDG knob.

(c) Push and hold the CWS button and pull the control wheel to the center of the pitch control range. Release the CWS button. Verify the autopilot clutches re-engage and hold the wheel stationary.

(d) Holding the control wheel lightly, move the NOSE UP key on the MFD to increase the pitch reference. Verify the command bars move up ~0.5 degree and the control wheel begins moving aft. (In some aircraft, the down spring may require manual assistance to get aft control stick movement).

(e) While holding the control wheel firmly, press and hold the CWS button to re-synchronize the pitch reference. Re-center the control wheel to wings level and mid-range elevator travel. Release the CWS button and check that servo clutches re-engage before releasing the control wheel.

(f) Press the NOSE DN key on the MFD. Verify the command bars command a pitch attitude ~0.5 degree lower and the control wheel begins moving forward. Hold the controls and press CWS to re-center the command bars and stop control wheel movement.

(g) With the Autopilot still engaged and the CWS button pressed, move the control wheel to its aft limit. Release the CWS button and apply continuous forward pressure, slowly moving the control wheel. After a brief delay, verify the trim wheel begins moving in a trim up direction.

(h) Grip the control wheel and press the CWS button. Verify trim motion stops. Move the control wheel to the forward limit and release the CWS button. Slowly move the control wheel aft. After a brief delay, verify the trim wheel begins to trim down. Relieve pressure on the wheel and verify the trim motion stops. Check that the trim wheel is free to turn. Hold the control wheel and press the A/P DISC TRIM INTER switch to disconnect the autopilot.

(12) VOR/LOC/GS Test

Check the VOR, ILS, and Glideslope functions with ramp test equipment. Operate the equipment according to the test equipment manufacturer's instructions.

NOTE: The PFD HSI does not show a course deviation bar unless a valid VHF NAV frequency is tuned.

(a) Simulate a VOR signal on a radial 360° relative to the airplane. Tune the NAV1 and NAV2 receivers to the simulation frequency.

(b) Set the HSI on PFD1 to VOR1 by pressing the CDI soft key until VOR1 is selected. Rotate CRS knob to set VOR1 course pointers to 360° relative to the airplane. (CDI Sync must be "OFF" on MFD Aux Group, page 5)

(c) Verify full scale deflection of VOR1 CDI by varying the selected course at least 10° left and right. Reset course pointers to 360° relative to the airplane.

(d) Engage the autopilot and press the NAV key on the MFD (if not already in NAV mode). Using the CRS knob, alter course by 10° to the right. Verify the Flight Director and aircraft controls respond by flying to the VOR course. Repeat to the left.

(e) Set CRS course pointer to 360° relative to the airplane. Disconnect Autopilot.

(f) Repeat steps (b) - (e) with VOR2.

(g) Simulate a Localizer/Glideslope signal. Tune this signal on NAV1 and NAV2 receiver.

(h) Set the PFD1 HSI to LOC1 by pressing CDI soft key until LOC1 is selected. Use the test equipment to center the deviation bars (localizer and glideslope) on the PFD.

(i) Engage AP. Press the APR key on the MFD. Verify that the LOC and GS annunciations are green on the PFD. Apply right/left and up/down localizer/glideslope signals using the test equipment. Verify that the Flight Director and flight controls respond appropriately. Disconnect AP.

(j) Repeat steps (h) and (i) with LOC2.

(13) ESP Functional Check (Optional)

This procedure is required only for aircraft with the optional Enhanced AFCS features.

NOTE: This check requires valid GPS reception.

(a) On the MFD, navigate to the SYSTEM SETUP page of the AUX group using the FMS knob. Press SETUP 2 softkey.

(b) Verify that there is a window for Stability & Protection and the status is “ENABLED”.

(c) Press Message softkey. Verify that there are no “ESP FAIL” or “ESP OFF” system messages on the PFDs.

(d) Cover both GPS1 and GPS2 antennas.

(e) Verify on PFD 1 and PFD 2 the ESP Roll Indices are displayed at 45° on the roll indicator.

(f) On the MFD, activate the cursor then rotate the large FMS knob to select Stability & Protection.

(g) Rotate the small FMS knob to change the status to “DISABLED”.

(h) Verify on PFD 1 and PFD 2 the ESP Roll Indices are not displayed at 45° on the roll indicator on the Attitude Display.

(i) Press Message softkey on PFD1 & PFD2. Verify that there is an “ESP OFF” system message on both PFDs.

(j) Remove the cover from both GPS1 and GPS2 antennas.

(k) Remove power to the aircraft.

(l) Wait ~15 seconds before re-applying aircraft power.

(m) Once AFCS PFT is complete and prior to GPS acquisition, verify on PFD1 and PFD2 the ESP Roll Indices are displayed at 45° on the roll indicator.

(n) After GPS has acquired satellites, verify on PFD1 and PFD2 the ESP Roll Indices are not displayed at 45° on the roll indicator on the Attitude Display.

(o) On the MFD, navigate to the SYSTEM SETUP page of the AUX group using the FMS knob.

(p) Press Setup 2 softkey. Verify there is a window for Stability & Protection and the status is “ENABLED”.

(q) Press Message softkey on PFD 1 and PFD 2. Verify that there are no “ESP FAIL” or “ESP OFF” system messages on the PFDs.

D. Inspection

(1) Twelve (12) Months

Each twelve months, perform the following inspections:

(a) Visually inspect:

1) the servos, connectors, support structures, and bridle cables to ensure that no corrosion, chafing, cracks, or other defects exist.

2) the servo bridle cables to ensure no fraying, corrosion, or other damage exists. Replace the cable if the condition is questionable.

3) the GFC 700 system wiring to ensure no chafing, wear, or other damage exists.

(b) Manually move the ailerons (for roll servo), stabilator (for pitch servo), stabilator trim wheel (for pitch trim servo), and rudder pedals (for yaw servo, if installed) from stop to stop and observe the servo, capstan, and control surface rigging. Ensure there is no binding in the bridle cabling, and that the capstan pulleys rotate freely.

(c) Check the tension of each servo bridle cable per “Figure 12” on page 221055, “Figure 13” on page 221058, or “Figure 14” on page 221061, as appropriate.

(d) Manually override the slip clutch every 12 months per the “Autopilot Clutch Overpower Check” on page 221043.

(2) First 1,000 Hours or Three (3) Years

At the first 1,000 hours or three years, whichever comes first; or any time the GSA 8X servo actuator is separated from the servo gearbox: clean and grease the servo actuator output gear with Mobilgrease 33.

E. Components

See “Figure 8” on page 221048.

(1) Garmin Servos

The Garmin servos are a two part system consisting of the GSA 8X Servo Actuator and the GSM 86 Servo Gearbox. See “Figure 9” on page 221049.

(a) GSA 8X Servo Actuator

The GSA 8X servo actuator is an electromechanical unit that provides automatic control of a single flight axis (pitch, roll, or yaw), or their associated trims.The GSA 8X receives serial RS-485 data packets from two GIA Integrated Avionics units, which contain data from the Flight Director, the GRS AHRS, and the GDC Air Data Computer. The drive clutch solenoid engages the GSM 86 drive clutch.The GSA 8X is mounted to the GSM 86 and can be removed and replaced without removing or de-rigging the GSM 86.

1) Removal

a) Turn the master switch off. Pull the AUTOPILOT (and, if appropriate, PITCH TRIM) circuit breaker(s).

b) Disconnect the servo actuator electrical harness connector.

c) Remove the bolts and washers securing the servo actuator to the servo gearbox and remove the servo actuator.

2) Installation

a) Grease the servo actuator output gear with Mobilgrease 33.

NOTE: Two O-ring seals are pressed into grooves in the back plate of the servo gearbox units. If during installation, removal, or reinstallation either of these O-rings becomes detached from the rear plate of the servo gearbox, carefully inspect and ensure there are no cuts or other damage to the O-ring(s). If no damage is found, reinstall the O-ring(s) in the rear plate, being careful not to stretch or compress it along its length while installing into the groove in the rear plate. If either O-ring is missing or damaged, replace with Garmin P/N 251-20031-00.

b) Position the servo actuator to the servo gearbox, secure with bolts and washers (4 ea.) as follows:

1] Insert all bolts and thread in approximately two turns by hand.

2] Starting with two opposite corner bolts, tighten all bolts until 1/64” to 1/32” (½ to 1 turn) from fully seated.

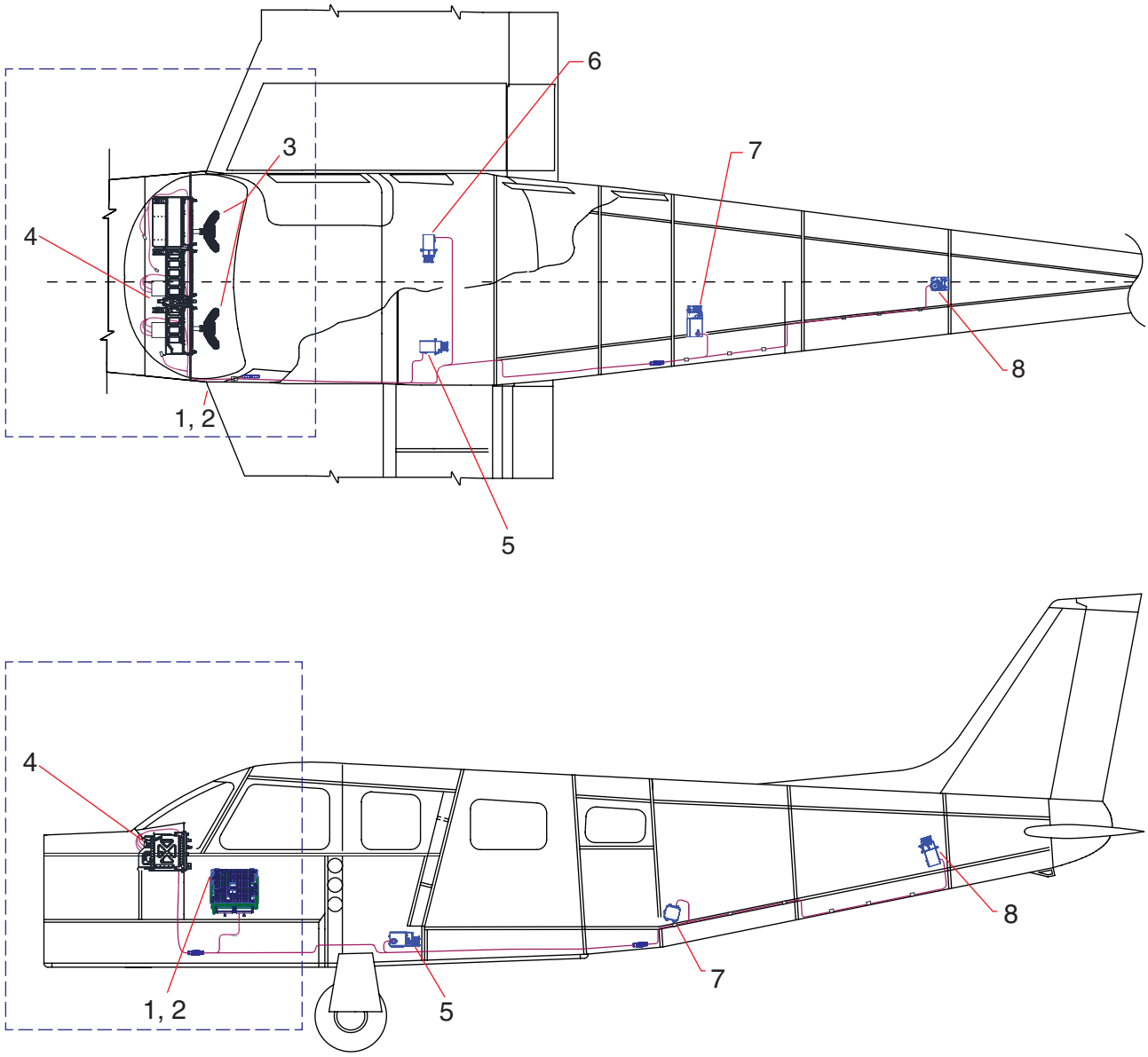
3] Verify that the GSA 8X is still slightly loose relative to the servo gearbox, ensuring there is still relative movement between the two units.

4] Starting with two opposite corner bolts, tighten all bolts to proper torque per “Figure 9” on page 221049.

c) Connect the servo actuator electrical harness connector.

d) Push-in the AUTOPILOT (and, if appropriate, PITCH TRIM) circuit breaker(s).

107866 A



1. CB334 CIRCUIT BREAKER (3 AMP) (REF)
2. CB351 CIRCUIT BREAKER (5 AMP) (REF)
3. CONTROL WHEEL SWITCH (BOTH SIDES)
4. DISPLAYs (PFD's, MFD) (REF)
5. ROLL SERVO
6. PITCH SERVO
7. PITCH TRIM SERVO
8. YAW SERVO

Autopilot Component Locator
Figure 8

(b) GSM 86 Servo Gearbox

See “Figure 10 (Sheet 1 of 2)” on page 221050.

The GSM 86 servo gearbox is mounted to the aircraft structure, via a custom mounting bracket, and is responsible for transferring the output torque of the GSA 8X servo actuator to the mechanical flight control surface linkage. The GSM 86 houses its internal components within an cast aluminum housing. There are multiple variations of the GSM 86, all designed to accommodate various flight-control system interfaces and torque requirements. The slip-clutch settings and type of output capstan are described elsewhere in this chapter.

Removal and Installation procedures are under the individual functional servo paragraphs, below.

(c) Return to Service

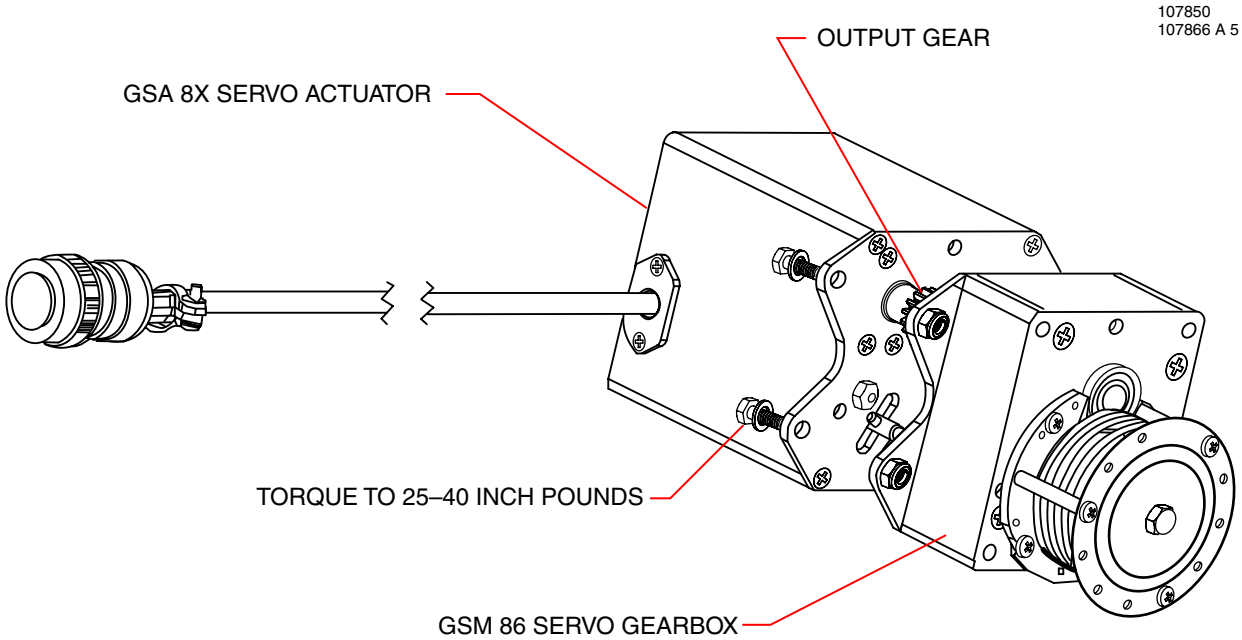
1) Original Servo(s) Reinstalled

No software loading is required if the removed servo(s) is re-installed. Continue to “Ground Checks” on page 221039.

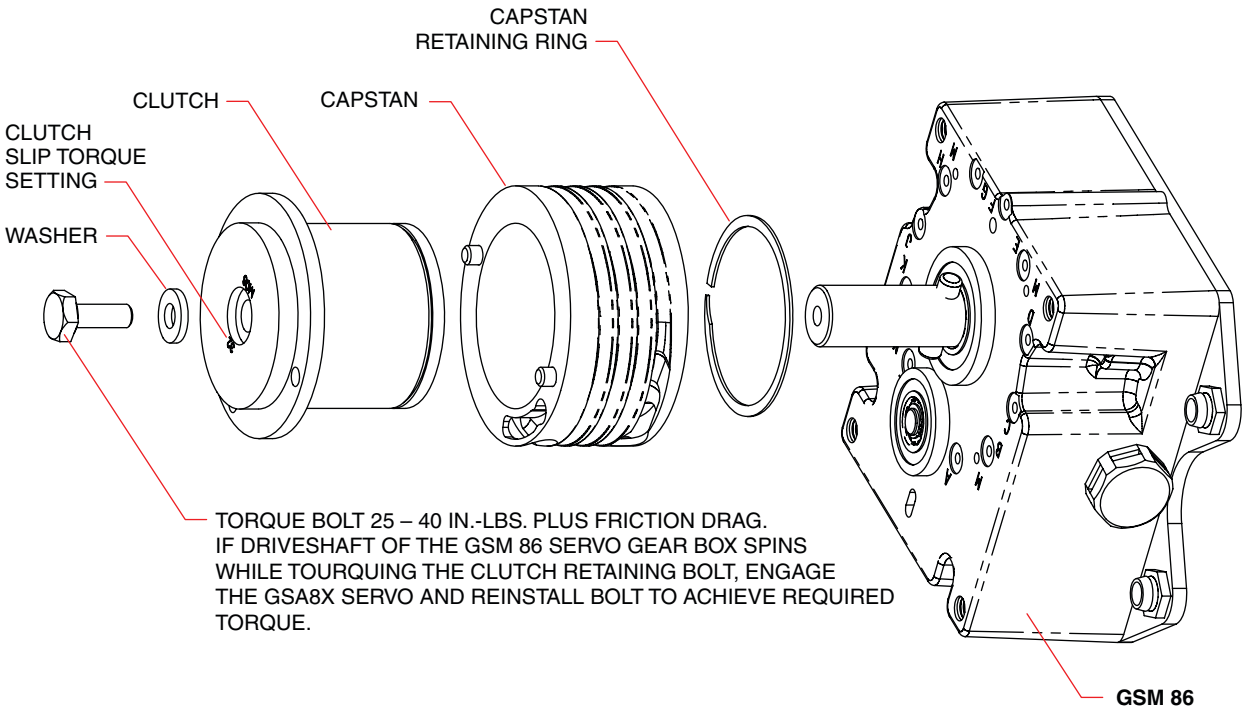
NOTE: This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) New, Repaired, or Exchanged Servo(s) Installed

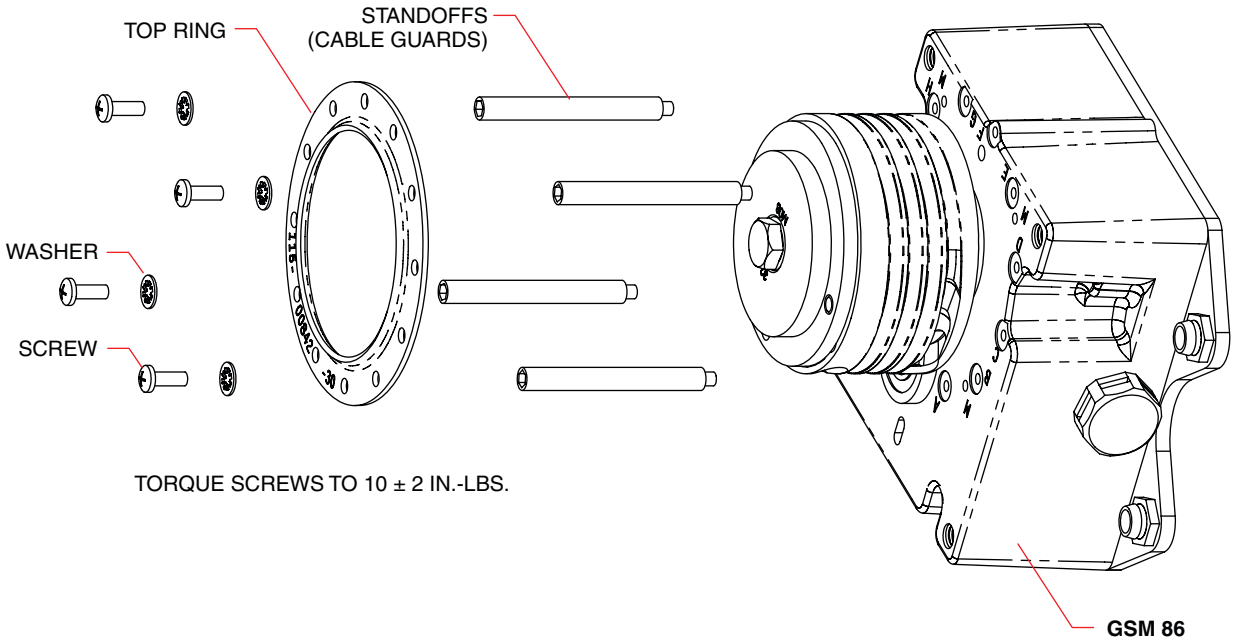
If a new, repaired, or exchanged servo is installed, the correct software files and certification gains must be loaded to the unit. See G1000 Software/Configuration Procedure in 34-25-01 or G1000 NXi Software/Configuration Procedure in 34-25-02, and then continue to “Ground Checks” on page 221039.



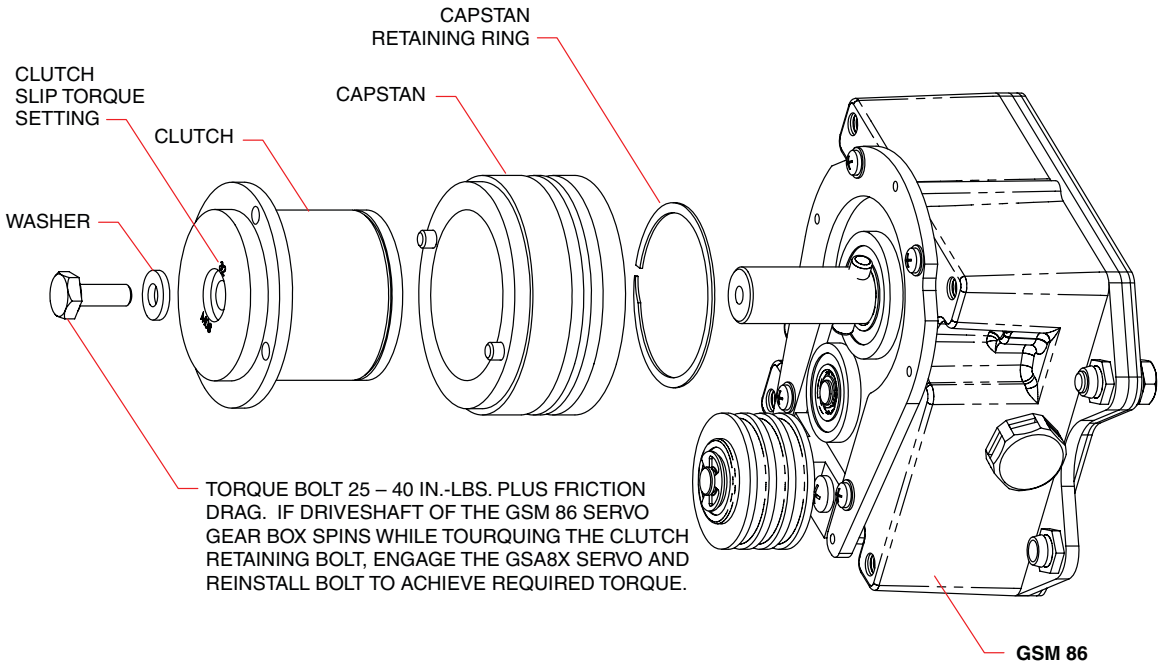
GSA 8X/GSM 86 (Typical)
Figure 9



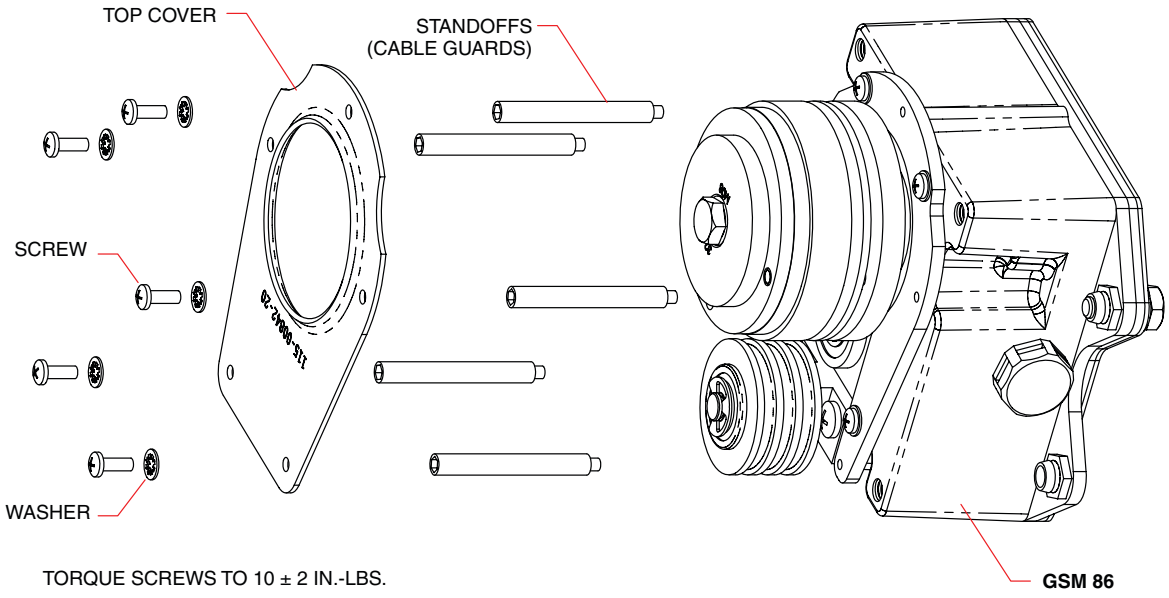
NOTE: RECORD NUMBER AND CLOCKING OF STANDOFFS PRIOR TO DISASSEMBLY.



GSM 86 Servo Gearbox
Figure 10 (Sheet 1 of 2)



NOTE: RECORD NUMBER AND CLOCKING OF STANDOFFS PRIOR TO DISASSEMBLY.



GSM 86 Servo Gearbox with Idler
Figure 10 (Sheet 2 of 2)

(2) Flight Control Cable Wrapping

NOTE: This procedure incorporates Piper Service Letter No. 1337.

This cable wrapping guidance applies to all autopilot servos: roll, pitch, trim, and, if installed, yaw.

(a) Autopilot Servo Bridle Cables

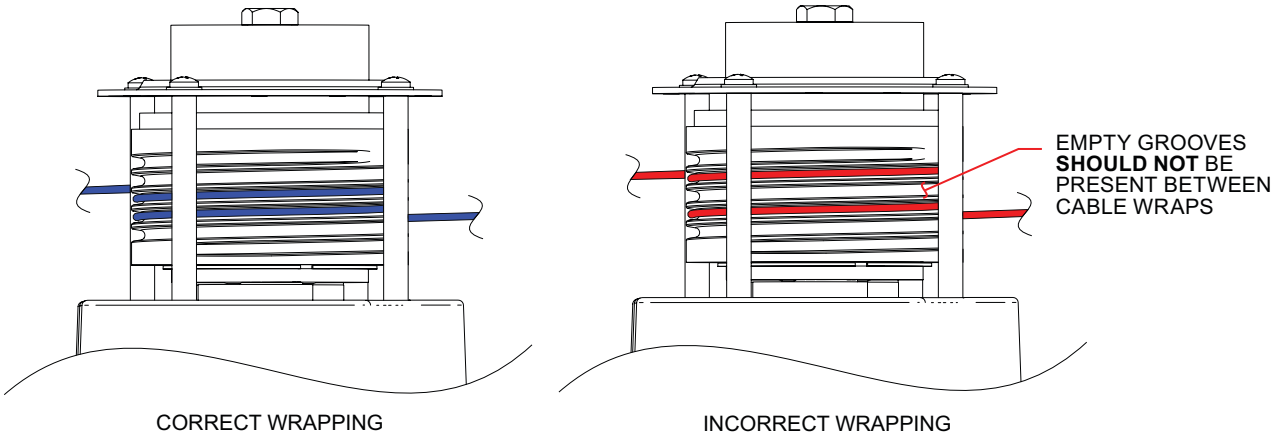
WARNING: IMPROPER WRAPPING CAN RESULT IN DAMAGE TO CABLES AND SERVOS ALONG WITH TIGHTENING, BINDING, OR COMPLETE LOSS OF FLIGHT CONTROLS DURING OPERATION.

NOTE: The following are general instructions and examples of servo cable wrapping to facilitate installation. Refer to the specifics provided in the individual servo instructions and illustrations for each servo installation.

See “Figure 11”, below, and the figures that follow which are associated with each servo, for examples of proper bridle cable wrapping.

During installation of the bridle cable, the following is recommended:

- When cable is at the center of travel, the locating ball should be at the center of the cable wrap.
- Verify that the locating ball is firmly pressed into the ball slot in the capstan. At no time during wrapping, routing, or installation of the cable should the locating ball be allowed to exit the ball slot.
- The capstan may move one full turn in either direction allowing for a maximum cable travel of 12.4” between the ball and cable exit. No less than 90° (¼ wrap) of cable should remain between the cable exit and the capstan location of ball at full travel in either direction.
- To ensure full engagement between the bridle cable and servo, at no time may the cable exit from the ends of the capstan where the grooves are not at full depth.
- When routing and securing the bridle cable to the flight control cable, the bridle cable should exit from the capstan without coming in contact with the cable guards.
- After installing the bridle cable, inspect the servo at full travel in both directions for proper and unrestricted movement of the cable. Verify that no grooves in the capstan were skipped.



Autopilot Servo Bridle Cable Wrapping – Correct vs. Incorrect
Figure 11

(b) Trim Servo Control Cables

See “Figure 14” on page 221061 for examples of trim cable wrapping.

During installation of the flight control cable, the following is recommended:

- When routing the control cable, the cable should exit from the capstan without coming in contact with the cable guards.
- After installing the control cable, inspect the servo at full travel in both directions for proper and unrestricted movement of the cable.

(3) Roll Servo

The roll servo is mounted underneath the left aft facing passenger seat. See “Figure 8” on page 221048, “Figure 9” on page 221049, and “Figure 12” on page 221055.

(a) Removal

1) Remove the left aft facing passenger seat.
2) Remove adjacent carpet.
3) Remove screws securing floor panel and remove panel.
4) Disconnect autopilot harness connector.
5) Remove nuts and bolts (2 ea.) securing each cable clamp (2) and remove cable clamps from aileron balance and autopilot bridle cables.
6) Remove bolts and washers (4 ea.) securing roll servo (GSA 80/GSM 86) to mounting brackets and remove roll servo with attached bridle cable.

(b) Installation

1) Rig ailerons per Aileron Control Rigging and Adjustment, 27-10-00.
2) Place the control column tee bar in full forward position and secure by use of a suitable tool or by placing weights on the aft side of the stabilator, if stabilator cables have been previously tensioned.
3) Lock the ailerons in neutral (i.e. - aligned with flaps) position using a suitable contour fixture at the inboard ends of the ailerons and the outboard ends of the flaps. Verify control wheels are centered and secure in that position.
4) Remove screws (4) and remove capstan cover and cable guards from servo.
5) Wrap autopilot bridle cable, as described in “Figure 12” on page 221055 and align the capstan.

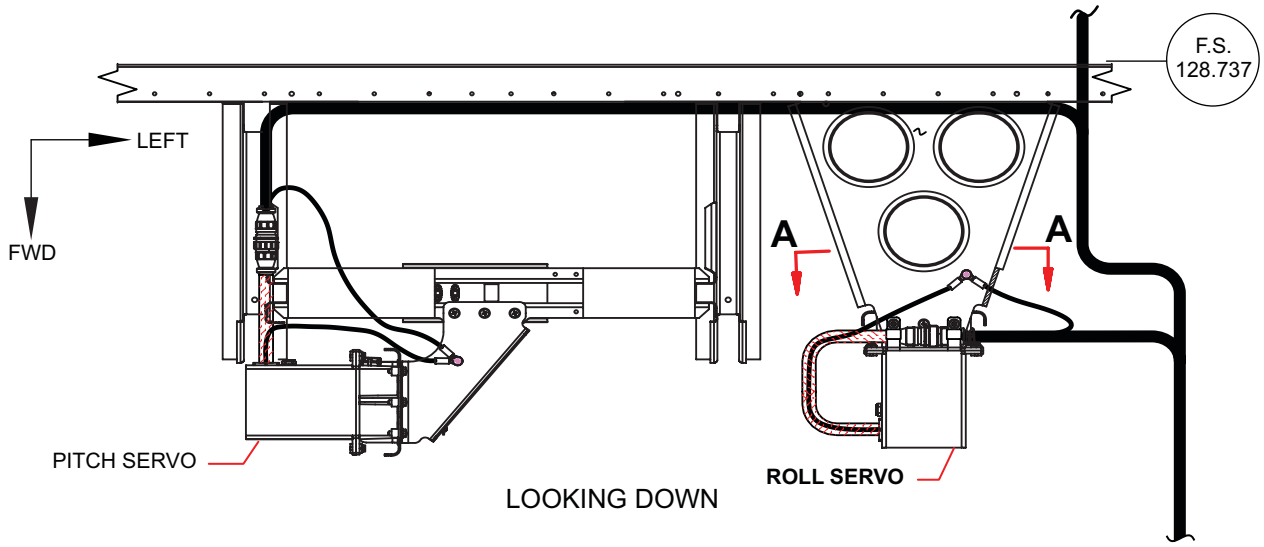
   NOTE: See “Autopilot Servo Bridle Cables” on page 221052 for guidance on the correct installation and wrapping of cables.

6) Replace cable guards and capstan cover, secure with screws (4).
7) Position roll servo (GSA 80/GSM 86) as shown in “Figure 12” on page 221055 and secure with bolts and washers (4 ea.) holding servo to mounting brackets. Torque bolts as specified in “Figure 12”.
8) Position cable clamps (2) as shown in “Figure 12” on page 221055 and tighten nuts and bolts (2 ea.). Adjust cable clamps in or out along the aileron balance cable to obtain the bridle cable tension specified in “Figure 12”.
9) Torque cable clamp bolts as specified in “Figure 12” on page 221055, Detail C.
10) Remove the locking fixtures at the inboard ends of the ailerons. Aileron neutral (i.e., aligned with flaps) position should be maintained with the control wheels in neutral. A droop of 1/8 inch is allowable.
11) Remove the control wheel/tee bar locks. Check to ensure that the left aileron up and right aileron down stops are contacted simultaneously and vice versa. Adjust stops as required.

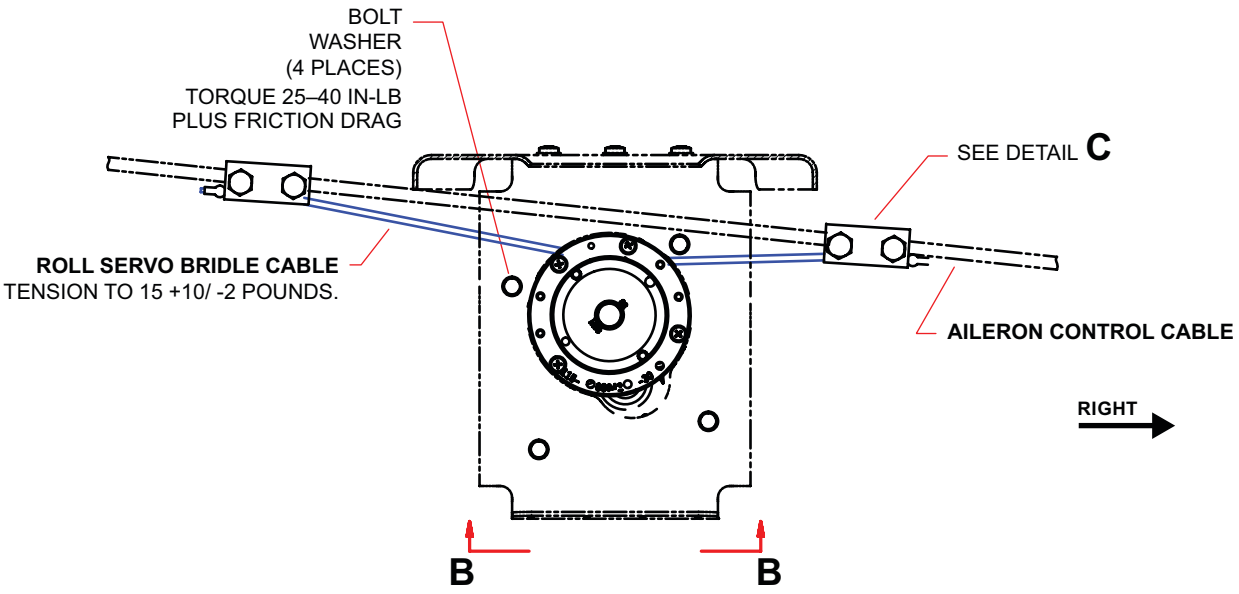
12) Rotate the left (pilot's) control wheel in each direction until the bellcranks contact the stops. The sprocket stops on the tee bar shall not be contacted until additional "override" movement (cushion) of the wheel occurs. A "cushion" on 0.030 to 0.040 inches is to be maintained as measured between the sprocket pin and adjustable control wheel stop bolts.
13) Place the ailerons in the neutral (aligned with the flaps) position. For each aileron, from the neutral position, check that the "up" travel and the "down" travel are within the limits as specified in 27-00-00, Chart 2.
    a) Center bubble of a protractor over surface of aileron at neutral position. Note reading.
    b) Move aileron full up and down. Check degree of travel in each direction. Degree of travel on protractor is determined by taking the difference between protractor reading at neutral and up, and neutral and down. Bubble must be centered at each reading.

       When measuring "down" travel from the neutral position, a light "up" pressure shall be maintained at the center of the aft edge of the aileron. When measuring "up" travel from the neutral position, a light "down" pressure shall be maintained at the center of the aft edge of the aileron (at the "up" position only), just sufficient to remove the slack between the bellcrank and the aileron.
14) If either step 10) thru 13) reveal the aileron controls out of rig, repeat steps 1) thru 8), above.
15) Connect autopilot harness.
16) Check aileron controls for free and correct movement.
17) Check autopilot system for binding and proper operation.
18) Perform "Ground Checks" on page 221039.
19) Replace floor panel and secure with screws.
20) Replace carpeting.
21) Replace the left aft facing passenger seat.
22) Review "Return to Service" on page 221049 to determine if software needs to be reloaded.

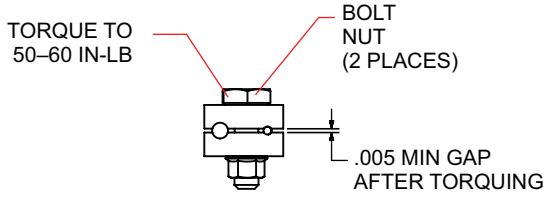
107850 C 1
107866 E 5



LOOKING DOWN

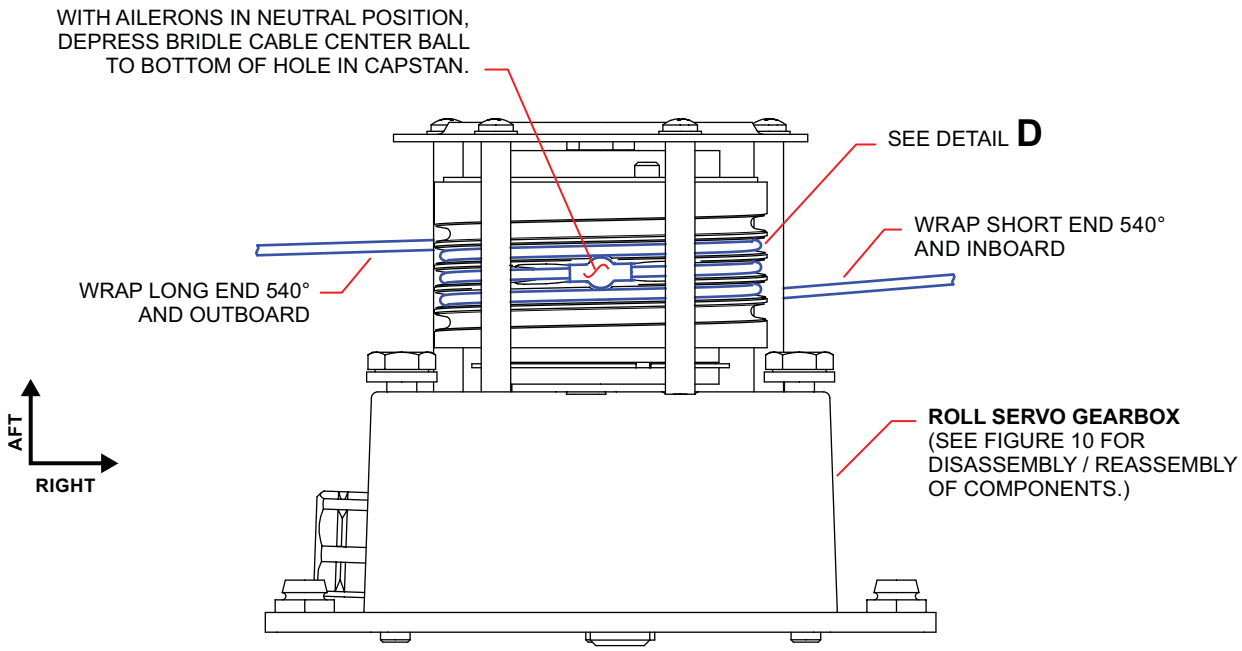


SECTION A–A
LOOKING FORWARD

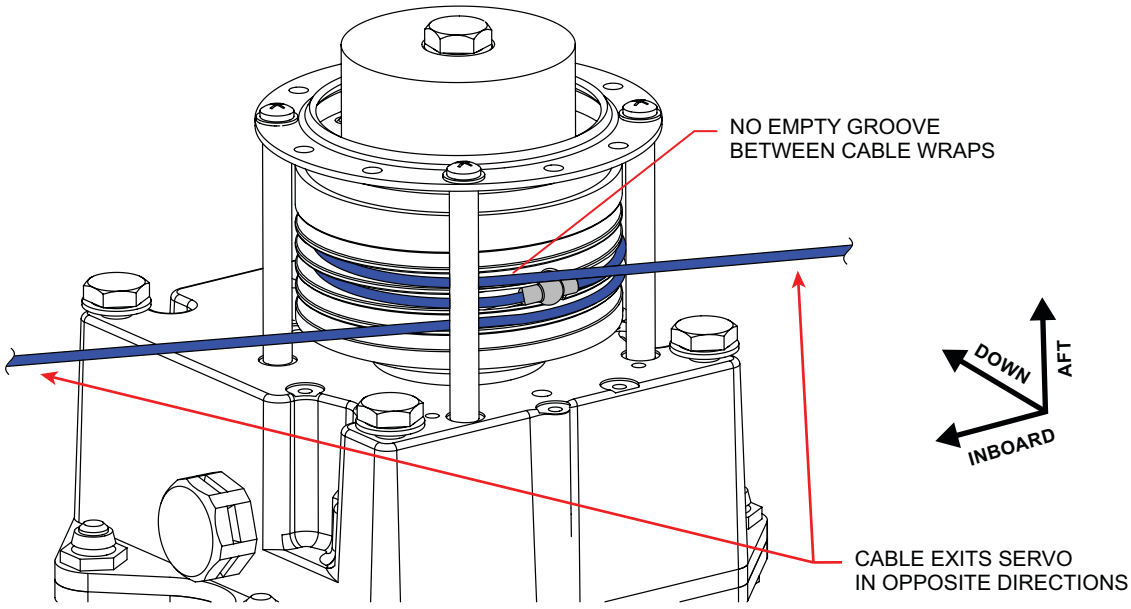


DETAIL C

Roll Servo Installation
Figure 12 (Sheet 1 of 2)



SECTION B–B
LOOKING UP



DETAIL D
ROLL SERVO CABLE WRAPPING

Roll Servo Installation
Figure 12 (Sheet 2 of 2)

(4) Pitch Servo

The pitch servo is mounted underneath the right aft facing passenger seat. A bridle cable and clamps attach the servo capstan to the right (upper) stabilator cable. See “Figure 8” on page 221048 and “Figure 13” on page 221058.

(a) Removal

1) Remove the right aft facing passenger seat.
2) Remove carpet and floor panel.
3) Disconnect autopilot harness connector.
4) Remove nuts and bolts (2 ea.) securing each cable clamp (2) and remove cable clamps from right (upper) stabilator cable and autopilot bridle cable.
5) Remove bolts and washers (4 ea.) securing pitch servo (GSA 81/GSM 86) to mounting bracket and remove pitch servo with attached bridle cable.

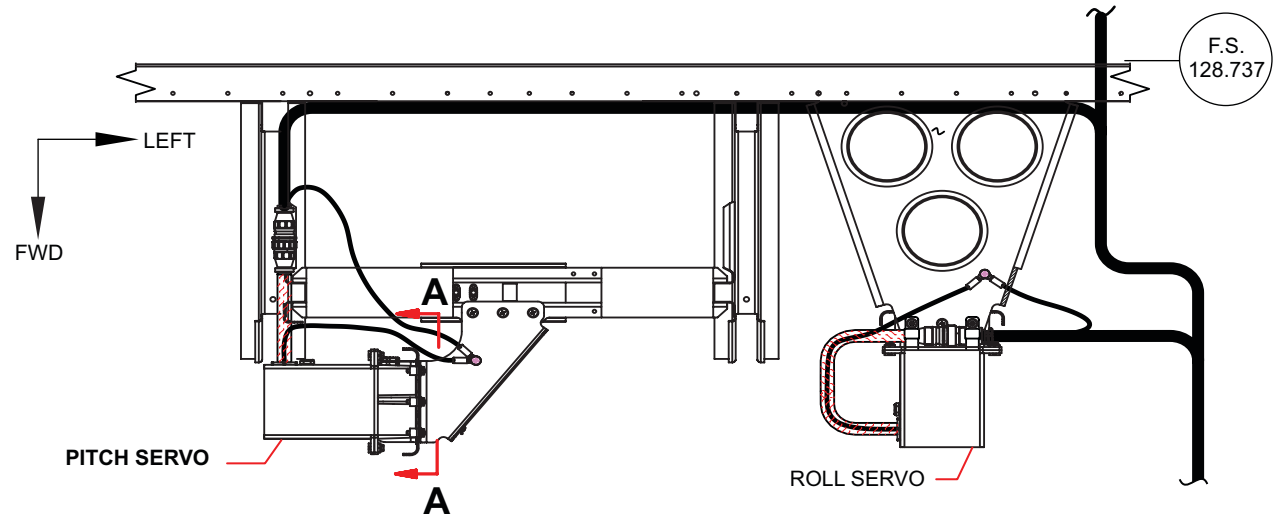
(b) Installation

1) Rig stabilator controls per Stabilator Controls Rigging and Adjustment, 27-30-00.
2) Remove screws (4) and remove capstan cover and cable guards from servo.
3) Wrap autopilot bridle cable, align capstan, and press center ball in place in “Figure 13” on page 221058.

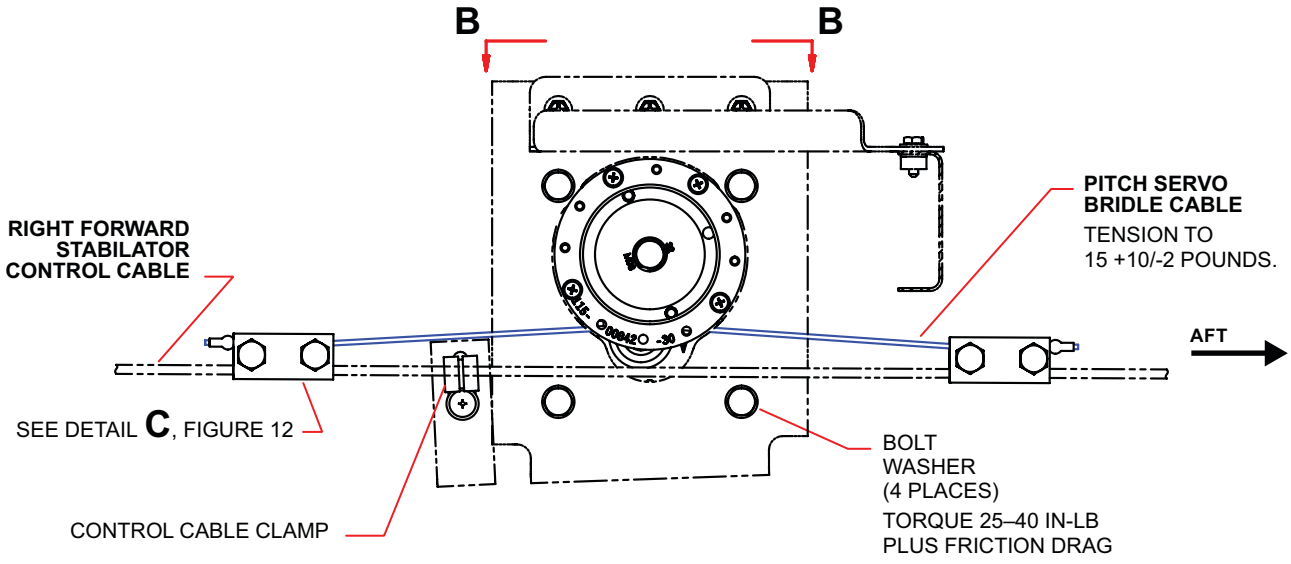
NOTE: See “Autopilot Servo Bridle Cables” on page 221052 for guidance on the correct installation and wrapping of cables.

4) Replace cable guards and capstan cover, secure with screws (4).
5) Position pitch servo (GSA 81/GSM 86) as shown in “Figure 13” on page 221058 and secure with bolts and washers (4 ea.) holding servo to mounting brackets. Torque bolts as specified in “Figure 13”.
6) Position cable clamps (2) as shown in “Figure 13” on page 221058 and tighten nuts and bolts (2 ea.).
7) Adjust cable clamps in or out along the stabilator cable to obtain the bridle cable tension specified in “Figure 13” on page 221058.
8) Torque cable clamp bolts as specified in “Figure 12” on page 221055, Detail C.
9) Connect autopilot harness.
10) Check stabilator controls for free and correct movement.
11) Check autopilot system for binding and correct operation.
12) Perform “Ground Checks” on page 221039.
13) Reinstall floor panel and carpet.
14) Reinstall the right aft facing passenger seat.
15) Review “Return to Service” on page 221049 to determine if software needs to be reloaded.

107850 C 1
107866 E 5

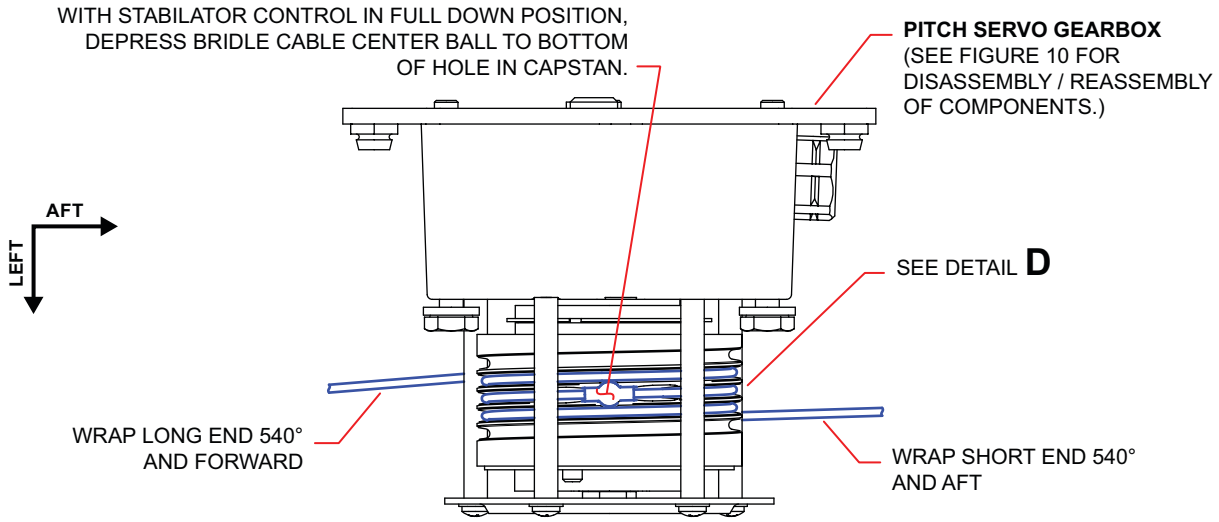


LOOKING DOWN

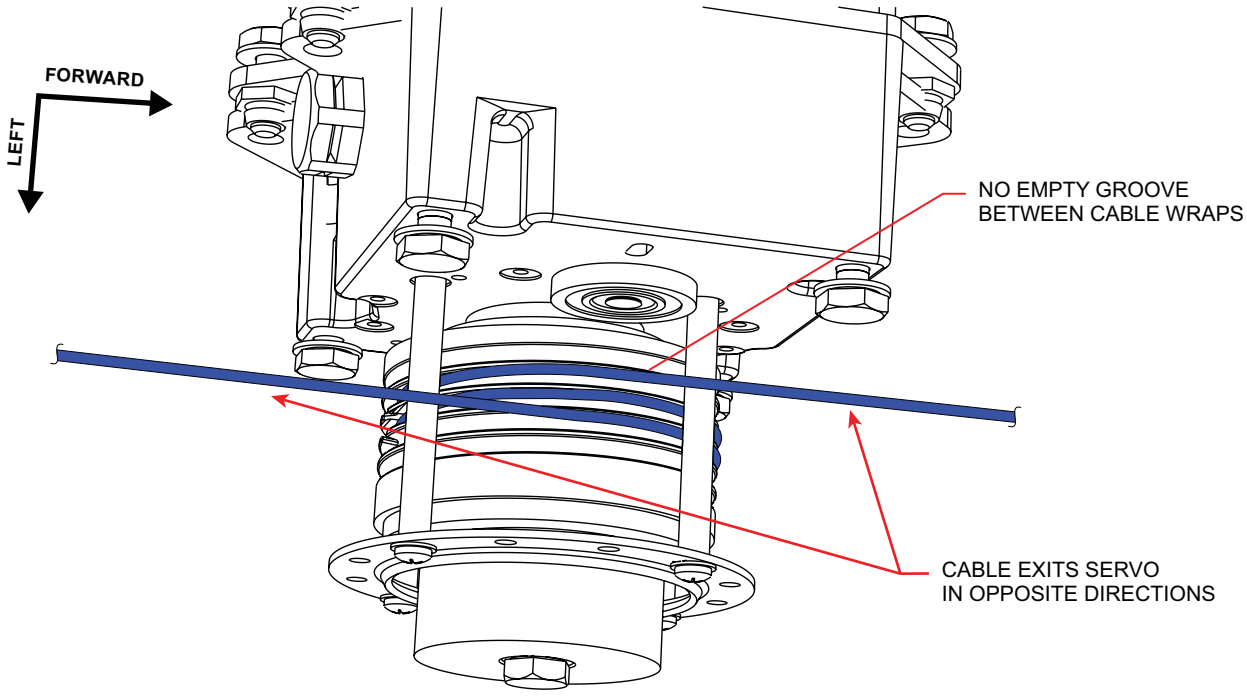


SECTION **A–A**
LOOKING RIGHT

Pitch Servo Installation
Figure 13 (Sheet 1 of 2)



SECTION **B–B**
LOOKING DOWN



DETAIL D
PITCH SERVO CABLE WRAPPING

Pitch Servo Installation
Figure 13 (Sheet 2 of 2)

(5) Trim Servo

The trim servo is located just left of the aircraft centerline, just aft of the bulkhead at F.S. 187.80 and is accessed through the aft fuselage access door. The left stabilator trim cable wraps around the servo idler pulley and servo capstan. See “Figure 8” on page 221048 and “Figure 14” on page 221061.

(a) Removal

1) Remove aft fuselage access door.
2) Tie a pull rope to the left stabilator trim cable forward of the servo and tie-off the pull rope to structure aft.

NOTE: This may be easier to accomplish by removing the rear seats, carpet and floor panel.

3) Tie a pull rope to the left stabilator trim cable aft of the turnbuckle in the rear fuselage aft of the trim servo and tie-off the pull rope to structure forward.
4) NOTE: The pull ropes apply tension to the trim cables to prevent the cables from unwrapping from the trim wheel drum or the trim barrel, and to prevent the cables from fouling at any of the pulleys.
5) Slack-off the turnbuckle in the left stabilator trim cable segment aft of the trim servo sufficient to relieve tension on the left stabilator trim cable as it wraps around the trim servo idler pulley and capstan.
6) Disconnect the autopilot harness.
7) Remove the capstan cover and cable guards (5) by removing the retaining screws (5).
8) Remove the cable from capstan and pulley
9) Remove the bolts and washers (4 ea.) securing the trim servo (GSA 80/GSM 86) to its mounting bracket and remove the trim servo.

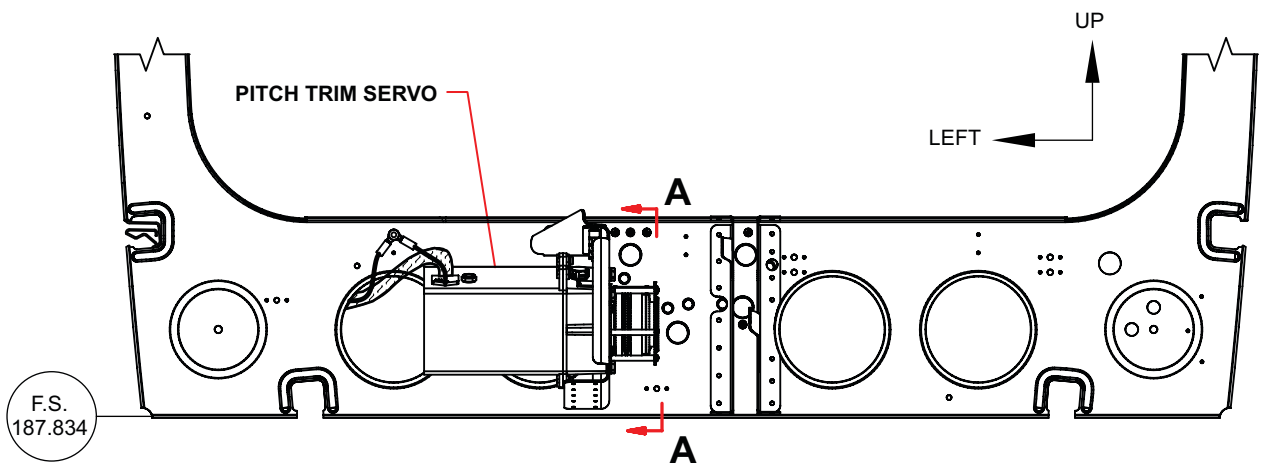
(b) Installation

1) With the capstan cover and cable guards removed, position the trim servo (GSA 80/GSM 86) as shown in “Figure 14” on page 221061. Secure with bolts and washers (4 ea.) holding servo to mounting brackets. Torque bolts as specified in “Figure 14”.
2) Drape the slack left trim cable over the servo capstan.
3) Wrap the cable around the capstan and pulley two (2) complete turns while pressing the trim cable into its groove as shown in “Figure 14” on page 221061.

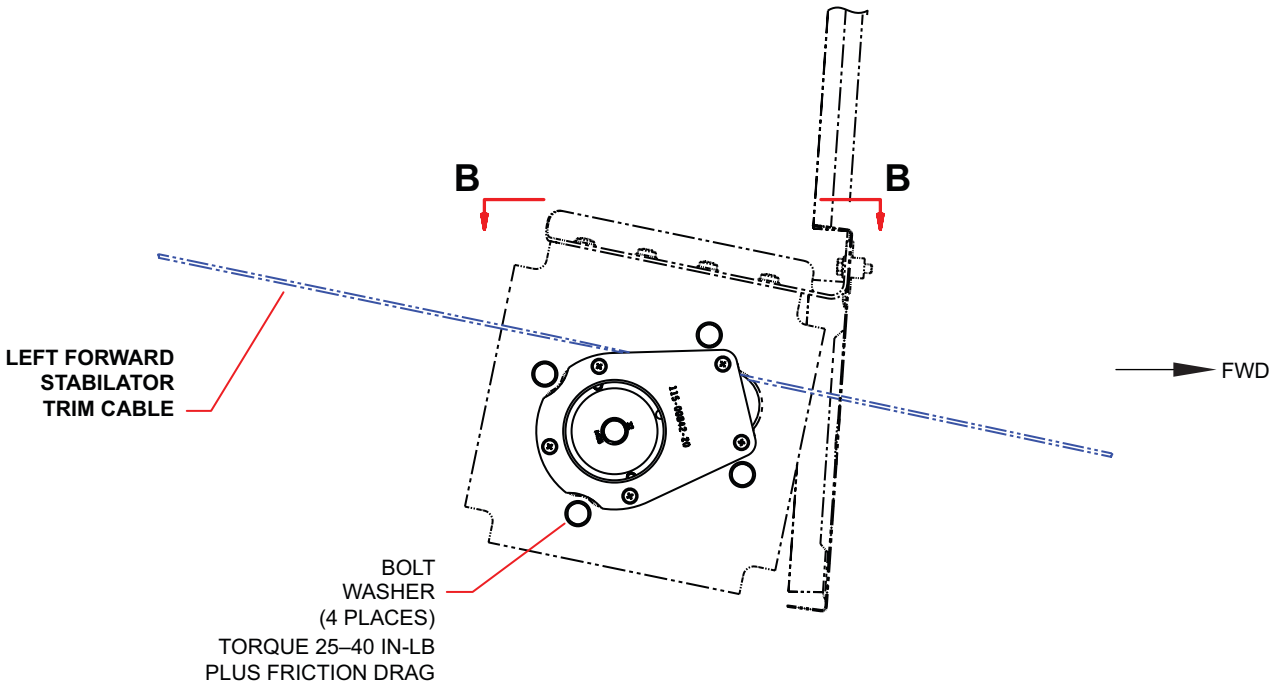
NOTE: See “Trim Servo Control Cables” on page 221053 for guidance on the correct installation and wrapping of cables.

4) Tighten turnbuckles and remove tie ropes.
5) Rig stabilator trim per Stabilator Trim Rigging and Adjustment, 27-30-00.
6) Check stabilator trim controls for free and correct movement.
7) Connect the autopilot harness.
8) Check autopilot system for binding and proper operation.
9) Perform “Ground Checks” on page 221039.
10) Install the capstan cover and cable guards (5) as shown in “Figure 14” on page 221061.
11) Reinstall aft fuselage access door. If removed, reinstall floor panel, carpet, and rear seats.
12) Review “Return to Service” on page 221049 to determine if software needs to be reloaded.

107850 C 2
107866 E 5

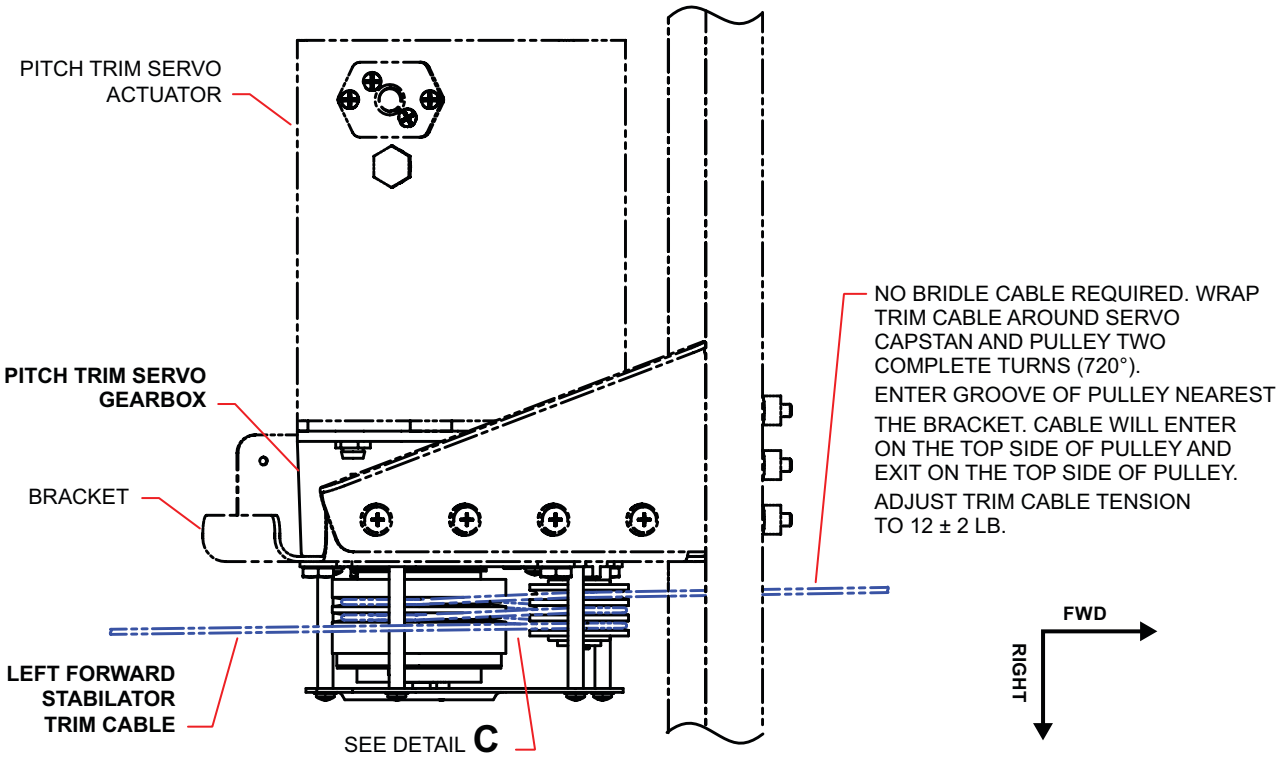


LOOKING FORWARD

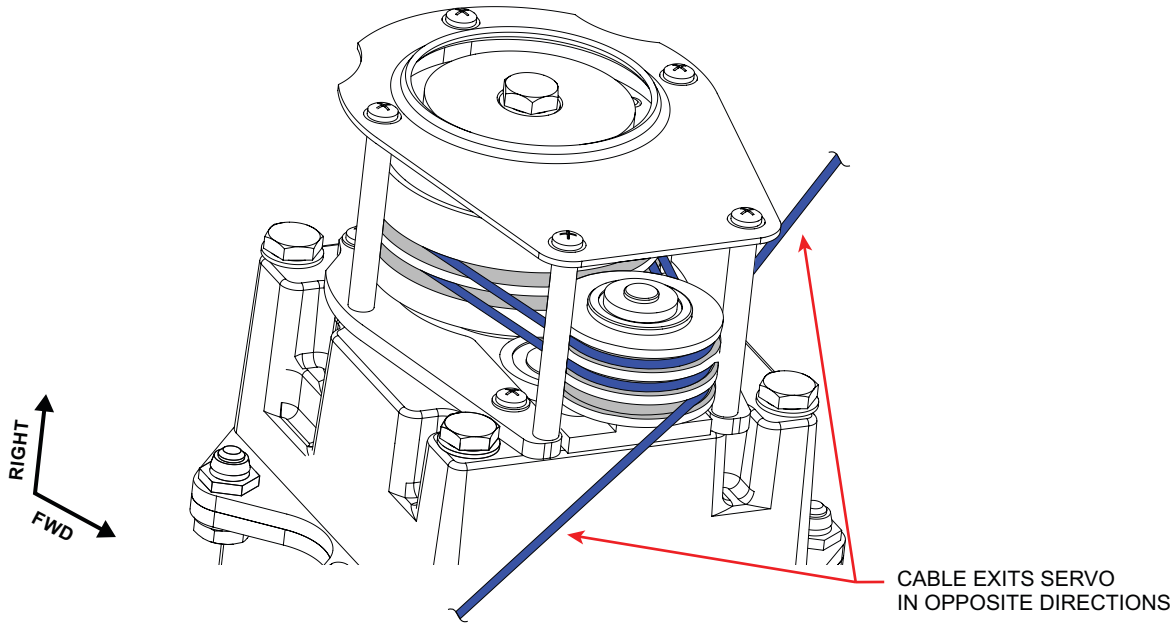


SECTION **A–A**
LOOKING LEFT

Trim Servo Installation
Figure 14 (1 of 2)



SECTION **B–B**
LOOKING DOWN



DETAIL **C**
PITCH TRIM SERVO CABLE WRAPPING

Trim Servo Installation
Figure 14 (2 of 2)

(6) Yaw Servo

The yaw servo, if installed, is located in the aft fuselage. See “Figure 8” on page 221048 and “Figure 15” on page 221064.

(a) Removal

1) Remove the aft fuselage access panel.
2) Disconnect autopilot harness.
3) Remove nuts and bolts (2 ea.) securing each cable clamp (2) and remove cable clamps from rudder cables and autopilot bridle cables.
4) Remove bolts and washers (4 ea.) securing yaw servo (GSA 81/GSM 86) to mounting bracket and remove yaw servo with attached bridle cable.

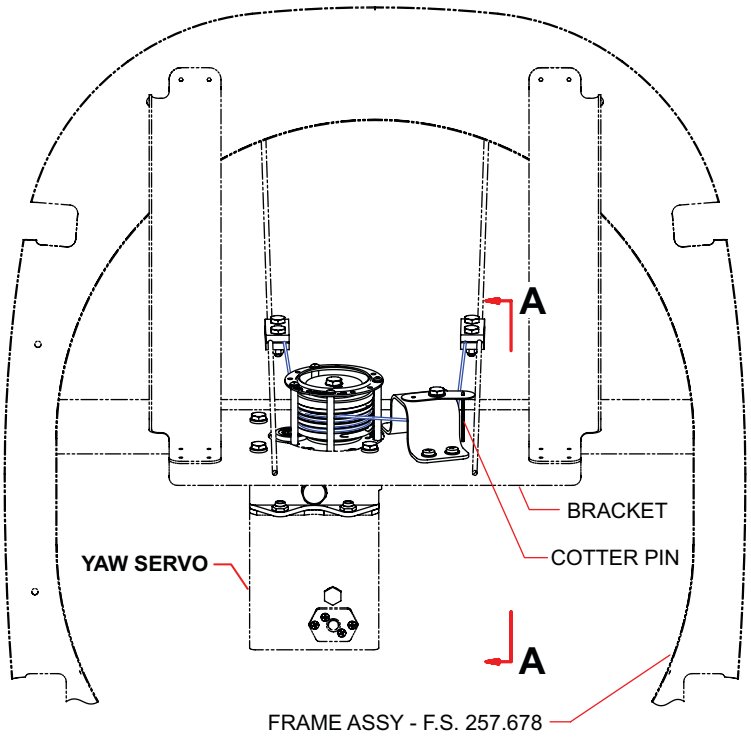
(b) Installation

1) Rig rudder controls per Rudder Control Rigging, 27-20-00.
2) Lock rudder in its neutral (streamlined) position.
3) Remove screws (4) and remove capstan cover and cable guards from servo.
4) Wrap autopilot bridle cable, as shown in “Figure 15” on page 221064.

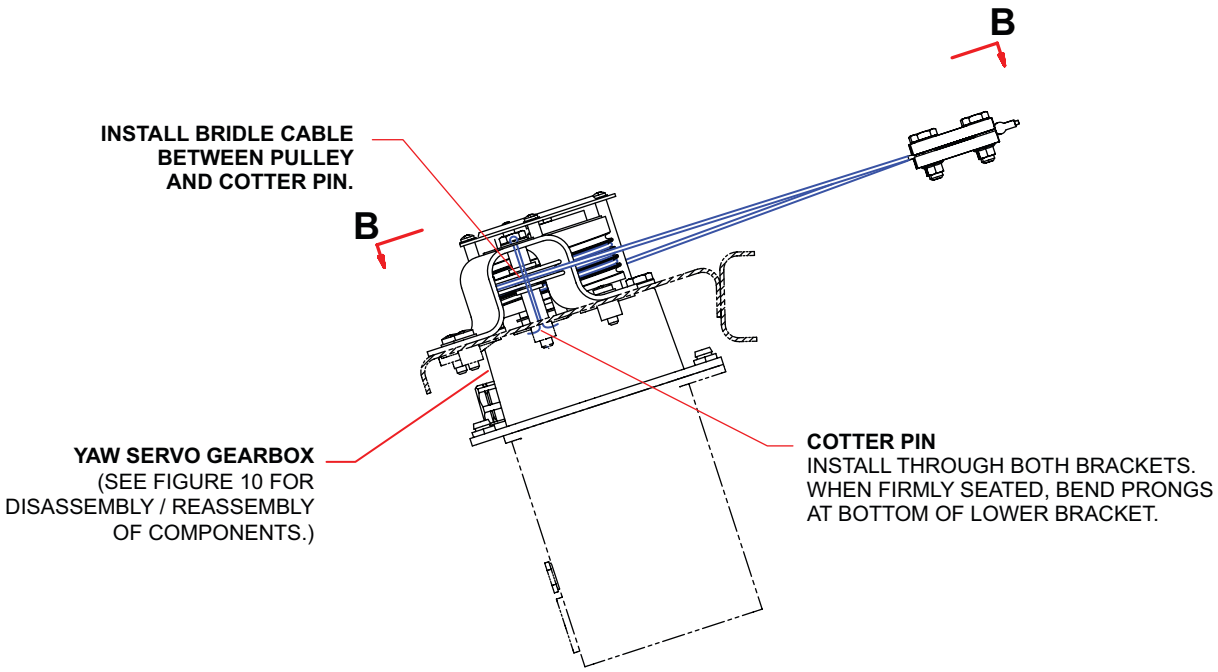
NOTE: See “Autopilot Servo Bridle Cables” on page 221052 for guidance on the correct installation and wrapping of cables.

5) Replace cable guards and capstan cover, secure with screws (4).
6) Position yaw servo (GSA 81/GSM 86) as shown in “Figure 15” on page 221064 and secure with bolts and washers (4 ea.). Torque bolts as specified in “Figure 15”.
7) Attach left and right autopilot bridle cables to the left and right rudder cables with cable clamps (2). Center ball of the cable must be positioned as shown in “Figure 15” on page 221064.
8) Torque cable clamp bolts as specified in “Figure 12” on page 221055, Detail C.
9) Connect autopilot harness.
10) Remove rudder lock.
11) Check rudder controls for free and correct movement.
12) Check autopilot system for binding and proper operation.
13) Perform “Ground Checks” on page 221039.
14) Review “Return to Service” on page 221049 to determine if software needs to be reloaded.

107850 C 2
107866 E 5

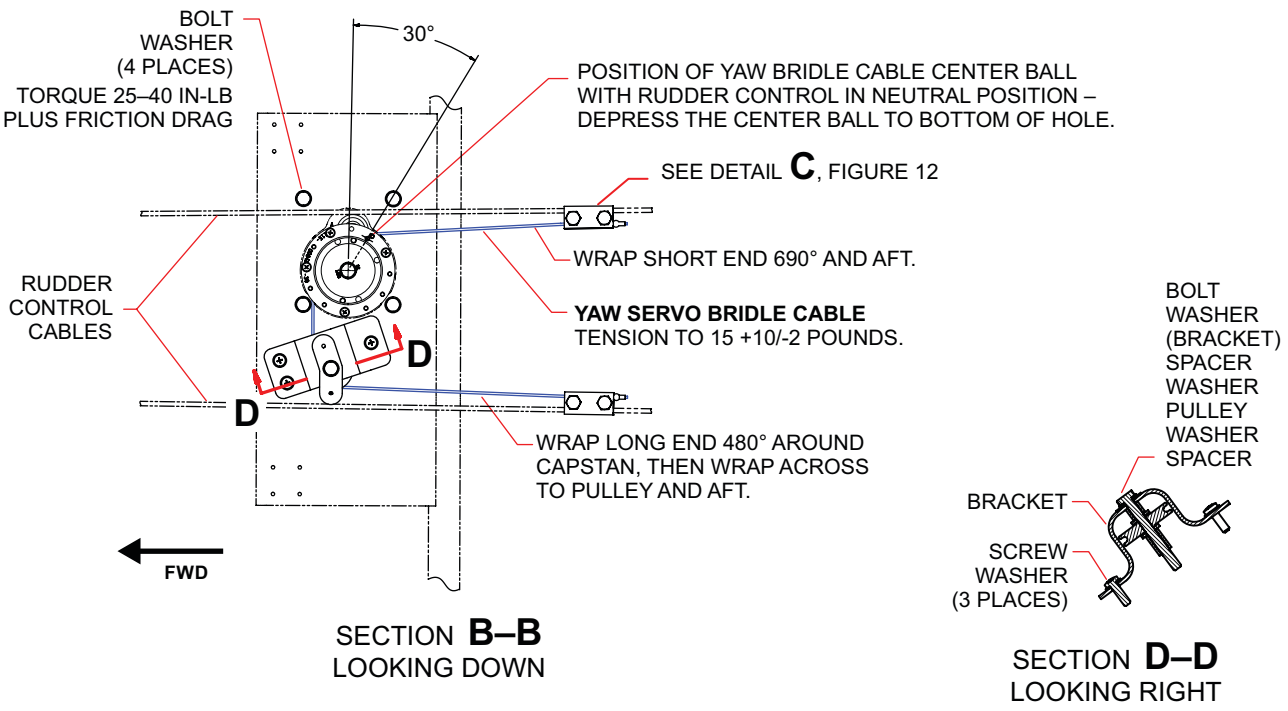


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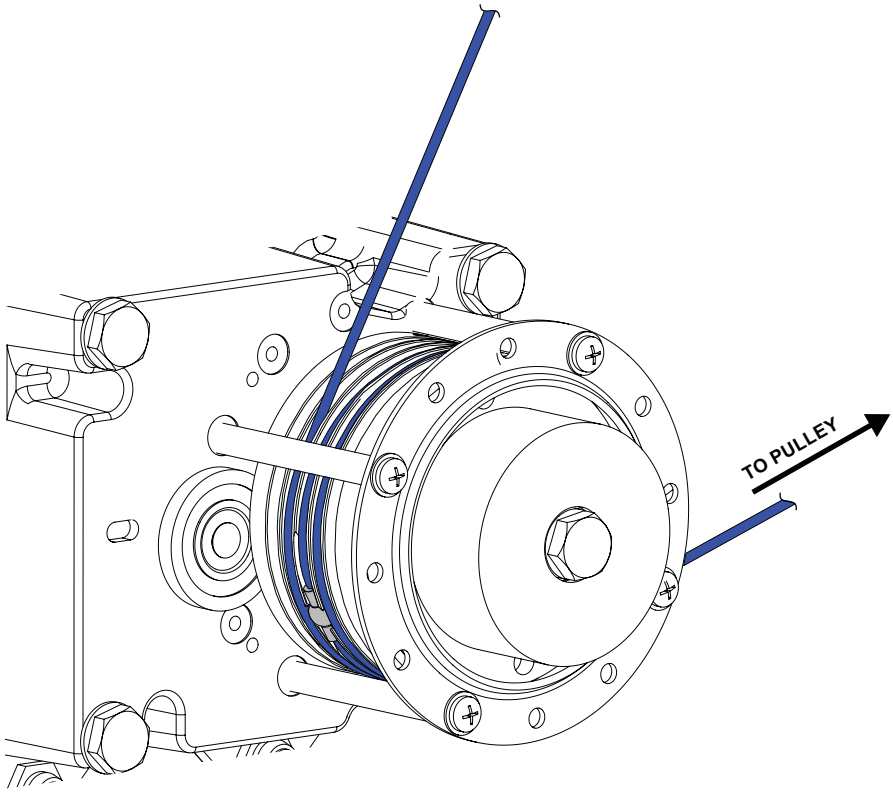
SECTION **A–A**
LOOKING RIGHT
(RUDDER CONTROL CABLES OMITTED)

Yaw Servo Installation (Optional)
Figure 15 (1 of 2)



SECTION **B–B**
LOOKING DOWN

SECTION **D–D**
LOOKING RIGHT



VIEW E–E
YAW SERVO
BRIDLE CABLE WRAP

Yaw Servo Installation (Optional)
Figure 15 (2 of 2)

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WIRING

Airplanes equipped with Garmin GFC 700 Automatic Flight Control Systems have additional EMI shielding on some harness assemblies. If repair is required, see EMI Shielding in 20-97-00.

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CHAPTER

23

COMMUNICATIONS

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CHAPTER 23

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	2	Sep 30/18			
23-00-00	1	Sep 30/18			
	2	Sep 30/18			
23-50-00	1	Sep 30/18			
	2	Sep 30/18			
	3	Sep 30/18			
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	5	Sep 30/18			
	6	Sep 30/18			
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	8	Sep 30/18			
	9	Sep 30/18			
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23-60-00	1	Sep 30/18			
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CHAPTER 23 - COMMUNICATIONS

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GENERAL

1. Radio Master Switch

   A separate master switch for the radios is located in the switch panel in the center of the instrument panel, and, in Seneca V, in the overhead switch panel. Both this switch and the battery master switch must be “ON” for the radios to operate.

2. Avionics Master Switch (with G1000 / G1000 NXi installation)

   A separate master switch for the avionics is located, in Seneca Vs, in the overhead switch panel. Both this switch and the battery master switch must be “ON” for the radios to operate.

   Avionics Bus #1 and #2 are fed directly from the tie bus, via the avionics master solenoids, which in turn are energized by the radio/avionics master switch. Refer to Chapter 24 and to schematics in 91-24-30.

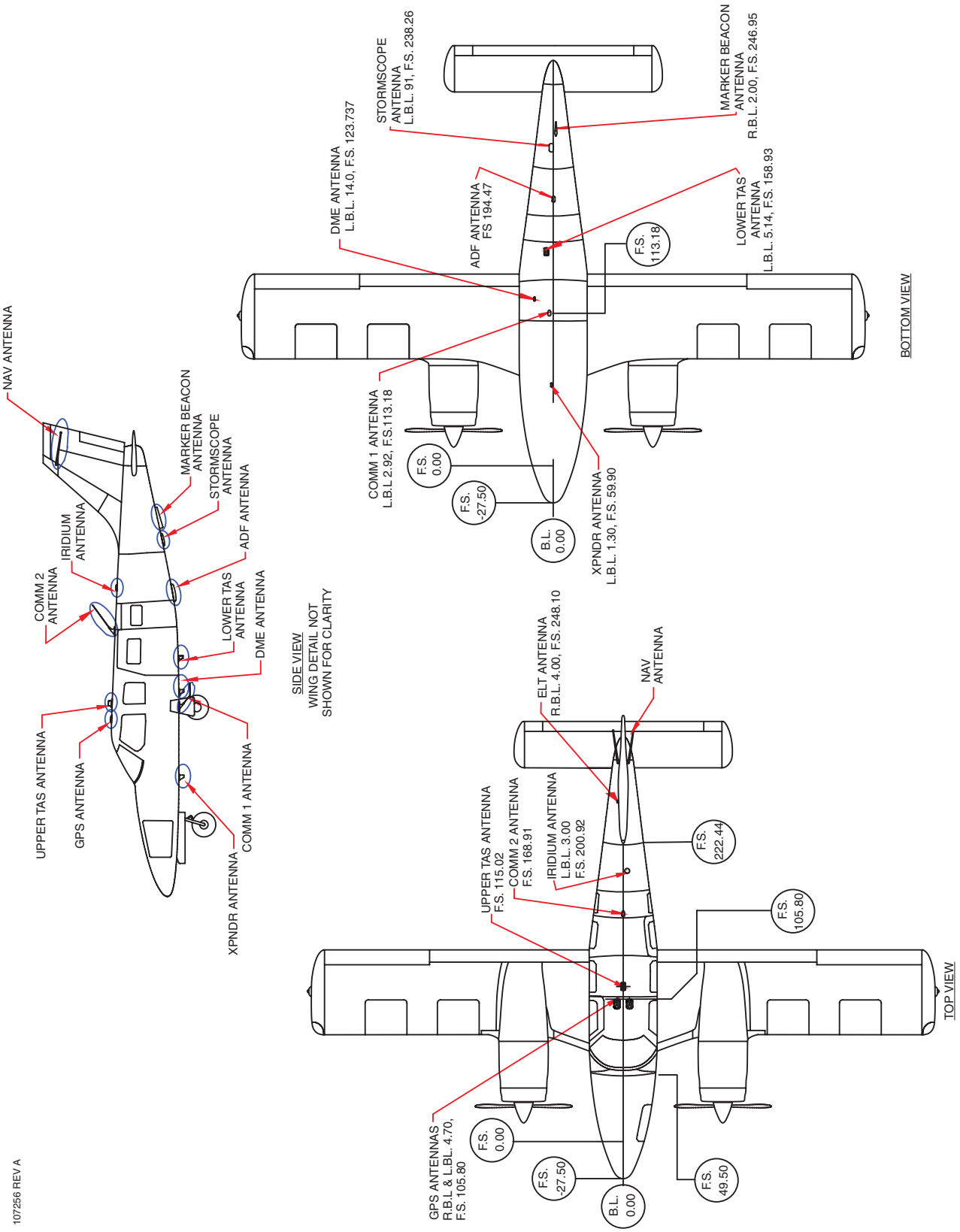
3. Emergency Switch (with G1000 / G1000 NXi installation)

   The EMER switch will activate the #1 COMM/NAV/GPS, pilot’s audio panel, along with #1 PFD in reversionary mode with AHRS 1 data, ADC1 data, a subset of engine parameters.

4. Antennas

   See “Figure 1” on page 23002 for antenna locations.

   S/N’s 3449459, 3449467 and up.



Antenna Locator
Figure 1

AUDIO INTEGRATING

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

1. Garmin GMA-340 Audio Control Panel

The Garmin GMA-340 Audio Panel was available as optional equipment in Seneca Vs S/N's 3449152 thru 3449177, less 3449161 and is installed as standard equipment in S/N's 3449161, 3449178 and up. It integrates the audio switching, amplifier, and intercom system with a marker beacon receiver. Maintenance of the GMA-340 is "on condition" only and, with the exception of swapping complete units, should be performed only by a qualified avionics shop in accordance with the GMA-340 Audio Panel Maintenance Manual (Garmin P/N 190-00149-02).

Information provided in this manual is intended to aid the removal and installation of the GMA-340 unit and its associated wiring and to permit basic system functional test and adjustment.

A. Removal and Installation

See Rack-Mounted Avionics, Removal and Installation, 39-10-00.

B. System Functional Test

See "Figure 1".

(1) On Ground

The following known good equipment is required prior to performing this test: microphone, headset, speaker and avionics receivers.

(a) Lamp Test

1) Apply power to the unit by rotating the pilot intercom knob clockwise.

2) The test button checks the internal LED annunciator and marker beacon lamps. Press TEST to confirm operation of the LED's. Cover the photocell with a finger and observe that the LED annunciators dim automatically. Check the front panel back-lighting and dimming function. Each annunciator contains a lamp for illumination.



GMA-340 Audio Panel
Figure 1

Effectivity
Seneca V

(b) Fail-safe Operation Check

1) Turn the unit off by rotating the pilot intercom knob counterclockwise.
2) Check fail-safe operation by exercising the COM 1 microphone, microphone key and audio over the headphones.
3) Turn the unit back on to continue testing.

(c) Transceiver Operational Check

1) Perform a ramp test radio check by exercising the installed transceivers, microphone, microphone key and audio over the headphones and speaker.
2) Verify that communications are loud and clear and push-to-talk (PTT) operation is correct.

(d) Intercom System (ICS) Check

1) Set the intercom to the ALL mode (Crew and Pilot LED's off).
2) Plug in headsets at each ICS position.
3) Adjust squelch and volume for each position and verify that the ICS is working properly.
4) Check Pilot and Copilot ICS positions for isolation and proper operation of volume and squelch controls.
5) Press the PA button. Verify that microphone audio is heard over the speaker.

(e) Aircraft Receivers Check

1) Select the audio source corresponding to each installed avionics unit and check for audio over the headsets.
2) Check for Pilot/Copilot audio isolation when pressing the COM 1/2 button.
3) Press the SPKR button and verify that any selected audio is heard over the speaker.

(f) Music System Check

Connect a stereo music source to MUSIC 2. Press the CREW button to set the ICS to the crew mode. Verify that stereo audio is heard in the passenger headsets only.

(2) In Flight

Verify proper operation of the marker lamps and marker audio, including the marker audio mute function. Check proper operation of the marker sensitivity selection (using the SENS button) by flying towards the outer marker position initially using HI sensitivity. When the OM audio is just barely audible in the headset, switching to LO sensitivity should reduce or eliminate the audio.

C. Adjustment

CAUTION: USE ONLY A 2 MM (MAX BLADE WIDTH) FLAT-BLADE NON-CONDUCTIVE SCREW DRIVER AS AN ADJUSTMENT TOOL. BE CAREFUL WHEN INSERTING ADJUSTMENT TOOL THROUGH THE TOP COVER. THE UNIT MAY BE DAMAGED IF AN ADJUSTMENT TOOL IS ACCIDENTALLY FORCED AGAINST UNINTENDED COMPONENTS OR CIRCUIT BOARD PATHS.

The following adjustments can be made through access holes in the top cover of the GMA 340:

(1) Marker Beacon Audio Level.

Counter-clockwise adjustment increases the marker audio level.

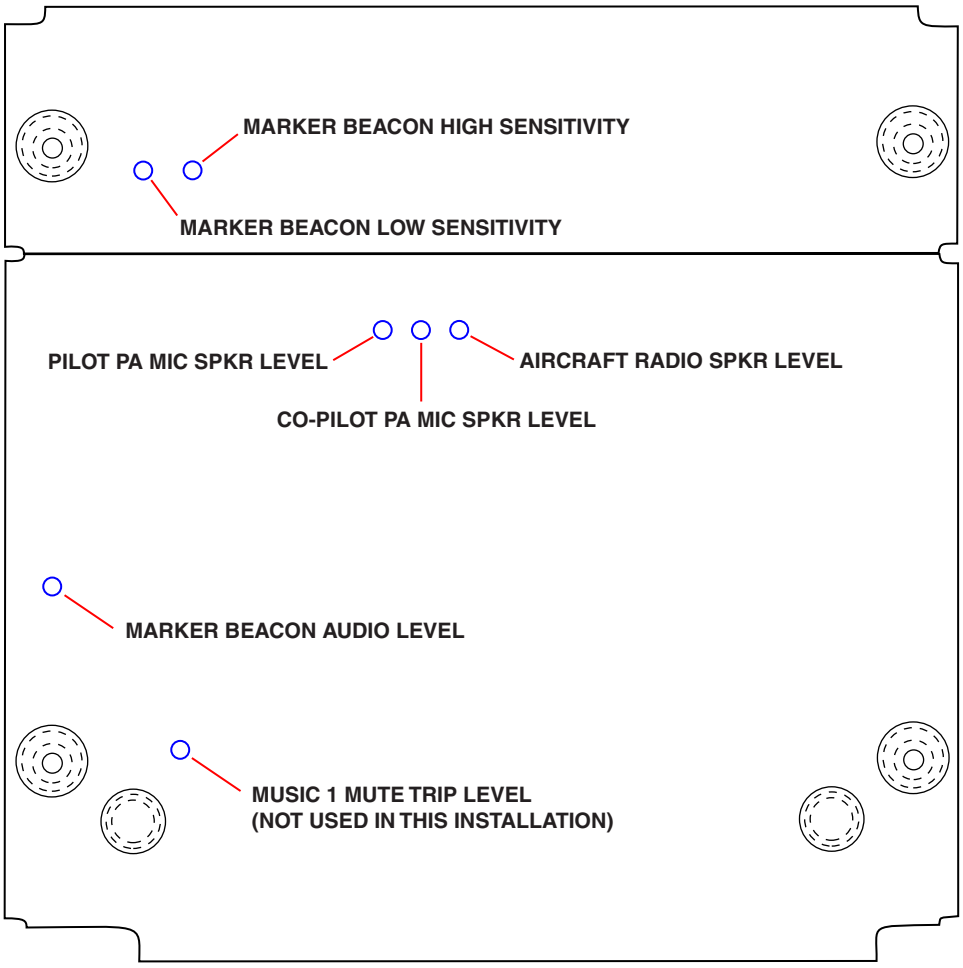
(2) Marker Beacon Sensitivity.

Clockwise (CW) adjustment increases the sensitivity. LOW sensitivity can be adjusted without affecting HIGH sensitivity setting. Adjusting HIGH sensitivity will, however, affect the LOW sensitivity. If the HIGH sensitivity setting is adjusted, then the LOW sensitivity setting should be checked and adjusted afterwards, as needed. If your GMA-340 top cover does not have the marker beacon sensitivity adjustment access holes as indicated in “Figure 2”, and you need to adjust the sensitivity, contact Garmin for instructions.

(3) Aircraft radio speaker output level.

(4) Pilot PA microphone speaker output level.

(5) Copilot PA microphone speaker output level.



GMA-340 Audio Panel Adjustments
Figure 2

Effectivity
Seneca V

2. M1091 Summing Amplifier

Optional in S/N's 3449301 thru 3449322 only, and standard equipment with Garmin G600. (PIR-PPS60205, Rev. A.)

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

The Baker Electronics M1091 Summing Amplifier is mounted on the underside of an access cover in the cabin floor, just aft of F.S. 100.00, on the pilot's side. The Summing Amp. is connected to the GMA-340 Audio Panel, the audio switching amplifier, and intercom system. It converts warning tones from up to six audio channels into one audio output. Maintenance is "on condition" only and, with the exception of swapping complete units, should be performed only by a qualified avionics shop.

Information provided herein is intended to aid the removal and installation of the M1091 Summing Amp. and its associated wiring and to permit basic system functional test and adjustment. In addition, an electrical schematic is provided in 91-23-50.

When replacing the Summing Amplifier Assy. the only adjustment required is to ensure that any unused channels are turned off. Proceed as follows:

A. Setup

(1) Remove the four #4-40 x 3/8 pan head phillips screws and four #4 lockwashers located on the sides of the M1091.

(2) Remove the two D-Subminiature connector male screw locks located on either side of the connector.

(3) Remove the case of the M1091.

(a) Remove the case by tilting the side of the case opposite the connector up at a 30 degree or greater angle.

(b) Slide the case toward the connector and remove it.

B. Adjustment

See "Chart 1" and "Figure 3" on page 23505.

(1) Input channels not used on the M1091 will be turned OFF by adjusting the corresponding channel potentiometer fully counter clockwise.

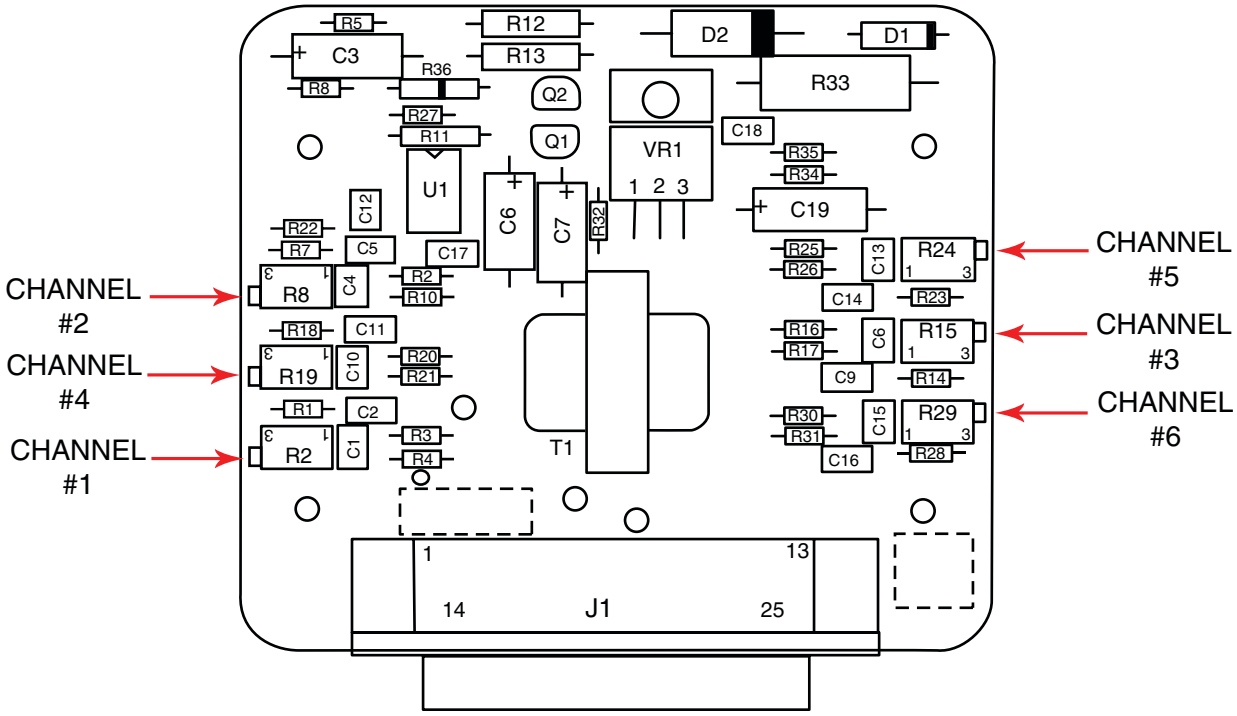
(2) Input channels that are used on the M1091 will be turned on by adjusting the corresponding channel potentiometer fully clockwise.

(3) After all channels have been adjusted, reinstall the unit case and the case and connector screws previously removed.

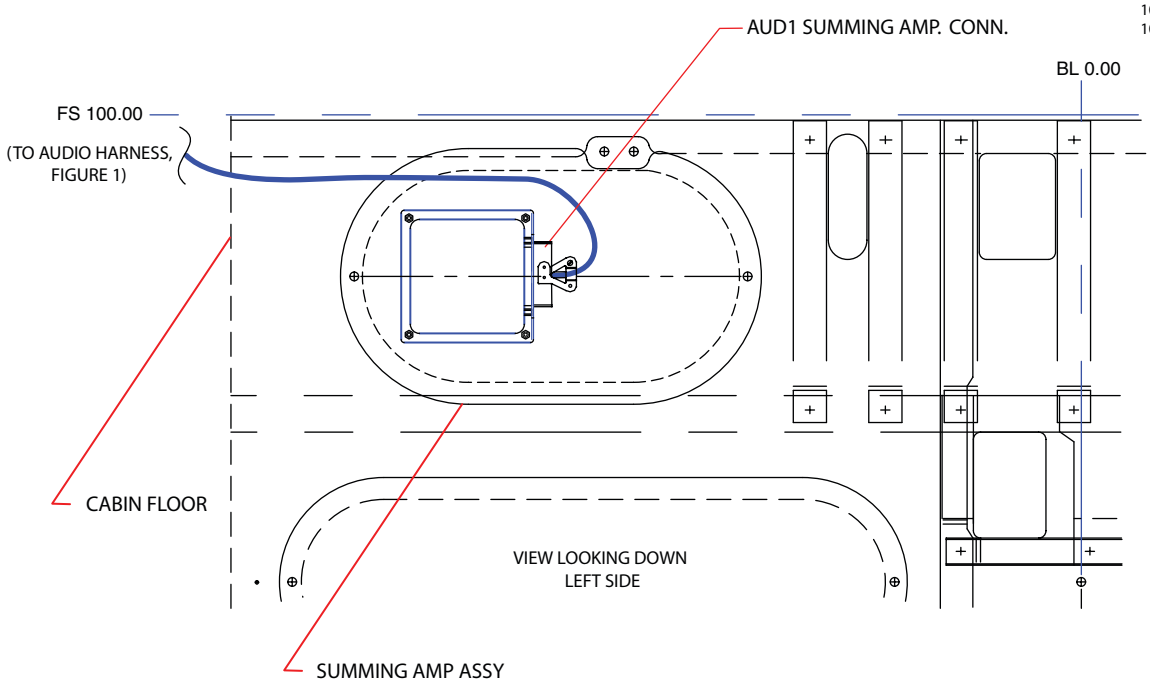
CHART 1
M1091 SUMMING AMPLIFIER CONFIG PAGE

MODEL	
PA34-220T	**AUDIO SOURCE**
CHANNEL 1	MODE S TRANSPONDER
CHANNEL 2	RADAR ALTIMETER
CHANNEL 3	IHAS OR TAWS B
CHANNEL 4	UNUSED
CHANNEL 5	UNUSED
CHANNEL 6	UNUSED

PPS60205



105121 W
105802 E



M1091 Summing Amplifier
Figure 3

Effectivity
Seneca V

C. System Test

Activate each audio source and verify that the desired audio output is produced.

3. Garmin GMA 1347 Audio Control Panel

S/N's 3449179 thru 3449508

The GMA 1347 integrates NAV/COM digital audio, intercom system and marker beacon controls. The Audio Panel communicates with both GIA 63Ws using RS-232 digital interface. The GIA 63W contains the GPS receiver, VHF COM/NAV receivers.

Maintenance of the GMA 1347 is "on condition" only and, with the exception of swapping complete units, should be performed only by a qualified avionics shop. (See 34-25-01.)

Information provided in this manual is intended to aid the removal and installation of the GMA 1347 unit and its associated wiring and to permit basic system functional test. In addition, a detailed electrical schematic is provided in 91-23-50.

A. Removal and Installation

See 34-25-01, Integrated Avionics System - Garmin, Components, GMA 1347, Removal and Installation.

B. System Functional Test

See "System Functional Test" on page 23501 and "Figure 1" on page 23501.

See GMA 1347 Test in 34-25-01.

4. Garmin GMA 1360 Audio Control Panel

S/N's 3449509 and up

The GMA 1360 integrates NAV/COM digital audio, intercom system and marker beacon controls. The Audio Panel communicates with both GIA 64s using RS-232 digital interface. The GIA 64 contains the GPS receiver, VHF COM/NAV receivers.

Maintenance of the GMA 1360 is "on condition" only and, with the exception of swapping complete units, should be performed only by a qualified avionics shop. (See 34-25-02.)

Information provided in this manual is intended to aid the removal and installation of the GMA 1360 unit and its associated wiring and to permit basic system functional test. In addition, a detailed electrical schematic is provided in 91-23-50.

A. Removal and Installation

See 34-25-02, Integrated Avionics System - Garmin, Components, GMA 1360, Removal and Installation.

B. System Functional Test

See "System Functional Test" on page 23501 and "Figure 1" on page 23501.

See GMA 1360 Test in 34-25-02.

5. Passenger No. 2 Push-to-Transmitt (PTT) (Optional)

A. Description

S/N's 3449479 and up may be equipped with an aft passenger intercom with two-way radio communications capability. This system is integrated into the GMA 1347 (G1000) / GMA 1360 (G1000 NXi) Audio Panels and consists of Mic / Phone jack assemblies located above the armrest in the right interior side panel adjacent to the right aft-facing seat. A push-to-transmit switch is co-located with the jacks. The system is designed and wired such that the pilot has transmit priority over the passengers/copilot.

B. Troubleshooting

See Chart 9 and Chart 10 in 34-25-01 (G1000) or in 34-25-02 (G1000 NXi).

C. Push-to-Transmit (PTT) Switch and Phone/Mic Jacks

The Push-to-Transmit (PTT) Switch and Phone/Mic Jacks are mounted in the interphone cover plate assembly in the right lower interior side panel adjacent to the right aft facing seat.

(1) Interphone Cover Plate Assembly

The cover plate assembly is secured in the interior panel by friction at the bottom and spring clips at the top.

(a) Removal

1) Remove electrical power from the airplane.
2) Grasping the top of the cover plate assembly with your fingertips, press down and pull out to free the spring clips.
3) Lift the cover plate assembly out of the interior panel sufficient to disconnect the harness connectors.
4) Remove the cover plate assembly.

(b) Installation

1) Place the cover plate assembly adjacent to the opening in the interior panel and connect the two harness connectors.
2) Install the cover plate assembly in the interior panel bottom first and secure it by pressing the spring clips home at the top.
3) Conduct an operational test of the intercom system.

(2) Push-to-Transmit (PTT) Switch

The PTT switch (Passenger 2) is press fit into the Interphone Cover Plate Assembly. Its wiring is soldered at the switch and to Pin 2 of the adjacent Mic Jack.

(a) Removal

1) Remove the Interphone Cover Plate Assembly per Removal, above.
2) Cut the wire leading from the PTT switch to the Mic Jack.
3) Press the PTT switch up and out of its hole in the cover plate assembly.

(b) Installation

NOTE: If installing a new PTT switch, solder wire pigtails onto Pins 1 and 2 of the new switch. Use 22 AWG wire unless otherwise specified in 91-23-50 Figure 5, Sheet 4.

1) Feed wire pigtails through hole in cover plate assembly and press switch into hole.
2) Splice wire pigtails to wires from the Mic Jack.
3) Reinstall Interphone Cover Plate Assembly per Installation, above.

(3) Mic and Phone Jacks

The Mic and Phone jacks are mounted in the Interphone Cover Plate Assembly and secured by an external jamnut.

(a) Removal

1) Remove the Interphone Cover Plate Assembly per “Removal” on page 23507.
2) Remove the jamnut and remove the jack.
3) Cut the wire leading from the PTT switch to the jack.

(b) Installation

NOTE: If installing a new jack, solder wire pigtails from the old harness connector as shown in 91-23-50 Figure 5, Sheet 4 for the appropriate jack. If a mic jack, solder additional wire pigtails onto Pins 1 and 2 of the new jack to accomodate the PTT switch wiring. Use 22 AWG wire unless otherwise specified in 91-23-50 Figure 5, Sheet 4.

1) Postion the jack in the appropriate hole as shown in “Figure 4” on page 23509.
2) Secure the jack with a jamnut.
3) Splice the wire pigtails to wires to the PTT switch.
4) Reinstall Interphone Cover Plate Assembly per “Installation” on page 23507.

D. AFT Passenger Transmit Functional Checkout Procedure

(PIR-107455, Rev. New.)

The only passenger that has transmit capability is the AFT Passenger #2 (right seat facing aft). There is no setup required to enable the transmit function using the AFT Passenger #2 PTT button located on the headset jack bezel. It should be understood that pressing the AFT Passenger #2 PTT button will allow the passenger to transmit on whatever VHF radio the copilot is setup for on the GMA 1347 (G1000) or GMA 1360 (G1000 NXi) Audio Panels. There is no means to mute or isolate the AFT Passenger #2 PTT button. The AFT Passenger #2 has all the capability normally available through the GMA 1347 (G1000) or GMA 1360 (G1000 NXi) Audio Panels in addition to the PTT which directly connects to the copilot transmit system. The copilot can override the AFT Passenger #2 transmission by depressing the yoke mounted copilot PTT. All AFT Passenger Transmit failure modes enable the copilot to have full functionality.

Transmit using the follow checkout procedure:

(a) Plug a headsets into the AFT Passenger #2 position (right seat facing aft) and copilot’s respective phone and microphone jacks. Configure the GMA 1347 (G1000) or GMA 1360 (G1000 NXi) and PFD as desired to transmit to another aircraft, station, or handheld radio from the copilots position.

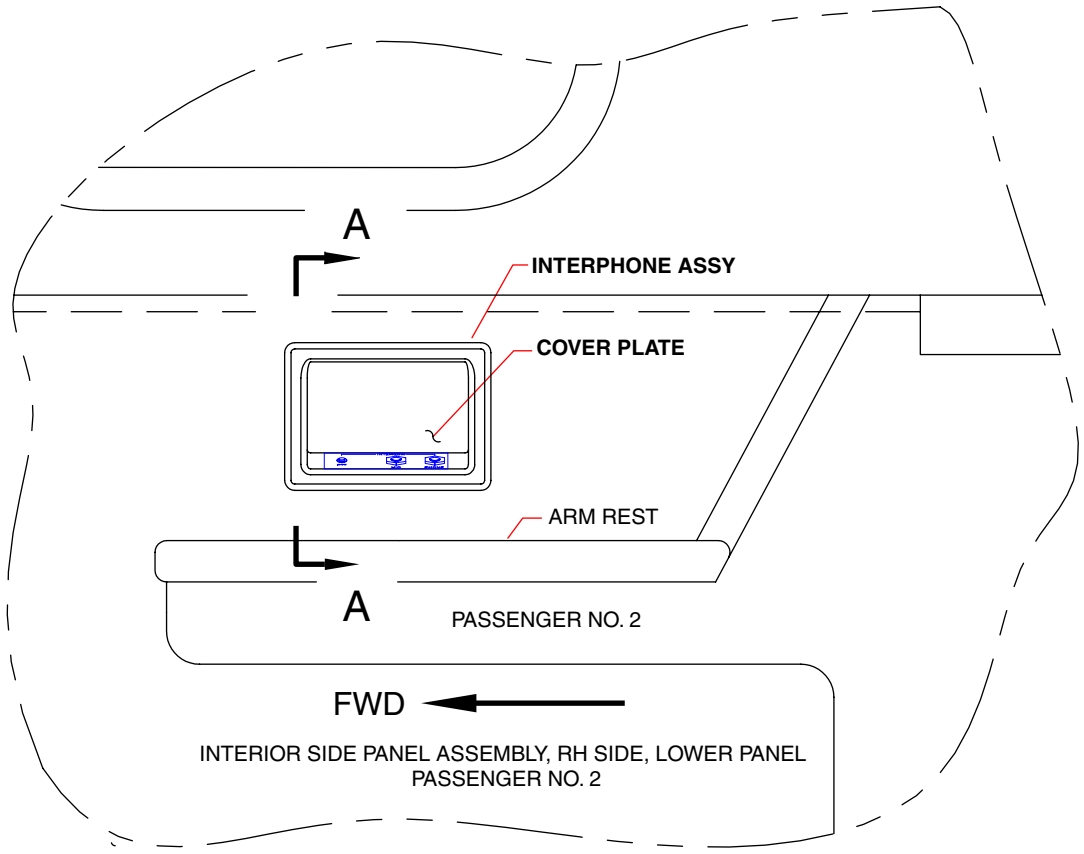
(b) Transmit from the AFT Passenger #2 position (right seat facing aft) by pressing the PTT button. Ensure the transmission is clearly received.

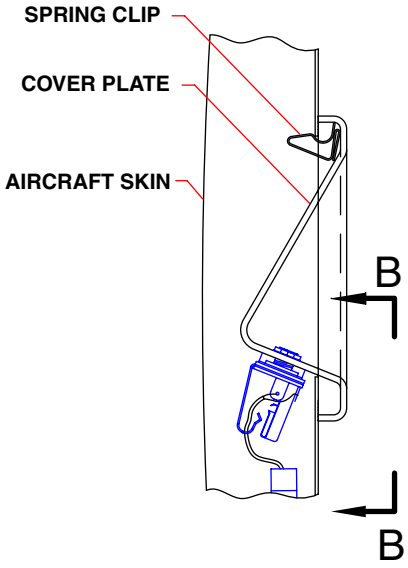
(c) Using two technicians, transmit a long count from the AFT Passenger #2 position (right seat facing aft) by continuously depressing the PTT button. While transmitting from the AFT Passenger #2 position have the second technician transmit from the copilot’s position. Ensure the copilot’s transmission is immediate and clearly interrupts the AFT Passenger #2 transmission.

(d) If either steps (b) or (c) are not functioning correctly troubleshoot and repeat until correct and clear.

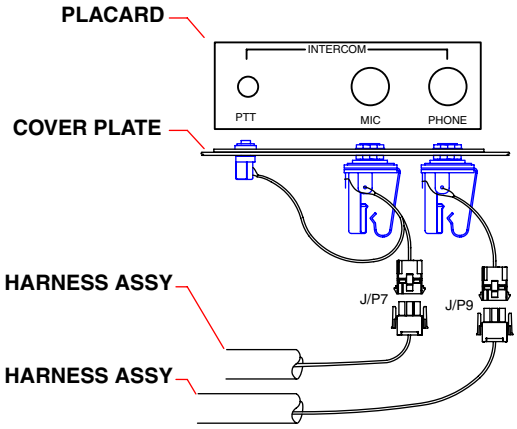
(e) If both steps (b) and (c) are operating correctly and clearly, this functional test is complete.

107435 A 1





VIEW A - A



VIEW B - B
(PARTS REMOVED FOR CLARITY)

Push-to-Transmit Switch and Phone/Mike Jacks
Figure 4

Effectivity
3449479 and up

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STATIC DISCHARGING

1. Static Wicks

   A. Description

   These airplanes are equipped with nine (9) static wicks: one (1) on the outboard end of each aileron; one (1) on the inboard end of each aileron; one (1) in the center of each flap; one (1) on each outboard end of the stabilator; and one (1) on the left-hand side of the rudder at the bottom.

   B. Inspection

   Each 100 hours or annually, whichever comes first, inspect static wicks as follows:

   (1) General appearance and physical condition.
   (2) Security of attachment to airframe.
   (3) Discharge points visible.
   (4) Resistance = 0.5 ± 20% megohms (500 to 1,000 volt megohmmeter).
   (5) Base resistance to airframe (one (1) ohm maximum).

2. Bonding Straps

   NOTE: Perform Electrical Bonding - On Condition Inspection, 51-80-00, whenever a bonding strap is installed/reinstalled.

   To aid in dissipating static electricity buildup the ailerons, stabilator, stabilator tab, and rudder are bonded to either the control's hinge or spar.

   When replacing the jumper assemblies (bonding straps), secure the end of the jumper that mounts to the control's hinge or spar as follows:

   A. Clean an area of 1 1/2 times the diameter of the jumper's washer down to bare metal.

   B. Attach the jumper and washer to the control's hinge or spar.

   C. Seal the cleaned area with waterborne, chromated, fluid resistant, epoxy primer (i.e. - PRC Desoto) and acrylic lacquer.

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CHAPTER

24

ELECTRICAL POWER

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CHAPTER 24

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GENERAL

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

This chapter contains instructions for correcting difficulties which may arise in the operation of the electrical system. It includes a general description and function of each part of the system along with test and adjustments of the various components.

1. Description and Operation

The electrical system is 28 Vdc, negative ground, consisting of:

A.	Two engine-driven 60-amp, alternators (Seneca IV)	standard, or
	Two engine-driven 85-amp, alternators (Seneca V)	standard
B.	Two Lamar Alternator Control Units	standard
C.	One Digital volt meter (See Note below)	standard
D.	Two ammeters (See Note below)	standard
E.	Dual main power distribution bus/tie bus	standard
F.	One 19 amp-hour lead acid battery-manifold type-Gill Seneca IV, Seneca V - S/N's 3449001 thru 3449466 less 3449459	standard
	One 19 amp-hour valve-regulated sealed lead acid battery-Concorde Seneca V - S/N's 3449459, 3449467 and up	standard
	One 6 amp-hour valve-regulated sealed lead acid battery-Concorde Seneca V - S/N's 3449459, 3449467 and up	standard
G.	One acid recovery jar with venting overboard Seneca IV, Seneca V - S/N's 3449001 thru 3449466 less 3449459	standard
H.	External power provisions	standard
I.	Visual annunciation with alternator "OUT'' sensors	standard

NOTE: Voltmeter and Ammeter are displayed on the MFD when either Avidyne Entegra or Garmin G1000 / G1000 NXi is installed.

The electrical generating system on Seneca IV models consists of two engine-driven, 60 ampere alternators, that are individually protected by 70 amp circuit breakers. The electrical generating system on Seneca V models consists of two engine-driven, 85 ampere alternators, that are individually protected by 90 amp circuit breakers.

Two solid state Lamar B 00382-1 alternator control units maintain effective alternator load sharing while regulating the system bus voltage at 28.8-volts. Also, incorporated in the alternator control units are overvoltage relays; one for each alternator circuit which prevents damage to electrical and avionics equipment in case the regulating function fails. Unregulated voltage that exceeds 32 volts automatically takes the malfunctioning alternator circuit off line. A warning light on the annunciator panel will illuminate if either alternator fails to produce current, accompanied by a zero indication on the individual ammeter. A low voltage warning light on the annunciator panel will illuminate when the electrical bus drops below 25 ± .3 volts. The loads from the electrical bus system are protected by manual reset type circuit breakers mounted on the lower right-hand instrument panel.

A 24-volt, 19 ampere hour battery is incorporated in the system to furnish power for starting and as a reserve power source in case of alternator failure; On Seneca IV models, the battery is located in a compartment located on the right side of the nose section of the airplane. On Seneca V models, the battery is located under the floor in the aft baggage compartment.

NOTE: Voltmeter and Ammeter are displayed on the MFD when Garmin G1000 / G1000 NXi is installed.

On Seneca V G1000 equipped aircraft, a hot battery bus, located in the aft baggage compartment, provides a continuous source of power for the hour meter, ELT (emergency locator transmitter) switch, forward baggage compartment light, and aft cabin courtesy lights. Because the battery bus is connected directly to the battery, power is available for these functions even when the Battery Master switch is OFF. The battery bus contains fuses to protect these circuits.

The EMERGENCY bus (See “Chart 1”) is activated by ARMing the EMER BATT switch on the overhead switch panel when main bus is de-energized. The bus is tied directly to the emergency battery via arming relay and a normally-closed relay and may be activated only when the main bus is de-actived.

The EMERGENCY bus provides power to items in “Chart 1”, and It is used for providing emergency power to systems required to land the aircraft in the event of a total electrical failure.

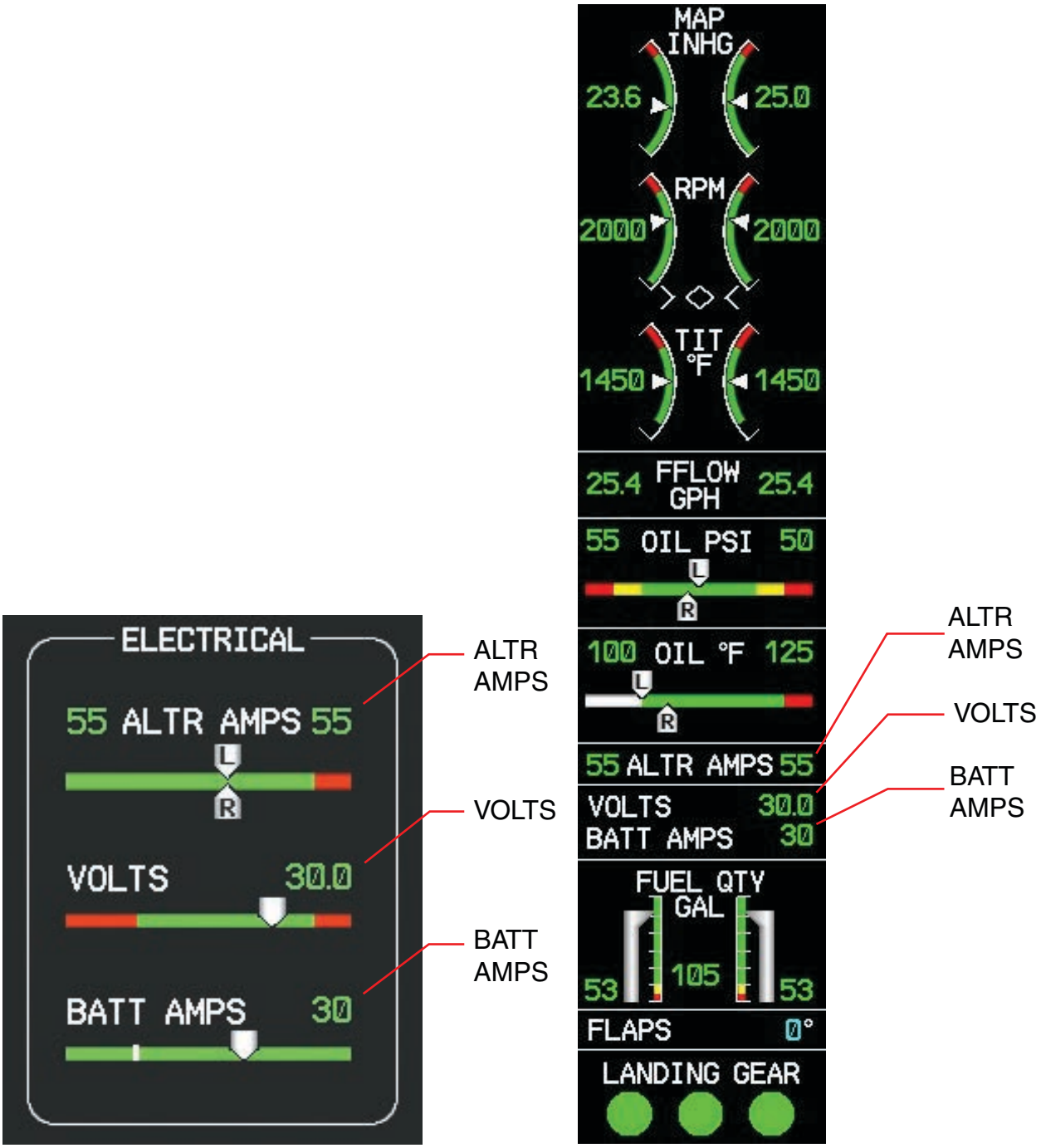
When the Battery Master switch, located on the overhead switch panel, is turned ON, the battery contactor closes, enabling current to flow from the battery to both the start contactor and the tie bus, located on the lower left of the pilots instrument panel. Should the airplane’s battery be depleted, a receptacle (located below the aft baggage compartment door) permits using an external 24 Vdc power source for engine start. With the Battery Master switch OFF, connecting an appropriate external source completes a circuit that closes the external power contactor, permitting current to flow from the external source direct to the starter contactor and the tie bus. Whether using the airplane’s battery, or external power, tie bus overcurrent protection is provided by a 250 amp in-line current limiter fuse.

Alternator amps, battery amps, and voltage are all displayed on the MFD (See “Figure 1” on page 24003) in Normal Mode or the PFD when the display backup (Reversionary) mode is enabled.

NOTE: When using only the airplane’s battery or a 24 volt external power source, the “VOLTS” CAS message will be displayed on the MFD (Garmin G1000 / G1000 NXi installation). Check the appropriate display for actual voltage.

CHART 1
EMERGENCY BUS SYSTEM (G1000 INSTALLATION)

EMERGENCY BUS
AHRS 1
ADC 1
AVIONICS LIGHTS
AUDIO / MKR
COM 1
GEA 1
GEA 2
INTEG AV 1
PFD 1
STBY INSTR



Normal Mode (MFD)

Reversionary Mode (PFD)

G1000 Display - Electrical
Figure 1

2. Contactor Maintenance

WARNING: A STARTER CONTACTOR THAT IS DAMAGED AS DESCRIBED BELOW CAN FAIL UNEXPECTEDLY IN THE CLOSED POSITION, CAUSING AN UNCOMMANDED ROTATION OF THE PROPELLER THE NEXT TIME THE MAIN POWER BUS IS ENERGIZED.

Contactor terminals may be damaged if correct procedure is not followed when loosening or tightening connections. Specifically, wrenching torque may be transmitted to components inside the contactor, causing damage that may go undetected.

Service spares replacement contactors are configured out-of-the-box as shown in “Figure 2”. The contactor is labeled with a note in red text that reads: “DO NOT REMOVE INNER NUT WHEN ASSEMBLING. USE WRENCH TO HOLD INNER NUT IN PLACE WHEN APPLYING OUTER NUT.”

“INNER NUT” refers to the nut(s) that are supplied with the contactor. When installing wire terminals onto the threaded posts, two wrenches must be used: One wrench (the back-up wrench) is placed on the inner nut to hold it stationary, while the other wrench turns and tightens the outer nut.

Contactors are manufactured by either White-Rodgers or Lamar. The correct nut torques, by manufacturer, are:

A. White-Rodgers

(1) On the main contactor terminal nuts (5/16 – 24 thread), use 45 to 55 inch-pounds (in-lb).

(2) On the control coil terminals nuts (#10 – 32 thread), use 12 to 18 in-lb.

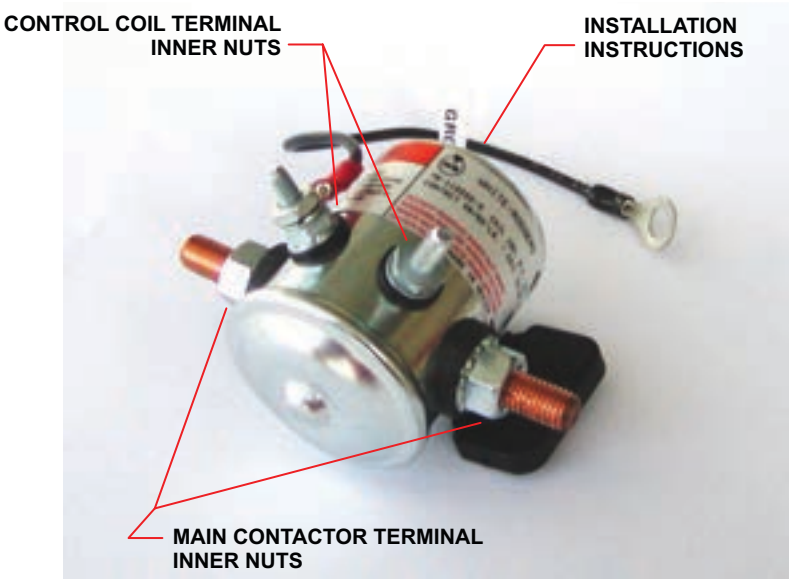
B. Lamar

(1) On the main contactor terminal nuts (5/16 – 24 thread), use 35 to 45 in-lb.

(2) On the control coil terminals nuts (#10 – 32 thread), use 10 to 15 in-lb.

NOTE: The proper installation torque for nuts installed on the threaded terminal posts is also labeled on the contactor.

NOTE: Fastener torques shown above reflect the specifications of the contactor manufacturer at the time of publication of this maintenance alert service letter, and are subject to change. Adhere to vendor installation instructions provided with any Piper approved replacement contactor.



Contactor (Typical)
Figure 2

D.C. GENERATION

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

1. Alternator System

The alternators are mounted on the accessory case on the rear of the engine (Seneca IV) or front of the engine with brackets (Seneca V) of each engine. Many advantages, both in operation and maintenance, are derived from this system.

The alternators have no armature or commutator and only a small pair of carbon brushes which make contact with a pair of copper slip rings. The rotating member of the alternator, known as the rotor, is actually the field windings. The rotor draws only 1/20th of the current output. Therefore, there is very little friction and negligible wear and heat in this area. The alternating current is converted to direct current by diodes pressed into the end bell housing of the alternator. The diodes are highly reliable solid state devices but are easily damaged if current flow is reversed through them.

The alternator system does not require a reverse current relay because of the high back resistance of the diodes and the inability of the alternator to draw current or motorize. A current regulator is unnecessary because the windings have been designed to limit the maximum current available. Therefore, the voltage control is the only control needed.

The circuit breaker panel contains two 5 ampere circuit breakers marked ALT FIELD left and right. If the field circuit breakers trip, it will result in a complete shutdown of power from the particular generating system. After a one or two minute cool-down period, the breakers can be reset manually. If tripping reoccurs, then a short exists in the alternator field.

Unlike previous systems, the ammeters do not indicate battery discharge but displays the load in amperes placed on the particular generating system. With all electrical equipment off (except master), the ammeters will indicate the amount of charging current demanded by the battery. This amount will vary, depending on the percentage of charge in the battery at the time. As the battery becomes charged, the amount of current displayed on the ammeters will reduce to approximately two amperes. The amount of current shown on the ammeters will tell immediately whether or not the alternator systems are operating normally if the following principles are kept in mind.

NOTE: The amount of current shown on the ammeter is the load in amperes that is demanded by the electrical system from the alternator. As a check, take for example a condition where the battery is demanding 10 amperes charging current; then switch on the anti-collision light.

NOTE: Take the value in amperes placarded on the panel for the anti-collision light circuit breaker (10 amps) and multiply this by 80 percent; you will arrive at a current of 8 amperes. This is the approximate current drawn by the anti-collision light. Therefore, when the anti-collision light is switched on, there will be an increase of current from 10 to 18 amperes displayed on the ammeter. As each unit of electrical equipment is switched on, the currents will add up and the total, including the battery, will appear on the ammeter.

NOTE: Voltmeter and Ammeter are displayed on the MFD when either Avidyne Entegra or Garmin G1000 is installed.

2. Troubleshooting

WARNING: ALL CHECKS AND ADJUSTMENTS OF THE ALTERNATOR AND/OR ITS COMPONENTS SHOULD BE MADE WITH THE ENGINE STOPPED. THEREFORE, TO COMPLETE SOME CHECKS OR ADJUSTMENTS, IT WILL BE NECESSARY TO REMOVE THESE UNITS FROM THE AIRPLANE AND PLACE THEM ON A TEST STAND.

Troubles peculiar to the electrical system and battery are listed in “Chart 1” on page 24303 and “Chart 2” on page 243013 along with their probable causes and suggested remedies. The wiring diagrams included in Chapter 91 will give physical breakdown of the different electrical circuits used in this airplane.

After the trouble has been corrected, check the entire electrical system for security and operation of its components.

CHART 1 (Sheet 1 of 4)
TROUBLESHOOTING ALTERNATOR

Trouble	Cause	Remedy
Zero output indicated on ammeter regardless of rpm (refer to alternator system test procedure).	Open field circuit.	With battery switch turned on, check for battery voltage from airplane's main buss field circuit to alternator field terminal. Measure voltage from ground (-) to the following points (+) in sequence; bus bar, output circuit breaker, field circuit breaker (5A), field terminals of master switch voltage regulator and alternator field terminal. Interruption of voltage through any of these points isolates the faulty component or wire which must be replaced. (See wiring schematic, Chapter 91.)
	Open output circuit.	With battery switch turned on, check for battery voltage from airplane's main buss through entire output circuit to alternator battery post. Measure voltage from ground (-) to the following points (+) in sequence; buss bar, output circuit breaker, ammeter, and alternator battery post. Interruption of voltage through any of these points isolates the faulty component or wire which must be replaced. (See wiring schematic, Chapter 91.)

NOTE: The Voltmeter and Ammeter are displayed on the MFD when either Avidyne Entegra or Garmin G1000 / G1000 NXi is installed.

CHART 1 (Sheet 2 of 4)
TROUBLESHOOTING ALTERNATOR

Trouble	Cause	Remedy
Zero output indicated on ammeter regardless of rpm (refer to alternator system test procedure). (continued)	Open field winding in alternator.	Disconnect field terminal of alternator from field wiring and check for continuity from field terminal to ground with ohmmeter (20-100 ohms) depending on brush contact resistance.
		WARNING: TURN MAGNETO SWITCH TO OFF BEFORE TURNING PROPELLER.
		(Pull propeller slowly by hand turning alternator rotor through 360° of travel.)
		If resistance is high, check brushes for spring tension and excessive wear and replace if necessary. If brushes are okay and field reads open, replace alternator.
Output indicated on ammeter does not meet minimum values specified in alternator system test procedure.	Faulty voltage regulator.	Start engine, turn on load (ref: alternator test procedure), set throttle at 2300 rpm. Check voltage at buss bar.
		Voltage should be 27.5-volts minimum. If voltage is below this value, replace regulator.
	High resistance connections in field or output circuit.	Check visually for loose binding posts at the various junction points in system, alternator battery post, lugs on ammeter, connections at voltage regulator, circuit breaker, etc. (See wiring schematic, Chapter 91.) Examine crimped terminal ends for signs of deterioration at crimp or strands of broken wire at crimp. Tighten any loose binding posts or replace bad wire terminals.
NOTE: The Voltmeter and Ammeter are displayed on the MFD when either Avidyne Entegra or Garmin G1000 / G1000 NXi is installed.		

CHART 1 (Sheet 3 of 4)
TROUBLESHOOTING ALTERNATOR

Trouble	Cause	Remedy
Output indicated on ammeter does not meet minimum values specified in alternator system test procedure. (continued)	Open rectifier.	If any of the six rectifiers pressed into the rear bell housing of the alternator open up internally, it will result in a definite limitation on the current that can be drawn from the alternator. After having checked the previous causes of low output it can be assumed that a faulty rectifier exists. See paragraph titled Inspection and Testing of Components.
Field circuit breaker trips.	Short circuit in field circuit.	Disconnect field wiring at terminal of alternator. Turn on master switch. It breaker continues to trip, proceed to disconnect each leg of field circuit, working from the alternator towards the circuit breaker until breaker can be reset and will hold. Replace component or wire which was isolated as defective. (See wiring schematic, Chapter 91.)
	Short circuit in field winding of alternator.	Disconnect field wiring at terminal of alternator. Turn on master switch. Reset breaker and if breaker fails to retrip, this isolates short circuit to field of alternator itself. Check brush holders for shorting against frame. If there are no obvious signs of a physical short circuit at field terminal or brush holder, replace alternator. (Note: intermittent short circuit.)

NOTE: The Voltmeter and Ammeter are displayed on the MFD when either Avidyne Entegra or Garmin G1000 / G1000 NXi is installed.

CHART 1 (Sheet 4 of 4)
TROUBLESHOOTING ALTERNATOR

Trouble	Cause	Remedy
WARNING: TURN MAGNETO SWITCH TO OFF BEFORE TURNING PROPELLER.		
Field circuit breaker trips. (continued)		Internal short circuiting of the field can occur at various positions of the rotor, therefore, reconnect field, reset breaker, pull propeller slowly by hand turning alternator rotor through 360° of travel. Observe circuit breaker for signs of tripping.
Battery installed that was charged with reversed polarity.	Battery charged backwards.	Remove battery. Connect load such as landing light lamp or similar load and discharge battery. Recharge with correct polarity and test each cell for signs of damage due to reversed charging.
NOTE: This type of condition can only occur in a case where a discharged battery has been removed from the airplane and put on a charger with the polarity reversed. This reversal in polarity cannot occur in the airplane due to any fault in the alternator system.		
Excessive ammeter fluctuation.	Excessive resistance in field circuit.	Check all connections and wire terminals in field circuit for deterioration such as loose binding posts, broken wire strands at terminals, etc. Tighten all connections and replace faulty terminals.
	High field circuit resistance.	If problem persists, jump across terminals of the following components one at a time until the faulty unit is isolated. a. Field 5 amp (alternator circuit protector. b. Alternator switch.
	Defective voltage regulator.	Replace voltage regulator.
NOTE: The Voltmeter and Ammeter are displayed on the MFD when either Avidyne Entegra or Garmin G1000 / G1000 NXi is installed.		

3. Alternator

A. Description

The principal components of the alternator are the brush holder assembly, front housing assembly, rectifier assembly, stator and coil assembly, rotor assembly, and the rear housing assembly. Since the alternator is a part of the power plant assembly, maintenance and inspection procedures can be obtained from the manufacturer of the engine or that particular alternator. In addition, on Seneca IV models, a shroud designed to aid in cooling covers the alternator.

B. Precautions

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

CAUTION: DISCONNECT THE BATTERY BEFORE REMOVING OR REPLACING ANY UNIT OR WIRING. ACCIDENTAL GROUNDING WILL CAUSE SEVERE DAMAGE TO THE UNITS AND/OR WIRING.

CAUTION: THE ALTERNATOR MUST NOT BE OPERATED ON AN OPEN CIRCUIT WITH THE ROTOR WINDING ENERGIZED.

CAUTION: NO POLARIZATION OF THE ALTERNATOR IS REQUIRED. ANY ATTEMPT TO DO SO MAY RESULT IN DAMAGE TO THE ALTERNATOR, REGULATOR OR CIRCUITS.

CAUTION: GROUNDING OF THE ALTERNATOR OUTPUT TERMINAL WILL CAUSE POSSIBLE DAMAGE TO THE ALTERNATOR AND/OR ITS CIRCUIT AND COMPONENTS.

CAUTION: REVERSED BATTERY CONNECTIONS WILL DAMAGE THE RECTIFIERS, WIRING OR OTHER COMPONENTS OF THE CHARGING SYSTEM. BATTERY POLARITY SHOULD BE CHECKED WITH A VOLTMETER BEFORE CONNECTING THE BATTERY. THIS AIRCRAFT IS NEGATIVE GROUND.

CAUTION: IF A BOOSTER BATTERY OR FAST CHARGER IS USED, ITS POLARITY MUST BE CONNECTED CORRECTLY TO PREVENT DAMAGE TO THE ELECTRICAL SYSTEM COMPONENTS.

C. Checking Alternator System

An ammeter is installed to enable independent output checks of each alternator, as well as electrical output/input of the battery. Should the ammeters show zero output from both alternators, check the alternators' electrical system. (Refer to Power Distribution Schematic in 91-24-30.)

(1) Ensure that the ammeter is operating properly.
(2) Disconnect the battery lead (+) at the alternator.
(3) Disconnect the field leads at the alternator.
(4) Ensure that all electrical units are off and battery is full charged.
(5) Turn on the battery switch.
(6) To check the alternator output circuit, connect a voltmeter or 24-volt test light to the battery lead and to ground. If a reading of approximately 24-volts registers on the voltmeter or the test lights, the battery circuit is operational.
(7) Should there be no indication of voltage, trace back through the output circuit until voltage is indicated. A component that allows no voltage to pass through it should be replaced.

D. Checking Belt Tension

If properly installed, tensioned, and checked periodically, the alternator drive belt will give very satisfactory service. An improperly tensioned belt, on the other hand, will wear rapidly and may slip and reduce alternator output. Accordingly, a belt should be checked for proper tension when installed, after the initial 25 hours of operation, and each 100 hours thereafter.

Using a calibrated torque wrench at the alternator nut, tension the alternator drive belt as follows:

(1) New belt (at initial installation) - 135 to 145 in.-lbs.
(2) Existing belt (after run-in) - 110 to 120 in.-lbs.

E. Removal (Seneca V only)

(1) Remove the cowling per 71-10-00.
(2) Disconnect the aircraft battery ground cable.
(3) Remove baffling as desired.
(4) Disconnect electrical leads at rear of alternator.
(5) Make note of the washer/shim stackup at the top and bottom alternator mounting bolts.
(6) Remove bolt, washers, shims, and nut from bottom mounting ear of alternator.
(7) Support the alternator. Remove bolt, washers, shims, nut, and alternator from engine.

F. Installation (Seneca V only)

(1) Position the top alternator mounting ear at the top alternator mount arm. Secure with bolt, washers, shims, and nut.

NOTE: Take care to ensure that the washer/shim stackup is installed as noted during removal.

(2) Install bolt, washers, shims, and nut at the bottom mounting ear of alternator.
  (a) Adjust belt tension as described in Check Belt Tension, above.
  (b) Torque the alternator bottom mounting bolt 220 to 260 in.-lbs. when setting belt tension.
(3) Connect electrical leads at rear of alternator. If alternator ground cable was also disconnected from the engine, reconnect and torque 90 to 110 in.-lbs.
(4) If previously removed, reinstall baffling.
(5) Connect the aircraft battery ground cable.
(6) Install the cowling per 71-10-00.

G. Overhaul

When repairing the alternator, complete disassembly may not be required. In some cases, it will only be necessary to perform those operations which are required to effect the repair. However, when servicing or inspecting vendor equipment installed in Piper airplanes, it is the user's responsibility to refer to the applicable vendor publication.

NOTE: (Seneca IV only) The drive assembly used on these alternators are not manufactured or serviced by the vendor, but are available from the engine manufacturer.

4. Alternator Control Unit — Lamar B-00382-1

A. Adjustment

The only adjustment necessary to maintain the alternator system is the adjustment of the voltage control on the alternator control unit. A voltage of 28.8 Vdc is automatically maintained. All other adjustments are made at the time of manufacture at the factory and need not be reset.

B. Bench Test — Lamar B-00382-1

CAUTION: IN-AIRCRAFT TESTING WITH ALTERNATOR RUNNING IS NOT RECOMMENDED. DO NOT BYPASS REGULATOR BUS TO FIELD AS A MEANS OF CHECKING ALTERNATOR OR OVERVOLTAGE PROTECTION WITH ALTERNATOR RUNNING.

C. Setting Up Test Equipment

(1) Remove unit from aircraft.

(2) Set up the following equipment as shown in “Figure 1” on page 243011:

Power:	Pure dc regulated power supply (A) 28.8V @ 5A Adjustable 6-12V power supply (B) @ 0.05A		
Resistors:	R1	1	
	R2	650	fine adjustment
	R3	7.2	dummy load 4A
Ammeter:	0-5A dc		
Voltmeters:	Precision meter between pins#1 and #8; between pin #3 and power (B) ground: 50 MV precision meter between shunts at pins #6 and #7.		
Switch:	Between R1 and R2		
Indicator:	28V light (Mazda #1829 or equivalent)		

(3) Set up load resistance as follows:

(a) Jumper pins #1 and #2.

(b) Set dummy load resistance for 4A @ 28.8V (approximately 7.2 ž).

(c) Remove jumper.

(4) Proceed to Voltage Control Unit Test.

D. Voltage Control Unit Test

(1) Turn power off.

(2) Mate unit with test connector.

(3) Open switch 1.

(4) Apply regulated power supply (A). Hold constant @ 28.8V ± 0.025V.

NOTE: Power supply should be adequate for load. If fluctuation occurs, assist with 28V battery to stabilize current.

(5) Allow unit 2-minute warm-up.

(6) Reset adjustment on alternator control unit for 1.0A field current.

(7) Decrease power supply (A) to approximately 28.6V until ammeter shows 3A field current.

(8) Check alternator inoperative indicator by varying power supply (B) to 10.25V ± 0.75V.

E. Equalizer Test

(1) Apply voltmeter at pins #5 and #8 (equalizer and ground). Equalizer voltage should read 5.75V ± 0.1V.

NOTE: Equalizer voltage outside 5.65 - 5.85V limits may still be acceptable because of meter calibration differences and temperature. If error is several times the stated 0.1V tolerance, have the unit rechecked by a fully equipped test facility.

(2) Set R2 for maximum resistance (650 ž).

(3) Close switch 1.

(4) Adjust R2 for 50 MV across R1.

(5) Read equalizer voltage.

(6) Subtract from reading in step 1. Difference should be 10.0V ± 0.25V

F. Overvoltage Protection Test

Refer to “Figure 2” on page 243011.

(1) Revise connection as follows:

(a) Disconnect dummy load.

(b) Add 28V relay (K1). No other connections should be on pin #2.

(c) Add a press-to-test (PTT) switch between power supply (A) and pin #2.

(d) Add resistor R3 and switch 2.

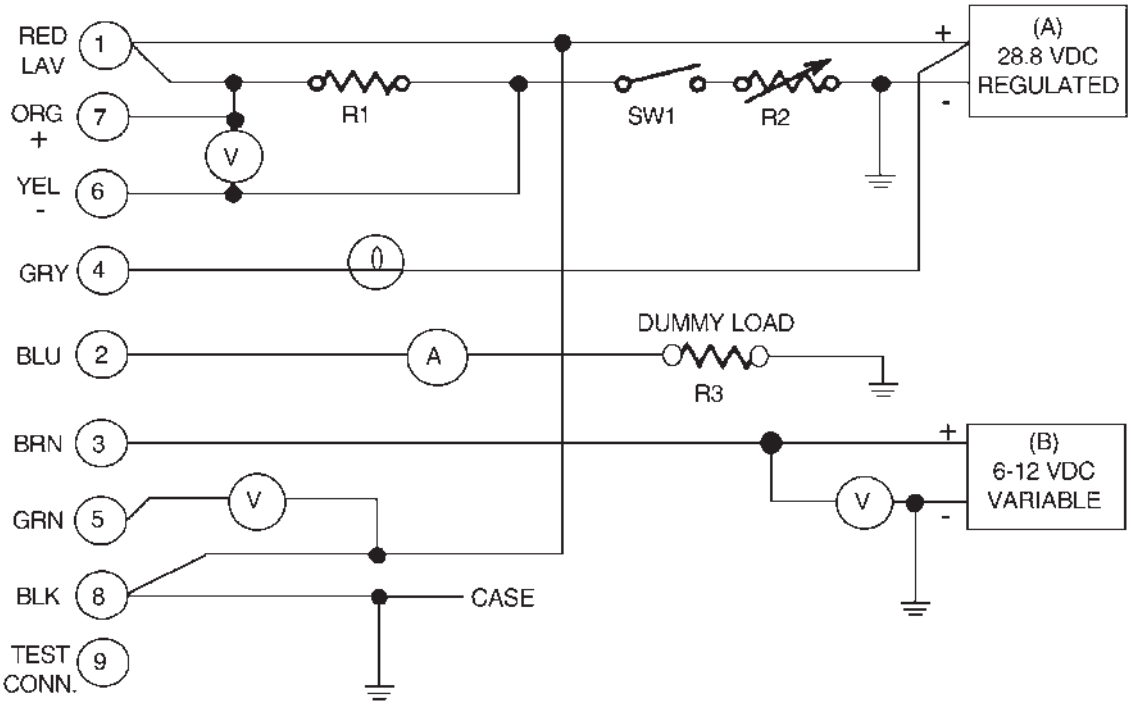
(2) Open switches 1 and 2. Set power supply (B) at 11V.

(3) Set power supply (A) to 31.8V. Depress PTT and hold for 5 seconds. Activation should occur.

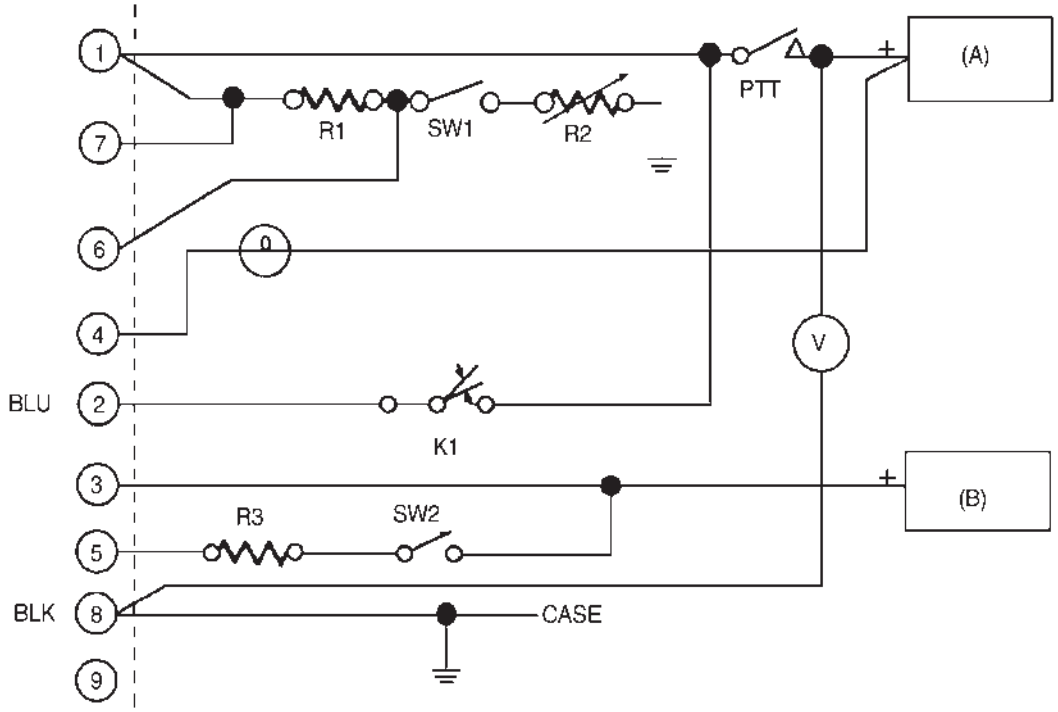
(4) Increase power (A) to 32.2V. Depress PTT for 5 seconds. Activation should occur.

(5) Close switch 2. Increase power to 33.8V. Depress PTT for 5 seconds. No activation should occur.

(6) Increase power to 34.5V. Depress PTT. Relay should activate almost instantly.



Bench Test of Alternator Control Unit — Lamar #B-00382-1
Figure 1



Overvoltage Test of ACU
Figure 2

5. Battery, Main (Dry-Charged Lead-Acid - Gill Model G-247)

Installed as standard equipment in Seneca IVs and Seneca V S/N's 3449001–3449458, 3449460–3449466.

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A. Servicing Battery

The battery is a Gill-247 19 amp hour, lead acid, manifold type. In the Seneca IV, it is mounted on a shelf in a compartment located in the right portion of the nose cone, and can be accessed via a removable access panel located on the right exterior side. In the Seneca V, it is mounted under the floor on the left side of the aft baggage compartment, and can be accessed via a removable access cover located over the floor. Fumes accumulated from the natural charging process are vented to the outside of the aircraft, through the recovery jar, requiring the vents be checked for any adverse corrosion. A positive and negative drain extends from the battery manifold and acid recovery jar that exit through vent tubes located on the bottom of the nose cone (Seneca IV see "Figure 3" on page 243015) or bottom of aft fuselage (Seneca V). Servicing of the battery should be completed every 50 operating hours or 90 days, whichever occurs first, and at every 100 hour inspection. The water level in the battery should be checked at every inspection. A check with a hydrometer should also be made. Make sure all connections are clean and tight.

B. Troubleshooting

See "Chart 2" on page 243013.

C. Removal

(1) Gain access to battery compartment.

CAUTION: ALWAYS REMOVE THE GROUND CABLE FIRST AND INSTALL LAST TO PREVENT AN ACCIDENTAL SHORT CIRCUIT OR ARCING.

(2) Disconnect the battery ground cable first; then the positive cable.
(3) Disconnect manifold overflow tube.
(4) Disconnect positive vent line.
(5) Remove battery hold down bolts.
(6) Remove battery.

D. Installation

(1) Ensure that all vent lines are free of kinks, cracks, and loose connections. Replace only with special hoses specified in Parts Catalog. (Do not replace with ordinary rubber hose.)
(2) Install battery in proper location.
(3) Connect positive vent line.
(4) Connect manifold overflow tube.
(5) Connect positive cable to battery first and secure. Connect ground cable to battery and secure.
(6) Install battery hold down bolts.
(7) Install and secure all access plates, panels, and covers.
(8) Close and secure baggage compartment door.

CHART 2 (Sheet 1 of 2)
TROUBLESHOOTING BATTERY

Trouble	Cause	Remedy
Discharged battery.	Battery worn out.	Replace battery.
	Standing too long.	Remove and recharge battery if left in unused airplane three weeks or more.
	Equipment left on accidentally.	Remove and recharge.
	Impurities in electrolyte.	Replace.
	Short circuit (ground) in wiring.	Check wiring.
	Broken cell partitions.	Replace.
Battery life is short.	Overcharge due to level	Maintain electrolyte. of electrolyte being below top of plates.
	Sulfation due to disuse.	Replace.
	Impurities in electrolyte.	Replace battery.
	Low voltage	Adjust voltage regulator.
Cracked cell jars.	Hold-down bracket loose.	Replace battery and tighten.
	Frozen battery.	Replace.
Compound on top of battery melts.	Charging rate too high.	Reduce charging rate by adjusting voltage regulator or replace regulator.
Electrolyte runs out of vent plug and/or into acid recovery jar. Excessive corrosion inside container.	Too much water added to battery and charging rate too high. Spillage from overfilling.	Drain and keep at proper level and adjust voltage regulator. Use care in adding water.
	Vent lines leaking or clogged.	Repair or clean.
	Charging rate too high.	Adjust voltage regulator.
Battery freezes.	Discharged battery.	Replace.
	Water added and battery not charged immediately.	Always recharge battery for 1/2 hour following addition of water in freezing weather.
Battery polarity reversed.	Connected backwards on charger.	Battery should be slowly discharged completely and then charged correctly and tested.

CHART 2 (Sheet 2 of 2)
TROUBLESHOOTING BATTERY

Trouble	Cause	Remedy
Battery consumes excessive water.	Charging rate too high (if in all cells).	Correct charging rate.

E. Cleaning

(1) Remove all accumulated contamination from the battery exterior with a stiff bristle brush. (Do not use a metal brush or abrasive materials.) Wipe exterior of battery and interior of manifold, including manifold top cover, with a cloth saturated with a solution of bicarbonate of soda mixed - one part soda to twenty parts of water. (Check that cell plugs are tight - do not allow soda solution to enter any cells.)

(2) Wash entire battery with clear water and dry thoroughly.

(3) Wash down the battery support and floor area, hold down supports, connectors and cable ends with a soda solution followed by clear water. Dry entire area and component parts thoroughly. Apply fresh acid resistant paint if required.

F. Battery Acid Recovery Jar

(1) Removal

(a) Remove battery.

(b) Keeping the jar upright in a vertical position, remove the acid recovery jar by removing the two (2) bracket screws that secure jar to bracket.

(2) Cleaning

(a) Visually inspect all vent lines for kinks, cracks, flexibility, and loose connections. Replace only with special hoses from parts manual. (DO NOT REPLACE WITH ORDINARY RUBBER HOSE.)

(b) (Seneca IV) Slowly pour the soda solution into the vent hoses, still attached to the bottom of the nose cone surface, using a small funnel. The solution will flow out the bottom nose cone vents.

(Seneca V) Slowly pour the soda solution into the vent hoses, still attached to the bottom of the aft fuselage surface, using a small funnel. The solution will flow out the bottom vents.

(c) Follow with a final purge of clear water to flush the vent lines and then blow dry with low pressure air. This ensures that the vent line is not kinked or restricted and that it is neutralized.

(d) (Seneca IV) Wipe down the lower right nose cone area surrounding the vents with soda solution and clear water. Apply a fresh coat of high quality wax to entire area.

(Seneca V) Wipe down the bottom aft fuselage area surrounding the vents with soda solution and clear water. Apply a fresh coat of high quality wax to entire area.

(e) Unscrew the bottom of the recovery jar and separate from the top. Remove jar pad. Observing environmental regulations, empty jar contents into a suitable container for safe disposal.

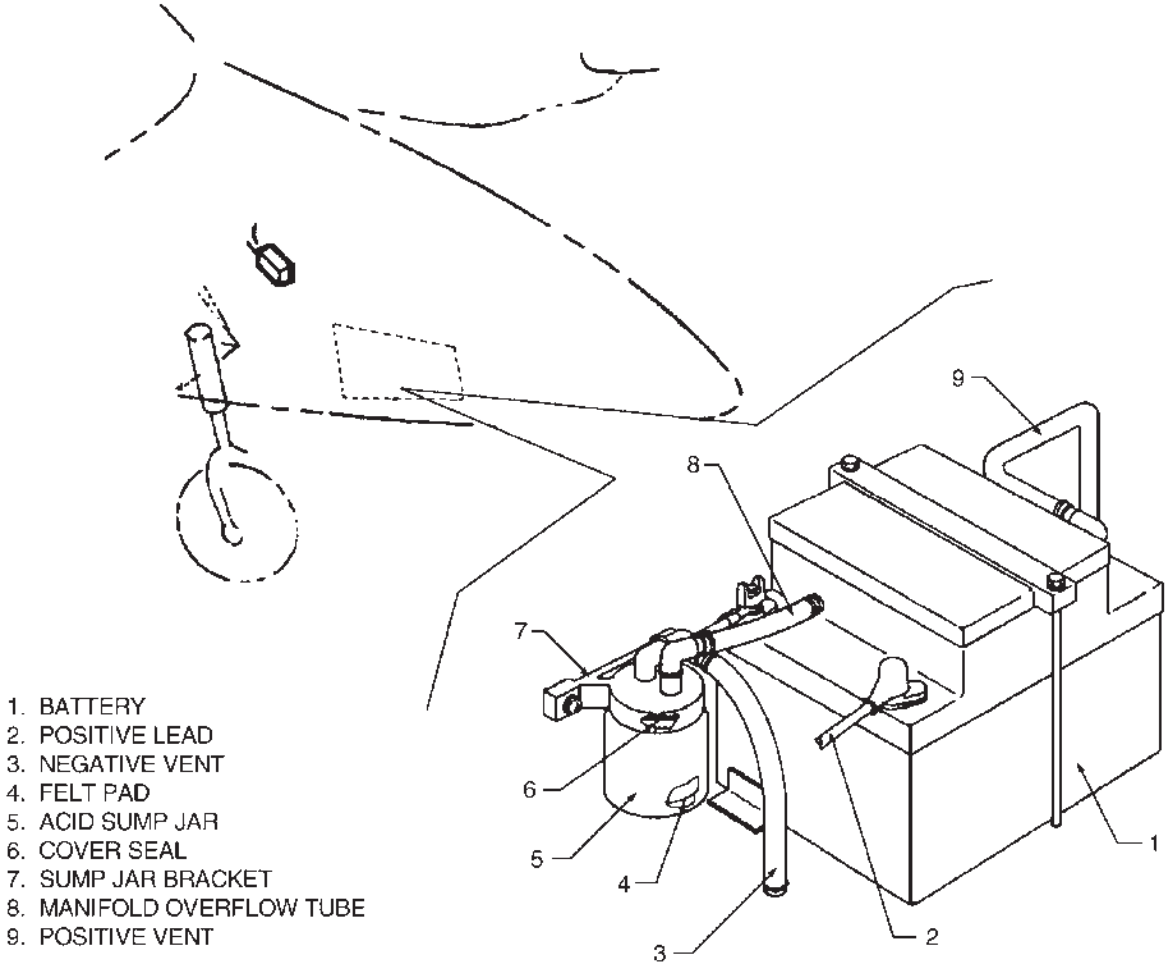
(f) Thoroughly wash and neutralize the jar, pad, top (including bracket), and the short length of vent hose still attached to the jar top with soda solution and clear water rinse.

(g) Thoroughly dry all components and recharge the jar with 0.75 bicarbonate of soda. Place dry jar pad in the jar on top of the soda charge.

(h) Screw jar back together and keep it in a vertical position.

(i) Install jar in aircraft.

(j) Install battery.



Battery Installation
Figure 3

Effectivity
Seneca IV only

Effectivity
Seneca IV - All
Seneca V - 3449001–3449458
3449460–3449466

G. Battery Charging - (Gill G-247)

CAUTION: NEVER ALLOW LEAD ACID BATTERIES, OR TOOLS USED ON THEM, TO COME IN CONTACT WITH, OR BE NEAR NI-CAD BATTERIES AND NI-CAD BATTERY TOOLS.

CAUTION: IF CHARGING IS NECESSARY, WEAR EYE PROTECTION. ENSURE THE CHARGING AREA IS WELL VENTILATED. IF CENTRAL AIR CONDITIONING IS USED, THE BATTERY CHARGING AREA SHOULD BE VENTED TO THE OUTSIDE AIR TO PREVENT HYDROGEN GASSES FROM BEING CIRCULATED THROUGHOUT THE BUILDING.

The National Electric Code forbids charging batteries installed in aircraft or within 10 feet of fuel tank areas. The battery must be removed from the aircraft for charging. Further, an aircraft battery should not be allowed to deteriorate to a point where safety of flight is jeopardized. The battery's emergency capacity should be sufficient to power the bus for thirty minutes.

(1) Remove cell plugs and ensure that vents in plugs are open and that vent valves operate freely.

(2) Check that the electrolyte level in each cell is at the bottom of the split ring.

(3) A hydrometer check of each cell should be accomplished. (Refer to Hydrometer Reading and Battery Charge in this chapter.)

(4) It is recommended that vent caps be left on the battery while charging. In addition, a wet cloth should be placed over the vent caps within the manifold.

(5) The battery may be charged at any rate, in amperes, not to exceed that point which would produce bubbling and gassing of the electrolyte or a cell temperature of 115°F in any case.

NOTE: If a cell temperature reaches the 115°F limit, the charging rate shall be reduced and the charge completed at 3 amperes or less. DO NOT CHARGE AT A HIGHER RATE WHEN CELLS ARE GASSING. Refer to Gill Service Manual G.S.M. - 682 for alternate charging methods and service procedures.

(6) If a constant current (recommended) charge is available, the charge should be started at 3 amperes and reduced in half if and when cells start gassing until fully charged.

(7) As charging occurs, if any cells sputter or flood, the electrolyte level is too high and the excess must be removed. In any case, the electrolyte level shall be adjusted at the end of the charge. The level will rise due to acid returning to the electrolyte mix, normal gassing, and expansion due to temperature rise.

(8) Thoroughly clean battery after charging to prevent remaining acid bridges which can form during charging.

H. Hydrometer Reading and Battery Charge

See “Chart 3” on page 243017.

Whenever checking the battery, ensure that all connections are clean and tight and that the fluid level is above the baffle plates. If necessary to add fluid, full cell with distilled water to the bottom of the split ring. After adding water, charge the battery until gassing before taking a hydrometer reading. Otherwise, the water and electrolyte will not be mixed, giving a false reading. Temperatures different from the established norm will effect the hydrometer readings. See “Chart 5” on page 243018 for the temperature corrections. Specific gravity values for a fully charged battery are shown in “Chart 4” on page 243017.

To adjust low specific gravity, charge the battery (see “Battery Charging - (Gill G-247)”) until it is gassing and until the specific gravity rises to no higher over a 3-hour period, the remove some electrolyte and replace with 1.300 specific gravity electrolyte. Repeat this step if, after one hour of charging, the specific gravity is still too low. DO NOT ADJUST A CELL THAT DOES NOT GAS.

CHART 3
HYDROMETER READING AND BATTERY CHARGE PERCENT

Hydrometer Reading	Percent of Charge
1280	100
1250	75
1220	50
1190	25
1160	Very little useful capacity
1130 or below	discharged

CHART 4
SPECIFIC GRAVITY AT FULL CHARGE VS TEMPERATURE

Electrolyte Temperature	Specific Gravity
47°F (8°C)	1,280 to 1,300
77°F (24°C)	1,280 to 1,290
107°F (42°C)	1,260 to 1,280
Temperature change of 30°F changes the reading 0.010.	

To adjust high specific gravity, charge the battery (see “Battery Charging - (Gill G-247)” on page 243016) until it is gassing and until the specific gravity rises no higher over a 3-hour period. Remove some electrolyte and replace with distilled water. Repeat this step if, after one hour of charging, the specific gravity is still too high.

I. Battery Discharge

The capacity of a storage battery is measured in units of ampere hours, which is the product of the electrical current in amperes multiplied by the time in hours. Although current may be obtained after the end of the time, the voltage of the battery has dropped to a point beyond which is not very useful. The ampere hours which may be obtained from a battery are greater for a long low-rate or intermittent rate discharge than for a short high-rate discharge because the voltage will drop faster at the higher discharge rate. The maximum permissible rate of discharge is limited only by the current-carrying ability of the wiring, motor, or other apparatus to which the battery is connected or by the current-carrying ability of the cell terminals and connectors and not by the plates themselves. Chart 6 lists recommended discharge rates.

J. Battery Temperature Considerations

Operation of storage batteries beyond their ambient temperature or charging voltage limits will result in excessive cell temperatures leading to electrolyte boiling, rapid deterioration of the cell and finally battery failure. The relationship between the maximum charging voltage and the number of cells in the battery is also significant, since this will determine (for a given ambient temperature and state of charge) the rate at which energy is absorbed as heat within the battery. The maximum voltage per cell should not exceed 2.35-volts and the maximum temperature should not exceed 115°F (46°C).

Low electrolyte temperatures temporarily reduce the battery capacity and the freezing point depends on the specific gravity. To prevent freeze damage, maintain the specific gravity at a reasonably high level as indicated by “Chart 7” on page 243018.

NOTE: Lead-acid batteries are subject to a constant discharge due to the internal chemical action.

CHART 5
SPECIFIC GRAVITY TEMPERATURE CORRECTION

Electrolyte Temperature °C	Electrolyte Temperature °F	Corrections	
60	140	+1.024	
55	130	+1.020	
49	120	+1.016	Add to
43	110	+.012	Reading
38	100	+.008	Add to
33	90	+.002	Reading
27	80	.000	
23	70	-.004	
15	60	-.008	
10	50	-.012	
5	40	-.016	Subtract
-2	30	-.020	From
-7	20	-.024	Reading
-13	10	-.028	
-18	0	-.032	
-28	-10	-.036	
-20	-20	-.040	
-30	-30	-.044	

CHART 6
DISCHARGE RATES

TELEDYNE Battery Type	Volts	Ampere Hours
GILL-G-247	24	19.0

CHART 7
ELECTROLYTE FREEZING POINTS

Specific Gravity	Freezing Point °C	Freezing Point °F
1.300	-70	-95
1.275	-62	-80
1.250	-52	-62
1.225	-37	-35
1.200	-26	-16
1.175	-20	-4
1.150	-15	+5
1.125	-10	+13
1.100	-8	+19

K. Battery Repairs, Storage and Service Tips

The internal parts of the battery have been designed to wear at approximately the same rate, making it uneconomical to replace any of the parts with new ones. Replacing the entire battery is simpler and cheaper.

Before storing the battery, it should be properly charged, the vent plugs put tightly in place, and the leads disconnected to prevent use during idle periods. The battery should be charged at intervals during the idle period. Before returning the battery to service, it should be thoroughly charged. The battery will be sufficiently charged when, after a 3-hour period, the specific gravity does not rise any higher with the electrolyte gassing and a charging rate of 1-1/2 amperes.

Long battery life and trouble-free service is obtained from the battery if the following simple tips are observed:

(1) Keep it clean.
(2) Keep it charged.
(3) Maintain proper electrolyte levels.
(4) Keep specific gravity equal among all cells.

L. Preparing New Dry Charged Battery for Installation

CAUTION: PRIOR TO INSTALLING A NEW DRY-CHARGED BATTERY, FOLLOW THE PREPARATION AND INSTALLATION INSTRUCTIONS FURNISHED WITH THE BATTERY BY TELEDYNE.

NOTE: The aircraft battery must be removed from the airplane if it is to be charged with a ground DC supply.

The dry-charged, Model GILL G-247 Battery shall be stored as received from the vendor. Do not remove vents seals, add acid, nor attempt to charge a dry-charged battery until the time arrives to install the battery into the airplane.

M. Corrosion Prevention

The battery should be checked for spilled electrolyte or corrosion at least each 50 hour inspection or at least every 30 days, whichever comes first. Should this be found in the box, on the terminals or around the battery, the battery should be removed and both the shelf area and battery cleaned by the following procedure:

CAUTION: DO NOT ALLOW SODA SOLUTION TO ENTER BATTERY.

(1) Clean the battery and the shelf. Corrosion effect may be neutralized by applying a solution of baking soda and water mixed to a consistency of thin cream. The application of this mixture should be applied until all bubbling action has ceased.
(2) Rinse the battery and shelf area with clean water and dry thoroughly.
(3) As necessary, paint the battery shelf area with an acid resistant paint. Allow paint to dry thoroughly.
(4) Install battery.

6. Battery, Main (Valve-Regulated Sealed Lead-Acid - Concorde Model RG24-20)

Installed as standard equipment in S/N's 3449459, 3449467 and up.

The Concorde RG24-20 valve-regulated sealed lead-acid (VRSLA) non-maintainable battery is a 24 volt battery with a rated capacity of 19.0 ampere hours, 225 cold cranking amps. Mounted under the floor on the left side of the aft baggage compartment, it can be accessed by removing the access cover which doubles as the baggage compartment floor. These batteries are fully charged when they leave the factory.

NOTE: See also the appropriate Concorde Component Maintenance Manual (CMM). See Vendor Publications under Supplementary Publications in the Introduction.

A. Troubleshooting

See "Chart 8" on page 243021.

B. Inspections

(1) Each 50 Hours or 30 Days, whichever comes first.

Visually check the battery and battery compartment for general condition, electrolyte venting, and corrosion. As required, clean per Cleaning, below.

(2) Capacity Check

(a) Airplanes operating less than 1000 hours per year:

1) Initial check at 12 months after initial installation (± 1 month).

2) As long as the capacity is above 90%, subsequent capacity checks every 12 months in service (± 1 month).

3) If the capacity is between 85 and 90%, subsequent checks every 6 months in service (± 1 month).

4) If the capacity is less than 85%, remove battery from service.

(b) Airplanes operating 1000 hours per year or more:

1) Initial check at 1000 hours after initial installation (± 100 hours).

2) As long as the capacity is above 90%, subsequent capacity checks every 1000 hours in service (± 100 hours).

3) If the capacity is between 85 and 90%, subsequent checks every 500 hours in service (± 100 hours).

4) If the capacity is less than 85%, remove battery from service.

C. Definitions

(1) Valve Regulated lead-acid (VRLA) battery - A lead-acid battery in which there is no free electrolyte and the internal pressure is regulated by a pressure relief valve. This battery requires no maintenance of the liquid level and recombines the gases formed on charge within the battery to reform water. The battery may be used in any attitude without danger of leakage or spilling of electrolyte.

(2) Rated Capacity (C_1) - Quantity of electricity in Ampere-hours (Ah) which the cell or battery is capable of delivering in 1 hour, throughout its normal service life, after full charge, under conditions defined with regard to temperature and end discharge voltage.

(3) C_1 Rate - The rate, in amperes, equal to the battery's rated C_1 capacity. For example, the C_1 rate of a battery rated at 3.5Ah is 3.5 amperes.

(4) End Point Voltage (EPV) - The voltage at which the discharge current is terminated when measuring battery capacity. Unless otherwise stated, the EPV is equal to 20.0 volts for 24 volt batteries.

(5) Open Circuit Voltage (OCV) - The voltage of the battery at rest (no charging or discharging current present). A stable OCV requires a rest of at least four hours.

CHART 8
TROUBLESHOOTING (BATTERY SYSTEM)

Trouble	Cause	Remedy
Low Voltage or No Voltage.	Battery Partially or Fully Discharged.	Perform capacity test.
	Battery fuse blown or circuit breaker tripped.	Inspect and replace fuse or reset circuit breaker.
Battery capacity less than 85% of rating	Battery beyond serviceable life.	Replace Battery.
Battery does not hold charge.	Battery beyond serviceable life.	Replace Battery.
Battery gets hot while recharging.	Battery beyond serviceable life.	Replace Battery.

D. Precautions

WARNING: LOW CAPACITY HAZARD. AIRCRAFT BATTERIES ARE CERTIFIED TO HAVE CERTAIN MINIMUM CAPACITY FOR EMERGENCY OPERATIONS IN THE EVENT OF A ELECTRICAL GENERATOR SYSTEM FAILURE. NEVER USE A BATTERY THAT HAS LESS THAN 80% OF RATED CAPACITY.

WARNING: BATTERIES ON CHARGE OR DISCHARGE PRODUCE HYDROGEN GAS, WHICH CAN EXPLODE IF IGNITED. DO NOT SMOKE, USE AN OPEN FLAME, OR CAUSE SPARKING NEAR A BATTERY. CHARGE, SERVICE OR TEST THE BATTERY ONLY IN A WELL VENTILATED AREA. THE USE OF EXHAUST FANS MAY REDUCE THE RISK OF EXPLOSION.

WARNING: BATTERIES CONTAIN SULFURIC ACID WHICH WILL CAUSE BURNS. DO NOT TOUCH EYES AFTER TOUCHING BATTERY. DO NOT GET ACID IN YOUR EYES, ON YOUR SKIN, OR IN YOUR CLOTHING. IN THE EVENT OF ACID IN THE EYES, FLUSH THOROUGHLY WITH CLEAN COOL WATER FOR SEVERAL MINUTES. GET PROFESSIONAL MEDICAL ATTENTION. REFER TO BATTERY MATERIAL SAFETY DATA SHEET (MSDS) FOR ADDITIONAL INFORMATION.

WARNING: ONLY CONSTANT POTENTIAL CHARGING MAY BE DONE ON THE AIRCRAFT. DO NOT PERFORM A CONDITIONING CHARGE WHILE BATTERY IS INSTALLED ON THE AIRCRAFT. THERE MAY BE A SERIOUS RISK OF INJURY TO PERSONNEL AND / OR DAMAGE TO THE AIRCRAFT OR AIRCRAFT SYSTEMS DUE TO HIGH VOLTAGE AND GENERATION OF EXPLOSIVE GASES WHEN CHARGING CONSTANT CURRENT.

WARNING: WEAR PROPER EYE, FACE, AND HAND PROTECTION AT ALL TIMES WHEN WORKING WITH BATTERIES. KNOW THE LOCATION AND USE OF EMERGENCY EYEWASH AND SHOWER NEAREST THE BATTERY CHARGING AREA.

WARNING: ELECTRIC SHOCK HAZARD. DO NOT TOUCH UNINSULATED PORTION OF THE CONNECTOR OR THE BATTERY TERMINALS. A POSSIBILITY OF SERIOUS ELECTRICAL SHOCK EXISTS

WARNING: ELECTRIC SHOCK HAZARD. DO NOT LAY TOOLS OR OTHER METAL OBJECTS ON THE BATTERY AS ARCING OR EXPLOSION COULD OCCUR. REMOVE CONDUCTIVE JEWELRY BEFORE WORKING AROUND BATTERY, CHARGER, OR TEST EQUIPMENT.

CAUTION: ELECTRIC BURN HAZARD. DO NOT WEAR CONDUCTIVE RINGS, BELT BUCKLES, OR OTHER JEWELRY WHEN WORKING WITH BATTERIES, CHARGERS, OR TEST EQUIPMENT. DO NOT LAY TOOLS OR OTHER METAL OBJECTS ON THE BATTERY AS ARCING AND SEVERE BURNS COULD OCCUR.

CAUTION: AIRCRAFT BATTERIES ARE CERTIFIED TO HAVE CERTAIN MINIMUM CAPACITY FOR EMERGENCY OPERATIONS IN THE EVENT OF AN ELECTRICAL GENERATOR SYSTEM FAILURE. NEVER "JUMP START" AN AIRCRAFT THAT HAS A DISCHARGED OR 'DEAD' BATTERY.

CAUTION: TO PREVENT DAMAGE TO THE CONNECTOR, ARC BURNS, OR EXPLOSION, BATTERIES SHOULD NEVER BE CONNECTED OR DISCONNECTED WHILE BEING CHARGED OR DISCHARGED. BATTERIES MUST BE CONNECTED OR DISCONNECTED ONLY WHEN THE CIRCUIT IS OPEN. ENSURE THE AIRCRAFT BATTERY SWITCH, EXTERNAL POWER SOURCE, OR THE CHARGER/ANALYZER IS IN THE "OFF" POSITION BEFORE CONNECTING OR DISCONNECTING THE BATTERY. BATTERY TERMINAL PROTECTORS SHOULD BE INSTALLED WHENEVER THE BATTERY IS NOT CONNECTED IN THE AIRCRAFT OR TO THE TEST EQUIPMENT.

CAUTION: BATTERIES CONTAIN HAZARDOUS MATERIALS. KNOW THE LOCATION AND PROPER USE OF EMERGENCY RESPONSE MATERIALS. REFER TO BATTERY MATERIAL SAFETY DATA SHEET (MSDS) FOR ADDITIONAL INFORMATION.

E. Removal

See "Figure 4" on page 243023 and "Figure 5" on page 243024.

(1) Unfasten the eight (8) quarter-turn fasteners and remove the battery cover (i.e., baggage compartment floor) from the aft baggage compartment.

CAUTION: REMOVE NEGATIVE CABLE FIRST, BEFORE EXPOSING AND LOOSENING POSITIVE TERMINAL NUT, TO AVOID SHORT CIRCUITS AND ELECTRICAL BURNS.

(2) Pull back rubber battery terminal boot, loosen nuts and disconnect battery cables.

WARNING: BATTERIES ARE HEAVY. USE APPROPRIATE LIFTING DEVICES OR EQUIPMENT. USE BATTERY HANDLES WHERE PROVIDED.

(3) Remove battery hold down bolts. Lift battery strap off battery and place away from carpeting, etc.

(4) Lift battery out of compartment.

F. Installation

See "Figure 4" on page 243023 and "Figure 5" on page 243024.

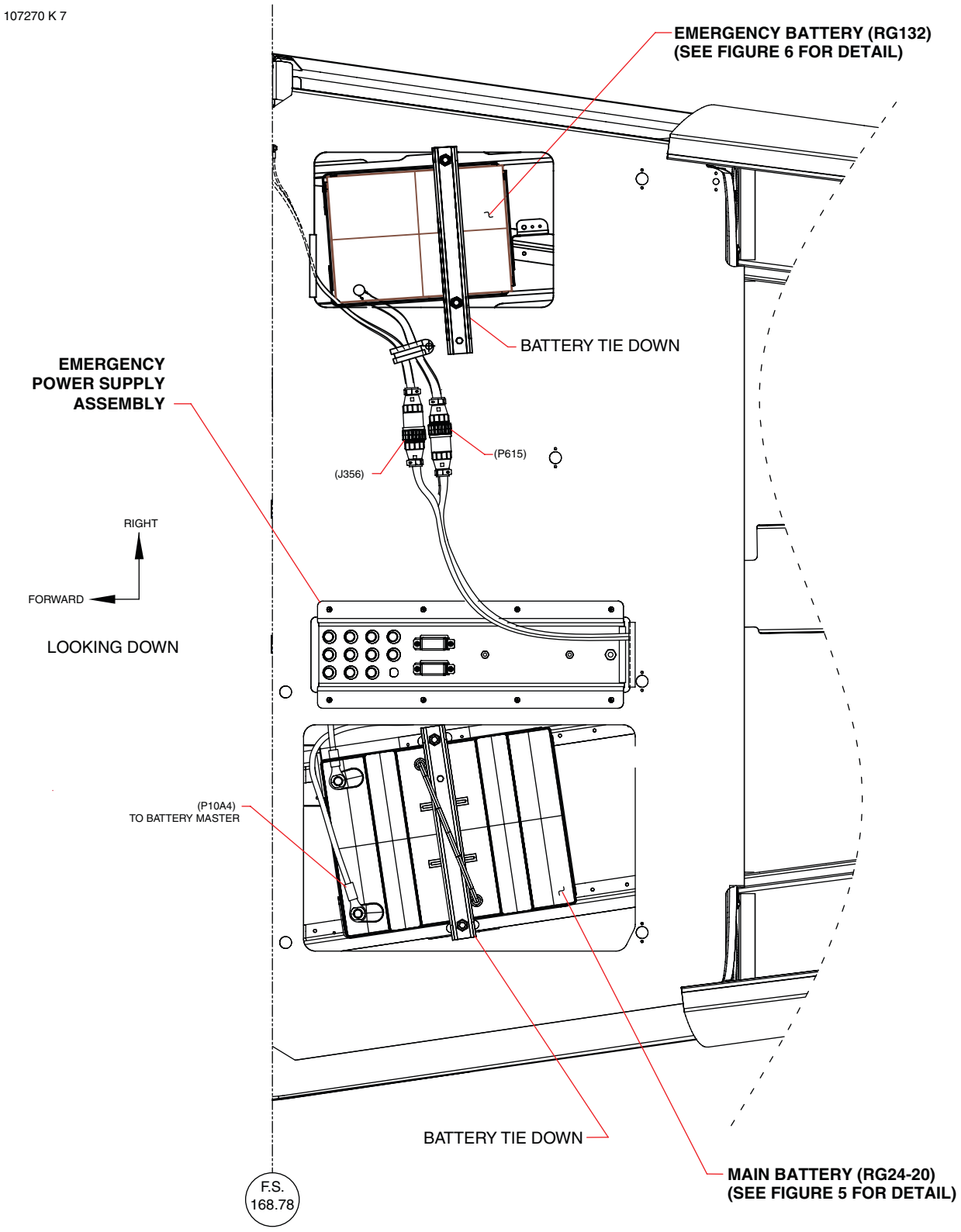
WARNING: BATTERIES ARE HEAVY. USE APPROPRIATE LIFTING DEVICES OR EQUIPMENT. USE BATTERY HANDLES WHERE PROVIDED.

(1) Place the battery into its tray under the aft baggage compartment.

(2) Place battery tie-down on battery and install battery hold-down bolts.

(3) Connect battery cables, tighten nuts and push rubber battery terminal boot back on.

(4) Install battery cover (i.e., baggage compartment floor) in of aft baggage compartment. Secure with eight (8) quarter-turn fasteners.



Battery Installations
Figure 4

G. Cleaning

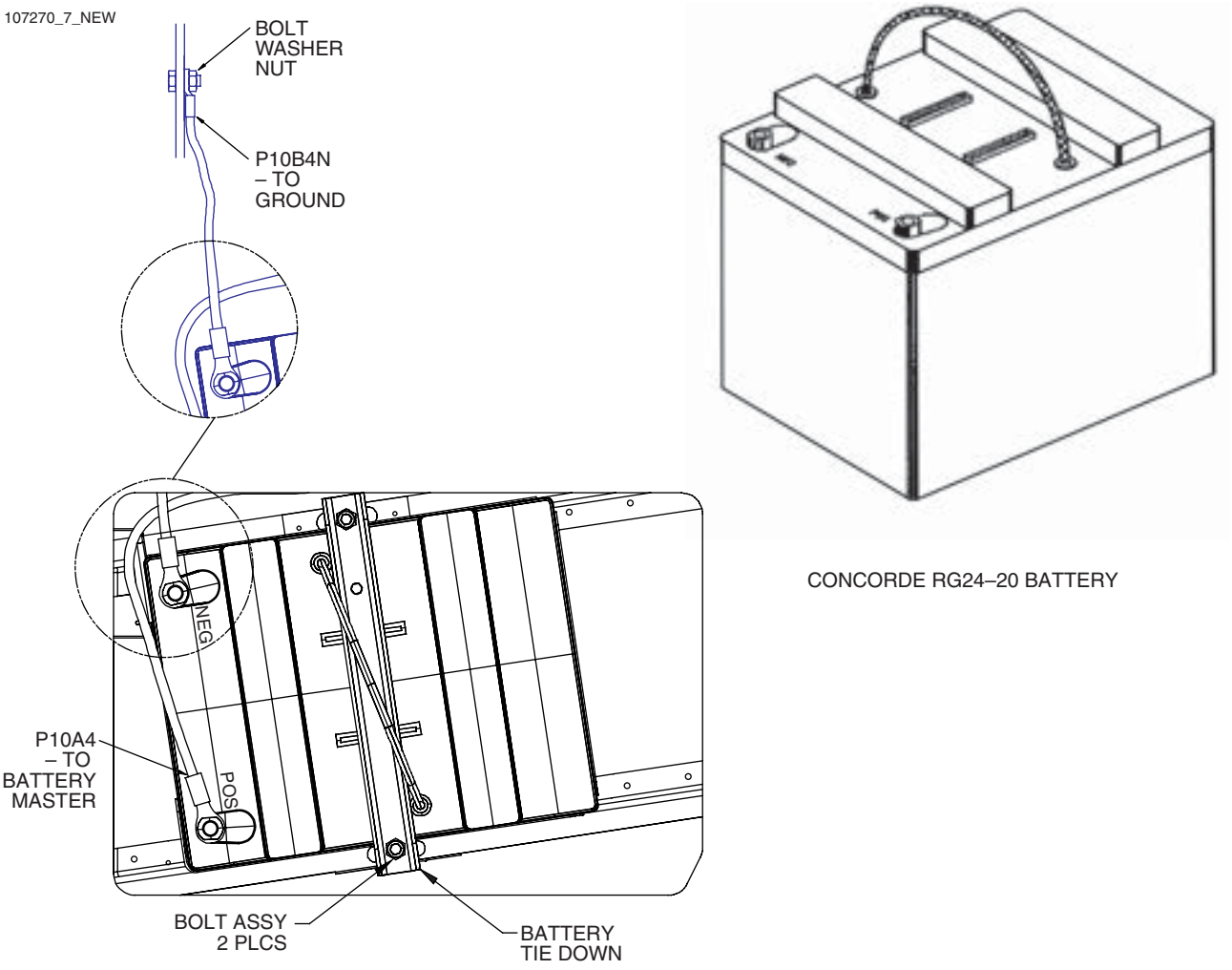
CAUTION: DO NOT USE ANY TYPE OF SOLVENT TO CLEAN THE BATTERY. SOLVENTS MAY DAMAGE THE BATTERY.

(1) Clean the outside surfaces of the battery with a lint-free shop cloth that is clean, dry, and free of oil.

(2) If the battery has caked-on dirt or grime, use a cloth dampened with tap water, then wipe dry.

(3) Wipe down the battery support and floor area, hold-down supports, connectors and cable ends.

(4) If there is any evidence of electrolyte venting / corrosion clean with a cloth saturated with a solution of baking soda and water. Rinse with a clean cloth wet with plain water. Dry thoroughly. Touchup paint as required.



Main Battery Installation
Figure 5

Effectivity
Seneca V
3449459, 3449467 and up

H. Charging - (Concorde Model RG24-20)

CAUTION: WEAR EYE PROTECTION WHEN CHARGING A BATTERY - DO NOT TAKE CHANCES. MAKE SURE THE CHARGING AREA IS WELL VENTILATED. IF CENTRAL AIR CONDITIONING IS USED, THE BATTERY CHARGING AREA SHOULD BE VENTED TO THE OUTSIDE AIR TO PREVENT HYDROGEN GASES FROM BEING CIRCULATED THROUGHOUT THE BUILDING.

The National Electric Code forbids charging batteries installed in aircraft or within 10 feet of fuel tank areas. The battery must be removed from the aircraft for charging. Further, an aircraft battery should not be allowed to deteriorate to the point where safety of flight is jeopardized. The batteries emergency capacity should be sufficient to power the essential bus for a minimum of thirty minutes.

Check battery voltage and date of manufacture prior to installation. If the open circuit voltage is less than 25.0 volts, or if 90 days has passed since date of manufacture, the battery should be charged before placing into service.

NOTE: If a battery gets very hot (external case temperature greater than 55°C (130°F) during this charge, stop the current and allow the battery to cool to room temperature before continuing the charge.

(1) If the battery voltage is below 20.0 volts, the battery should be recharged.

(2) Remove the battery from the aircraft and recharge in a well-ventilated area.

NOTE: Non-maintainable means that this type of battery cannot be opened to add fluid. Periodic maintenance is still required in the form of boost charging if the battery becomes discharged.

(3) Constant Potential (Tapering) Charge:

NOTE: Constant Potential Charging is the preferred method of charging the battery. If the battery does not pass the capacity test, the Conditioning Charge Procedure should then be used.

(a) Connect the battery terminals to the constant potential charging equipment.

(b) Apply a constant potential of 28.25 ± 0.25 volts (for 24 volt batteries) with a current capability of at least 0.2C1 amperes.

NOTE: The output current of the charging equipment should be set as high as possible.

(c) Continue charging until the charge current stabilizes (within 10%) for three consecutive hourly readings.

(4) Conditioning Charge Procedure:

WARNING: CHARGE THE BATTERY ONLY IN A WELL VENTILATED AREA BECAUSE A SIGNIFICANT AMOUNT OF HYDROGEN GAS MAY BE RELEASED FROM THE BATTERY.

(a) Discharge the battery at the C1 rate to the EPV. If battery has already been discharged to the EPV, skip this step.

(b) Connect the battery terminals to the constant current charging equipment.

(c) Charge at a constant current rate of C1/10 until the voltage on charge is 31.0 volts or higher for four (4) hours, then discontinue charging.

NOTE: If a battery gets very hot (external case temperature greater than 55°C (130°F) during this charge, stop the current and allow the battery to cool to room temperature before continuing the charge.

(d) After charging, allow the battery to cool down for at least eight (8) hours before any other tests are performed.

I. Battery Capacity Test - (Concorde Model RG24-20)

(1) Special Tools:

CAUTION: CONTACT CONCORDE FOR DETERMINATION OF EQUIVALENT TEST EQUIPMENT. SOME BRANDS OF BATTERY CHARGERS WILL DESTROY THE BATTERY.

(2) Test Procedure:

(a) If the battery is cold, warm it up to at least 20° C (68° F) before testing.

(b) Depending on the type of charger available, charge the battery Constant Potential (CP). See Constant Potential (Tapering) Charge, above.

(c) Connect the battery to the discharge equipment and discharge at the C1 rate on the label.

(d) Discharge the battery to the EPV.

(e) Record the time to EPV. The battery is acceptable for continued use if the ampere hour capacity (actual hours of discharge x ampere rate of discharge) is 51 minutes or greater (85% of rated C1 capacity or greater).

1) If the battery passes the capacity test, see Return to Service, below.

2) If the battery fails the capacity test, perform the conditioning procedure under Conditioning Charge Procedure, above. After the battery has been conditioned, perform a second capacity test.

3) If the battery passes the second capacity test, return it to service. If the battery fails the second capacity test, repeat the conditioning charge and repeat the capacity test.

4) If the battery passes the third capacity test, return it to service. If the battery fails the third capacity test, the battery should be replaced.

NOTE: If a battery gets very hot (external case temperature greater than 55°C (130°F) during charging, replace the battery.

J. Return to Service

(1) Charge the battery per “Charging - (Concorde Model RG24-20)” on page 243025.

(2) Install the battery per “Installation” on page 243022.

7. Battery, Emergency (Valve-Regulated Sealed Lead-Acid - Concorde Model RG-132)

Installed as standard equipment in S/N's 3449459, 3449467 and up.

NOTE: Charging these batteries requires a locally-made charging harness. See the appropriate Concorde Component Maintenance Manual (CMM) per Vendor Publications under Supplementary Publications in the Introduction. The charging harness requires a special connector, AMP 206153-1, four terminals AMP 66098-6, and a clamp AMP 206062-1 which can be procured locally or ordered from Piper as P/N's 555-806, 588-792 (4 ea.), and 554-612, respectively.

The Concorde RG-132 valve-regulated sealed lead-acid (VRSLA) non-maintainable battery is a 24 volt battery with a rated capacity of 6.0 ampere hours. It is mounted under the floor on the right side of the aft baggage compartment, and can be accessed by removing the access cover which doubles as the baggage compartment floor. These batteries are fully charged when they leave the factory.

A. Troubleshooting

See "Chart 8" on page 243021.

B. Removal

See "Figure 4" on page 243023 and "Figure 6" on page 243028.

(1) Unfasten the eight (8) quarter-turn fasteners and remove the battery cover (i.e., baggage compartment floor) from the aft baggage compartment.

(2) Disconnect the connector to the emergency power supply assembly.

WARNING: BATTERIES ARE HEAVY. USE APPROPRIATE LIFTING DEVICES OR EQUIPMENT. USE BATTERY HANDLES WHERE PROVIDED.

(3) Remove battery hold down bolts. Lift battery strap off battery and place away from carpeting, etc.

(4) Lift battery out of compartment.

C. Installation

WARNING: BATTERIES ARE HEAVY. USE APPROPRIATE LIFTING DEVICES OR EQUIPMENT. USE BATTERY HANDLES WHERE PROVIDED.

(1) Place the battery into its tray under the aft baggage compartment.

(2) Place battery tie-down on battery and install battery hold-down bolts.

(3) Connect the connector to the emergency power supply assembly.

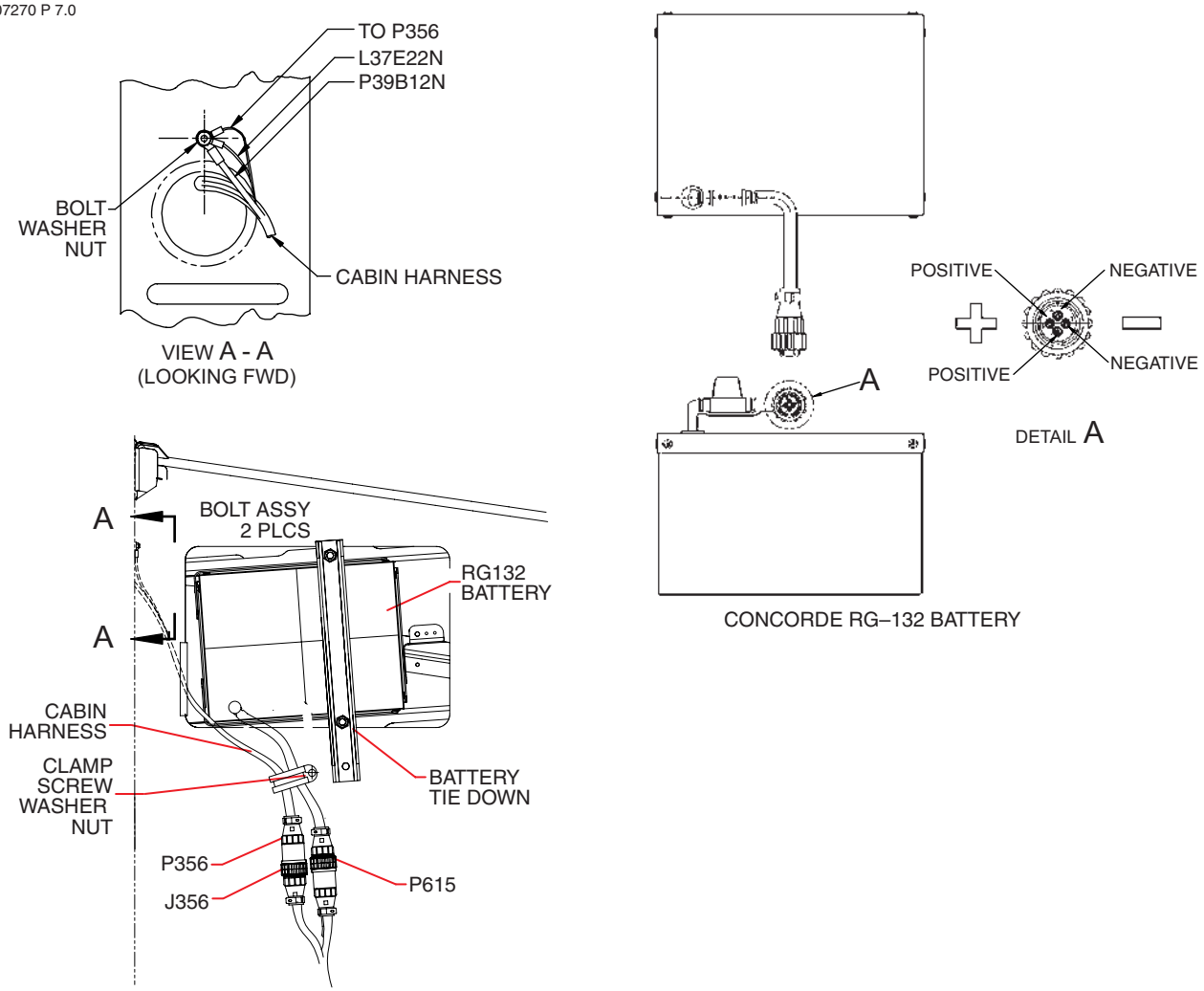
(4) Install battery cover (i.e., baggage compartment floor) in of aft baggage compartment. Secure with eight (8) quarter-turn fasteners.

D. Servicing

See Inspections, Definitions, Precautions, Cleaning, Charging, Capacity Test, and Return to Service instructions under paragraph "Battery, Main (Valve-Regulated Sealed Lead-Acid - Concorde Model RG24-20)" on page 243020, above.

NOTE: See also the appropriate Concorde Component Maintenance Manual (CMM). See Vendor Publications under Supplementary Publications in the Introduction.

107270 P 7.0



Emergency Battery Installation
Figure 6

Effectivity
Seneca V
3449459, 3449467 and up

EXTERNAL POWER

CAUTION: EXTERNAL POWER IS SUPPLIED TO THE ELECTRICAL BUS. TURN OFF ALL ELECTRICAL EQUIPMENT BEFORE APPLYING OR REMOVING EXTERNAL POWER.

1. Description

On Seneca IV models, the external power receptacle is located on the left side of the nose.

On Seneca V models, the external power receptacle is located on the right side of the fuselage below the aft cargo door.

A. Ground Power

CAUTION: WHEN USING AN AUXILIARY POWER UNIT, ENSURE THE VOLTAGE AND POLARITY ARE SET TO CORRESPOND WITH THE AIRCRAFT SYSTEM VOLTAGE AND POLARITY.

(1) Set the ground power unit to the proper voltage and polarity or, if using a single voltage ground power unit (or battery) ensure it is the proper voltage and polarity.

NOTE: If using a battery, connect the RED lead of jumper cable to the POSITIVE (+) terminal of external 24 volt battery, as appropriate, and the BLACK lead goes to the NEGATIVE (-) terminal.

(2) Turn aircraft Master Switch and all electrical equipment OFF.

(3) Insert the plug of the jumper cable into the socket located per Description, above.

(4) When using external power for operation of any of the airplane's equipment, the Master Switch must be ON.

B. Starting Through External Power Receptacle

WARNING: ENSURE THAT THE PROPELLER AREA IS CLEAR AND THAT THE INDIVIDUAL MONITORING THE START FROM THE GROUND POWER UNIT REMAINS CLEAR AS THE ENGINE STARTS.

WARNING: DO NOT ATTEMPT ANY FLIGHT IF THERE IS NO INDICATION OF ALTERNATOR OUTPUT.

CAUTION: BEFORE USING ANY EXTERNAL POWER FOR GROUND POWER, A HYDROMETER READING SHOULD BE TAKEN. IF THE READING INDICATES LESS THAN 1190, THE BATTERY SHOULD BE RECHARGED OR REPLACED BEFORE CONTINUING.

NOTE: For all normal operations using the external power jumper cables, the master switch should be OFF. However, it is possible to use the ship's battery in parallel with the battery switch ON, which will not increase amperage at that point but will give longer cranking capabilities.

(1) If the airplane's battery is nearly depleted, the following procedure should be used when using a 24 volt battery for external power:

(a) If the battery switch is being left in the OFF position, the battery need not be disconnected. If the battery switch is switched ON, it is recommended that the airplane's battery be disconnected at the negative terminal to prevent excessive loading of the external battery.

(b) Check to ensure that all electrical equipment is turned off.

(c) With the external battery connected to the receptacle, start the RIGHT engine only with normal starting procedures.

(d) Disconnect the external battery and, if applicable, reconnect the ship's battery.

(e) Switch on the battery switch and check ammeter for battery charging current.

(2) A power cart can be used instead of a battery in which case all but step a of the above procedure is applicable. The power cart (or APU) should be able to start the system even through a dead battery.

WARNING: ENSURE THAT THE PROPELLER AREA IS CLEAR AND THAT THE INDIVIDUAL MONITORING THE START FROM THE GROUND POWER UNIT REMAINS CLEAR AS THE ENGINE STARTS.

CAUTION: EXERCISE GREAT CARE DISCONNECTING GROUND POWER PLUG. DISCONNECT THE PLUG ONLY FROM THE PILOT'S SIDE OF THE AIRPLANE.

(a) Start engine and move ground power unit well away from aircraft.

(b) Turn ON Battery Master Switch.

(c) Observe ammeter indicates alternator electrical current to ammeter on both left and right systems.

(d) With the alternator(s) on line, observe charging current on ammeter.

CAUTION: IF AIRCRAFT BATTERY IS WEAK, CHARGING CURRENT WILL BE HIGH. DO NOT TAKEOFF UNTIL CHARGING CURRENT FALLS BELOW 20 AMPS.

(e) Do not takeoff until charging current falls below 20 amps.

ELECTRICAL LOAD DISTRIBUTION

1. Electrical Load Analysis

The following charts are provided to aid in determining available electrical capacity when adding or removing equipment over the life of the airplane.

A. Seneca IV

See “Chart 1” on page 24605.

B. Seneca V

(PIR-VB-2235)

(1) Standard, Meggitt EFIS, Avidyne Entegra IFDS, or Garmin G600 Options
S/N’s 3449001–3449458, 3449460–3449466 see “Chart 2” on page 24606.

(2) Garmin G1000 and S-TEC Autopilot Option
S/N’s 34494467–3449491 see “Chart 3” on page 24607.

(3) Garmin G1000 and GFC 700 Autopilot
S/N’s 3449459, 3449492–3449508 see “Chart 4” on page 24609.

(4) Garmin G1000 NXi
S/N’s 3449509 and up see “Chart 5” on page 246011.

2. USB Charging Ports

Six USB charging outlets (2 in cockpit, 4 in cabin) are available in S/N’s 3449509 and up. One is installed in each side the instrument panel and four in the passenger compartment, see 11-30-00 Figure 4.

A. Description

The Dual USB Charging Ports convert 28 VDC ship’s current to standard 5V power for any electronic product that charges using a USB connector. Two Universal Serial Bus-A (USB-A) ports are at each location and are designed as DCPs (Dedicated Charging Ports) to industry-standard protocol per the USB Battery Charging 1.2 Compliance Plan. Both ports can provide up to 2.1 amps of power simultaneously to charge any USB device.

B. Operation

(1) Electrical Performance

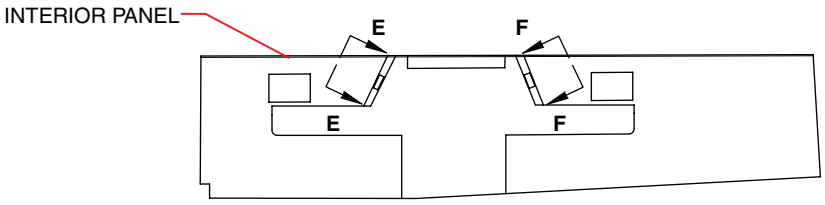
The Dual USB Charging Port converts an aircraft (DC) input voltage within the range specified to a 5V (DC) output. This output power is applied to a dual USB-A connector in accordance with the USB Implementers Forum.

The USB D+ and D- data lines communicate with the USB portable device to tell the device it is a dedicated charging port (DCP), capable of a higher current than a standard USB port. This allows the USB portable device to draw up to 2.1 Amps.

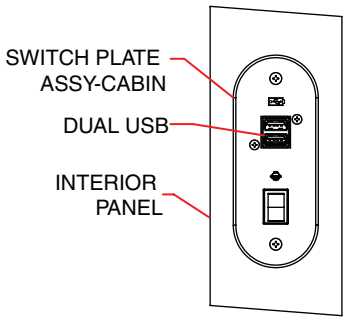
The unit is designed as a DC-to-DC converter with a series switch on each output to regulate current applied to that output. Each series switch independently reduces the output current to a safe level if the USB portable device draws excess current, is shorted or has a fault.

If the temperature of the unit becomes elevated due to a fault or excessive load, the device will seamlessly communicate with the USB portable device to lower the charge current. This allows the device to continue charging while the unit returns to a temperature within designed limits. When the temperature returns to a safe level the unit will automatically reestablish the higher charge current level with the device and continue charging.

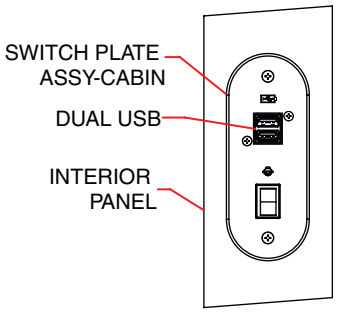
34F24A011 NEW 2.0



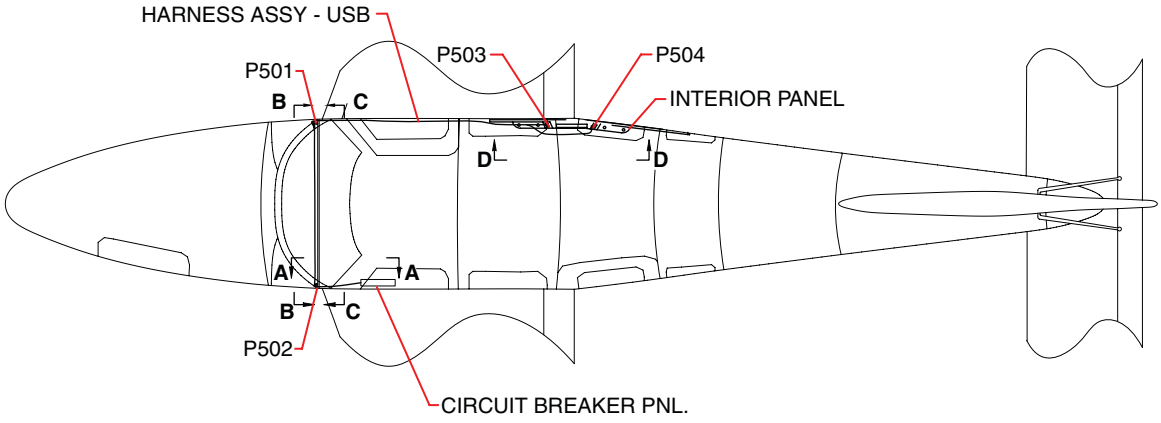
VIEW D-D
VIEW LOOKING OUTBD



VIEW E-E
VIEW LOOKING DOWN
LHS FWD



VIEW F-F
VIEW LOOKING DOWN
LHS AFT



PLAN VIEW

Effectivity
with Garmin G1000 / G1000 NXi

USB Installation
Figure 1

(2) Protective Features

(a) Short Circuit Protection

The unit is capable of surviving a short circuit event without permanent damage. The unit goes into an over-current condition so that the average current is significantly reduced and the device is protected.

(b) Over-Current Protection

The unit monitors the current draw individually on each port. During an over-current condition the voltage is reduced. If the voltage falls below 3.8 VDC the output is turned off for a period of 12 seconds. The output is then checked for continued over-current conditions every 16 milliseconds. This condition is referred to as a hiccup mode. The device stays in this mode until the over-current condition is removed, then returns to normal operation.

(c) Low Input Voltage Shutdown

If the input voltage applied to the unit drops below 10 VDC the unit will shut down until the applied voltage returns to a level within range.

(d) Over-Temperature

When the temperature of the unit becomes elevated, the unit communicates with the USB portable device to reduce the charge current output (1 amp limit). When the temperature returns to an acceptable level the unit automatically returns to a higher charge current as required (up to 2.1 amps).

C. Troubleshooting

Troubleshooting the Dual USB Charging Ports is limited to verifying correct harness wiring: 10–28 VDC at Pin A and a good ground at Pin B; and that the harness is then connected to the unit. If the harness has been verified and is properly connected and energized and the unit fails to function properly - remove and replace the unit.

D. Removal

Each unit is face-mounted to the instrument panel. Both units are removed and installed by the same procedure.

(1) Disconnect the negative (ground) battery cable.

(2) From behind the instrument panel, disconnect the electrical harness from the USB unit.

(3) At the front of the instrument panel, remove the two screws securing the USB unit to the panel and remove the unit.

E. Installation

(1) Position charging port unit behind the instrument panel and secure with two screws.

(2) Reconnect wiring harness to the USB unit.

(3) Reconnect the negative (ground) battery cable.

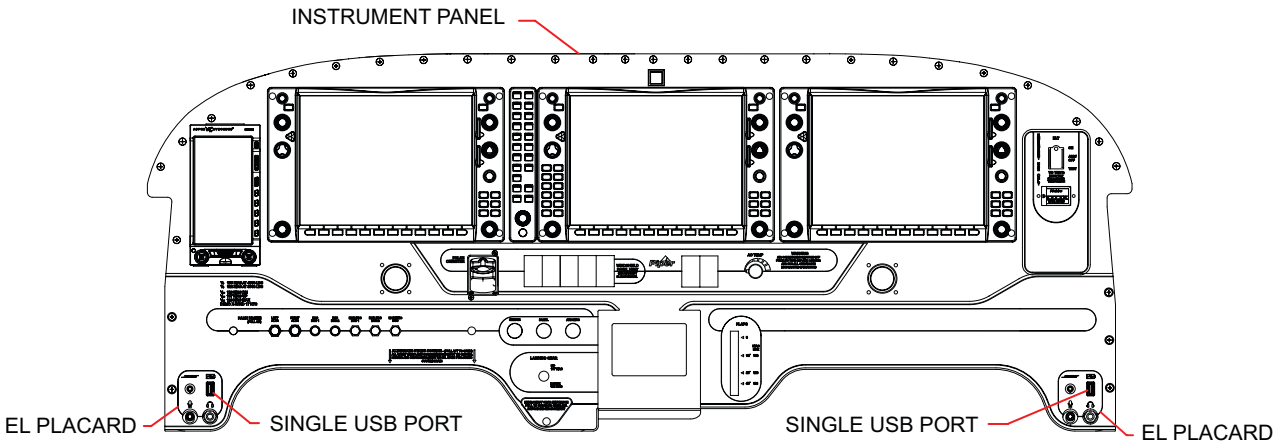
(4) Turn ON the master switch.

(5) Connect a USB device to each charging port and verify charging.

(a) If charging occurs at each port, install is complete.

(b) If charging does not occur at one or both ports, troubleshoot per “Troubleshooting”.

34F24A011 NEW 3.0



VIEW C-C
VIEW LOOKING FWD

Effectivity
with Garmin G1000 / G1000 NXi

USB Installation
Figure 2

CHART 1
ELECTRICAL SYSTEM COMPONENT LOADS (SENECA IV)

Duty Cycle		Equipment	Circuit Breaker	Load (Amps)	Optional
Cont.	Inter.				
X	X	Alternator Field (2)	5	1.57	
		Anti-Collision (Strobe)	10	2.0	
	X	Cabin Lights (4)	5	2.0	
	X	Cigar Lighter	10	4.0	
	X	Combustion Heater	15	11.0	
	X	Defroster Blower	10	3.0	
	X	Fuel Pump (2)	7.5	5.0	
	X	Heated Windshield	15	7.0	X
	X	Hydraulic Pump	25	7.5	
X		Instrument Lights	7.5	3.66	
	X	Landing Lights (2)	10	8.4	
X		Master Contactor	—	0.6	
	X	Pitot Heat	15	6.0	
X		Position Lights	10	1.5	
	X	Prop Deice		18.0	X
		Red Flood Lights	5	2.0	
	X	Stall Warning Cluster	5	1.0	
	X	Stall Warning Heat		7.5	X
	X	Starter		175.0	
	X	Starter Solenoid	10	10.0	
X		Turn & Bank	5	0.3	
	X	Electric Flaps	15	6.0	
X	X	Alternators	70		

CHART 2
ELECTRICAL SYSTEM COMPONENT LOADS (SENECA V)

Duty Cycle		Circuit Equipment	Load Breaker	(Amps)	Optional
Cont.	Inter.				
X	X	Alternator Field (2)	5 ea.	1.57 ea.	
		Anti-Collision (Strobe)	10	2.0	
	X	Cabin Lights (4)	5	2.0	
	X	Combustion Heater	15	11.0	
	X	Fresh Air Blower	15	7.0	
	X	Fuel Pump (2)	7.5 ea.	5.0 ea.	
	X	Heated Windshield	15	7.0	X
	X	Hydraulic Pump	25	7.5	
X		Panel & Switch Lights	7.5	3.66	
	X	Landing Lights	10	8.4	
X		Master Contactor	—	0.6	
	X	Pitot Heat	15	6.0	
X		Position Lights	10	1.5	
	X	Prop Deice	20	18.0	X
		Cockpit Flood Lights	5	2.0	
	X	Stall Warning Heat	15	8.0	X
	X	Starter		175.0	
X		Turn & Bank	5	0.3	
	X	Electric Flaps	15	6.0	
X	X	Alternators	90		
X		Entertainment Console	10		X

CHART 3 (Sheet 1 of 2)
ELECTRICAL SYSTEM COMPONENT LOADS (SENECA V - Garmin G1000 and S-TEC Autopilot Option)

ITEM	# UNITS	AMP /UNIT	DURATION	TIME START	TAXI	T/O	CRUISE	LAND	REMARKS	AMPS * MINUTES START	TAXI	T/O	CRUISE	LAND
Starter	2	110	INT	0.25	0	0	0	0	15 second crank time	55	0	0	0	0
Starter Contactor	2	1.47	INT	0.25	0	0	0	0	Same time as starter engaged	0.735	0	0	0	0
Primary battery contactor	1	0.6	CONT	1	15	15	60	10		0.6	9	9	36	6
Emergency power contactor	1	0.25	CONT	0	0	0	60	0		0	0	0	15	0
ALT Field	2	1.5	CONT	0	15	15	60	10		0	45	45	180	30
ALT Reg	2	0.07	CONT	0	15	15	60	10		0	2.1	2.1	8.4	1.4
Avionics Contactor	2	0.7	INT	1	0	0	0	0	Denergized when avionics “ON”	1.4	0	0	0	0
Flight Hour Meter (Hobbs)	1	0.05	CONT	1	15	15	60	10		0.05	0.75	0.75	3	0.5
Fuel Pump	2	5	INT	1	0	3	0	3	T/O, Landing and Emergency Use	10	0	30	0	30
Electric Flaps	1	6	INT	0	0	0.25	0	0.25		0	0	1.5	0	1.5
Hydraulic Pump	1	26	INT	0	0	0.25	0	0.25	Gear extend / retract approx. 6 to 7 sec.	0	0	6.5	0	6.5
Hydraulic Relay	1	0.35	INT	0	0	0.25	0	0.25	Gear extend / retract approx. 6 to 7 sec.	0	0	0.0875	0	0.0875
Strobe Lights	2	2	CONT	0	0	15	60	10		0	0	60	240	40
Position Lights	2	0.25	CONT	0	15	15	60	10		0	7.5	7.5	30	5
Taxi-Recognition Lights	2	4.2	INT	0	15	0	0	5		0	126	0	0	42
Landing Lights	2	4.2	INT	0	0	5	0	5		0	0	42	0	42
Instrument Panel Lights	1	0.025	CONT	1	15	15	60	10		0.025	0.375	0.375	1.5	0.25
Avionics Lights	1	0.25	CONT	1	15	15	60	10		0.25	3.75	3.75	15	2.5
Switch Lights	1	2.7	CONT	1	15	15	60	10		2.7	40.5	40.5	162	27
Cabin Reading Lights	4	2	INT	0	0	0	60	0	50% Utilization	0	0	0	480	0
Cockpit Flood Lights	2	1	INT	0	5	0	0	0		0	10	0	0	0
Stall Warning Heat	1	8	CONT	0	15	15	60	10		0	120	120	480	80
Pitot Heat	1	6	CONT	0	15	15	60	10		0	90	90	360	60
Combustion Heater	1	11	CONT	0	15	15	60	10		0	165	165	660	110
Defroster	1	1.6	INT	0	15	7	60	7		0	24	11.2	96	11.2
Fresh Air Blower	1	7	INT	0	15	7.5	15	5	Not typically required in cruise flight	0	105	52.5	105	35
Prop Sync	1	1.5	CONT	0	0	0	60	0	Prop Sync off during T/O and Landing	0	0	0	90	0
Prop De-Ice	1	18	CONT	0	0	15	60	10		0	0	270	1080	180
Flight Surface De-Ice	1	0.5	CONT	0	0	15	60	10		0	0	7.5	30	5
Windshield Heat	1	7	CONT	0	0	15	60	10		0	0	105	420	70
Wing Ice Light	1	0.7	INT	0	1	1	3	1		0	0.7	0.7	2.1	0.7
A/C Compressor	1	2	CONT	0	15	7	0	5		0	30	14	0	10
A/C Clutch	1	0.5	CONT	0	15	7	0	5		0	7.5	3.5	0	2.5
Avionics Cooling Fan	4	0.4	CONT	1	7	7	30	5	50% duty cycle worst case	1.6	11.2	11.2	48	8
STBY Instrument (ASPEN)	1	1.5	CONT	1	15	15	60	10	Low power cycle observed	1.5	22.5	22.5	90	15
STBY Instrument (ASPEN)	1	0.9	INT	1	5	5	20	3	Hi power cycle (33% duty additional current observed)	0.9	4.5	4.5	18	2.7
GDU 1040	3	1.3	CONT	1	15	15	60	10		3.9	58.5	58.5	234	39
GDU 1040	3	1.2	INT	1	15	5	0	0	worst case temp with heaters ON (additional amps)	3.6	54	18	0	0
GDC 74A	2	0.235	CONT	1	15	15	60	10		0.47	7.05	7.05	28.2	4.7
GRS 77	2	0.3	CONT	0	15	15	60	10		0	9	9	36	6
GRS 77 (Init)	2	1	INT	1	0	0	0	0	Peak occurs momentarily at start-up	2	0	0	0	0
GMA 1347	1	1.75	CONT	1	15	15	60	10		1.75	26.25	26.25	105	17.5

CHART 3 (Sheet 2 of 2)
ELECTRICAL SYSTEM COMPONENT LOADS (SENECA V - Garmin G1000 and S-TEC Autopilot Option)

ITEM	# UNITS	AMP /UNIT	DURATION	TIME START	TAXI	T/O	CRUISE	LAND	REMARKS	AMPS * MINUTES START	TAXI	T/O	CRUISE	LAND
GIA 63W (INTEG AV)	2	1.25	CONT	0	15	15	60	10	includes superflag output draw from WOW relay	0	37.5	37.5	150	25
GIA 63W (COM)	2	0.3	CONT	1	15	15	60	10		0.6	9	9	36	6
GIA 63W (COM)	2	4	INT	0	2	2	6	1	10% duty cycle transmit	0	16	16	48	8
GEA	2	0.15	CONT	1	15	15	60	10		0.3	4.5	4.5	18	3
GEA	2	0.17	INT	1	5	5	20	3	estimated to be 1/3 of full potential tranducer load	0.34	1.7	1.7	6.8	1.02
S-TEC Autopilot	1	0.9	INT	0	15	15	60	7		0	13.5	13.5	54	6.3
Turn Coordinator	1	0.3	CONT	0	15	15	60	10	Used as autopilot sensor only	0	4.5	4.5	18	3
GWX 68 Radar	1	2	CONT	0	15	15	60	10		0	30	30	120	20
GTS 825 TAS	1	2	CONT	0	15	15	60	10	2.25 amp boot-up, 2.0 Max op	0	30	30	120	20
GTX 33 XPDR	1	0.85	CONT	0	15	15	60	10	maximum quiescent	0	12.75	12.75	51	8.5
GTX 33 XPDR	1	0.81	INT	0	7	7	30	5	maximum full TSO reply rate (50% transmit duty cycle)	0	5.67	5.67	24.3	4.05
WX-500 Stormscope	1	0.38	CONT	0	15	15	60	10	0.38 Maximum @ 28 volts	0	5.7	5.7	22.8	3.8
KR-87 ADF	1	0.625	CONT	0	15	15	60	10		0	9.375	9.375	37.5	6.25
KN-63 DME	1	0.61	CONT	0	15	15	60	10		0	9.15	9.15	36.6	6.1
GRT 10 XM Remote Transceiver	1	0.044	CONT	0	15	15	60	10		0	0.66	0.66	2.64	0.44
GDL 69A XM Receiver	1	0.35	CONT	0	15	15	60	10		0	5.25	5.25	21	3.5
GSR 56 IRIDIUM	1	0.571	CONT	0	15	15	60	10		0	8.565	8.565	34.26	5.71
GSR 56 IRIDIUM	1	1.2	INT	0	15	5	20	0	worst case temp with heaters ON (additional amps)	0	18	6	24	0
									TOTAL AMP * MINUTES	87.72	1201.99	1455.28	5878.1	1022.70
									UTILIZATION LOAD (AMPS)	87.72	80.13	97.02	97.97	102.27

CHART 4 (Sheet 1 of 2)
ELECTRICAL SYSTEM COMPONENT LOADS (SENECA V - Garmin G1000 and GFC 700 Autopilot)

ITEM	# UNITS	AMP /UNIT	DURATION	TIME START	TAXI	T/O	CRUISE	LAND	REMARKS	AMPS * MINUTES START	TAXI	T/O	CRUISE	LAND
Starter	2	110	INT	0.25	0	0	0	0	15 second crank time	55	0	0	0	0
Starter Contactor	2	1.47	INT	0.25	0	0	0	0	Same time as starter engaged	0.735	0	0	0	0
Primary battery contactor	1	0.6	CONT	1	15	15	60	10		0.6	9	9	36	6
Emergency power contactor	1	0.25	CONT	0	0	0	60	0		0	0	0	15	0
ALT Field	2	1.5	CONT	0	15	15	60	10		0	45	45	180	30
ALT Reg	2	0.07	CONT	0	15	15	60	10		0	2.1	2.1	8.4	1.4
Avionics Contactor	2	0.7	INT	1	0	0	0	0	Denergized when avionics “ON”	1.4	0	0	0	0
Flight Hour Meter (Hobbs)	1	0.05	CONT	1	15	15	60	10		0.05	0.75	0.75	3	0.5
Fuel Pump	2	5	INT	1	0	3	0	3	T/O, Landing and Emergency Use	10	0	30	0	30
Electric Flaps	1	6	INT	0	0	0.25	0	0.25		0	0	1.5	0	1.5
Hydraulic Pump	1	26	INT	0	0	0.25	0	0.25	Gear extend / retract approx. 6 to 7 sec.	0	0	6.5	0	6.5
Hydraulic Relay	1	0.35	INT	0	0	0.25	0	0.25	Gear extend / retract approx. 6 to 7 sec.	0	0	0.0875	0	0.0875
Strobe Lights	2	2	CONT	0	0	15	60	10		0	0	60	240	40
Position Lights	2	0.25	CONT	0	15	15	60	10		0	7.5	7.5	30	5
Taxi-Recognition Lights	2	4.2	INT	0	15	0	0	5		0	126	0	0	42
Landing Lights	2	4.2	INT	0	0	5	0	5		0	0	42	0	42
Instrument Panel Lights	1	0.025	CONT	1	15	15	60	10		0.025	0.375	0.375	1.5	0.25
Avionics Lights	1	0.25	CONT	1	15	15	60	10		0.25	3.75	3.75	15	2.5
Switch Lights	1	2.7	CONT	1	15	15	60	10		2.7	40.5	40.5	162	27
Cabin Reading Lights	4	2	INT	0	0	0	60	0	50% Utilization	0	0	0	480	0
Cockpit Flood Lights	2	1	INT	0	5	0	0	0		0	10	0	0	0
Stall Warning Heat	1	8	CONT	0	15	15	60	10		0	120	120	480	80
Pitot Heat	1	6	CONT	0	15	15	60	10		0	90	90	360	60
Combustion Heater	1	11	CONT	0	15	15	60	10		0	165	165	660	110
Defroster	1	1.6	INT	0	15	7	60	7		0	24	11.2	96	11.2
Fresh Air Blower	1	7	INT	0	15	7.5	15	5	Not typically required in cruise flight	0	105	52.5	105	35
Prop Sync	1	1.5	CONT	0	0	0	60	0	Prop Sync off during T/O and Landing	0	0	0	90	0
Prop De-Ice	1	18	CONT	0	0	15	60	10		0	0	270	1080	180
Flight Surface De-Ice	1	0.5	CONT	0	0	15	60	10		0	0	7.5	30	5
Windshield Heat	1	7	CONT	0	0	15	60	10		0	0	105	420	70
Wing Ice Light	1	0.7	INT	0	1	1	3	1		0	0.7	0.7	2.1	0.7
A/C Compressor	1	2	CONT	0	15	7	0	5		0	30	14	0	10
A/C Clutch	1	0.5	CONT	0	15	7	0	5		0	7.5	3.5	0	2.5
Avionics Cooling Fan	4	0.4	CONT	1	7	7	30	5	50% duty cycle worst case	1.6	11.2	11.2	48	8
STBY Instrument (ASPEN)	1	1.5	CONT	1	15	15	60	10	Low power cycle observed	1.5	22.5	22.5	90	15
STBY Instrument (ASPEN)	1	0.9	INT	1	5	5	20	3	Hi power cycle (33% duty additional current observed)	0.9	4.5	4.5	18	2.7
GDU 1040	3	1.3	CONT	1	15	15	60	10		3.9	58.5	58.5	234	39
GDU 1040	3	1.2	INT	1	15	5	0	0	worst case temp with heaters ON (additional amps)	3.6	54	18	0	0
GDC 74A	2	0.235	CONT	1	15	15	60	10		0.47	7.05	7.05	28.2	4.7
GRS 77	2	0.3	CONT	0	15	15	60	10		0	9	9	36	6
GRS 77 (Init)	2	1	INT	1	0	0	0	0	Peak occurs momentarily at start-up	2	0	0	0	0
GMA 1347	1	1.75	CONT	1	15	15	60	10		1.75	26.25	26.25	105	17.5

CHART 4 (Sheet 2 of 2)
ELECTRICAL SYSTEM COMPONENT LOADS (SENECA V - Garmin G1000 and GFC 700 Autopilot)

ITEM	# UNITS	AMP /UNIT	DURATION	TIME START	TAXI	T/O	CRUISE	LAND	REMARKS	AMPS * MINUTES START	TAXI	T/O	CRUISE	LAND
GIA 63W (INTEG AV)	2	1.25	CONT	0	15	15	60	10	includes superflag output draw from WOW relay	0	37.5	37.5	150	25
GIA 63W (COM)	2	0.3	CONT	1	15	15	60	10		0.6	9	9	36	6
GIA 63W (COM)	2	4	INT	0	2	2	6	1	10% duty cycle transmit	0	16	16	48	8
GEA	2	0.15	CONT	1	15	15	60	10		0.3	4.5	4.5	18	3
GEA	2	0.17	INT	1	5	5	20	3	estimated to be 1/3 of full potential tranducer load	0.34	1.7	1.7	6.8	1.02
GFC 700	1	0.9	INT	0	15	15	60	7		0	13.5	13.5	54	6.3
GWX 68 Radar	1	2	CONT	0	15	15	60	10		0	30	30	120	20
GTS 825 TAS	1	2	CONT	0	15	15	60	10	2.25 amp boot-up, 2.0 Max op	0	30	30	120	20
GTX 33 XPDR	1	0.85	CONT	0	15	15	60	10	maximum quiescent	0	12.75	12.75	51	8.5
GTX 33 XPDR	1	0.81	INT	0	7	7	30	5	maximum full TSO reply rate (50% transmit duty cycle)	0	5.67	5.67	24.3	4.05
WX-500 Stormscope	1	0.38	CONT	0	15	15	60	10	0.38 Maximum @ 28 volts	0	5.7	5.7	22.8	3.8
KR-87 ADF	1	0.625	CONT	0	15	15	60	10		0	9.375	9.375	37.5	6.25
KN-63 DME	1	0.61	CONT	0	15	15	60	10		0	9.15	9.15	36.6	6.1
GRT 10 XM Remote Transceiver	1	0.044	CONT	0	15	15	60	10		0	0.66	0.66	2.64	0.44
GDL 69A XM Receiver	1	0.35	CONT	0	15	15	60	10		0	5.25	5.25	21	3.5
GSR 56 IRIDIUM	1	0.571	CONT	0	15	15	60	10		0	8.565	8.565	34.26	5.71
GSR 56 IRIDIUM	1	1.2	INT	0	15	5	20	0	worst case temp with heaters ON (additional amps)	0	18	6	24	0
									TOTAL AMP * MINUTES	87.72	1197.50	1450.78	5860.1	1019.71
									UTILIZATION LOAD (AMPS)	87.72	79.83	96.72	97.67	101.97

CHART 5 (Sheet 1 of 2)
ELECTRICAL SYSTEM COMPONENT LOADS (SENECA V - Garmin G1000 NXi)

ITEM	# UNITS	AMP /UNIT	DURATION	TIME START	TAXI	T/O	CRUISE	LAND	REMARKS	AMPS * MINUTES START	TAXI	T/O	CRUISE	LAND
Starter	2	110	INT	0.25	0	0	0	0	15 second crank time	55	0	0	0	0
Starter Contactor	2	1.47	INT	0.25	0	0	0	0	Same time as starter engaged	0.735	0	0	0	0
Primary battery contactor	1	0.6	CONT	1	15	15	60	10		0.6	9	9	36	6
Emergency power contactor	1	0.25	CONT	0	0	0	60	0		0	0	0	15	0
ALT Field	2	1.5	CONT	0	15	15	60	10		0	45	45	180	30
ALT Reg	2	0.07	CONT	0	15	15	60	10		0	2.1	2.1	8.4	1.4
Avionics Contactor	2	0.7	INT	1	0	0	0	0	Denergized when avionics “ON”	1.4	0	0	0	0
Flight Hour Meter (Hobbs)	1	0.05	CONT	1	15	15	60	10		0.05	0.75	0.75	3	0.5
Fuel Pump	2	5	INT	1	0	3	0	3	T/O, Landing and Emergency Use	10	0	30	0	30
Electric Flaps	1	6	INT	0	0	0.25	0	0.25		0	0	1.5	0	1.5
Hydraulic Pump	1	26	INT	0	0	0.25	0	0.25	Gear extend / retract approx. 6 to 7 sec.	0	0	6.5	0	6.5
Hydraulic Relay	1	0.35	INT	0	0	0.25	0	0.25	Gear extend / retract approx. 6 to 7 sec.	0	0	0.0875	0	0.0875
Strobe Lights	2	0.5	CONT	0	0	15	60	10		0	0	15	60	10
Position Lights	2	0.17	CONT	0	15	15	60	10		0	5.1	5.1	20.4	3.4
Taxi-Recognition Lights	2	0.6	INT	0	15	0	0	5		0	18	0	0	6
Landing Lights	2	0.6	INT	0	0	5	0	5		0	0	6	0	6
Instrument Panel Lights	1	0.025	CONT	1	15	15	60	10		0.025	0.375	0.375	1.5	0.25
Avionics Lights	1	0.25	CONT	1	15	15	60	10		0.25	3.75	3.75	15	2.5
Switch Lights	1	2.7	CONT	1	15	15	60	10		2.7	40.5	40.5	162	27
Cabin Reading Lights	4	0.03	INT	0	7.5	7.5	30	5	50% Utilization	0	0.9	0.9	3.6	0.6
Cockpit Flood Lights	2	0.3	INT	0	5	0	0	0		0	3	0	0	0
Cockpit Flood Dimmer	2	0.056	CONT	0	15	15	60	10		0	1.68	1.68	6.72	1.12
Stall Warning Heat	1	8	CONT	0	15	15	60	10		0	120	120	480	80
Pitot Heat	1	6	CONT	0	15	15	60	10		0	90	90	360	60
Combustion Heater	1	11	CONT	0	15	15	60	10		0	165	165	660	110
Defroster	1	1.6	INT	0	15	7	60	7		0	24	11.2	96	11.2
Fresh Air Blower	1	7	INT	0	15	7.5	15	5	Not typically required in cruise flight	0	105	52.5	105	35
Prop Sync	1	1.5	CONT	0	0	0	60	0	Prop Sync off during T/O and Landing	0	0	0	90	0
Prop De-Ice	1	18	CONT	0	0	15	60	10		0	0	270	1080	180
Flight Surface De-Ice	1	0.5	CONT	0	0	15	60	10		0	0	7.5	30	5
Windshield Heat	1	7	CONT	0	0	15	60	10		0	0	105	420	70
Wing Ice Light	1	0.9	INT	0	1	1	3	1		0	0.9	0.9	2.7	0.9
A/C Compressor	1	2	CONT	0	0	0	0	0		0	0	0	0	0
A/C Clutch	1	1.42	CONT	0	0	0	0	0		0	0	0	0	0
Avionics Cooling Fan	4	0.19	CONT	1	15	15	30	5		0.76	11.4	11.4	22.8	3.8
STBY Instrument (ASPEN)	1	1.5	CONT	1	15	15	60	10	Low power cycle observed	1.5	22.5	22.5	90	15
STBY Instrument (ASPEN)	1	0.9	INT	1	15	5	20	3	Hi power cycle (33% duty additional current observed)	0.9	13.5	4.5	18	2.7
GDU 1050	3	0.9	CONT	0.33	15	15	60	10		0.9	40.5	40.5	162	27
GDU 1050	3	1	INT	0.33	15	5	20	3	worst case temp with heaters ON (additional amps)	1	45	15	60	9
GDC 72	2	0.21	CONT	0.5	15	15	60	10		0.21	6.3	6.3	25.2	4.2
GRS 79	2	0.24	CONT	0.5	15	15	60	10		0.24	7.2	7.2	28.8	4.8
GMA 1360	1	1	CONT	1	15	15	60	10		1	15	15	60	10

CHART 5 (Sheet 2 of 2)
ELECTRICAL SYSTEM COMPONENT LOADS (SENECA V - Garmin G1000 NXi)

ITEM	# UNITS	AMP /UNIT	DURATION	TIME START	TAXI	T/O	CRUISE	LAND	REMARKS	AMPS * MINUTES START	TAXI	T/O	CRUISE	LAND
GIA 64 (INTEG AV)	2	0.8	CONT	0.5	15	15	60	10	includes superflag output draw from WOW relay	0.8	24	24	96	16
GIA 64 (COM)	2	0.3	CONT	0.5	15	15	60	10		0.3	9	9	36	6
GIA 64 (Transmitting)	1	4.3	INT	0	2	2	6	1	10% duty cycle transmit	0	8.6	8.6	25.8	4.3
GEA 71B	2	0.15	CONT	1	15	15	60	10		0.3	4.5	4.5	18	3
GEA 71B	2	0.37	INT	1	15	15	60	10	100% transducer load	0.74	11.1	11.1	44.4	7.4
GSA Servos (autopilot)	4	1.61	INT	0	0	1.5	6	1	Estimated 10% duty cycle	0	0	9.66	38.64	6.44
GWX 75/80	1	2.5	CONT	0	15	15	60	10		0	37.5	37.5	150	25
GTS 825 TAS	1	1.7	CONT	0	15	15	60	10		0	25.5	25.5	102	17
GTX 345R	1	0.65	CONT	1	15	15	60	10	@ maximum reply rate	0.65	9.75	9.75	39	6.5
WX-500 Stormscope	1	0.38	CONT	0	15	15	60	10	0.38 Maximum @ 28 volts	0	5.7	5.7	22.8	3.8
Becker ADF 3500 system	1	0.3	CONT	0	15	15	60	10		0	4.5	4.5	18	3
KN-63 DME	1	0.61	CONT	0	15	15	60	10		0	9.15	9.15	36.6	6.1
GDL 69eA XM Receiver	1	0.35	CONT	0	15	15	60	10		0	5.25	5.25	21	3.5
GSR 56 IRIDIUM	1	0.651	CONT	0	15	15	60	10		0	9.765	9.765	39.06	6.51
GSR 56 IRIDIUM	1	1.2	INT	0	15	5	20	0	worst case temp with heaters ON (additional amps)	0	18	6	24	0
CO Guardian Pulse Oximeter	1	0.3	CONT	0	15	15	60	10		0	4.5	4.5	18	3
USB Ports (cockpit)	2	0.65	INT	0	7.5	7.5	30	5	50% max load assumed	0	9.75	9.75	39	6.5
USB Ports (cabin)	2	0.85	INT	0	7.5	7.5	30	5	50% max load assumed	0	12.75	12.75	51	8.5
									TOTAL AMP * MINUTES	80.06	978.77	1292.72	5013.42	876.00
									UTILIZATION LOAD (AMPS)	80.06	65.25	86.18	83.56	87.60

CHAPTER

25

EQUIPMENT / FURNISHINGS

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CHAPTER 25

LIST OF EFFECTIVE PAGES

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	2	Sep 30/18		29	Sep 30/18
				30	Sep 30/18
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FLIGHT/PASSENGER COMPARTMENT

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

The Seneca IV has the capacity of seating up to six passengers. The Seneca V has the capacity to seat up to six passengers, or five passengers with the addition of an entertainment/executive console, in place of the right hand aft facing seat.

1. Seats

See “Figure 1” on page 25102.

A. Cockpit

The pilot’s and copilot’s seats are on rails and can be removed as follows. Reverse this procedure for installation.

(1) Lift up on handle just under the front of the seat and move the seat to the center of its travel.

(2) Remove the screws from the clips (fore and aft) on the rails and move the control column full forward.

(3) Pull the release and move the seat forward until the forward seat legs are aligned with the holes in the rails. Rotate the seat backward just until the legs clear the rails.

(4) With the front legs clear, move the seat aft until the rear legs can also be moved clear.

CAUTION: UPON INSTALLATION, WITH SEAT OCCUPIED AND AT LOWEST VERTICAL POSITION, CHECK SEAT FOR PROPER ENGAGEMENT OF PINS TO SEAT TRACK.

B. Cabin

(1) Forward facing passenger seats:

(a) Removal

On the floor attachment plates where the rear legs are retained, a spring pin retains the seat by keeping them forward under the lips of the retainers. Push down on the pin with a drift, or other suitable tool, and slide the seats back until the legs can clear their retaining plates.

(b) Installation

Align the four legs of the chair with the openings in their retainers, push down on the part of the chair where the legs are over the keeper pin, and then push the chair forward into the retaining plate.

(2) Aft facing passenger seats:

(a) Removal

1) Disconnect safety belt.

2) Remove the two mounting bolts from the seat’s front legs.

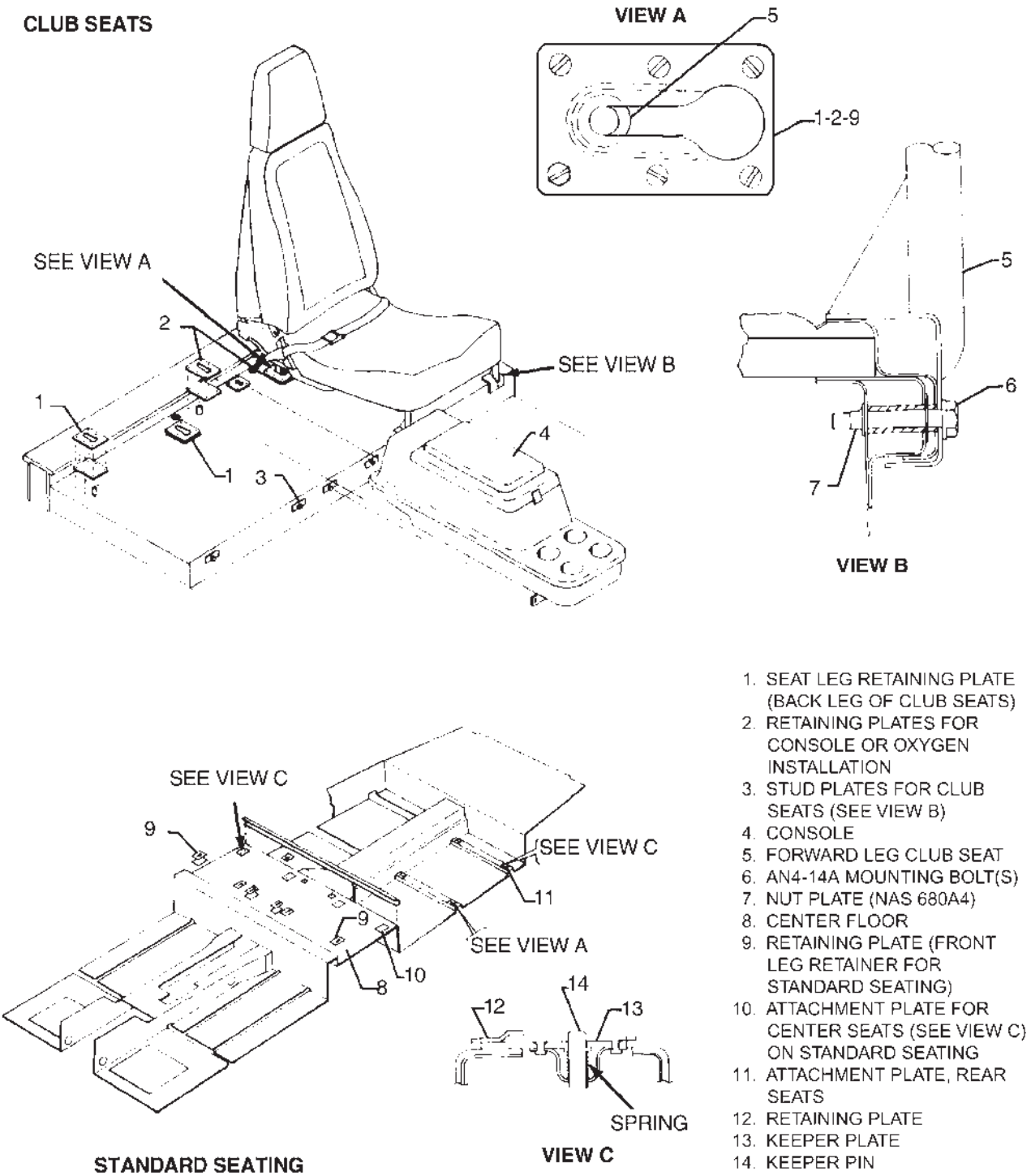
3) With the front legs of the seats disconnected, slide the seat aft until its back legs are clear of the retainers.

NOTE: Make sure the plates are clean of dirt that might prevent the leg foot from entering its retaining plate.

(b) Installation

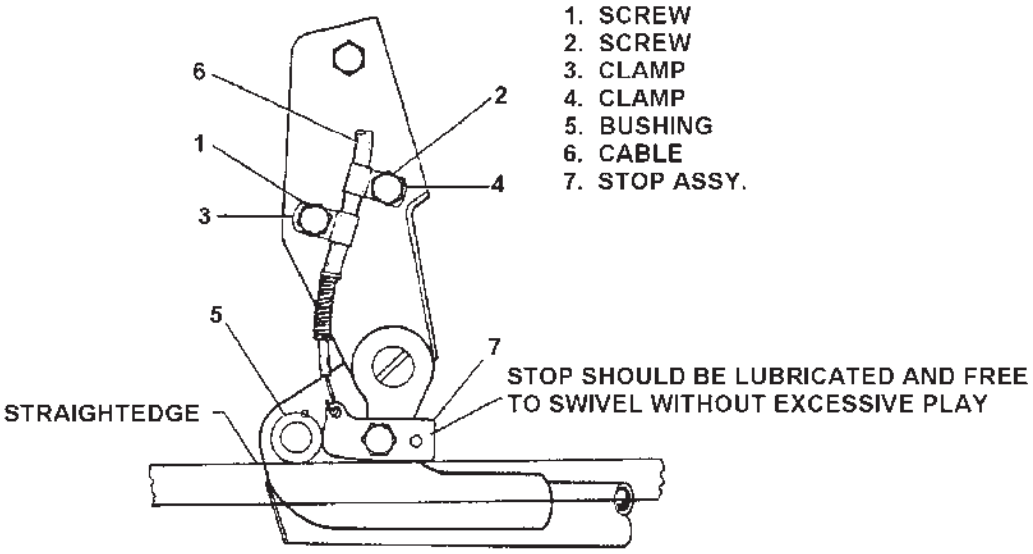
Aligning the seats’ legs in the retaining plates and install the mounting bolts.

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1. SEAT LEG RETAINING PLATE (BACK LEG OF CLUB SEATS)
2. RETAINING PLATES FOR CONSOLE OR OXYGEN INSTALLATION
3. STUD PLATES FOR CLUB SEATS (SEE VIEW B)
4. CONSOLE
5. FORWARD LEG CLUB SEAT
6. AN4-14A MOUNTING BOLT(S)
7. NUT PLATE (NAS 680A4)
8. CENTER FLOOR
9. RETAINING PLATE (FRONT LEG RETAINER FOR STANDARD SEATING)
10. ATTACHMENT PLATE FOR CENTER SEATS (SEE VIEW C) ON STANDARD SEATING
11. ATTACHMENT PLATE, REAR SEATS
12. RETAINING PLATE
13. KEEPER PLATE
14. KEEPER PIN

Seats and Center Console Installation
Figure 1



Seat Back Lock
Figure 2

C. Rigging Instructions - Seat Back Lock and Release

Refer to “Figure 2”.

(1) Loosen screws and ascertain that clamps are in a relaxed condition. (Push-pull cable is able to move within the clamps.)

(2) Place a straightedge along the lower surface of bushing of the seat back release.

(3) Adjust the push-pull cable by raising or lowering it until the lower surface of the stop assembly is parallel to the straightedge.

(4) Secure the push-pull cable in this position by tightening screws on clamps. The stop should be lubricated and free to swivel without excessive play.

(5) Push on seat back with stop assembly in an engaged position to check engagement. Rotate the seat back release handle and check for disengagement of seat back.

D. Lumbar Support

See “Figure 3” on page 25105.

(1) Description

In the optional interior, the pilot and co-pilot seats incorporate a Lumbar support feature. The installation consists of an inflatable bladder attached to the seat back filler and an inflation bulb located under and on the inboard side of each pilot and co-pilot seat.

(2) Removal

(a) Remove seat from airplane.

(b) Loosen velcro securing seat back filler cover.

(c) Remove only enough of seat back filer cover to expose lumbar bladder.

NOTE: Inflation tube may be removed before or after bladder is removed from seat back filler. Tube is not glued to nipple attachment; it can be removed by carefully pulling on tube.

(d) Remove inflation tube from bladder.

CAUTION: DO NOT USE A CHEMICAL SOLVENT TO REMOVE BLADDER. SOLVENT MAY DAMAGE SEAT BACK FILLER

CAUTION: TO AVOID OR MINIMIZE DAMAGE TO SEAT BACK FILLER DURING REMOVAL, USE ONE HAND TO RETAIN SEAT BACK FILLER IN PLACE, WHILE GENTLY REMOVING BLADDER WITH OTHER HAND.

(e) Starting at either right or left edge of bladder, carefully and slowly pull bladder and pad assembly from seat back filler.

(3) Assembly

(a) Apply a layer of 3M 847 cement to smooth side of bladder pad.

(b) Apply a layer of 3M 847 cement to back side of bladder (side away from inflation tube nipple).

(c) Attach bladder pad to bladder.

NOTE: While cement does not set immediately, there is no need to wait before attaching bladder and pad to seat back filler.

(4) Installation

(a) Apply a layer of 3M 847 cement to rough side of bladder pad.

(b) Apply a layer of 3M 847 cement to seat back filler where bladder is to be located.

(c) Attach bladder and pad assembly to seat back filler. Depending on temperature and humidity, allow 0:30 minutes to 1:00 hour for cement to set.

(d) Install seat back filler cover and secure velcro fastenings.

(e) Install seat in airplane.

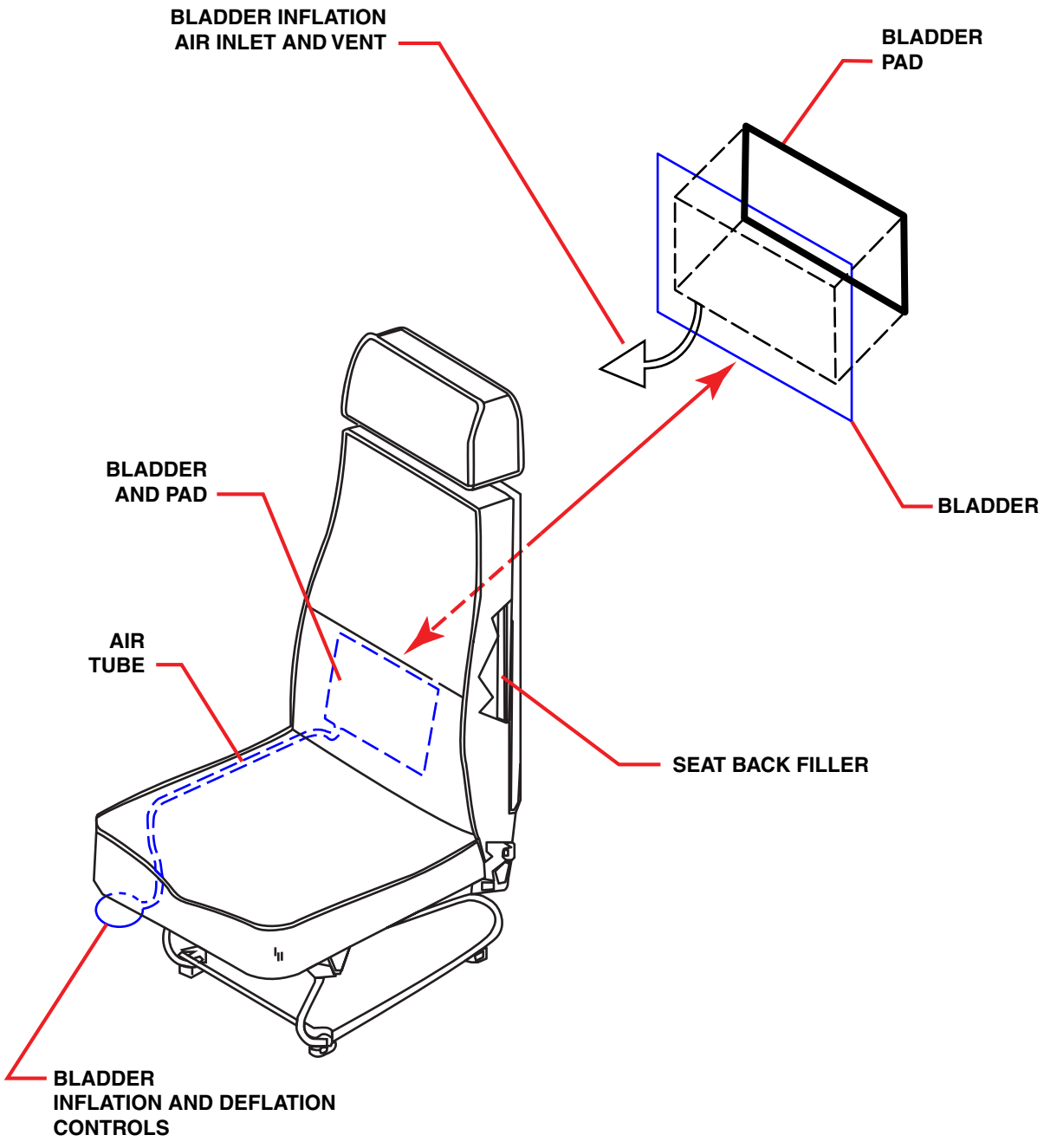
2. Center Console Removal and Installation

See “Figure 1” on page 25102.

The center console, if installed, is mounted to the center floor in the same manner as the club seats. Remove and install as follows:

A. Remove the bolts from the aft legs of the console that are held to the stud plates.

B. Slide the unit aft and lift up on the forward part of the console until the legs clear the retainers.



Lumbar Seat Bladder Installation
Figure 3

C. Install in the reverse order.

NOTE: If the console is not going to be installed any time soon, it is recommended that the retaining and stud plates be taped over to prevent dirt from fouling the assemblies.

3. Restraint System

An integrated shoulder harness / lap belt restraint system is installed in S/N's 3449165 and up. Earlier airplanes employ a traditional lap belt with a separate, detachable, shoulder harness.

A. Inspection

(1) In Seneca IV's and Seneca V S/N's 3449001 thru 3449164:

(a) Shoulder Harness

1) Inspect ends and attachment points for condition and security.

2) Inspect harness web material for condition and wear over its entire length. Particularly look for wear and fraying where harness web passes in and out of inertial reel. If excessively worn, replace.

3) Check inertia reel mechanism by pulling sharply on strap. Verify reel will lock in place under sudden stress.

(b) Lap Belt

1) Inspect ends and attachment points for condition and security.

2) Inspect harness web material for condition and wear over its entire length. Particularly look for wear and fraying where harness web passes in and out of adjustable buckle end. If excessively worn, replace.

3) Inspect shoulder harness keeper nylon bushing. If excessively worn or missing, replacement of that half of the lap belt is required.

(c) Additionally, in Seneca IV S/N's 3447001 thru 3447008 only.

4) Verify compliance with Piper Service Bulletin No. 990. If not yet complied with, then inspect seat belts at each position, as follows:

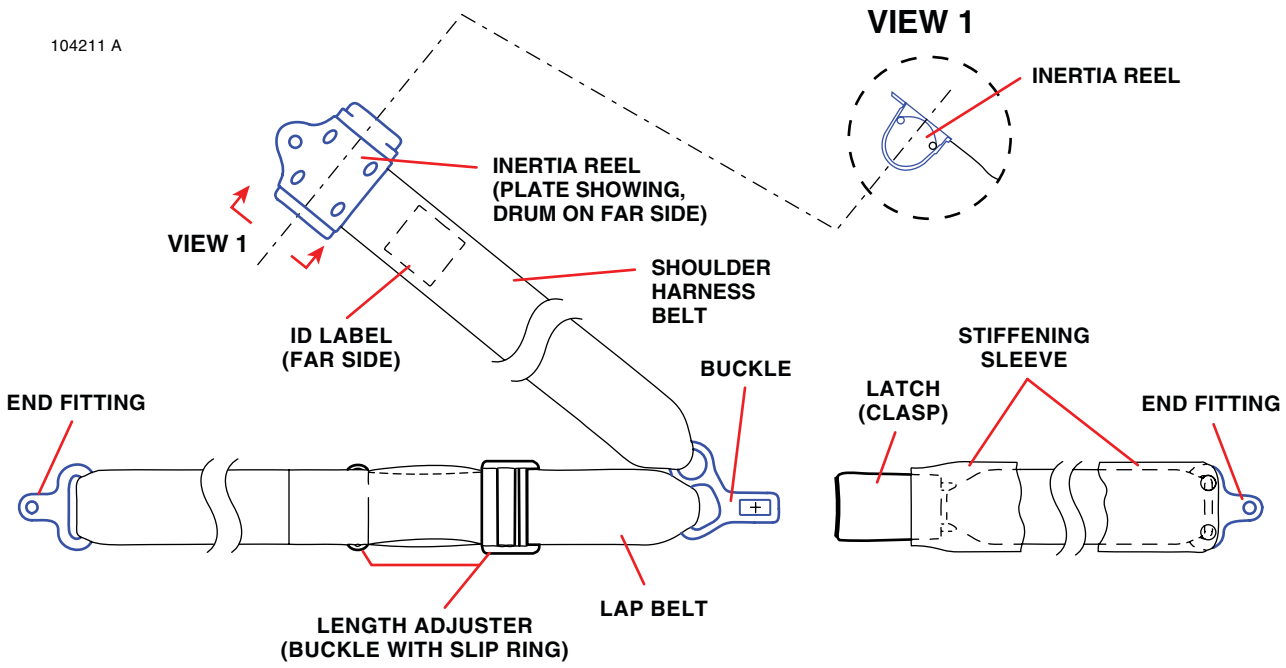
a) Replace all belts manufactured by Pacific Scientific and the seat belt has a date stamp of 6/94 thru 8/95, if the seat belt buckle end has an old Piper logo (large stylized "P") or no logo, and if the seat belt fitting end I.D. tag has a date stamp of 6/94 thru 8/95.

b) Replacement belts, fitting end and buckle end, should be ordered under the same part number(s) found on the suspect seat belts as no new part numbers were issued. Order the appropriate quantity of: Seat Belt, Fitting End - P/N 564- 867; Buckle End (short) - P/N 564-868; Buckle End (medium) - P/N 564-889; and/or, Buckle End (long) - P/N 564-862.

c) Check the new seat belts for operation and security, after installation.

d) Make an appropriate logbook entry of compliance with Service Bulletin No. 990.

5) Pending replacement of the suspect seat belts, the aircraft may be flown, provided the Pilot-In-Command inspects each seat belt being used for secure positive latching and that the seat belts are free of any anomalies prior to each flight and during each use.



Restraint System Belt Assembly (Typical)
Figure 4

(2) In Seneca V S/N's 3449165 and up:

Integrated Shoulder Harness / Lap Belt (Schroth)

(a) Inspect ends and attachment points for condition and security.

(b) Inspect harness web material for condition and wear over its entire length. Particularly look for wear and fraying where harness web passes in and out of inertial reel and in and out of the adjusting buckle. If excessively worn, replace.

(c) Check inertia reel mechanism by pulling sharply on strap. Verify reel will lock in place under sudden stress.

B. Removal

The crew restraint system upper attachments and inertia reels are located behind inertia reel cover. To service or remove the reels or upper attachments, you must remove the inertia reel covers.

Remove the lower end fittings first.

(1) Lower End Fitting Attachments

The lower end fittings of the crew restraint assemblies are attached to links or clevises, which are attached to the floor structure. Remove the lower end fittings from the link and clevis assemblies at either side of the seat as follows:

(a) Unlatch the seat belt buckle.

(b) Remove the bolt assemblies that attach the inside (aisle) end fittings that secure the belt to the link assemblies.

(c) Remove the bolt assemblies that attach the outboard side end fittings that secure the belt to the link assemblies.

(2) Upper Attachments and Inertia Reels
  (a) Remove the three screws and washers securing the reel cover.
  (b) Remove the bolts securing the inertia reel to the support channel.
  (c) Remove inertia reel.

C. Installation

(1) Upper Attachments and Inertia Reels
  (a) Align the inertia reel with the support channel and install mounting bolt.
  (b) Align reel cover with mounting holes and install washers and screws.
  (c) Check operation and security.
(2) Lower End Fitting Attachments
  (a) Align outboard side end fitting of the belt to the mount and install bolt to secure.
  (b) Align the inside (aisle) end fittings to the mount and install bolt to secure.
  (c) Check operation and security.

D. Inertial Reel Adjustment

The inertial reel locking feature prevents the shoulder strap from extending and holds occupant in place. For normal movement strap will extend and retract as required. If required, adjust inertial reel as follows:

(1) Allow harness to wind up on reel as much as possible.
(2) On end of reel, pry off plastic cover over spring. Make sure spring does not come out of plastic cover. Set aside plastic cover.
(3) Unwind the harness completely. Measure and mark the harness 24 inches from the reel center.
(4) Wind harness onto reel until the 24 inch mark is reached. Hold reel and place cap with spring over reel shaft end.
(5) Align slot in shaft with spring tang. Wind spring 6 1/2 turns and snap plastic cover into holes in reel end shaft.
(6) Release harness and allow harness to wind up. Extend harness several times to check reel for smooth operation.
(7) Hold inertia reel with reel completely wound and inertia mechanism end up. Pry off plastic cover over mechanism and set reel aside.
(8) Install nut in plastic cover so that stud in cover is flush with nut surface. Position cover over reel and snap cover into place. Extend harness several times to ensure reel operates smoothly.

4. Entertainment/Executive Console (Seneca V only)

See “Figure 5” on page 251010.

The Entertainment/Executive Console is available on Seneca V models in place of the right hand aft facing seat.
Some of the features of the console are: a horizontally sliding, pull out table, which when in the extended position, reveals a storage compartment; a cabinet designed to house a multimedia entertainment system, which is hidden away by a vertically sliding tambour door (controls for the entertainment system are located in this compartment); a fore and aft sliding beverage cooler drawer with a removable stainless steel container; a pilot’s reference material compartment; and a three-cup cup holder that hides away into the side of the console when not in use.

A. Removal

WARNING: REMOVE ALL ELECTRICAL POWER FROM THE AIRPLANE BEFORE REMOVING CONSOLE.

(1) Remove fire extinguisher from front of console (behind copilot seat).

(2) Disconnect electrical connections from console. (Electrical connections are standard connections, no special tools are necessary.)

(3) Remove four mounting bolts from aft side, bottom of console (the side facing the forward facing passenger seats). (Bolts may be hidden behind carpet.)

CAUTION: DO NOT LIFT CONSOLE BY PULLING UP ON TABLE. DAMAGE TO CONSOLE COULD RESULT.

CAUTION: WHEN MOVING CONSOLE, BE SURE TO SECURE ALL DOORS AND DRAWERS TO PREVENT THEM FROM SLAMMING OPEN AND CLOSED. DAMAGE TO CONSOLE COULD RESULT.

(4) Slide console aft until back legs are clear of retainers.

CAUTION: SET CONSOLE DOWN ON TO SURFACE GENTLY TO PREVENT BENDING ANGLE BRACKET.

(5) Carefully remove console from airplane.

NOTE: If the console is not going to be installed, it is recommended that the retaining and stud plates be taped over to prevent dirt from fouling the assemblies. Also enclose electrical connectors and secure harnesses.

B. Installation

CAUTION: DO NOT LIFT CONSOLE BY PULLING UP ON TABLE. DAMAGE TO CONSOLE COULD RESULT.

CAUTION: WHEN MOVING CONSOLE, BE SURE TO SECURE ALL DOORS AND DRAWERS TO PREVENT THEM FROM SLAMMING OPEN AND CLOSED. DAMAGE TO CONSOLE COULD RESULT.

(1) Align console into position, engaging back legs into retainers.

(2) Install four mounting bolts to aft side, bottom of console (the side facing the forward facing passenger seats).

WARNING: REMOVE ALL ELECTRICAL POWER FROM THE AIRPLANE BEFORE CONNECTING ELECTRICAL CONNECTIONS TO CONSOLE.

(3) Connect electrical connections to console. (Electrical connections are standard connections, no special tools are necessary.)

(4) Install fire extinguisher to front of console (behind copilot seat).





MOUNTING
BOLTS

5. Carpets

The carpets are individually fastened to the floor with Velcro fasteners and adhesive (Hysol EA9309 NA).

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EMERGENCY

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

This section contains information necessary to perform operational checks of the Emergency Locator Transmitter (ELT), with or without a pilot's remote switch. Included are the appropriate removal and installation instructions to facilitate battery replacement.

1. Narco 910 ELT

In Seneca IVs, the Narco 910 ELT or an "Artex ELT 110-4" was installed as standard equipment. See "Figure 1" on page 25603.

CAUTION: ALTHOUGH THE ELT IS INDEPENDENT OF THE AIRPLANE'S ELECTRICAL SYSTEM, REMOVE NEGATIVE CABLE FROM AIRPLANE'S BATTERY ANYTIME WORK IS TO BE DONE ON ANY ELECTRICAL OR ELECTRONIC EQUIPMENT.

CAUTION: INSPECT THE EXTERNAL WHIP ANTENNA FOR ANY DAMAGE. AVOID BENDING THE WHIP. ANY SHARPLY BENT OR KINKED WHIP SHOULD BE REPLACED. ANTENNA DAMAGE MAY CAUSE STRUCTURAL FAILURE OF WHIP IN FLIGHT.

A. Battery Removal and Installation

(1) Remove access panel on aft fuselage, right hand side.
(2) Set ON/OFF/ARM switch on transmitter to OFF.
(3) Disconnect antenna coaxial cable from ELT.
(4) Disconnect wiring harness connector from ELT.
(5) Remove ELT from its mounting tray.
(6) Remove 8 flat head screws from unit.
(7) Carefully separate unit into two sections.
(8) Unsnap battery connector (connector toward back end of circuit board).
(9) Carefully remove battery pack (contained in white foam jacket) from the ELT.
(10) Cut tape holding the two halves of foam together and remove old battery pack.
(11) Install new battery pack in foam jacket. Tape foam halves together with a good quality glass filament tape.
(12) Install battery pack assembly into ELT. Plug connector into circuit board.
(13) Slide the two unit section together. Ensure red gasket in header is sitting flat.
(14) Secure with 8 new screws provided with replacement battery. Ensure all 8 screws are snugged up.
(15) Install ELT into tray in airplane. Perform test of ELT.

B. Testing

CAUTION: ALL ELT “ON” TESTS SHOULD BE PERFORMED WITHIN THE FIRST FIVE (5) MINUTES AFTER THE HOUR UTC OR AS REQUIRED BY LOCAL OR NATIONAL AUTHORITIES. NOTIFY ANY NEARBY CONTROL TOWER OF YOUR INTENTIONS.

CAUTION: DO NOT ALLOW TEST DURATION TO EXCEED FIVE (5) SECONDS.

CAUTION: CONSULT FAA ADVISORY CIRCULAR AC 91-44A, LATEST REVISION, FOR DETAILED INFORMATION CONCERNING UNSHIELDED TESTING.

(1) Conduct test only during the first five minutes after any hour.

(2) If operational test must be made at any time other than the first five minutes after the hour, notify the nearest FAA traffic Control Tower or Flight Service Station prior to the test.

(3) Test should be no longer than three audio sweeps.

(4) If the antenna is removed, a dummy load should be substituted during the test.

(a) Remove access panel or cover to gain access to transmitter.

(b) Turn aircraft master switch ON. Turn the aircraft communications receiver ON and tune to 121.5 mhz.

(c) Turn receiver volume up until a slight background noise is heard. If equipped, automatic squelch must be overridden.

(d) If aircraft is not fitted with a communications receiver, request the nearest FAA facility to listen for E.L.T. signal.

(e) Set ON/ARM/OFF switch on the transmitter to the ON position for approximately 2 seconds. Return to OFF, then ARM position.

(f) Test transmission should be received by aircraft communications receiver and/or FAA facility. During cold weather, there may be a slight delay before transmission occurs.

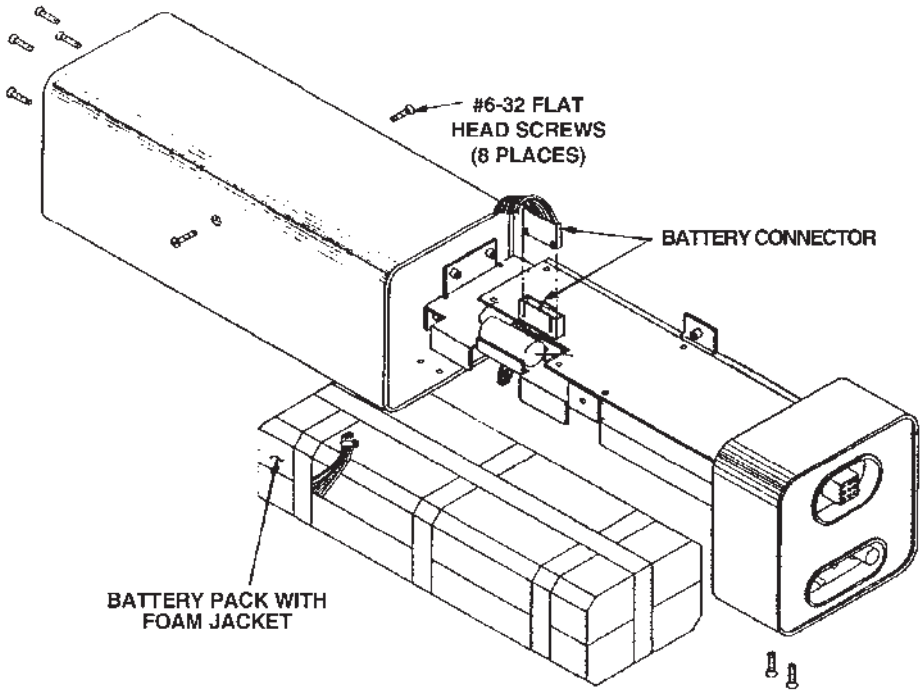
(g) A properly functioning transmitter emits a characteristic downward swept tone.

(h) When test is completed, ensure transmitter ON/ARM/OFF is in the ARM position. Whenever unit is checked by moving transmitter ON/ARM/OFF switch from ARM to ON position, it must first be moved to OFF position before resetting to ARM position.

(5) Install access panel on dorsal fin aft of fuselage station 259.30 and secure with the appropriate screws.

C. Electrical Schematic

See 91-25-60.



Narco 910 Battery Pack
Figure 1

2. Artex ELT 110-4

A. Description

This ELT was installed as standard equipment in Seneca V S/N's 3449001 through 3449358. See "Figure 2" on page 25605. In Seneca IVs either it or a "Narco 910 ELT" was installed as standard equipment.

The Artex ELT 110-4 transmits on 121.5 mHz and 243.0 mHz, and is designed to meet or exceed the requirements of TSO C91a and FAR Part 91. Electrical power for the ELT transmissions is totally supplied by its own self-contained battery. The battery must be replaced if the transmitter has been used in an emergency situation or if accumulated test time exceeds one hour, or no later than the replacement date marked on the transmitter label, whichever comes first.

The Artex 110 cannot be accidentally activated by dropping the unit, handling it roughly, or during shipping. However, when properly mounted, and locked into its mounting tray, the ELT will activate in a crash, regardless of the cockpit remote switch and ELT switch position. The normal position of the ELT switch is in the down or OFF position. The normal position of the remote cockpit switch is in down or ARM position

Whenever the ELT is activated, a red light located just above the remote cockpit switch will blink to alert the pilot or maintenance personnel. Should the ELT be activated accidentally, it must be reset. To reset:

(1) Position the remote cockpit switch to ON, then immediately reposition it to ARM, or;
(2) Position the switch on the ELT to ON, then immediately reposition it to OFF.

B. Inspections

(1) 100 Hour

Inspect per the 100 hour inspection (including the Antenna Test) under Artex ME406 ELT, above.

(2) Annual

Every twelve (12) months, inspect the ELT installation as follows:

NOTE: The following inspection satisfies the requirements of FAR 91.207.

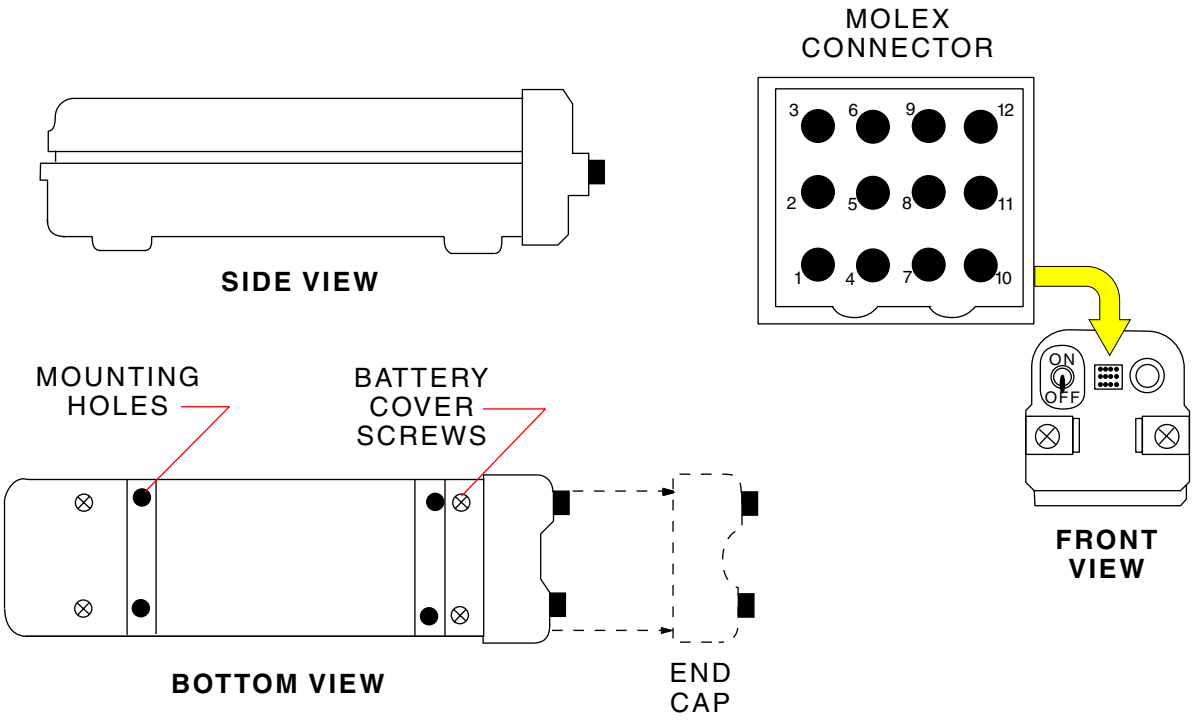
(a) Disconnect and remove ground (negative) battery cable from battery. Gain access as follows:
  1) Seneca IV, remove the access panel on the right side of the nose.
  2) Seneca V, remove the rear closeout panel in the cabin.
(b) At right side of empennage, remove four (4) screws and remove ELT access panel.
(c) Remove ELT Connections

CAUTION: DO NOT USE CONTACT CLEANER ON ELT COMPONENTS. SUCH CHEMICAL AGENTS CAN BE HIGHLY DESTRUCTIVE TO THE MOUNTING HARDWARE AND ELT HOUSING.

  1) Loosen the thumbscrews and remove the Molex and coax connectors.
  2) Visually inspect and confirm proper seating of all connector pins. Special attention should be given to coaxial center conductor pins which are prone to retracting into the connector housing.

(d) Remove ELT from its mounting tray per Removal, below.

Inspect the mounting hardware. Ensure the hardware is free of cracks or other obvious damage.



Artex ELT 110-4
Figure 2

(e) Battery Inspection

1) Remove the four (4) securing screws from the bottom of the ELT.
2) Position the ELT product label (arrow) side down, and carefully lift the battery pack away from the ELT and lay it alongside the ELT.
3) Carefully disconnect the harness from the connnector in the black plastic housing. Use a flat-bladed screwdriver to pry the connector out of its mating plug.
4) Inspect the battery pack and ELT chassis. The battery cells, components and connectors should be free of corrosion.
5) Inspect for broken wires, connections, or damage.
6) Ensure the battery housing is free of cracks or other visible damage.
7) Verify the battery expiration date. If the battery pack has not expired it may be reinstalled. The battery pack must be replaced with a new one:
   a) After use in an emergency;
   b) After an inadvertent activation of unknown duration;
   c) When the total of all known transmissions exceeds one (1) hour; or
   d) On or before the battery replacement (expiration) date.

(f) Perform G-switch Check under Testing, below.

(g) Reinstall ELT into airplane per Installation, below.

(h) Perform Antenna Test per the procedure under Artex ME406 ELT, above.

(i) Reset ELT by turning ELT switch to “ON” then to ‘OFF/ARM” position.

(j) Make an appropriate logbook entry documenting completion of this inspection and whether or not the ELT passed or failed.

C. Removal

(1) In cabin, remove rear closeout panel.
(2) Disconnect and remove negative cable from battery.
(3) At right side of empennage, remove four (4) screws and remove ELT access panel.
(4) Remove ELT from the airplane by:
   (a) Loosening the two screws on the front of the mounting tray and pull mounting tray cap off.
   (b) Disconnecting coax (antenna) cable.
   (c) Disconnecting the Molex cable from the ELT unit.
   (d) Remove unit from airplane. Lift unit from the connector end. Carefull use of a flat-bladed screwdriver as a lever makes this step easier.

D. Installation

(1) Install unit into mounting tray:
   (a) Connect molex and coax cables to ELT unit.
   (b) Install mounting tray cap and secure to front of mounting tray with the two screws.
(2) Replace ELT access panel and secure with four (4) screws.
(3) In cabin, connect negative cable to battery.
(4) Replace cabin rear closeout panel.
(5) Test transmitter.

E. Battery Replacement

See “Figure 2” on page 25605.

(1) Removal

(a) If required, remove the ELT from the airplane per Removal, above.

(b) Remove the four screws on the bottom of the ELT securing the battery pack.

(c) Disconnect battery pack connector from main unit. Use a flat-bladed screwdriver to pry the connector out of its mating plug.

(d) Remove battery pack from unit.

(2) Installation

(a) Securely plug in new battery pack connector to main unit.

(b) Immediately reset unit by positioning unit switch to ON, then to OFF.

(c) Fit new battery pack into place. Ensure all gaskets are properly aligned.

(d) Replace the four screws. Dress wires away from standoffs to avoid pinching wires between standoffs and the battery pack.

(e) Install ELT into airplane per Installation, above.

F. Testing

CAUTION: ALL ELT “ON” TESTS SHOULD BE PERFORMED WITHIN THE FIRST FIVE (5) MINUTES AFTER THE HOUR UTC OR AS REQUIRED BY LOCAL OR NATIONAL AUTHORITIES. NOTIFY ANY NEARBY CONTROL TOWER OF YOUR INTENTIONS.

CAUTION: DO NOT ALLOW TEST DURATION TO EXCEED FIVE (5) SECONDS.

CAUTION: CONSULT FAA ADVISORY CIRCULAR AC 91-44A, LATEST REVISION, FOR DETAILED INFORMATION CONCERNING UNSHIELDED TESTING.

Always perform the tests within the first five (5) minutes of the hour. Notify any nearby control tower of your intentions. If outside of the US, always follow all local or national regulations for testing ELTs. Do not allow test duration to exceed five (5) seconds. Any time the ELT is activated it is transmitting a 121.5 MHz distress signal.

The transmitter operates on the emergency frequencies of 121.5 and 243.0 MHz; both of these frequencies are monitored by the various FAA installations. Before performing any operational test of the ELT, the following precautions should be observed:

- Test should be no longer than three audio sweeps.
- Test should be conducted only within the time period made up of the first five minutes after any hour.
- If the operational tests must be made at a time not included within the first five minutes after the hour, the test should be coordinated with the closest FAA Tower or Flight Service Station.

(1) Installed Transmitter Test (Self-Test) (Every 90 Days)

(a) Turn both the airplane master switch and the radio master switch ON.

(b) Tune airplane communications receiver to 121.5 mHz and select SPKR on the audio panel.

(c) Turn the ELT aircraft panel switch to “ON”, wait for three (3) sweeps on the receiver, which takes about one (1) second, and then turn the switch back to the “ARM” position.

(d) To pass the test, you must hear the three (3) sweeps and see the front panel light immediately begin to flash continuously until the unit is switched to “ARM”.

1) During the “ON” to “ARM” transition, the microprocessor in the ELT checks the “G-Switch” (automatic activation switch) latching circuit, pins 5 and 8 on the tray connector.

2) If there is a problem, the processor will not immediately turn on the cockpit light during those first few sweeps. For example, if the jumper between pins 5 and 8 was not installed or was open, the cockpit light would flash momentarily upon ELT activation and the stay off for approximately three (3) seconds before beginning to flash continuously.

3) Repairs should be done only by a licensed aviation radio repair shop.

(2) Antenna Test (Each 100 Hours)

Use the procedure under “Artex ME406 ELT” above.

(3) G-switch Check (Annually)

CAUTION: AS WITH ALL BEACON-TESTING, THIS TEST SHOULD BE PERFORMED WITHIN THE FIRST FIVE (5) MINUTES OF THE HOUR, AND ANY LOCAL CONTROL TOWER SHOULD BE ADVISED OF THE TEST.

A basic test of G-switch operation can be performed as follows:

(a) Setup

1) Remove the ELT from the airplane per Removal, above.

2) Fabricate a “shorting plug” by obtaining an appropriate connector and installing a jumper (i.e. - short) between Pins 5 and 8. (Artex sells a test plug for this purpose under part number 151-2012.)

NOTE: Be sure the correct pins are shorted. Some connections will force the ELT to activate. Others will keep the unit from activating in any circumstance. No combination of shorts will cause permanent damage to the ELT, however all wrong pin combinations erroneously indicate a faulty ELT.

If the ELT activates without any pins shorted it is defective and should be returned for repair.

3) Install the “shorting plug” on the ELT.

4) For a more thorough test, monitor the transmission with an AM receiver tuned to 121.5 MHz, as described in Transmitter Test, below.

(b) Procedure

The ELT should remain OFF until an acceleration of three (3) Gs or more is applied axially, in the rearward direction. This action should activate the unit, transmitting immediately on 121.5 MHz.

CAUTION: MAINTAIN A FIRM GRIP ON THE ELT.

1) Apply acceleration greater than three (3) Gs to the ELT, in the rearward direction. This can be achieved by using a rapid forward (throwing) motion in the direction of the label arrow, then rapidly reversing the direction.

2) Monitor ELT activation by observing the ELT LED (and AM receiver, if desired).

3) Allow ELT to transmit only long enough to verify operation.

4) Reset ELT by turning ELT switch to “ON” then to ‘OFF” position.

G. Electrical Schematic

See 91-25-60.

3. Artex ME406 ELT

See “Figure 3” on page 256012.

A. Description and Operation

The Artex ME406 ELT is installed as standard equipment in Seneca V’s S/N’s 3449359 and up. It’s a fully TSO-C126 / ETSO-2C126 certified, type AF (automatic fixed) beacon 406 MHz ELT. All functions of the ME406 are under microprocessor control. A self-test routine checks ELT operation and installation, then presents the results as visual and auditory “error codes” to aid in troubleshooting and to indicate status. The ME406 is pre-programmed at the factory using a short message format. The following User Protocols are supported: Serial Number; Tail Number; 24-Bit Aircraft Address; and Aircraft Operator Designator/Serial Number.

The battery pack consists of two D-size lithium cells mounted in a cover assembly, and is field replaceable.Rated life is one hour of use or the expiration date on the battery pack, whichever comes first, as specified by FAR 91.207(c).

In the event of a crash, the ME406 activates automatically, and transmits the standard swept tone on 121.5 MHz lasting until battery power is gone. In addition, for the first 24 hours of operation, a 406 MHz signal is transmitting at 50 second intervals. This transmission lasts 440 ms and contains identification data programmed into the beacon and is received by Cospas-Sarsat satellites. Position accuracy of the 121.5 MHz signal is approximately 15-20 km radius about the transmitter. Accuracy of the 406 MHz signal is within about a 3 km radius.

In a crash, an acceleration activated crash sensor (G-switch) turns the ELT “ON” automatically when the ELT experiences a change in velocity (or deceleration) of 4.5 fps ± 0.5 fps. Activation is also accomplished by means of the cockpit mounted remote switch or the local switch on the ELT. To deactivate the ELT, set either switch to the “ON” position, then back to ‘ARM’.

The ELT does not have an “OFF” position. Instead, a jumper between two pins on the front D-sub connector must be in place for the G-switch to activate the unit. The jumper is installed on the mating half of the connector so that when the connector is installed, the beacon is armed. This allows the beacon to be handled or shipped without “nuisance” activation (i.e. - front connector removed).

NOTE: The ELT can still be manually activated using the local switch on the front of the ELT. Care should be taken when transporting or shipping the ELT not to move the switch or allow packing material to become lodged such as to toggle the switch.

If the ELT is inadvertently activated, deactivate it by setting either switch to the “ON” position and then back to “ARM”.

B. Troubleshooting

See “Chart 1” on page 256010.

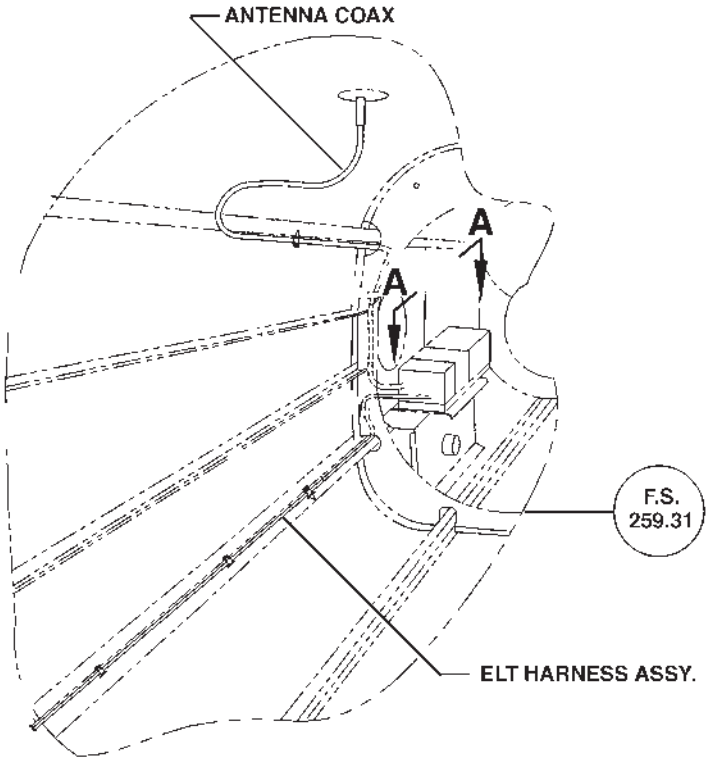
CHART 1 (Sheet 1 of 2)
TROUBLESHOOTING ME406 ELT

Trouble	Cause	Remedy
3 Flash Error after performing Self Test.	Bad load detect. Detects open or short condition on the antenna output or cable.	Check that the RF cable is connected and in good condition.
		Perform continuity check of center conductor and shield. Check for a shorted cable.
		Check for intermittent connection in the RF cable.
		If this error code persists there may be a problem with the antenna installation. Check this with a VSWR meter. Check the antenna for opens, shorts, resistive ground plane connection.
4 Flash Error after performing Self Test.	Low power detected. Occurs if output power is below about 33 dBm (2 watts) for the 406 signal or 17 dBm (50 mW) for the 121.5 MHz output.	Verify battery voltage. Replace battery if low voltage (~5.6 VDC) or if 7 Flash error is also present.
	Also may indicate that 406 signal is off frequency.	Verify 406 MHz frequency. If bad, return for repair / replacement.
5 Flash Error after performing Self Test.	Indicates that the ELT has not been programmed.	Read ELT 406 MHz signal to verify programming.
6 Flash Error after performing Self Test.	Indicates that G-switch loop between pins 5 and 12 at the D- sub connector is not installed. ELT will not activate during a crash.	Check that the harness D-sub jumper is installed iby verifying less thant one (1) ohm of resistance between pins 5 and 12. If missing, install jumper wire.
7 Flash Error after performing Self Test.	Indicates that the ELT battery has too much accumulated operation time (more than one hour) per regulation.	Replace battery.
	May also indicate damage to the battery circuit.	If error does not clear after battery replacement, check continuity of battery circuit and correct function of circuit components.

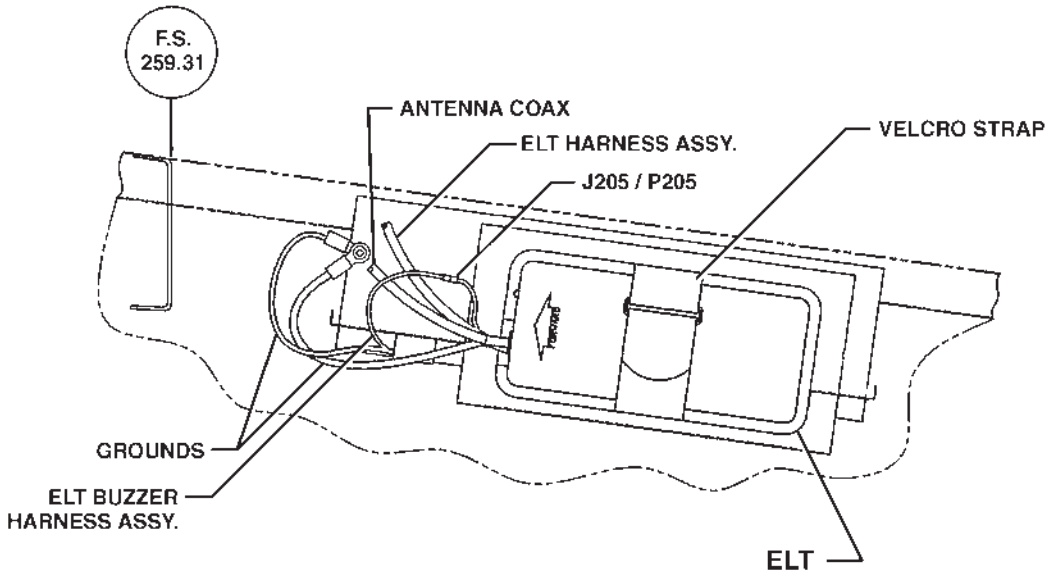
CHART 1 (Sheet 2 of 2)
TROUBLESHOOTING ME406 ELT

Trouble	Cause	Remedy
Remote Switch LED always ON (steady).	Wiring error or frayed wires shorting out pins on back of Remote Switch.	Verify wiring. Verify integrity of all crimp or solder connections on harness.

106648 NEW



LOOKING AFT, RIGHT SIDE



VIEW A - A

Effectivity
3449359 and up

Artex ME406 ELT Installation
Figure 3

Effectivity
Seneca V
with Artex ME406 ELT

C. Inspections

(1) 100 Hour

Each 100 hours time-in-service, inspect the ELT installation as follows:

(a) Inspect the ELT unit and mount for proper installation and secure mounting.

(b) Inspect wiring and conduits for proper routing, secure mounting, and obvious defects.

(c) Inspect bonding and shielding for proper installation and condition.

(d) Inspect antenna for condition, secure mounting, and proper operation (per Antenna Test under Testing, below).

(2) Annual

Every twelve (12) months, inspect the ELT installation as follows:

NOTE: The following inspection satisfies the requirements of FAR 91.207.

(a) In the cabin, remove the rear closeout panel.

(b) Disconnect and remove negative cable from battery.

(c) At right side of empennage, remove four (4) screws and remove ELT access panel.

(d) Remove ELT Connections

1) Loosen the thumbscrews and remove the D-sub and RF connectors.

2) Visually inspect and confirm proper seating of all connector pins. Special attention should be given to coaxial center conductor pins which are prone to retracting into the connector housing.

(e) Remove ELT from its mounting tray per Removal, below.

Inspect the mounting hardware. Ensure the hardware is free of cracks or other obvious damage.

(f) Battery Inspection

NOTE: The battery pack contains static sensitive parts, take Electrostatic Discharge (ESD) precautions before handling.

1) Remove the eight (8) securing screws from the battery-side cover.

NOTE: Battery pack is identified by the embossed text: “BATTERY ACCESS ON THIS SIDE”.

CAUTION: DO NOT PULL ON THE FLEXIBLE PORTION OF THE CABLE - USE THE RIGID SECTION OF THE FLEX CIRCUIT AT THE CONNECTOR AS A HANDLE.

2) Carefully lift the battery cover (battery pack) away from the ELT and unplug the flex-cable connected to the pack.

3) Inspect the battery pack and ELT chassis. The battery cells, components and connectors should be free of corrosion.

4) Inspect flex-circuit for broken connections or damage.

5) Ensure the battery housing is free of cracks or other visible damage.

6) Verify the battery expiration date. If the battery pack has not expired it may be reinstalled. The battery pack must be replaced with a new one:

a) After use in an emergency;

b) When the transmitter has been in use for more than one (1) cumulative hour (i.e. - 7 flash error);

c) After an inadvertent activation of unknown duration; or

d) On or before the battery replacement (expiration) date.

(g) Perform G-switch Check under Testing, below.

(h) Reinstall ELT into airplane per Installation, below.

(i) Perform Antenna Test under Testing, below.

(j) Perform Digital Message Verification under Testing, below.

(k) Reset ELT by turning ELT switch to “ON” then to ‘ARM” position.

(l) Check ELT for signs of registration. In the US, NOAA supplies a beacon registration label that is applied to the ELT when it is registered.

(m) Make an appropriate logbook entry documenting completion of this inspection and whether or not the ELT passed or failed.

D. Removal

(1) In the cabin, remove the rear closeout panel.

(2) Disconnect and remove negative cable from battery.

(3) At right side of empennage, remove four (4) screws and remove ELT access panel.

(4) Loosen the thumbscrews and remove the D-sub and RF connectors.

(5) Loosen the velcro strap and remove the ELT from its mounting tray.

E. Installation

(1) Insert the ELT into the mounting tray at an angle so that the locking ears at the end fit into the mounting tray locking slots.

(2) Fasten the Velcro strap around the ELT so that it is firmly held in place.

(3) Insert the D-sub and RF connectors ensuring that they are seated properly.

(4) Tighten the thumbscrews.

(5) At right side of empennage, place ELT access panel in position and secure with four (4) screws.

(6) In the rear cabin, connect negative cable to battery.

(7) Reinstall the rear closeout panel.

F. Battery Replacement

NOTE: When replacing the battery pack, a replacement kit containing the battery pack, replacement gasket, hardware and labels is available.

(1) Remove ELT per Removal, above.

(2) Remove the battery pack as follows:

NOTE: The battery pack contains static sensitive parts, take Electrostatic Discharge (ESD) precautions before handling.

(a) Remove the eight (8) securing screws from the battery-side cover.

NOTE: Battery pack is identified by the embossed text: “BATTERY ACCESS ON THIS SIDE”.

CAUTION: DO NOT PULL ON THE FLEXIBLE PORTION OF THE CABLE - USE THE RIGID SECTION OF THE FLEX CIRCUIT AT THE CONNECTOR AS A HANDLE.

(b) Carefully lift the battery cover (battery pack) away from the ELT and unplug the flex-cable connected to the pack.

(c) Discard/recycle the old battery pack.

(3) Lay the new battery pack on the work surface with the batteries facing up.

(4) Install a replacement seal in the slot along the perimeter of the housing.

(5) Leaving the battery as it is, position the ELT over the battery pack with one hand and plug the flex-cable connector into the battery assembly using the other. The cable should not be twisted and the connector should 'click' into place.

NOTE: The battery connector is keyed to prevent incorrect installation.

(6) Mate the ELT to the battery, making sure that the seal is positioned correctly during the process.

(7) Replace the eight (8) securing screws and torque to 10 - 12 inch-lbs.

(8) Install ELT per Installation, above.

(9) Perform Installed Transmitter Test (Self Test) per Testing, below.

(10) Enter pertinent battery replacement information in the aircraft log book and fill out any other documentation required by local authority.

G. Testing

CAUTION: ALL ELT "ON" TESTS SHOULD BE PERFORMED WITHIN THE FIRST FIVE (5) MINUTES AFTER THE HOUR UTC OR AS REQUIRED BY LOCAL OR NATIONAL AUTHORITIES. NOTIFY ANY NEARBY CONTROL TOWER OF YOUR INTENTIONS.

CAUTION: DO NOT ALLOW TEST DURATION TO EXCEED FIVE (5) SECONDS.

CAUTION: CONSULT FAA ADVISORY CIRCULAR AC 91-44A, LATEST REVISION, FOR DETAILED INFORMATION CONCERNING UNSHIELDED TESTING.

Always perform the tests within the first five (5) minutes of the hour. Notify any nearby control tower of your intentions. If outside of the US, always follow all local or national regulations for testing ELTs. Do not allow test duration to exceed five (5) seconds. Any time the ELT is activated it is transmitting a 121.5 MHz distress signal. If the unit operates for approximately 50 seconds, a 406 MHz distress signal is transmitted and is considered valid by the satellite system.

(1) Installed Transmitter Test (Self Test) (Every 90 Days)

(a) Turn both the airplane master switch and the radio master switch ON.

(b) Tune a receiver (usually the aircraft radio) to 121.5 MHz.

(c) Turn the ELT aircraft panel switch to "ON", wait for three (3) sweeps on the receiver, which takes about one (1) second, and then turn the switch back to the "ARM" position while paying special attention of the LED activity upon entering the 'ARM" condition.

(d) To pass the test, you must hear the three (3) sweeps and see the front panel light immediately begin to flash continuously until the unit is switched to "ARM".

1) During the "ON" to "ARM" transition, the microprocessor in the ELT checks the "G-Switch" (automatic activation switch) latching circuit, pins 5 & 12 on the D-sub connector at the ELT; the 406 MHz transmitter for proper RF output and a battery check.

2) If the ELT is working properly, the sequence following entry to the "ARM" condition will result in the panel LED staying illuminated for approximately one (1) second (one pulse), and then extinguishing. The buzzer should also sound once.

(e) If the panel LED and buzzer present more than one (1) pulse when switched to "ARM", determine the problem from the list in Chart 1, above.

(2) Antenna Test (Each 100 Hours)

Use a low quality AM broadcast receiver to determine if energy is being transmitted from the antenna.

(a) Hold the antenna of the radio (tuning dial on any setting) about six (6) inches from the activated ELT antenna. The ELT aural tone should be heard on the AM broadcast receiver.

(b) This is not a measured check, but it does provide confidence that the antenna is radiating sufficient power to aid search and rescue.

NOTE: Use of the aircraft's VHF receiver, tuned to 121.5 MHz, is not recommended. This receiver is more sensitive and could pick up a weak signal even if the radiating ELT antenna is disconnected. While it will confirm the ELT is active, it does not check the integrity of the ELT system

(3) G-switch Check (Annually)

A basic test of G-switch operation can be performed as follows:

CAUTION: EVEN WITHOUT AN ANTENNA CONNECTED, THE SIGNAL CAN BE RECEIVED BY A SATELLITE. AS WITH ALL BEACON-TESTING, THIS TEST SHOULD BE PERFORMED WITHIN THE FIRST FIVE (5) MINUTES OF THE HOUR, AND ANY LOCAL CONTROL TOWER SHOULD BE ADVISED OF THE TEST.

(a) Setup

1) The ELT must be removed from the airplane. See Removal, above.

2) Fabricate a "shorting plug" by obtaining a standard 15-pin D-sub connector and installing a jumper (i.e. - short) between Pins 5 and 12. (Artex sells a test plug for this purpose under part number 150-1130.)

NOTE: Be sure the correct pins are shorted. Some connections will force the ELT to activate. Others will keep the unit from activating in any circumstance. No combination of shorts will cause permanent damage to the ELT, however all wrong pin combinations erroneously indicate a faulty ELT.

If the ELT activates without any pins shorted it is defective and should be returned for repair.

3) Install the "shorting plug" on the ELT.

4) For a more thorough test, monitor the transmission with an AM receiver tuned to 121.5 MHz, as described in Transmitter Test, above.

(b) Procedure

The ELT should remain OFF until an acceleration of three (3) Gs or more is applied axially, in the rearward direction. This action should activate the unit, transmitting immediately on 121.5 MHz.

CAUTION: MAINTAIN A FIRM GRIP ON THE ELT.

1) Apply acceleration greater than three (3) Gs to the ELT, in the rearward direction. This can be achieved by using a rapid forward (throwing) motion in the direction of the label arrow, then rapidly reversing the direction.

2) Monitor ELT activation by observing the ELT LED (and AM receiver, if desired).

3) Allow ELT to transmit only long enough to verify operation.

4) Reset ELT by turning ELT switch to "ON" then to 'ARM" position.

(4) Digital Message Verification (Annually)

The ARTEX 406 MHz ELT transmits a 406 MHz message upon reset, which is encoded such that it will be ignored by the SAR satellite system. The 15-digit number contained in this transmission is used to register the ELT with the appropriate 406 MHz ELT registration authority. In the US, the National Oceanic and Atmospheric Administration (NOAA) maintains the database of registered ELT's. The information in this database provides the Search and Rescue system with aircraft identification data in the event an actual distress signal is transmitted.

Verify the 406 MHz digital message as follows:

(a) Required Equipment

A test set capable of receiving and decoding the message - i.e. - Artex ELT Test Set P/N 453-1000, or equivilent.

(b) Procedure

1) At right side of empennage, remove four (4) screws and remove ELT access panel.
2) Disconnect the antenna coax cable at the ELT, connect test set or terminate as applicable.
3) Perform all necessary steps to prepare Test Set to receive 406 MHz signal including (but not limited to) turning on power, activating program or any other steps required for the particular Test Set being used.
4) Perform the Installed Transmitter Test (Self Test) as described above.
5) Watch the screen on the Test Set to ensure that a message has been received. Repeat "Self Test" if necessary.
6) View message, ensure that all applicable information is correct (country code, aircraft ID, etc.).
7) The 15 digit ID hex ID (for example "ADC6492640D3411F1') should match what is shown on the ELT product label. This is the 15 digit hex ID (Unique Identification Number or 'UIN") that is used to register the ELT.
8) At right side of empennage, place ELT access panel in position and secure with four (4) screws.

H. Electrical Schematic

See 91-25-60.

4. Artex ELT 1000

Installed as standard equipment in S/N's 3449509 and up.

A. Description and Operation

The Artex ELT 1000 is a TSO-C126 / ETSO-2C126 certified and COSPAS-SARSAT TAC 251 approved, type AF (Automatic Fixed) beacon, which transmits on 121.5 and 406 MHz. When the 406 MHz signal is detected by the Cospas–Sarsat satellite LEO (Low Earth Orbit system), a position is calculated and the 121.5 MHz signal is used to home in on the crash site. The ELT 1000 is pre-programmed at the factory to use a transmitted short message format. The information contained in the message includes: (a) beacon serial number, (b) an aircraft identification or registration number, (c) a country of registration and country code and (d) position coordinates, if beacon is programmed to receive position data from the aircraft navigation system. ELT operation checks are performed by a series of self-tests with results as visual and auditory "error codes" to aid in troubleshooting and to indicate status.

The battery pack consists of two D-size lithium cells mounted in a covered assembly, and is field replaceable. Rated life is six (6) years unused or with total of transmissions exceeding one hour of use, whichever comes first, as specified by FAR 91.207(c). The lithium battery packs used on the Artex ELT 1000 are certified under TSO C142a.

In the event of a crash, the ELT 1000 activates automatically, and transmits the standard swept tone (homing signal) on 121.5 MHz lasting until battery power is gone, which with lithium cells is at least 50 hours. In addition, for the first 24 hours of operation, a 406 MHz signal is transmitting at 50 second intervals. This transmission lasts 440 ms and contains identification data programmed into the beacon and is received by Cospas-Sarsat satellites. Position accuracy of the 121.5 MHz signal is approximately 15–20 km radius about the transmitter. Accuracy of the 406 MHz signal is within about a 3 km radius. If position information is extracted from the aircraft navigation system, the accuracy improves to approximately 100 meters.



Artex ELT 1000 Emergency Locator Transmitter
Figure 4

In a crash, an acceleration activated crash sensor (G-switch) turns the ELT “ON” automatically when the ELT experiences a change in velocity (or deceleration) of 4.5 fps ± 0.5 fps. Activation is also accomplished by means of the cockpit mounted remote switch or the local switch on the ELT. To deactivate the ELT, set either switch to the “ON” position, then back to “ARM”.

The ELT does not have an “OFF” position. Instead, a jumper between two pins on the front D-sub connector must be in place for the G-switch to activate the unit. The jumper is installed on the mating half of the connector so that when the connector is installed, the beacon is armed. When this connector is removed the ELT can not be activated. This allows the beacon to be handled or shipped without “nuisance” activation (i.e., front jumper connector removed).

NOTE: The ELT can still be manually activated using the local switch on the front of the ELT. Care should be taken when transporting or shipping the ELT not to move the switch or allow packing material to become lodged such as to toggle the switch.

If the ELT is inadvertently activated, deactivate it by setting either switch to the “ON” position and then back to “ARM”. If the switch is already in the “ON” position, move it to the “ARM” position.

The ELT is located in the tailcone and accessed through a large panel on the right lower side. The ELT antenna is to the right of the tail at R.B.L. 4.00, F.S. 248.10. (See 23-00-00 Figure 1.)

B. Troubleshooting

See “Chart 2”.

C. Inspections

(1) 100 Hour

Each 100 hours time-in-service, inspect the ELT installation as follows:

(a) Inspect the ELT unit and mount for proper installation and secure mounting.

(b) Inspect wiring and conduits for proper routing, secure mounting, and obvious defects.

(c) Inspect bonding and shielding for proper installation and condition.

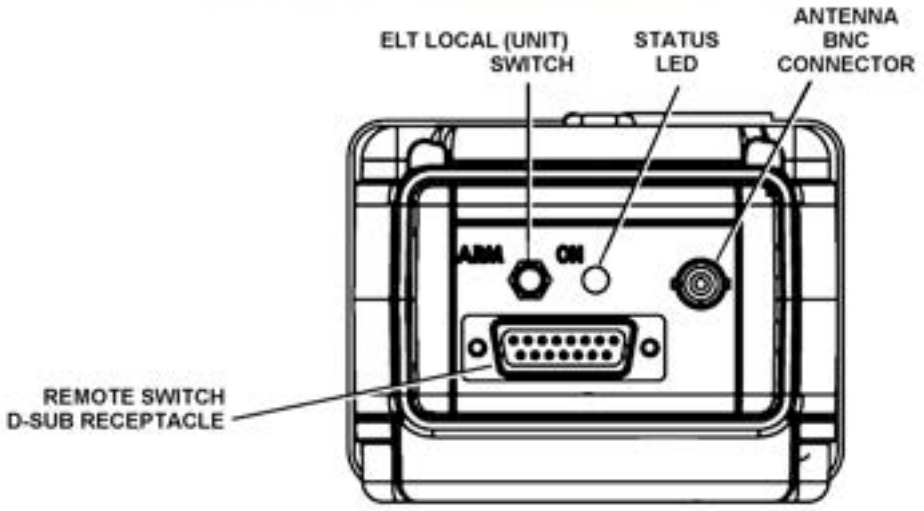
CHART 2
TROUBLESHOOTING ELT 1000

Trouble	Cause	Remedy
Remote switch LED always on (Steady).	Improper wiring.	Verify wiring.
	Short circuit.	Verify integrity of all crimp and solder connections.
ELT Will Not Turn Off.	**Turn ELT “OFF” Using Local Switch. If ELT turns OFF, then:**	
	Defective remote switch harness reset circuit wiring.	Check wiring continuity and repair as necessary.
	Defective remote switch.	Replace remote switch.
	If ELT Does Not Turn “OFF” Using Local Switch. then:	
	ELT defective.	Remove battery pack to disable ELT and return ELT to manufacturer for servicing.
Flash Errors.	Self-Test.	See “Chart 3” on page 256029.



COCKPIT (REMOTE)
SWITCH





ELT 1000 Manual Activation Switches
Figure 5

(d) Inspect antenna for condition, secure mounting, and proper operation (per “Antenna Test” on page 256027).

(e) Check ELT for signs of registration. In the US, NOAA supplies a beacon registration label that is applied to the ELT when it is registered.

(f) Make an appropriate logbook entry documenting completion of this inspection and whether or not the ELT passed or failed.

(2) Annual

Every twelve (12) months, inspect the ELT installation as follows:

NOTE: The following inspection satisfies the requirements of FAR 91.207.

(a) Remove ELT from its mounting tray per Removal, below.

Inspect the mounting hardware. Ensure the hardware is free of cracks or other obvious damage.

(b) Battery Inspection

NOTE: The battery pack contains static sensitive parts, take ESD precautions before handling.

1) Remove the eight (8) securing screws from the battery-side cover.

NOTE: Battery pack is identified by the embossed text:
“BATTERY ACCESS ON THIS SIDE”.

CAUTION: DO NOT PULL ON THE FLEXIBLE PORTION OF THE CABLE - USE THE RIGID SECTION OF THE FLEX CIRCUIT AT THE CONNECTOR AS A HANDLE.

2) Carefully lift the battery cover (battery pack) away from the ELT and unplug the flex-cable connected to the pack.

3) Inspect the battery pack and ELT chassis. The battery cells, components and connectors should be free of corrosion.

4) Inspect flex-circuit for broken connections or damage.

5) Ensure the battery housing is free of cracks or other visible damage.

6) Verify the battery expiration date. If the battery pack has not expired it may be reinstalled. The battery pack must be replaced with a new one:

a) After use in an emergency;

b) When the transmitter has been in use for more than one (1) cumulative hour (i.e., 7 flash error);

c) After an inadvertent activation of unknown duration; or

d) On or before the battery replacement (expiration) date.

(c) Perform G Switch Check

(d) Reinstall ELT into airplane per Installation

(e) Perform Antenna Test

NOTE: For a fee, http://www.406Test.com allows testing of the ELT through-the-satellite.

(f) Perform “Digital Message Verification” on page 256031.

(g) Reset ELT by turning ELT switch to “ON” then to ‘ARM” position.

(h) Check ELT for signs of registration. In the US, NOAA supplies a beacon registration label that is applied to the ELT when it is registered.

(i) Make an appropriate logbook entry documenting completion of this inspection and whether or not the ELT passed or failed.

D. Removal

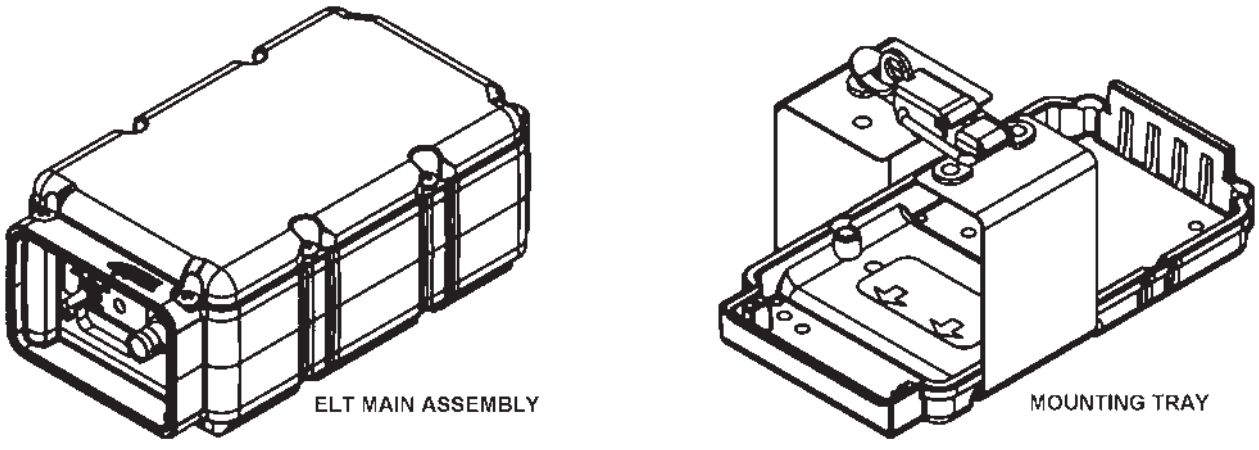
See “Figure 6” and “Figure 7” on page 256023.

(1) In the cabin, remove the rear closeout panel.
(2) Disconnect and remove negative cable from battery.
(3) At right side of empennage, remove four (4) screws and remove ELT access panel.
(4) Loosen the thumbscrews and remove the D-sub and RF connectors.
(5) Remove ELT Connections:
  (a) Loosen the thumbscrews and remove the D-sub and RF connectors.
  (b) Visually inspect and confirm proper seating of all connector pins. Special attention should be given to coaxial center conductor pins which are prone to retracting into the connector housing.
(6) Unlock the retention strap and remove the ELT from its mounting tray.

E. Installation

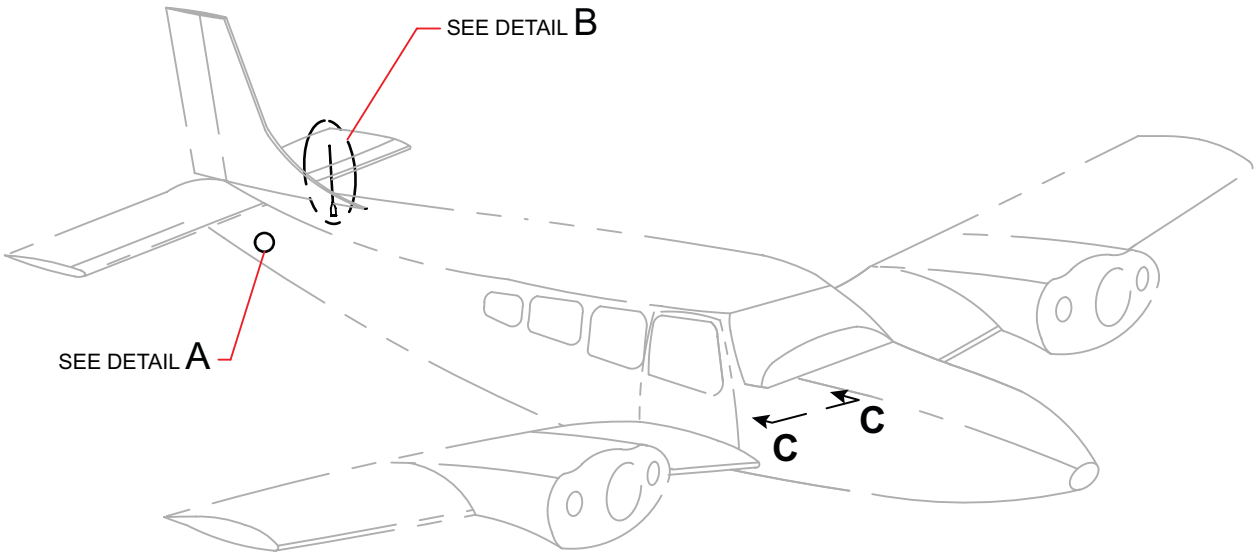
See “Figure 6” and “Figure 7” on page 256023.

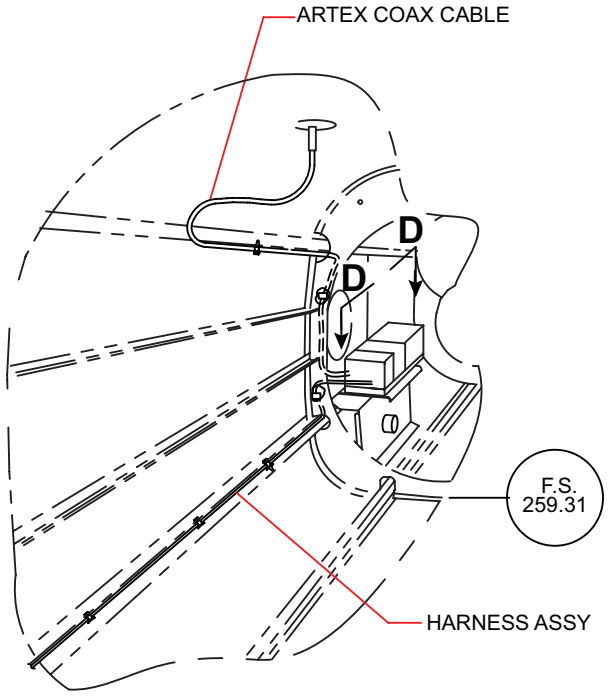
(1) Insert the ELT into the mounting tray at an angle so that the locking ears at the end fit into the mounting tray locking slots.
(2) Lock the retention strap around the ELT so that it is firmly held in place.
(3) Insert the D-sub and RF connectors ensuring that they are seated properly.
(4) Tighten the thumbscrews.
(5) At right side of empennage, place ELT access panel in position and secure with four (4) screws.
(6) In the rear cabin, connect negative cable to battery.
(7) Reinstall the rear closeout panel.



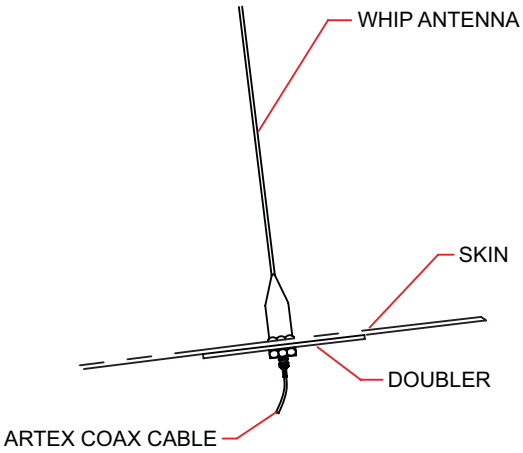
ELT 1000 Assembly
Figure 6

34F34A009 NEW





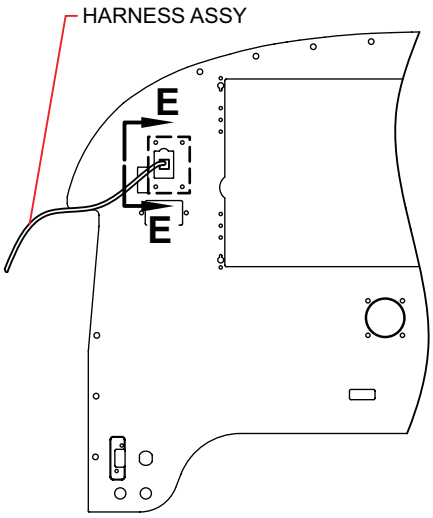
DETAIL A
RHS LOOKING AFT



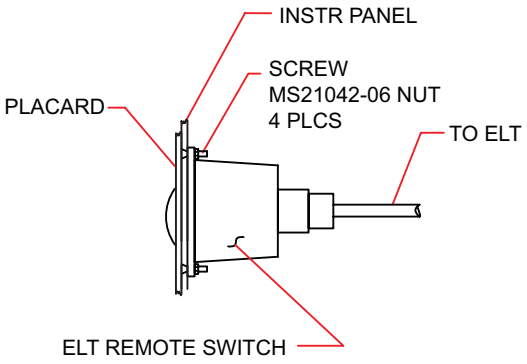
DETAIL B
RHS LOOKING INBD

ELT 1000 Installation
Figure 7 (Sheet 1 of 2)

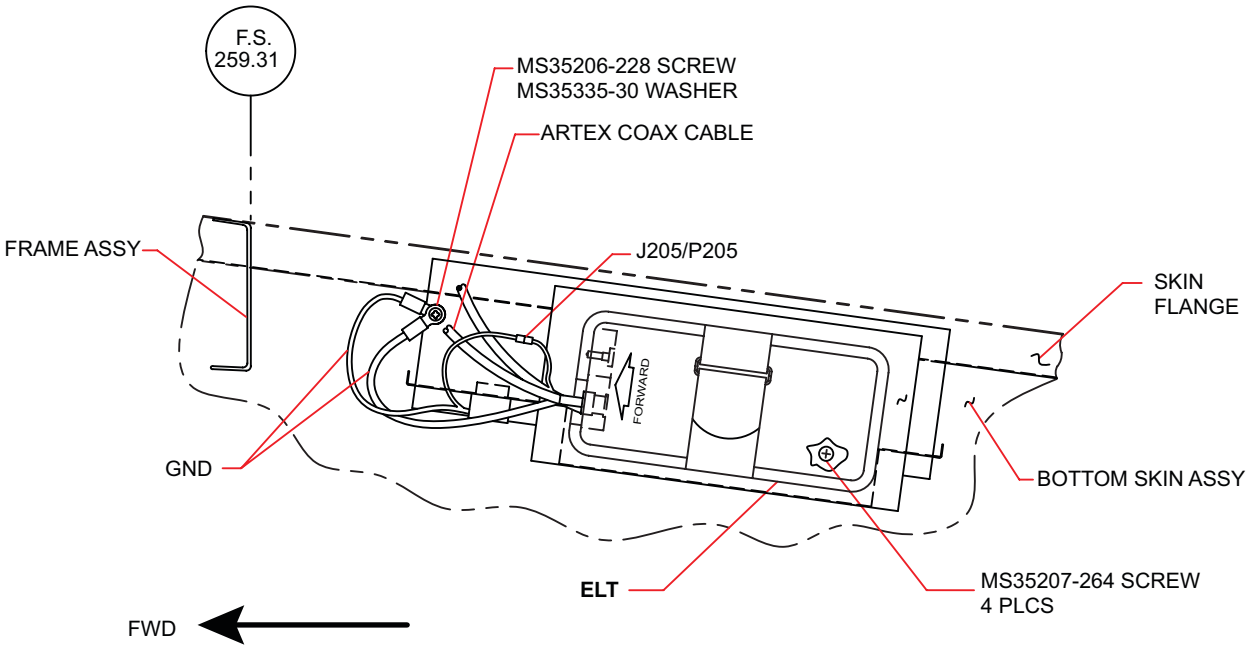
34F34A009 NEW



VIEW C - C
RHS LOOKING AFT



VIEW E - E
RHS LOOKING INBD



VIEW D-D
ELT 1000 RADIO INSTL
ROTATED 45˚ CW
RHS LOOKING DN

ELT 1000 Installation
Figure 7 (Sheet 2 of 2)

F. Battery Replacement

See “Figure 8” on page 256026.

NOTE: When replacing the battery pack, a replacement kit containing the battery pack, replacement gasket, hardware and labels is available.

(1) Remove ELT per “Removal” on page 256022.

(2) Remove the battery pack as follows:

NOTE: The battery pack contains static sensitive parts, take ESD precautions before handling.

(a) Remove the eight (8) securing screws from the battery-side cover.

NOTE: Battery pack is identified by the embossed text: “BATTERY ACCESS ON THIS SIDE”.

CAUTION: DO NOT PULL ON THE FLEXIBLE PORTION OF THE CABLE - USE THE RIGID SECTION OF THE FLEX CIRCUIT AT THE CONNECTOR AS A HANDLE.

(b) Carefully lift the battery cover (battery pack) away from the ELT and unplug the flex-cable connected to the pack.

(c) Discard/recycle the old battery pack.

(3) Lay the new battery pack on the work surface with the batteries facing up.

(4) Install a replacement seal in the slot along the perimeter of the housing.

(5) Leaving the battery as it is, position the ELT over the battery pack with one hand and plug the flex-cable connector into the battery assembly using the other. The cable should not be twisted and the connector should ‘click’ into place.

NOTE: The battery connector is keyed to prevent incorrect installation.

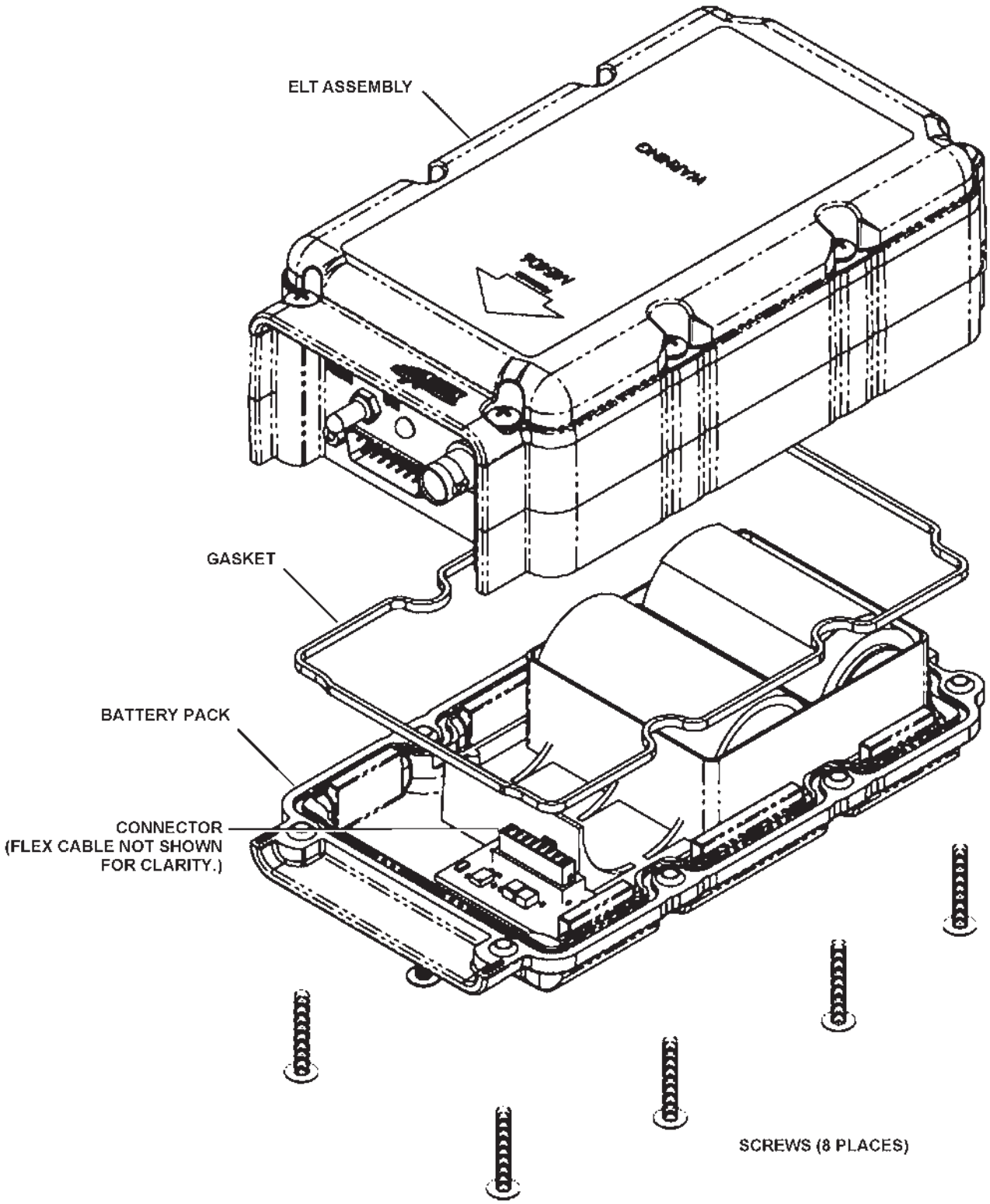
(6) Mate the ELT to the battery, making sure that the seal is positioned correctly during the process.

(7) Replace the eight (8) securing screws and torque to 10 - 12 inch-lbs.

(8) Install ELT per Installation

(9) Perform “Installed Transmitter Test (Self Test)” on page 256027.

(10) Enter pertinent battery replacement information in the aircraft log book and fill out any other documentation required by local authority.



ELT 1000 Battery Assembly
Figure 8

G. Testing

CAUTION: ALL ELT "ON" TESTS SHOULD BE PERFORMED WITHIN THE FIRST FIVE (5) MINUTES AFTER THE HOUR UTC OR AS REQUIRED BY LOCAL OR NATIONAL AUTHORITIES. NOTIFY ANY NEARBY CONTROL TOWER OF YOUR INTENTIONS.

CAUTION: DO NOT ALLOW TEST DURATION TO EXCEED FIVE (5) SECONDS.

CAUTION: CONSULT FAA ADVISORY CIRCULAR AC 91-44A, LATEST REVISION, FOR DETAILED INFORMATION CONCERNING UNSHIELDED TESTING.

Always perform the tests within the first five (5) minutes of the hour. Notify any nearby control tower of your intentions. If outside of the US, always follow all local or national regulations for testing ELTs. Do not allow test duration to exceed five (5) seconds. Any time the ELT is activated it is transmitting a 121.5 MHz distress signal. If the unit operates for approximately 50 seconds, a 406 MHz distress signal is transmitted and is considered valid by the satellite system.

(1) Installed Transmitter Test (Self Test)

Every 90 Days

(a) Turn both the airplane master switch and the radio master switch ON.

(b) Tune a receiver (usually the aircraft radio) to 121.5 MHz.

(c) Turn the ELT aircraft panel switch to "ON", wait for three (3) sweeps on the receiver, which takes about one (1) second, and then turn the switch back to the "ARM" position while paying special attention of the LED activity upon entering the 'ARM" condition.

(d) To pass the test, you must hear the three (3) sweeps and see the front panel light immediately begin to flash continuously until the unit is switched to "ARM".

1) During the "ON" to "ARM" transition, the microprocessor in the ELT checks the "G-Switch" (automatic activation switch) latching circuit, pins 5 & 12 on the D-sub connector at the ELT; the 406 MHz transmitter for proper RF output and a battery check.

2) If the ELT is working properly, the sequence following entry to the "ARM" condition will result in the panel LED staying illuminated for approximately one (1) second (one pulse), and then extinguishing. The buzzer should also sound once.

(e) If the panel LED and buzzer present more than one (1) pulse when switched to "ARM", determine the problem from the list in "Chart 3" on page 256029.

(2) Antenna Test

Each 100 Hours use a low quality AM broadcast receiver to determine if energy is being transmitted from the antenna.

(a) Hold the antenna of the radio (tuning dial on any setting) about six (6) inches from the activated ELT antenna. The ELT aural tone should be heard on the AM broadcast receiver.

(b) This is not a measured check, but it does provide confidence that the antenna is radiating sufficient power to aid search and rescue.

NOTE: Use of the aircraft's VHF receiver, tuned to 121.5 MHz, is not recommended. This receiver is more sensitive and could pick up a weak signal even if the radiating ELT antenna is disconnected. While it will confirm the ELT is active, it does not check the integrity of the ELT system.

(3) G-switch Check

Test operation of the G-switch annually as follows:

CAUTION: EVEN WITHOUT AN ANTENNA CONNECTED, THE SIGNAL CAN BE RECEIVED BY A SATELLITE. AS WITH ALL BEACON-TESTING, THIS TEST SHOULD BE PERFORMED WITHIN THE FIRST FIVE (5) MINUTES OF THE HOUR, AND ANY LOCAL CONTROL TOWER SHOULD BE ADVISED OF THE TEST.

(a) Setup

1) The ELT must be removed from the airplane. See Removal

2) Fabricate a “shorting plug” by obtaining a standard 15-pin D-sub connector and installing a jumper (i.e. - short) between Pins 5 and 12. (Artex sells a test plug for this purpose under part number 150-1130.)

NOTE: Be sure the correct pins are shorted. Some connections will force the ELT to activate. Others will keep the unit from activating in any circumstance. No combination of shorts will cause permanent damage to the ELT, however all wrong pin combinations erroneously indicate a faulty ELT.

If the ELT activates without any pins shorted it is defective and should be returned for repair.

3) Install the “front connector” on the ELT.

4) For a more thorough test, monitor the transmission with an AM receiver tuned to 121.5 MHz, as described in “Installed Transmitter Test (Self Test)”.

(b) Procedure

The ELT should remain OFF until an acceleration of three (3) Gs or more is applied axially, in the rearward direction. This action should activate the unit, transmitting immediately on 121.5 MHz.

CAUTION: MAINTAIN A FIRM GRIP ON THE ELT.

1) Apply acceleration greater than three (3) Gs to the ELT, in the rearward direction. This can be achieved by using a rapid forward (throwing) motion in the direction of the label arrow, then rapidly reversing the direction.

2) Monitor ELT activation by observing the ELT LED (and AM receiver on 121.5 MHz, if desired). Verify activation by listening for the aural sweep tone on the receiver. The LED on the ELT housing will also begin to flash, indicating activation. For proof, the AM radio will verify that the RF output is operational.

3) Allow ELT to transmit only long enough to verify operation.

CAUTION: ENSURE ELT IS RESET BEFORE 50 SECONDS HAS ELAPSED FROM SUCCESSFUL ACTIVATION. FAILURE TO DO SO WILL PERMIT A “LIVE” SIGNAL TO BE SENT TO THE SEARCH AND RESCUE SATELLITES.

4) Reset ELT by turning ELT switch to “ON” then to ‘ARM” position.

CHART 3 (Sheet 1 of 2)
ELT 1000 SELF-TEST

Beeps/Flashes	Result	Indication/Action
1	System OK.	One pulse two seconds long of the LED, remote light and buzzer.
2	Code not used in ELT 1000.	
3	Code not used in ELT 1000.	
4		Low output power.
	Bad load or faulty ELT unit	1) Terminate antenna cable at antenna connection end w/50Ω load. Repeat self-test. If error clears proceed to step 3.
		2) Terminate antenna cable at antenna connection end w/50Ω load. Repeat self-test. If error clears proceed to step 3.
		3) Check coax cable and connectors for shorts and opens, bad center pin, etc. Disconnect, then reconnect cable. Repeat self-test.
		4) Check antenna ground for low resistance (should be <100 m Ω).
		5) Antenna should be 3 feet away from any vertical metal.
		6) Have antenna checked by factory.
5		No position data present (long message protocol only. Error is suppresed for short messages).
	Aircraft navigation system “OFF”.	Turn ON navigation system.
	Faulty system interface wiring or connections.	Check integrity of wiring and connections.
6		G-Switch loop between pins 5 and 12 is not present.
	Jumper open.	Verify D-Sub connector jumper is installed by checking for less than 1Ω between pins 5 and 12. Repair as necessary.
	Jumper missing.	Install G-SW enable jumper between pins 5 and 12 in D-sub harness connector.
7		Battery Check.
	Battery operating time is >1 hr.	Replace battery pack.
	Battery voltage low.	Replace battery pack.
	Battery memory read-write error.	Repeat self-test. Replace battery pack if 7-flash persists.

CHART 3 (Sheet 2 of 2)
ELT 1000 SELF-TEST

Beeps/Flashes	Result	Indication/Action
8		Protocol Programming Data.
	Missing data such as ELT S/N or aircraft tail number.	ELT requires reprogramming w/appropriate data.

(4) Digital Message Verification

The ARTEX ELT 1000 transmits a 406 MHz message upon reset, which is encoded such that it will be ignored by the SAR satellite system. The 15-digit number contained in this transmission is used to register the ELT with the appropriate 406 MHz ELT registration authority. In the US, the National Oceanic and Atmospheric Administration (NOAA) maintains the database of registered ELT's. The information in this database provides the Search and Rescue system with aircraft identification data in the event an actual distress signal is transmitted.

Annually verify the 406 MHz digital message as follows:

(a) Required Equipment

A test set capable of receiving and decoding the message - i.e., Artex ELT Test Set P/N 453-1000, or equivalent.

(b) Procedure

1) At right side of empennage, remove four (4) screws and remove ELT access panel.
2) Disconnect the antenna coax cable at the ELT, connect test set or terminate as applicable.
3) Perform all necessary steps to prepare Test Set to receive 406 MHz signal including (but not limited to) turning on power, activating program or any other steps required for the particular Test Set being used.
4) Perform the Installed Transmitter Test (Self Test) .
5) Watch the screen on the Test Set to ensure that a message has been received. Repeat Self Test if necessary.
6) View message, ensure that all applicable information is correct (country code, aircraft ID, etc.).
7) The 15 digit ID hex ID (for example "ADC6492640D3411F1') should match what is shown on the ELT product label. This is the 15 digit hex ID (Unique Identification Number or 'UIN") that is used to register the ELT.
8) At right side of empennage, place ELT access panel in position and secure with four (4) screws.

H. Programming

(PIR-107994 D)

This procedure is used when installing a new or exchanged Artex ELT 1000 emergency locator transmitter.

NOTE: Emergency locator reprogramming is required on all ELT 1000 equipped aircraft. Perform the following.

(1) Plug programming cable into the ELT and connect to the Handheld Programmer.

(2) Power on Handheld. Press “Home”. Main Screen View should appear.

NOTE: Press “Home” until Artex Icon appears on the Main Screen View.

(3) From the Handheld Programmer Main Screen View, Select “ARTEX”.

(4) Artex ELT Programmer page, select “Program”.

(5) Artex ELT Programmer page.

NOTE: Artex Programmer may default to previously entered Model, Protocol and Country Code.

(a) In the dropdown menu, select ELT Model “ELT1000”.

(b) Program GPS Baud Rate to 4800:

1) Select “Options”.

2) Select the “Test” position on the ELT 1000 and release within two (2) seconds after selecting “Get Options” on the Artex Programmer.

3) Select the Dropdown Menu.

4) Select the GPS Baud “4800”.

5) Select the “Test” Position on the ELT 1000 and release within two (2) seconds after selecting “Set” on the Artex Programmer.

6) Verify “Successfully Programmed Speed” is displayed on the Artex Programmer.

7) Select “Return”.

(c) Dropdown menu Protocol

- For U.S. aircraft and aircraft bound for export country that requires Tail number registration on the ELT:
  a) Select “Aviation User/Tail No./Loc (L)”.
  b) Dropdown menu, select desired Country Code.
  c) Enter the tail number of the aircraft (N number), which may by up to seven alphanumeric characters. Use stylus to select “abc” (for alphabet) and select “123” (for numbers). (If keyboard was opened, press “Done”)
  d) When finished, select “Enter”.
- For aircraft bound for export country that requires 24 bit address registration:
  a) Select “Std Loc/Aircraft 24-bit Addr (L)”.
  b) Select the desired number system (Octal: 8 digits from 0-7 or Hex: 6 digits from 0-9 or A-F) to match the format of the 24-bit address provided.
  c) Enter the 24-bit address. (If keyboard was opened, press “Done”)
  d) When finished, select “Enter”.

NOTE: The 24 digit hex code is shown at the bottom of the screen with the abbreviated 15 digit hex code below it.

(d) Select the test position on the ELT 1000 switch and release within 2 seconds after selecting “Program” on the programmer.

(e) Verify that the screen shows “Programming Successful Verify ELT Using Sarcalc”.

(6) Label Printing

(a) Connect the Handheld Programmer to the applicable printer suing the specified cable and adapter.

(b) After the ELT has been successfully programmed, select “Print”, from Printing ELT Data, select “Combined Label”. After printing is complete, select “Return”.

NOTE: Place new label over current ELT’s label that reads, “This ELT is Programmed For and Must be Registered in Country, Three Digit Country Code, 15-Digit Hex ID”.

(7) Programming Verification

NOTE: For a Direct Verification, connect the Handheld coax cable to the ELT and for a Remote Verification, remove coax cable and connect Handheld Whip Antenna. For a Remote Verification, a distance of 0-30 feet from the ELT antenna is recommended.

(a) Select “Sarcalc”.

(b) Select “Measure”.

(c) The screen will display “Waiting for Data”.

(d) Turn the ELT switch to “TEST” then release.

(e) Verify the message contains the correct information for the aircraft.

(f) When completed, turn the Handheld OFF.

I. Installation Verification

NOTE: Perform the following after ELT installation is completed in the aircraft.

(1) Perform the following functional check within the first five (5) minutes after the hour (UTC), as required by FAA AC 43.13-1, Chapter 12, § 12-22, Note 3.

(2) Tune an AM receiver to 121.50 MHz and locate it approximately 20 to 30 feet from the ELT antenna.

(3) Initiate the self-test by momentarily pressing the cockpit remote switch into the TEST position for approximately one (1) second.

(4) On the AM receiver, listen for three (3) audible sweeps, which takes about one (1) second.

(5) Note the LED activity on the cockpit remote switch or the audio beeps from the ELT. If the ELT is working properly, the LED will stay on for approximately one (1) second and then turn OFF accompanied by a short beep and a long beep.

J. Electrical Schematic

See 91-25-60.

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CHAPTER

26

FIRE PROTECTION

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CHAPTER 26

LIST OF EFFECTIVE PAGES

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CHAPTER 26 - FIRE PROTECTION

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Monthly		1
100 Hour / Annual		2
Gauged, Class 2B:C Fire Extinguisher Model No. A344T		2
Description		2
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Maintenance		2
Annual		2
Six-Year Maintenance		3
Twelve-Year Hydrostatic Test		3
Replacement Interval		3

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EXTINGUISHING

WARNING: AFTER DISCHARGE OF EXTINGUISHER AVOID EXPOSURE TO SMOKE, VAPORS AND OTHER BY-PRODUCTS OF FIRE. DO NOT INCINERATE EXTINGUISHER.

CAUTION: EXTINGUISHER IS A PRESSURE VESSEL. PROTECT FROM CORROSIVE CONDITIONS. IF THERE IS ANY CORROSION OR DAMAGE, EXTINGUISHER SHOULD BE CAREFULLY EMPTIED AND DISCARDED. USE ONLY AS DIRECTED.

1. Portable Fire Extinguisher

Factory installed in S/N's 3447001 through 3447029 and 3449001 through 3449500.

A. Disposable, Class 2B:C Fire Extinguisher Model No. RT-A600

(1) Description

Containing 1.2 lbs. (550 grams) of Halon 1211/1301 blend, The extinguisher is located right behind the co-pilots seat. It is secured by a quick release bracket. See "Figure 3" on page 26203.

To operate the extinguisher, remove it from the quick-release bracket, hold it upright in either hand by the handgrip, with the spray nozzle pointing forward. Remove the safety pin, direct the nozzle towards the base of the fire source, depress the lever. Maximum extinguishing effect is obtained if the fire fighter uses side to side motion and keeps moving in towards the base of the fire source as it is extinguished. Releasing the lever closes a secondary seal inside the operating head. This interrupts the flow of extinguishant, thus retaining part of the charge, for dealing with a flash back or re-ignition should they occur, without waste or leakage. A partly or totally discharged extinguisher should be replaced immediately after use.



Model RT-A600
(Piper P/N 459-887)
Figure 1

(2) Inspection

Disposable type fire extinguishers should be maintained and inspected in accordance with the nameplate instructions.

(a) Before Each Flight

RT-A600 extinguishers with date codes 2012 and later are subject to H3R Service Bulletin No. 2015-01 (Piper Service Bulletin No. 1280A). These extinguishers should be labeled "Weigh this unit before each flight" and must be weighed before each flight. Weight must be above the minimum specified on the nameplate.

(b) Monthly

Inspect monthly or more frequently. Ensure nozzle is not obstructed and safety seal is intact. Inspection is a "quick check" that an extinguisher is available and will operate. It is intended to give reasonable assurance that the extinguisher is fully charged and operable. This is done by seeing that it is in its designated place, that it has not been actuated (discharged) or tampered with, and that there is no obvious physical damage or condition to prevent operation. Determine fullness by weighing or "hefting."

(c) 100 Hour / Annual

Each 100 hours or annually, whichever comes first, weigh the extinguisher. Replace and return to manufacturer if gross weight is below the minimum specified on the nameplate.

B. Gauged, Class 2B:C Fire Extinguisher Model No. A344T

Factory installed in S/N's 3449501and up. Authorized service replacement in all Seneca IV's and V's.

(1) Description

Containing 1.25 lbs. (567 grams) of Halon 1211. The extinguisher is located right behind the co-pilots seat. See "Figure 3" on page 26203.

To operate the extinguisher, remove it from the quick-release bracket, hold it upright in either hand by the handgrip, with the spray nozzle pointing forward. Remove the safety pin, direct the nozzle towards the base of the fire source, depress the lever. Maximum extinguishing effect is obtained if the fire fighter uses side to side motion and keeps moving in towards the base of the fire source as it is extinguished. Releasing the lever closes a secondary seal inside the operating head. This interrupts the flow of extinguishant, thus retaining part of the charge, for dealing with a flash back or re-ignition should they occur, without waste or leakage. A partly or totally discharged extinguisher should be replaced immediately after use.

Gauged type fire extinguishers should be installed, inspected, and maintained in accordance with the owner's manual supplied with the extinguisher and NFPA No. 10, "Portable Fire Extinguishers." The following summarizes portions of those two documents.



Model A344T
(Piper P/N 459-750)
Figure 2

(2) Inspection

Check when initially placed in service and at minimum 30-day intervals or more frequently if circumstances dictate. Persons performing 30-day inspections are not be required to be certified.

Check to see that the extinguisher is undamaged; that the nozzle is unobstructed; that the gauge pressure is in the operable (green) range; that the lockpin and tamper seal is in place, and that the operating instructions are clearly visible. If the inspection reveals a deficiency in any of these conditions, corrective action must be taken.

(3) Maintenance

(a) Annual

Extinguishers shall be subjected to maintenance not more than one year apart or when specifically indicated by an inspection. Maintenance is a thorough examination of the extinguisher, covering mechanical parts, extinguishing agent and expelling means. It is intended to give maximum assurance that an extinguisher will operate effectively and safely, and should be done professionally. Most authorities require special tags be attached to the extinguisher to verify this service.

(b) Six-Year Maintenance

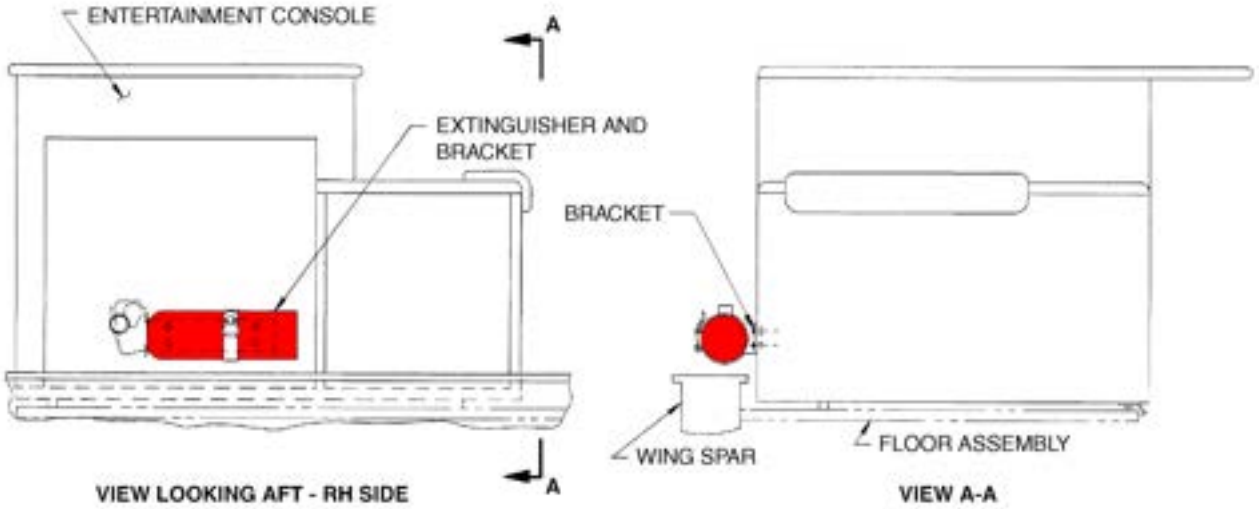
Every six years, stored pressure fire extinguishers that require a 12-year hydrostatic test shall be emptied and subjected to applicable maintenance procedures. This should be done professionally, and involves a thorough inspection and the replacement of certain parts. When these maintenance procedures are performed during periodic recharging or hydrostatic testing, the six year requirement shall begin from that date.

(c) Twelve-Year Hydrostatic Test

All gauged H3R Aviation fire extinguishers require a 12 year hydrostatic test performed by certified technicians as per NFPA 10. Simply stated, the Hydrostatic test confirms the integrity of the cylinder, and at the 12 year mark, is performed in conjunction with the 6 year maintenance.

C. Replacement Interval

Replace disposable-type (non-gauged) fire extinguishers as specified in 5-30-00.



Fire Extinguisher Locator
Figure 3

Effectivity
Seneca V
with Entertainment Console

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CHAPTER

27

FLIGHT CONTROLS

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CHAPTER 27

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CHAPTER 27 - FLIGHT CONTROLS

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GENERAL

1. Description and Operation

The airplane is controlled in flight by the use of three primary control surfaces, consisting of ailerons, stabilator, and rudder. Operation of these controls is through the movement of the control column-tee bar assembly and rudder pedals.

On the forward end of each control column is a sprocket assembly. A chain is wrapped around the sprockets to connect the right and left controls and then back to idler sprockets on the column's tee bar, which in turn connect to the aileron primary control cables. The cables operate the aileron bellcrank and push-pull rods. The stabilator is controlled by a cable connected to the bottom of the tee bar assembly and operates an aft fuselage bellcrank which controls a push rod connected to the balance arm of the stabilator. Cables also connect the rudder pedals with the rudder horn.

Provisions for directional and longitudinal trim control is provided by an adjustable trim mechanism for the stabilator and rudder. The stabilator trim is controlled by a wheel and drum mounted on the floor tunnel between the front seats. Cables routed aft from the drum to a screw assembly mounted above the stabilator attachment point. This screw assembly in turn moves the push rod which controls the stabilator trim tab. The rudder trim is controlled by a knob and screw assembly attached to the rudder pedal assembly. The flaps are electrically operated.

2. Standard Practices and Procedures

NOTE: For additional information, refer to Federal Aviation Administration Advisory Circular AC 43.13-1B, Chapter 7. Download from the FAA at https://www.faa.gov/.

WARNING: FAILURE TO VERIFY FREE AND CORRECT OPERATION OF THE FLIGHT CONTROL SYSTEM MAY RESULT IN THE DESTRUCTION OF THE AIRPLANE AND LOSS OF LIFE, IF THE AIRPLANE IS FLOWN.

IF THERE IS ANY UNCERTAINTY, DO NOT FLY AN AIRPLANE UNTIL FREE AND CORRECT OPERATION OF THE FLIGHT CONTROL SYSTEM IS VERIFIED.

CONTACT PIPER CUSTOMER SERVICE FOR ASSISTANCE AT (+1) 772-299-2141 OR CUSTOMERSERVICE@PIPER.COM. PIPER'S NORMAL BUSINESS HOURS ARE MONDAY THROUGH FRIDAY, 7:30 A.M. TO 4:30 P.M. (EASTERN).

A. General

(1) It is important to remove all necessary access panels to gain both physical and visual access to control cables to ensure and verify proper routing in accordance with applicable service or maintenance manual instructions.

(2) Pay particular attention to control cable part numbers per the applicable parts catalog, in order to verify the proper replacement. Route and locate cables per the applicable service or maintenance manual.

(3) Where practical, remove and install one flight control cable, motor, actuator, or pushrod at a time. This will help avoid incorrect reassembly and aid in the proper connection of the flight controls.

(4) Flight control cable rigging tensions specified in "Chart 2" on page 27004 must be corrected to account for ambient temperature, use "Chart 1" on page 27002.

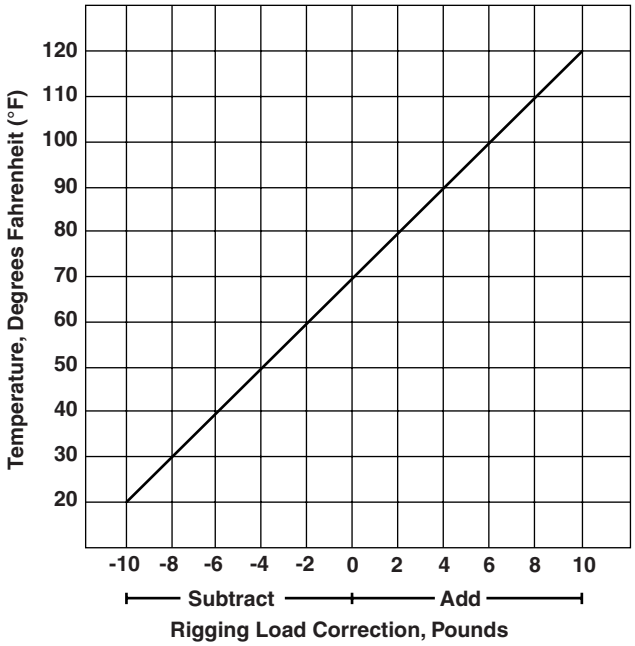
(5) Verify that the flight control system operates smoothly throughout the full range of motion.

(6) Visually confirm that all pulleys rotate freely. Visually confirm that all cables pass through all guides and fairleads without snagging or binding, and do not interfere with or chafe on any airframe or system components.

(7) Verify that the flight control surfaces move in the proper direction when commanded.

(8) Recheck the tension of replacement flight control cables upon reaching 50 hours of operation.

CHART 1
CABLE TENSION vs. AMBIENT TEMPERATURE



B. The following tips may be helpful in the removal, installation, and rigging of individual control system assemblies.

(1) It is recommended, though not always necessary, to level and place the airplane on jacks during rigging and adjustment.

(2) Remove turnbuckle barrels from cable ends before withdrawing the cables through the structures.

(3) Tie a cord to the cable end before withdrawing the cable through the structures to facilitate installation of cable.

(4) Mark cable ends, etc., before disconnecting, use a felt tip marking pen.

(5) Assemble and adjust the turnbuckles so that each terminal is screwed an approximately equal distance into the barrel. Do not turn the terminals in such a manner that will put a permanent “twist” into the cables.

(6) Cable tensions should be taken with the appropriate control surface in its neutral position.

NOTE: Cable rigging tensions specified must be corrected to ambient temperature in the area where the tension is being checked, using “Chart 1”. When installing new cables, initially tension them 20 to 30% over nominal tension, then loosen them to the “high side” of the tolerance. This “pre-stretching” will aid in maintaining specified tension after flight testing.

(7) Check all cable ball ends for proper seating in retainers after setting cable tension.

(8) After completion of each adjustment, check the turnbuckles to be sure not more than three terminal threads are visible outside the barrel. Install the locking clips, and check for proper installation by trying to remove the clips using fingers only. Both locking clips may be installed in opposite holes. Locking clips which have been installed and removed must be scrapped and not reused. Safety turnbuckles in accordance with AC 43.13-1, latest revision.

(9) When pushrods or rod ends are provided with an inspection hole, the screw must be screwed in far enough to pass the inspection hole. This can be determined visually or by feel, inserting a piece of wire into the inspection hole. If no hole is provided, there must be a minimum of 0.375 (3/8) of an inch thread engagement.

(10) When installing/adjusting rod end jamnuts, refer to 20-00-00, Installing Rod End Bearings, for the proper method.

(11) After completion of adjustments, each jamnut must be tightened securely.

NOTE: Torque all nuts in the flight control system (including nose wheel steering). Refer to 20-00-00, Torque Requirements.

(12) Ensure all pulley guard pins are properly installed and secured.

3. Flight Control Surface Travel

See “Chart 2” on page 27004 for specifications, see appropriate section for rigging instructions.

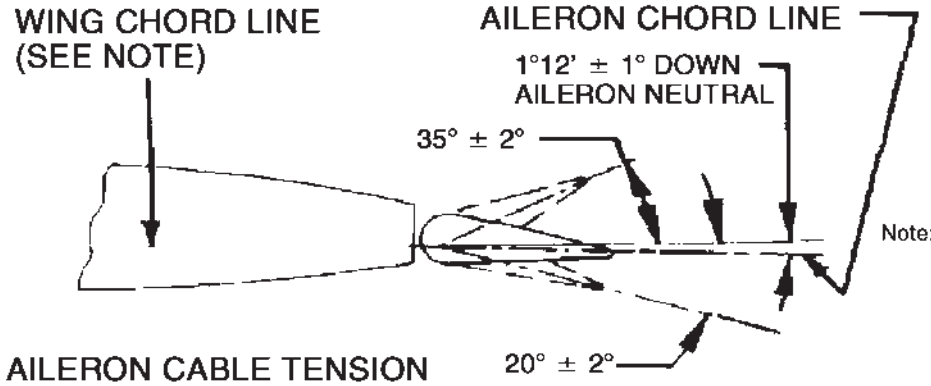
4. Flight Control Cable Tension

CAUTION: CABLE TENSIONS GIVEN IN “CHART 2” ON PAGE 27004 APPLY ONLY TO AIRPLANES WITHOUT AUTOPILOT BRIDLE CABLE INSTALLATIONS, UNLESS THE AUTOPILOT SYSTEM WAS FACTORY INSTALLED BY PIPER. IF AN AUTOPILOT USING BRIDLE CABLES HAS BEEN INSTALLED AFTER LEAVING THE FACTORY, CONSULT THE APPROPRIATE AUTOPILOT VENDOR DOCUMENTATION FOR CORRECT CABLE TENSIONS WITH AUTOPILOT BRIDLE CABLES ATTACHED.

A. See “Chart 2” on page 27004 for specifications, see appropriate section for rigging instructions.

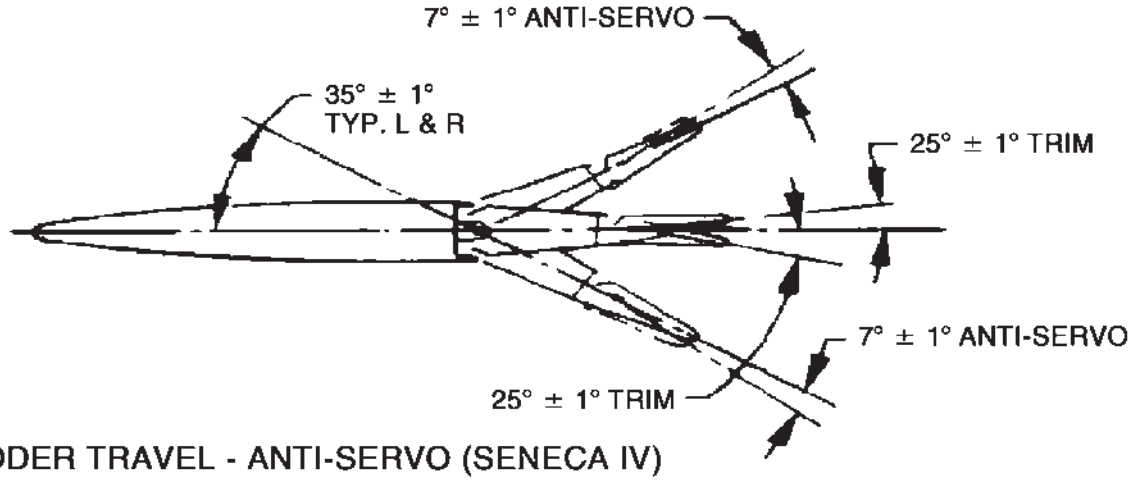
B. When a new cable is installed, cable tension must be rechecked after flight test.

CHART 2 (Sheet 1 of 2)
FLIGHT CONTROL SURFACES RIGGING LIMITS (PIR-5009-4 Rev. G, PPS-5009-5 Rev. P)

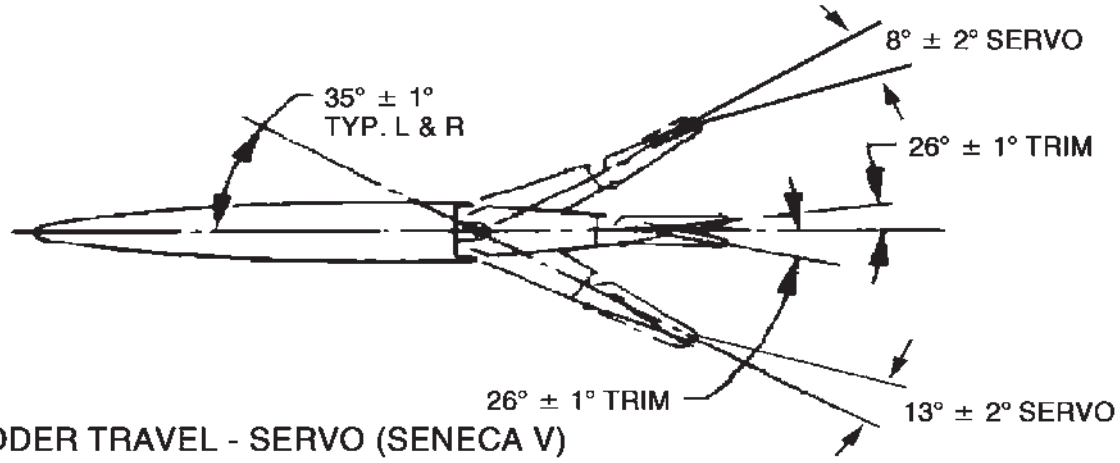


Note: Travels are measured from the position where the aileron chord is aligned with the wing chord (i.e. - aileron neutral.)

AILERON CABLE TENSION
40 ± 5 LB.



RUDDER TRAVEL - ANTI-SERVO (SENECA IV)

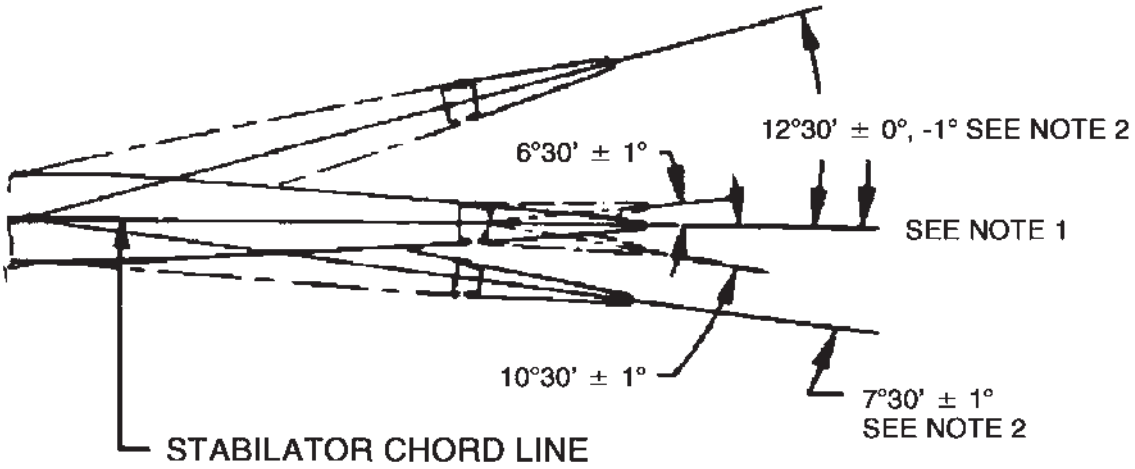


RUDDER TRAVEL - SERVO (SENECA V)

CABLE TENSIONS

RUDDER CABLES	40 ± 5 LB.
RUDDER TRIM CABLES	10 ± 2 LB.

CHART 2 (Sheet 2 of 2)
FLIGHT CONTROL SURFACES RIGGING LIMITS



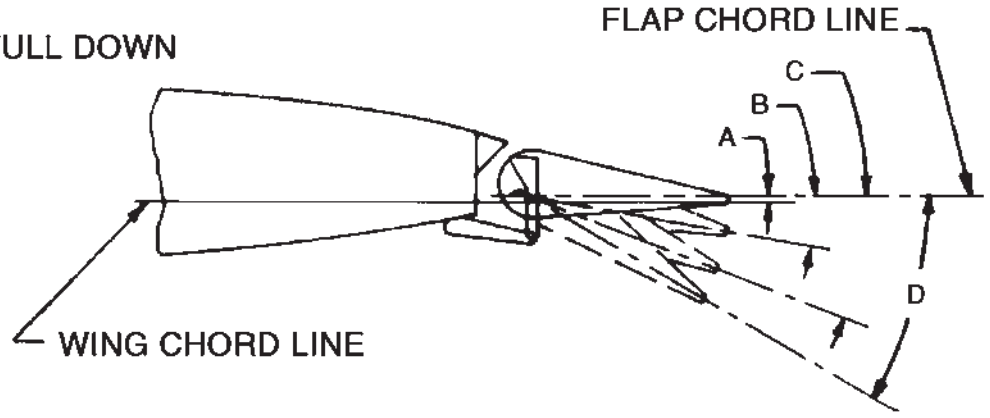
Notes:
1. The neutral position of the stabilator is when the stabilator chord line is parallel with the top of the seat track.
2. Total tab free play is not to exceed 0.06 inch.

CABLE TENSIONS

STABILATOR	40 ± 5 LB
STABILATOR TRIM TAB	10 ± 1 LB

FLAP TRAVEL

A = 0° ± 1° FULL UP
B = 10° ± 2°
C = 25° ± 2°
D= 40° ± 2° FULL DOWN



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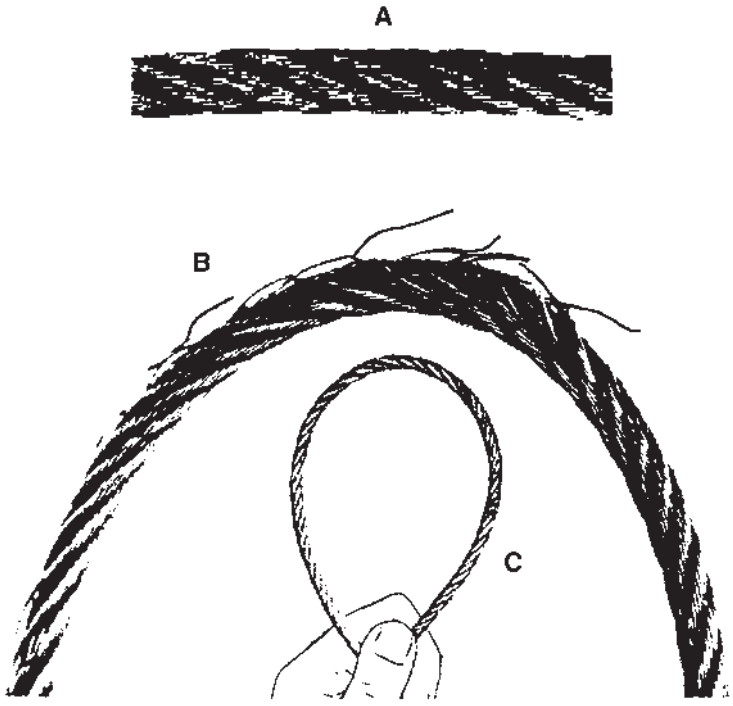
5. Control Cable Inspection

Aircraft control cable systems are subject to a variety of environmental conditions and forms of deterioration that, with time, may be easy to recognize as wire/strand breakage or the not-so-readily visible types of wear, corrosion, and/or distortion. The following data may help in detecting the presence of these conditions:

A. Cable Damage

Critical areas for wire breakage are sections of the cable which pass through fairleads and around pulleys. To inspect each section which passes over a pulley or through a fairlead, remove cable from aircraft to the extent necessary to expose that particular section. Examine cables for broken wires by passing a cloth along length of cable. This will clean the cable for a visual inspection, and detect broken wires, if the cloth snags on cable. When snags are found, closely examine cable to determine full extent of damage.

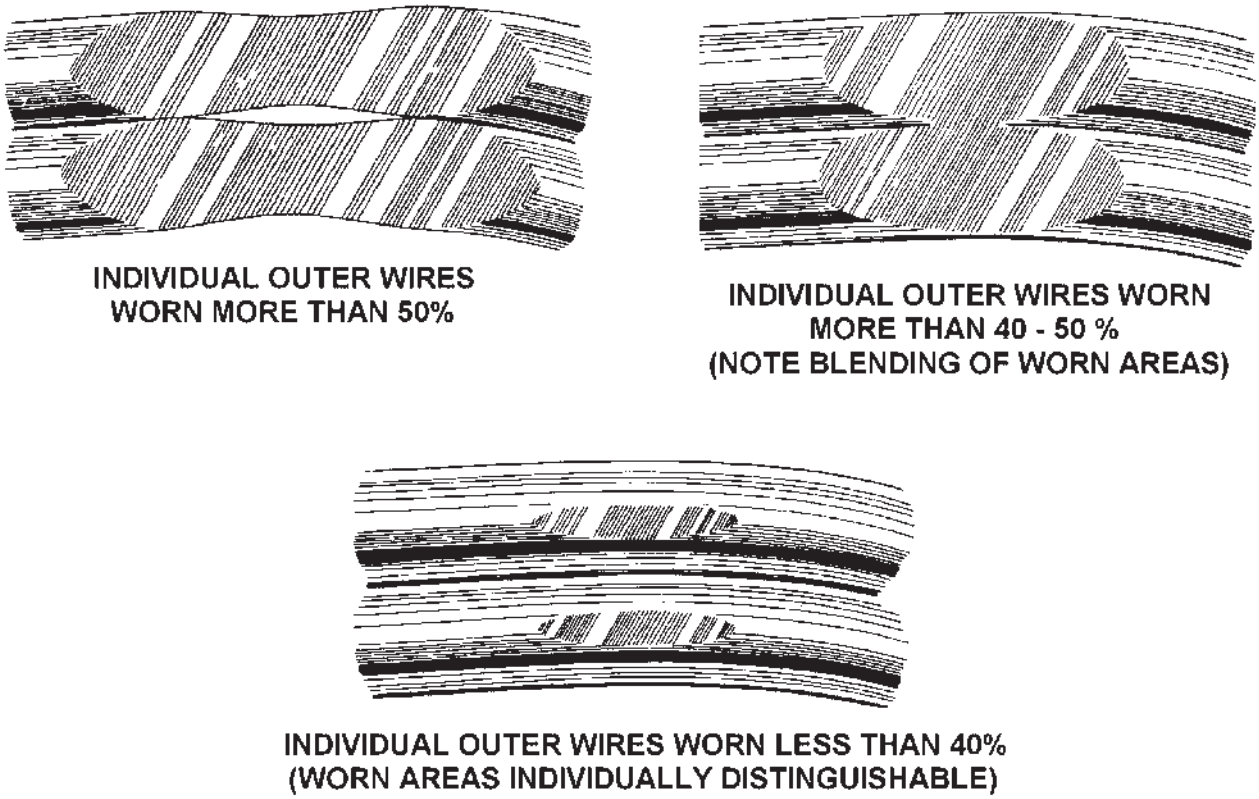
The absence of snags is not positive evidence that broken wires do not exist. “Figure 1” (A) shows a cable with broken wires that were not detected by wiping, but were found during a visual inspection. The damage became readily apparent (“Figure 1” (B)) when the cable was removed and bent using the techniques depicted in “Figure 1” (C).



Control Cable Inspection Technique
Figure 1

B. External Wear Patterns

Wear will normally extend along cable equal to the distance cable moves at that location. Wear may occur on one side of the cable only or on its entire circumference. Replace flexible and non-flexible cables when individual wires in each strand appear to blend together (outer wires worn 40-50 percent) as depicted in “Figure 2”.



Cable Wear Patterns
Figure 2

C. Internal Cable Wear

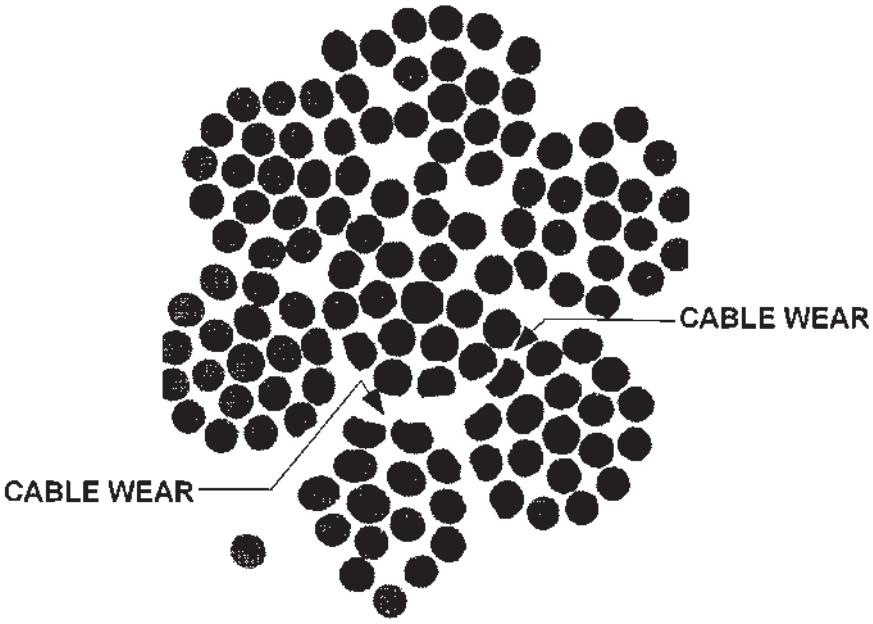
As wear is taking place on the exterior surface of a cable, the same condition is taking place internally, particularly in the sections of the cable which pass over pulleys and quadrants. This condition, shown in “Figure 3”, is not easily detected unless the strands of the cable are separated. Wear of this type is a result of the relative motion between inner wire surfaces. Under certain conditions the rate of this type wear can be greater than that occurring on the surface.

D. Corrosion

Carefully examine any cable for corrosion that has a broken wire in a section not in contact with wear producing airframe components such as pulleys, fairleads, etc. It may be necessary to remove and bend the cable to properly inspect it for internal strand corrosion as this condition is usually not evident on the outer surface of the cable. Replace cable segments if internal strand rust or corrosion is found.

Areas especially conducive to cable corrosion are battery compartments, lavatories, wheel wells, etc., where concentrations of corrosive fumes, vapors, and liquids can accumulate.

NOTE: Check all exposed sections of cable for corrosion after a cleaning and/or metal-brightening operation has been accomplished in that area.



Internal Cable Wear
Figure 3

E. Cable Maintenance

CAUTION: TO AVOID REMOVAL OF CORROSION-PREVENTATIVE COMPOUNDS AND CABLE INTERNAL LUBRICANT, DO NOT USE VAPOR DEGREASING, STEAM CLEANING, METHYLETHYLKETONE (MEK) OR OTHER SOLVENTS.

CAUTION: DO NOT OIL CONTROL CABLES.

Frequent inspections and preservation measures such as rust prevention treatments for bare cable areas will help to extend cable service life. Where cables pass through fairleads, pressure seals, or over pulleys, remove accumulated heavy coatings of corrosion prevention compound. Provide corrosion protection for these cable sections by lubricating as specified in the Lubrication Chart, 12-20-00.

F. Cable Fittings (Terminals, turnbuckles, etc.)

Clean the fittings thoroughly with a suitable solvent prior to inspection.

NOTE: Most control cables feature strands made of galvanized steel. By design, the galvanized coating corrodes sacrificially, protecting the steel strands underneath. This normal corrosion process creates zinc oxide powder, which can migrate along the length of the control cable, covering the surface of the fittings with an inert white speckled coating. This coating, along with any other contaminants such as dirt, oil or grease, must be removed in order to properly examine the fittings.

(1) 100 Hour Standard Inspection

Check swaged terminal reference marks for any indication of cable slippage within fitting. Inspect fitting assembly for distortion and/or broken strands at the terminal. Check that all bearings and swivel fittings (bolted or pinned) pivot freely to prevent binding and subsequent failure. Check turnbuckles for proper thread exposure and broken or missing safety wires/clips.

Pay particular attention to corrosion and “pitting” on cable terminals, turnbuckles and cable fittings. Any corrosion or pitting found requires replacement of the corroded fitting and/or cable.

(2) 100 Hour Special Inspection

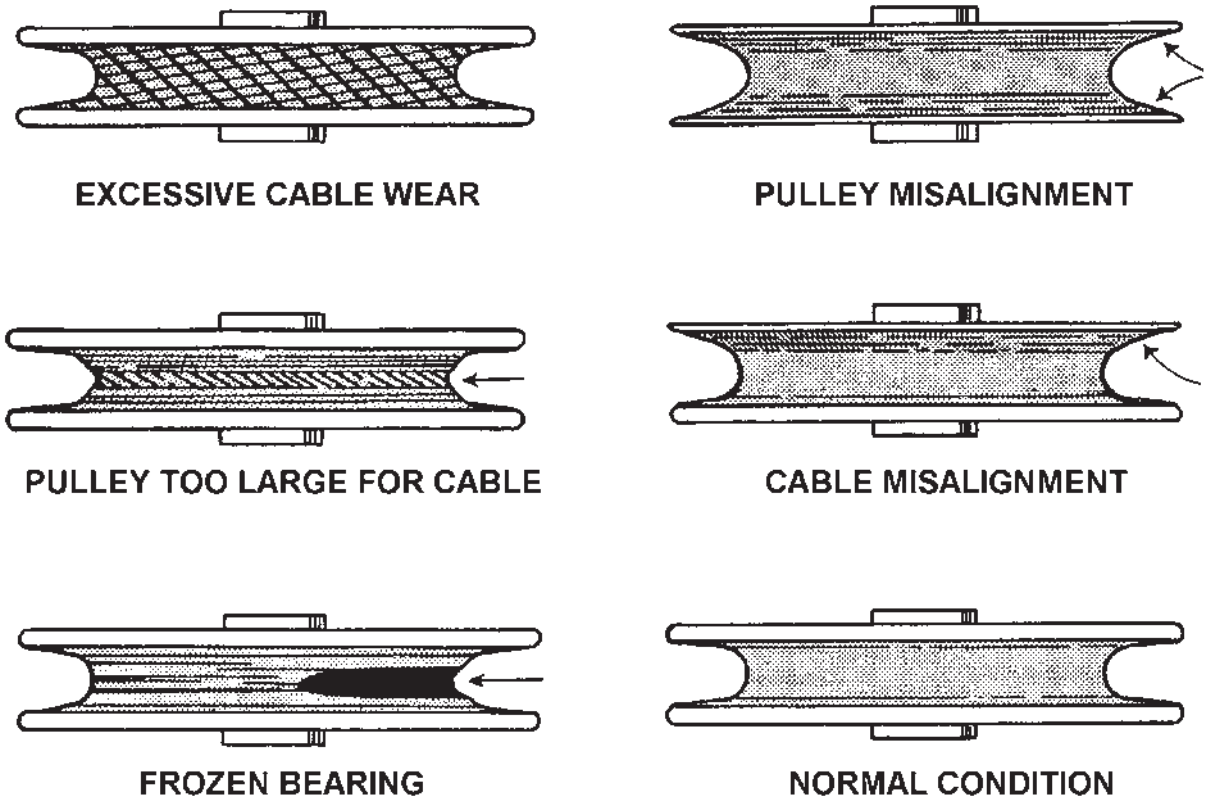
For airplanes 15 years old or older, using a 10X magnifier, visually inspect the entire surface of each cable terminal, turnbuckle, or other cable fitting for corrosion or cracking. Inspect under safety wire or clips wrapped around the cable or fitting. Any evidence of corrosion or cracking, however minute, is cause for replacement. A logbook entry documenting the replacement of a cable terminal, turnbuckle, or other cable fitting relieves the inspection requirement for that fitting only, until such time as that fitting has been in service for 15 years.

G. Pulleys

Inspect pulleys for roughness, sharp edges, and presence of foreign material embedded in the grooves. Examine pulley bearings to assure proper lubrication, smooth rotation, freedom from flat spots, dirt, and paint spray. Periodically rotate pulleys, which turn through a small arc, to provide a new bearing surface for the cable. Maintain pulley alignment to prevent the cable from riding on flanges and chafing against guards, covers, or adjacent structure. Check all pulley brackets and guards for damage, alignment, and security.

Pulley Wear Patterns

Various cable system malfunctions may be detected by analyzing pulley conditions. These include such discrepancies as too much tension, misalignment, pulley bearing problems, and size mismatches between cables and pulleys. Examples of these conditions are shown in “Figure 4” on page 270011.



Pulley Wear Patterns
Figure 4

H. Stabilator Control System Special Inspection

NOTE: This inspection incorporates the requirements of Piper Service Bulletin No. 1245A.

Beginning at the first regularly scheduled maintenance event after reaching fifteen (15) calendar years time-in-service and each two thousand (2,000) hours or seven (7) calendar years time-in-service thereafter, whichever occurs first, perform the following inspection.

NOTE: Documented replacement of a control cable or cable fitting (i.e., turnbuckle, terminal, etc.) relieves this inspection requirement for that component only until it reaches fifteen (15) calendar years time-in-service.

(1) Secure a jack stand to the tail skid (tie down).

(2) Locate the rear cabin bulkhead, which is behind the aft seats. The rear cabin bulkhead is accessible from inside the aircraft.

NOTE: Remove seats and/or other interior components as desired to ease accomplishment of this inspection.

(3) Remove and retain the access cover from the rear cabin bulkhead, in order to gain access to the tailcone area.

(4) Visually inspect the entire length of the stabilator flight control cable system (See “Figure 3” on page 27305), with special emphasis on the turnbuckle area, as described in the steps that follow.

(5) Identify the two (2) stabilator flight control cable turnbuckles (“Figure 3” on page 27305, Item 17) which are located inside the tailcone, directly aft of the rear cabin bulkhead.

(6) Disassemble the turnbuckle bodies to facilitate a complete inspection of the turnbuckle bodies and associated cable terminals.

NOTE: See Standard Procedures, above; and, Removal under Stabilator Controls in 27-30-00.

(7) Clean the turnbuckle bodies and associated cable terminals thoroughly per the instructions provided in Cable Fittings, above.

(8) Using a 10X magnifier, a mirror and a suitable light source, carefully examine the entire surface of each turnbuckle, cable terminal, and adjacent portion of the flight control cable, inspecting for cracks, corrosion, or broken cable strands. Any evidence of cracks or cable fraying, however minute, is cause for replacement. Any evidence of corrosion that remains after accomplishing the cleaning instructions above is cause for replacement.

(9) Reassemble the turnbuckle bodies and associated cable terminals. Rig and adjust stabilator travel and stabilator control cable tension.

NOTE: See Standard Procedures; and, Installation and Rigging and Ajustment under Stabilator Controls in 27-30-00.

(10) Reinstall the rear cabin bulkhead access cover.

NOTE: Reinstall seats and/or other interior components that were removed in Step (2).

(11) Remove the jack stand from the aircraft.

(12) Make a logbook entry indicating compliance with this inspection and documenting any components that were replaced.

AILERON CONTROLS

1. Troubleshooting

“Chart 1” lists troubles peculiar to aileron flight controls along with their probable causes and suggested remedies. When troubleshooting the aileron flight controls, additional reference may be obtained from Chapter 57 on control surface balancing, if required. After the trouble has been corrected, check the entire aileron flight control system for security and operation.

CHART 1 (Sheet 1 of 2)
TROUBLESHOOTING AILERON CONTROL SYSTEM

Trouble	Cause	Remedy
Lost motion between control wheel and aileron.	Cable tension too low.	Adjust cable tension.
	Linkage loose or worn.	Check linkage and tighten or replace.
	Broken pulley.	Replace pulley.
	Cables not in place on pulleys.	Install cables correctly. Check cable guards.
Resistance to control wheel rotation.	System not lubricated properly.	Lubricate system.
	Cable tension too high.	Adjust cable tension.
	Control column horizontal cable improperly adjusted.	Adjust cable tension.
	Pulleys binding or rubbing.	Replace binding pulleys and/or provide clearance between pulleys and brackets.
	Cables not in place on pulleys.	Install cables correctly. Check cable guards.
	Bent aileron and/or hinge.	Repair or replace aileron and/or hinge.
	Cables crossed or routed incorrectly.	Check routing of control cables.
Control wheels not synchronized.	Incorrect control column rigging.	Re-rig control column.
Control wheels not horizontal when ailerons are neutral.	Incorrect rigging of aileron system.	Re-rig aileron system.
Incorrect aileron travel.	Aileron control rods not adjusted properly.	Adjust control rods.
	Aileron bellcrank stops not adjusted properly.	Adjust bellcrank stops.

CHART 1 (Sheet 2 of 2)
TROUBLESHOOTING AILERON CONTROL SYSTEM

Trouble	Cause	Remedy
Correct aileron travel cannot be obtained by adjusting stops.	Incorrect rigging of aileron cables, and/or control wheel.	Re-rig controls.
Control wheel stops before control surfaces reach full travel.	Incorrect rigging between control wheel and control cables.	Re-rig controls.

2. Control Column Assembly

A. Removal

See “Figure 1” on page 27103.

(1) To remove either control wheel with tube, the following procedure may be used:

(a) Separate the control wheel tube from the flexible joint that is located on either side of the tee bar assembly by removing the nut, washer and bolt. Pull the tube from the flexible joint.

(b) If removing the left control tube, slide the stop from the tube.

(c) Should wires for the various Autopilot systems be installed in the control tube, disconnect them at the quick disconnect terminals behind the instrument panel. Draw the wires back into the tube and back out through the forward end of the tube.

(d) Remove the control wheel assembly from the instrument panel.

(2) The tee bar with assembled parts may be removed from the airplane by the following procedure:

(a) Remove the access panel to the aft section of the fuselage.

(b) Relieve cable tension from the stabilator control cables at one of the stabilator cable turnbuckles in the aft section of the fuselage.

(c) Relieve tension from the aileron control cables and chains at the turnbuckle that connects the chains at the top of the tee bar.

(d) Disconnect the control chains from the control cables where the chains and cables join by removing the cotter pins, nuts, bolts and bushings.

(e) If the control wheel assemblies have not been previously disconnected from the tee bar assembly, separate the control wheel tubes at the flexible joints by removing the nuts, washers and bolts.

(f) Disconnect the bobweight actuating rod at the tee bar.

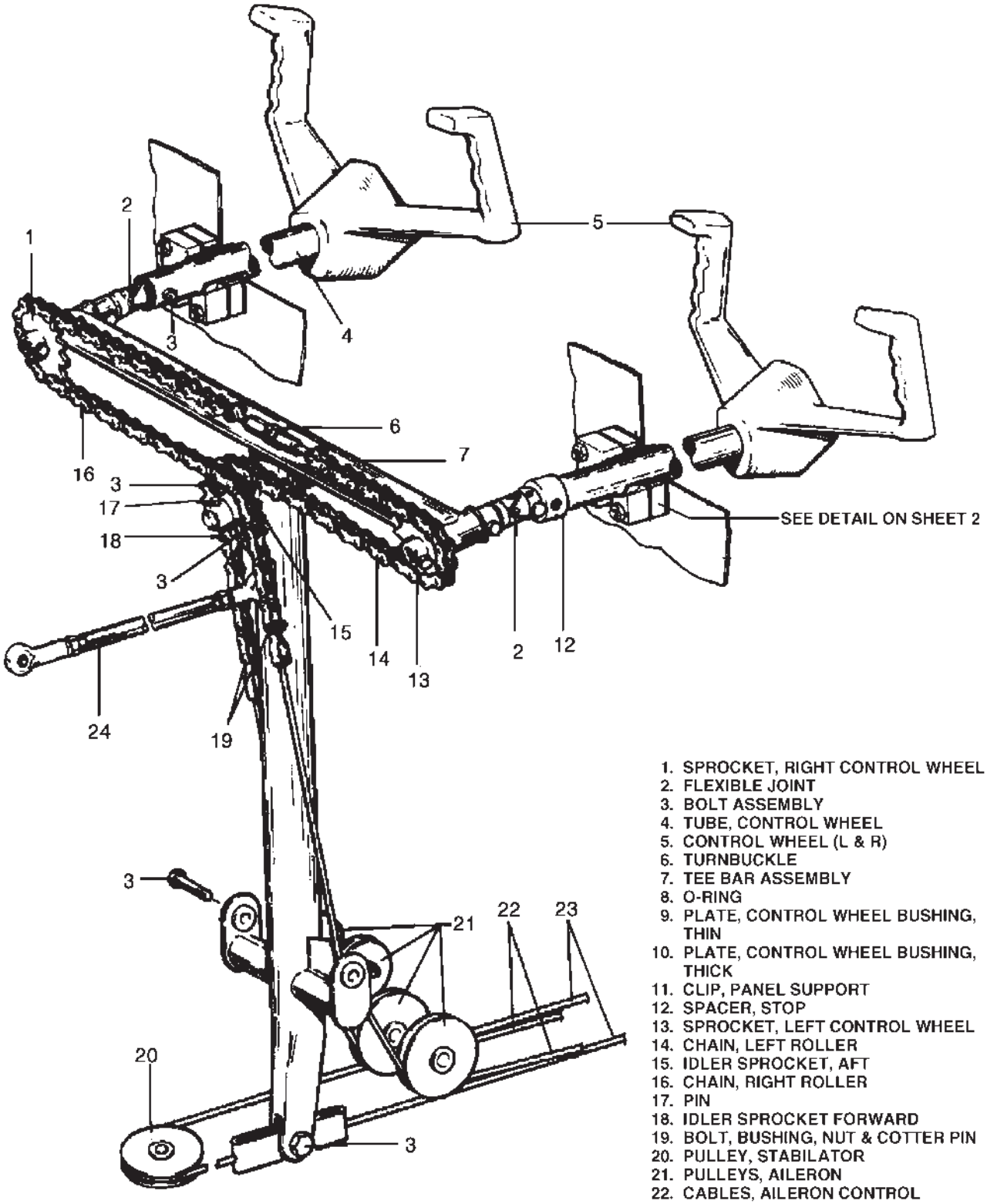
(g) Remove the tunnel plate just aft of the tee bar by laying back enough tunnel carpet to remove the plate attachment screws.

(h) Remove the two aileron control cable pulleys attached to the lower section of the tee bar by removing the pulley attachment bolt.

(i) Disconnect the stabilator control cables from the lower end of the tee bar assembly.

(j) Disconnect the necessary control cables, such as the propeller pitch control, mixture control, etc., that will allow the tee bar assembly to be removed.

(k) Remove the tee bar assembly by removing the attachment bolts with washers and nuts (which are through each side of the floor tunnel) and lifting it up and out through the right side of the cabin.

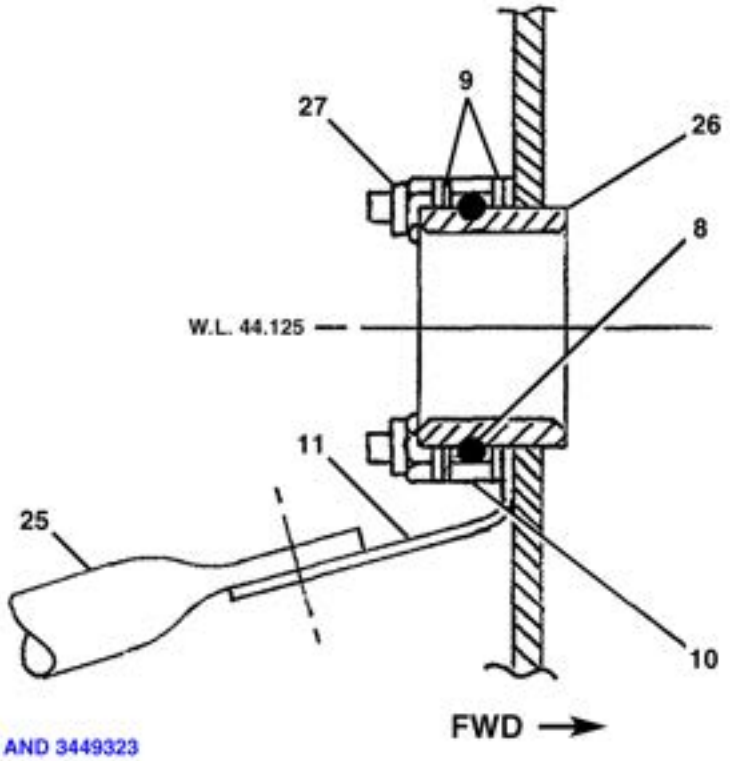


1. SPROCKET, RIGHT CONTROL WHEEL
2. FLEXIBLE JOINT
3. BOLT ASSEMBLY
4. TUBE, CONTROL WHEEL
5. CONTROL WHEEL (L & R)
6. TURNBUCKLE
7. TEE BAR ASSEMBLY
8. O-RING
9. PLATE, CONTROL WHEEL BUSHING, THIN
10. PLATE, CONTROL WHEEL BUSHING, THICK
11. CLIP, PANEL SUPPORT
12. SPACER, STOP
13. SPROCKET, LEFT CONTROL WHEEL
14. CHAIN, LEFT ROLLER
15. IDLER SPROCKET, AFT
16. CHAIN, RIGHT ROLLER
17. PIN
18. IDLER SPROCKET FORWARD
19. BOLT, BUSHING, NUT & COTTER PIN
20. PULLEY, STABILATOR
21. PULLEYS, AILERON
22. CABLES, AILERON CONTROL
23. CABLES, STABILATOR CONTROL
24. BOBWEIGHT ACTUATING ROD

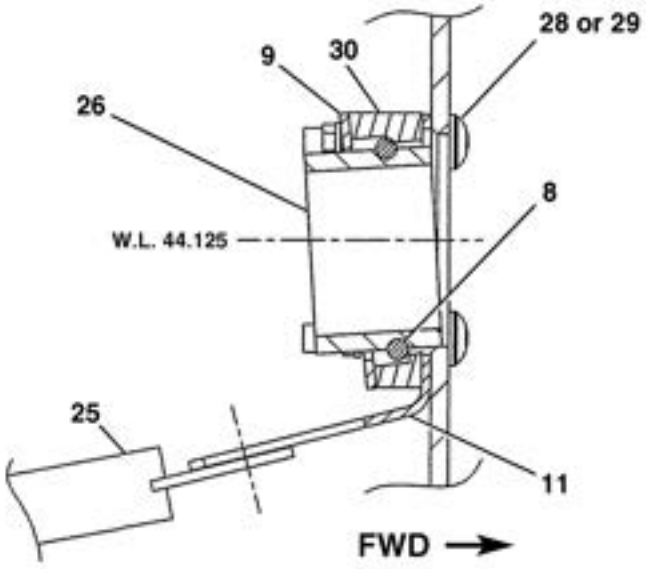
Control Column Installation
Figure 1 (Sheet 1 of 2)

100850 BR

25. SUPPORT
26. BUSHING
27. SCREW, WASHER, AND NUT (FOUR PLACES EACH SIDE)
28. SCREW AND NUT (FOUR PLACES EACH SIDE)
29. SCREW AND HY-LOK COLLAR (FOUR PLACES EACH SIDE)
30. GUIDE



S/N's 3449001 THRU 3449325, LESS 3449311 AND 3449323



S/N's 3449311, 3449323, 3449326 AND UP

Control Column Installation
Figure 1 (Sheet 2 of 2)

B. Installation

See “Figure 1” on page 27103.

(1) The tee bar assembly may be installed in the airplane by the following procedure:

(a) Swing the tee bar assembly into place from the right side of the cabin and secure with attachment bolts, washers, and nuts inserted in through each side of the floor tunnel.

(b) Connect the bobweight actuating rod to the tee bar. (See “Figure 2” on page 27106 for proper rigging of bobweight.)

(c) Connect the stabilator control cables to the lower end of the tee bar with bolt, washer, nut, and cotter pin. The cable ends must be free to rotate.

(d) Place the aileron control cables around the pulleys that attach to the lower section of the tee bar: position pulleys and secure with bolt, washers and nut.

(e) Install the control wheel:

(f) Place the control wheels in neutral (centered) position and install the aileron control chains on the control wheel sprockets and idler crossover sprockets. The turnbuckle must be centered between the two control wheel sprockets.

(g) Loosen the connecting bolts of the idler sprockets to allow the chain to fit snug around the control wheel sprockets and over the idler sprockets.

(h) Connect the aileron control cables to the ends of the chains with bolts, bushings, nuts and cotter pins.

(i) Adjust the chain turnbuckle between the two control wheel sprockets to allow the control wheels to be neutral and obtain proper cable tension as given in “Chart 1” on page 27002. It may be necessary in order to have both control wheels neutral to set the chain turnbuckle to neutralize the wheels and then set cable tension with the turnbuckles located under the floor panel aft of the main spar as instructed in “Rigging and Adjustment” on page 271020. Before safetying the turnbuckle, check that when the ailerons are neutral, the control wheels will be neutral and the chain turnbuckle centered. Also, the aileron bellcranks should contact their stops before the control wheel hits its stop. Models which have adjustable aileron tee bar stops must maintain .030 to .040 clearance between the sprocket pin and the adjustable stop bolts after the bellcranks contact their stops.

(j) Set stabilator cable tension with the turnbuckles in the aft section of the fuselage as described in the appropriate section of this chapter. Check safety of all turnbuckles upon completion of adjustments.

(k) Tighten the connecting bolts of the idler sprockets. (Torque 45 ± 5 in-lb.)

(l) Install the floor tunnel plate and secure with screws. Fasten the tunnel carpet in place.

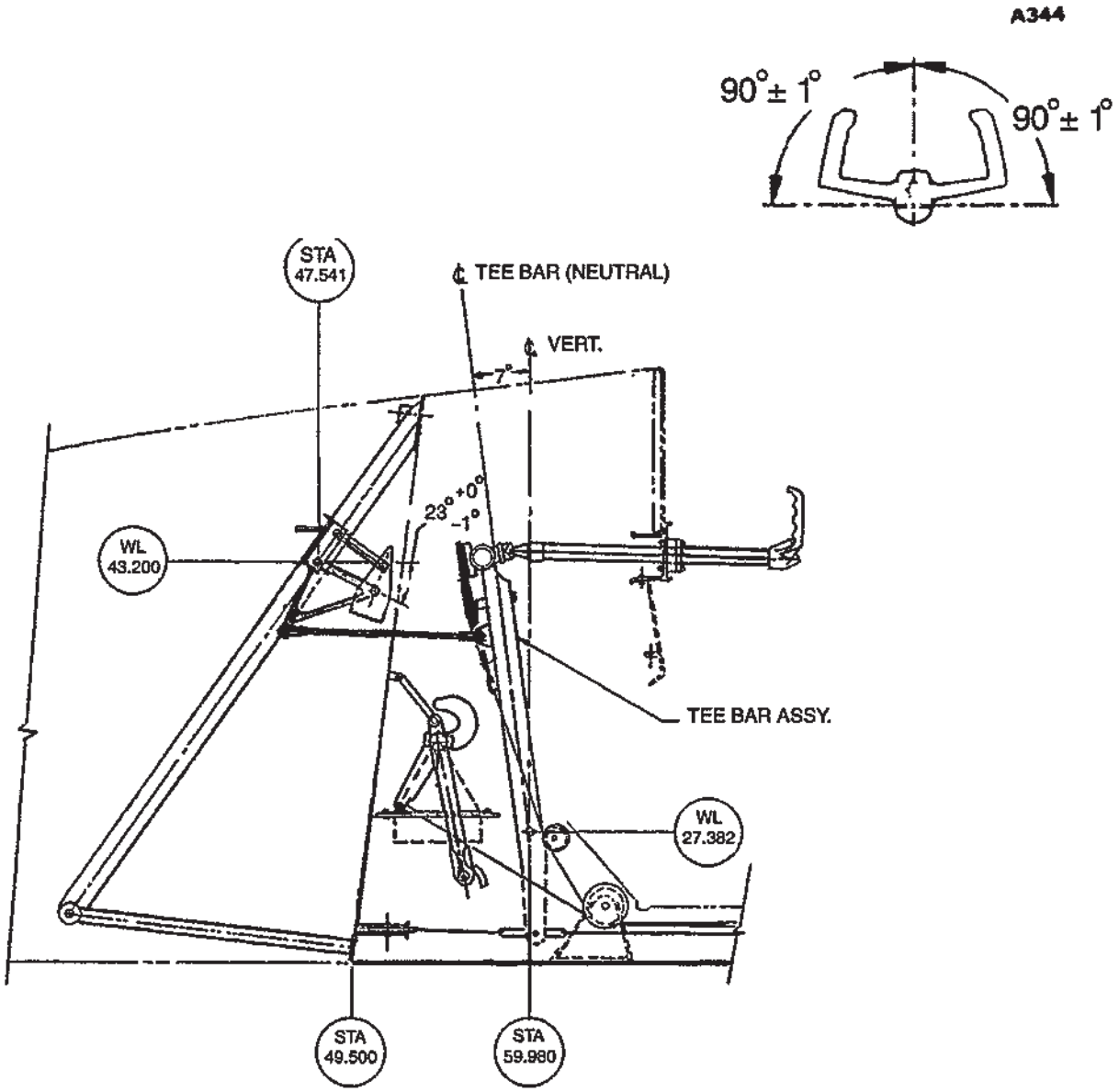
(2) Install either control wheel assembly as follows:

(a) Insert the control wheel tube through the instrument panel.

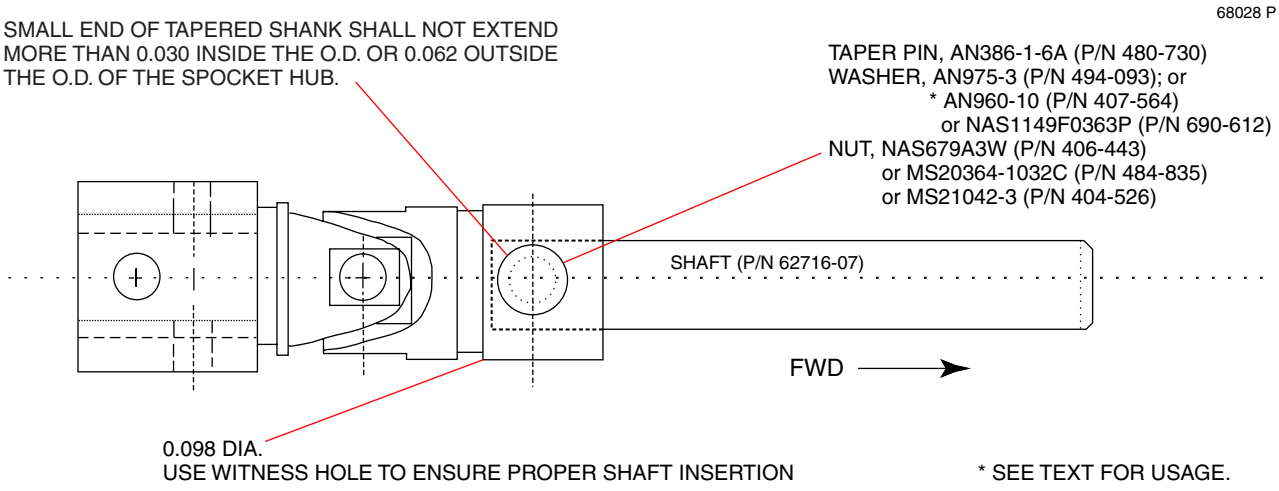
(b) Should wires for the various autopilot systems need to be installed in the control tube, route them through the hole in the forward side of the tube and out of the small hole in the side. Position the rubber grommet in the hole in the side of the tube.

(c) On the left control tube, install the stop.

(d) Connect the control wheel tube to the flexible joint of the tee bar assembly. If the control cables and or chains have not been removed or loosened, place the ailerons in neutral and install the control tube on the flexible joint to allow the control wheel to be neutral. Install bolt, washer and nut and tighten.



Control Column Rigging
Figure 2



Flex (Universal) Joint Assembly
Figure 3

(3) To Install Flex Joint Replacement

See “Figure 1” on page 27103 and “Figure 3”.

(a) Carefully lay out location for hole to be drilled in flex joint tube to match hole in control column shaft.

(b) Using a #5 (0.2055) drill bit, drill hole through flex joint tube at location determined above.

(c) Ream drilled hole, in steps, with a #1 reamer, checking to ensure proper depth for taper pin and sufficient pin thread protrusion for proper installation.

NOTE: Reamer may be purchased from your Piper Dealer as P/N 906-713.

(d) Install pin through tube and shaft.

1) If pin shoulder does not protrude past sprocket hub, install a AN960-10 washer.

2) If pin shoulder protrudes past sprocket hub, install a AN975-3 washer.

(e) Install nut. Torque 35–40 inch-pounds.

C. Control Wheel (S/N's 3449042 and up.)

See "Figure 4" on page 27109.

The retainer clip (P/N 104687-002) and MS24693-S40 screws (2 ea.) are factory installed in S/N's 3449302 and up. S/N's 3449042 thru 3449301 must individually procure and retrofit the retainer clip and screws.

(1) Removal

(a) Remove two (2) screws and retainer clip.

(b) Remove control wheel attachment screw and spacer.

(c) Slide control wheel off of tube.

(2) Installation

(a) Degrease aft end of tube and inside of control wheel using acetone or naphtha. Allow to dry.

(b) Degrease control wheel attachment screw, the spacer, and the nutplate (inside the tube) using acetone or naphtha. Allow to dry.

(c) Prime inside of control wheel with Loctite 7649 (Piper P/N 279-073). Allow to dry.

(d) Install control wheel onto tube using Loctite 27121 (Piper P/N 279-128). Take care to ensure screw hole in control wheel aligns with screw hole in tube.

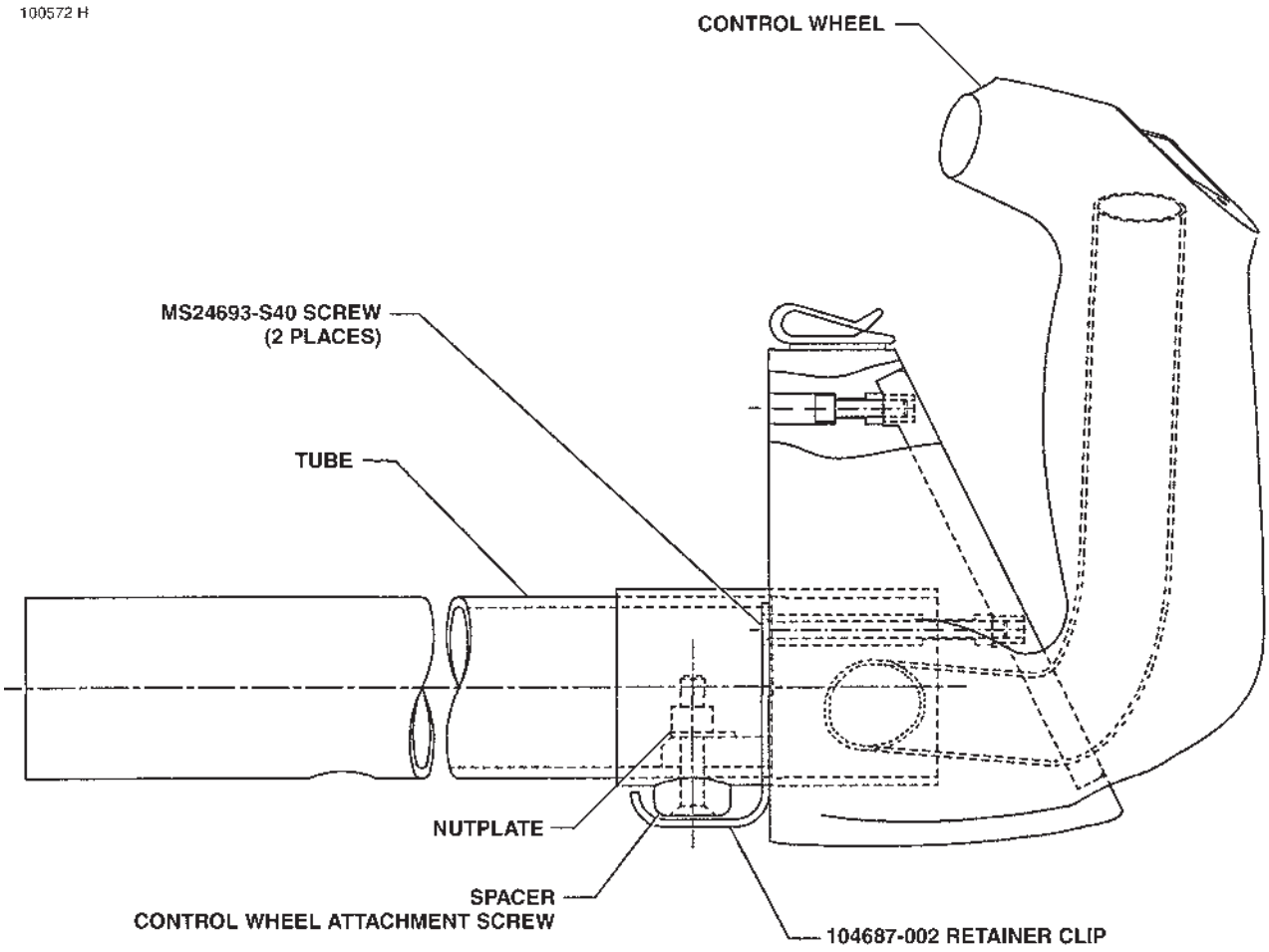
NOTE: This step must be accomplished promptly due to short cure time.

(e) Prime threads of control wheel attachment screw with Loctite 7649 (Piper P/N 279-073). Allow to dry.

(f) Install control wheel attachment screw (with spacer) into nutplate using Loctite 27121 (Piper P/N 279-128).

NOTE: This step must be accomplished promptly due to short cure time.

(g) Position retainer clip to capture spacer and control wheel attachment screw. Secure with screws (2), being careful not to overtighten the screws and damage the logo medallion.





Control Wheel Installation
Figure 4

Effectivity
3449042 & Up

3. Tee Bar Assembly Inspection

WARNING: ANY CRACKED COMPONENT MUST BE REPLACED PRIOR TO THE NEXT FLIGHT.

In airplanes with 3,500 or more hours time in service (TIS), conduct the following inspection each 100 hours TIS.

NOTE: The following procedures incorporate the requirements of Piper Service Bulletin No. 1355B.

NOTE: Installation of a replacement tee bar weldment assembly as shown in Figure 5, Detail B, relieves this recurring inspection requirement.

NOTE: Before proceeding, wipe surfaces clean using a suitable cleaning agent.

NOTE: For this procedure, use a 10X magnifier, a mirror and a suitable light source or other equipment capable of providing equal or better resolution.

A. Confirm the existing tee bar configuration. Compare the installed assembly to “Figure 5” on page 271011.

- If the local area conforms to Figure 5, Detail A, then inspections are required. Proceed to step (2).
- If the local area conforms to Figure 5, Detail B, then the installed tee bar assembly includes the required features (i.e., the weldment has a gusset as shown in Detail B). No recurring inspections are required. Proceed to step (4).

B. Examine the area shown in Detail A for cracks, with special emphasis on welded joints.

NOTE: For inspection and paint guidance, refer to FAA Advisory Circular AC 43.13-1B. This free document is available for download from the FAA at https://www.faa.gov/.

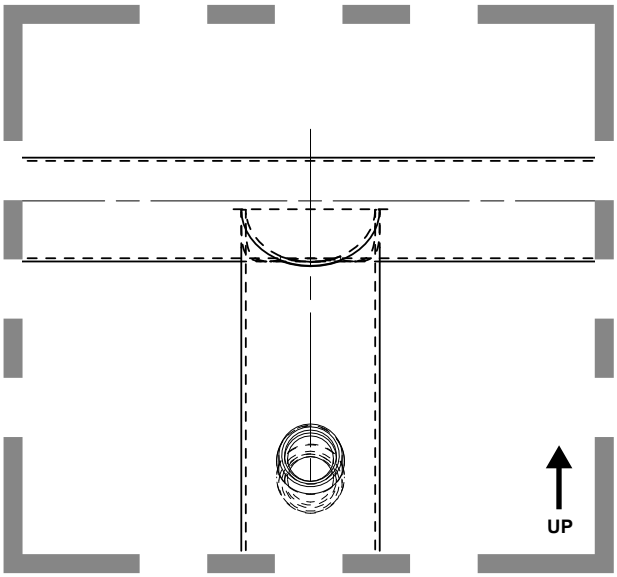
(1) If the paint is cracked, chipped or peeled in the vicinity of the inspection areas, or if there is evidence of corrosion (such as staining or blistering), then a fluorescent penetrant inspection is required. Prepare the surface by removing the paint using Cee-Bee A-202 paint stripper or an equivalent. Wipe off any residue, in the areas where the paint was removed, with isopropyl alcohol.

NOTE: Paint must be removed using chemical processes only. The use of abrasives or other mechanical methods to remove the paint will hide the existence of any discontinuities, making an accurate inspection impossible.

NOTE: Follow the paint stripper manufacturer’s instructions.

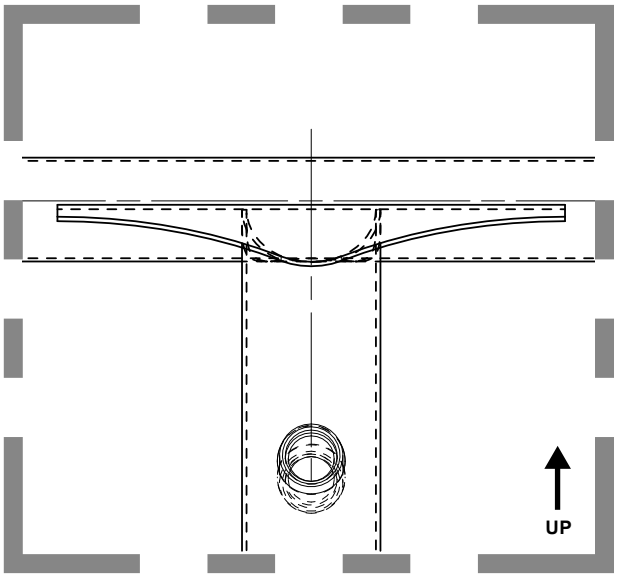
(2) Perform a fluorescent dye penetrant inspection per FAA Advisory Circular AC 43.13-1B, Chapter 5, Section 5.

- If any cracks are discovered, then the tee bar weldment must be replaced prior to the next flight. The appropriate replacement tee bar is specified in the PA-34-220T Parts Catalog, P/N 761-887. See step (3).
- If no cracks are discovered, apply primer and topcoat to any bare metal. Refer to FAA Advisory Circular AC 43.13-1B, Chapter 6, for guidance. Then, proceed to step (4).



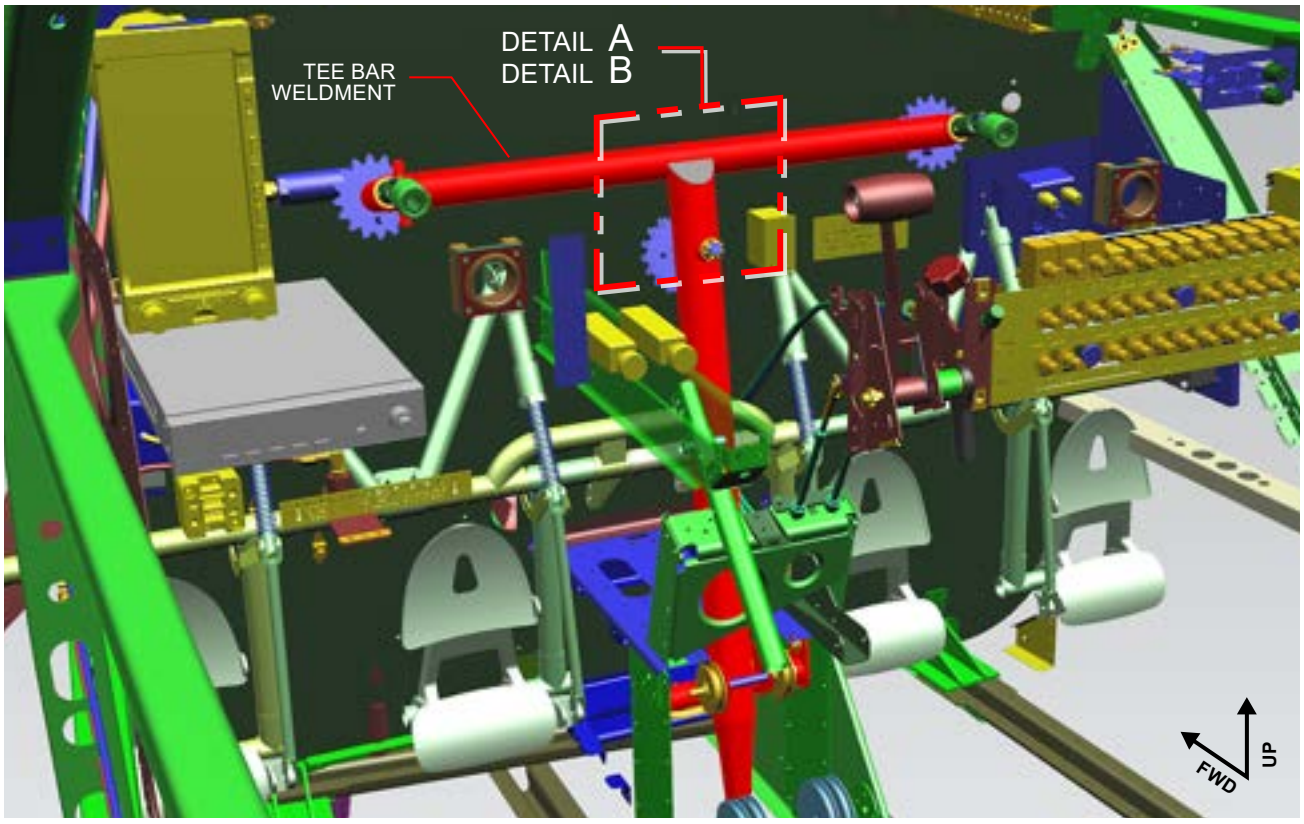
DETAIL A

LOOKING FORWARD AT
CONTROL COLUMN TEE BAR WELDMENT
WITHOUT GUSSET



DETAIL B

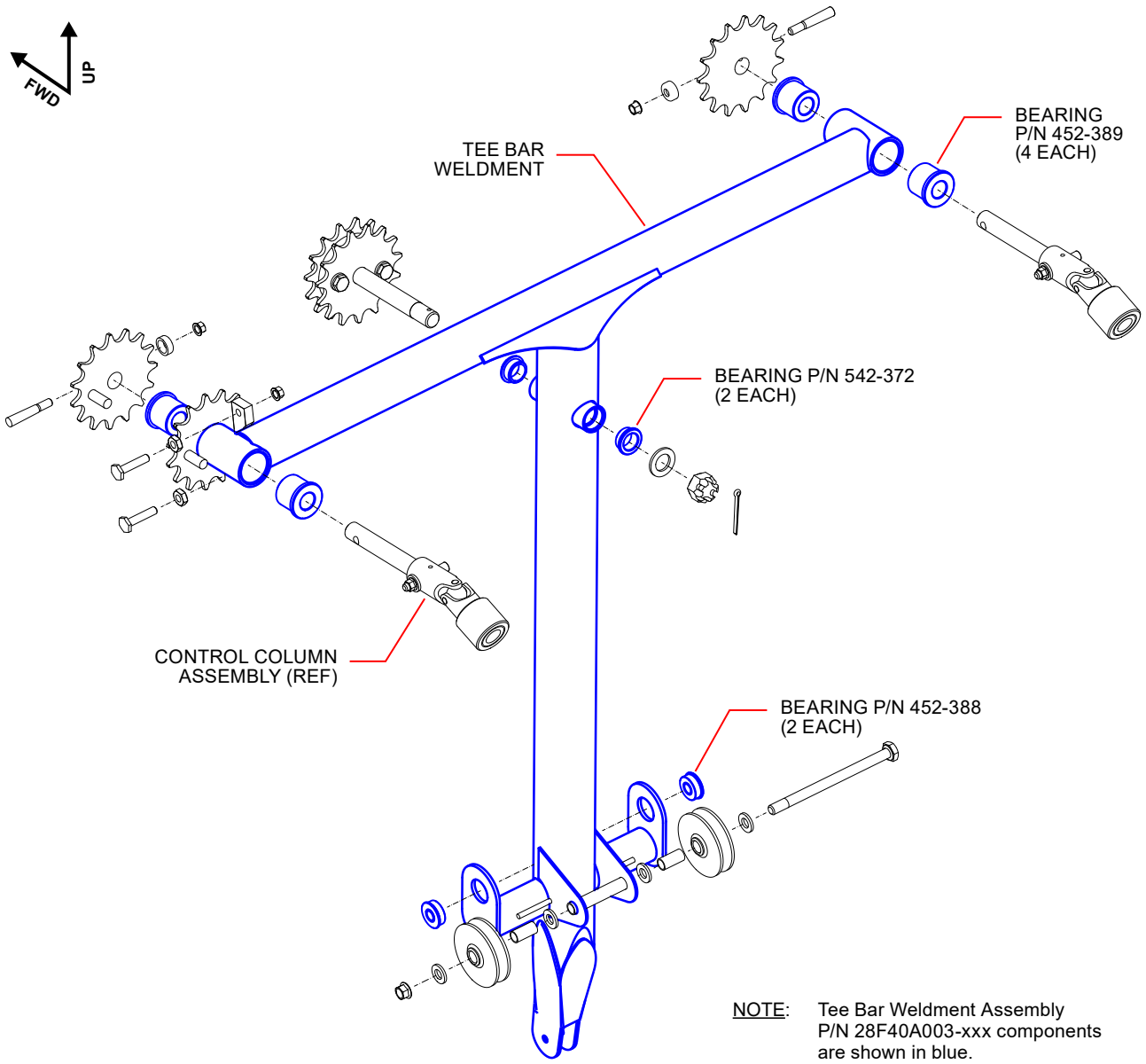
LOOKING FORWARD AT
CONTROL COLUMN TEE BAR WELDMENT
WITH GUSSET



TEE BAR ASSEMBLY

LOOKING FORWARD
SOME COMPONENTS REMOVED FOR CLARITY

Tee Bar Assembly – Weldment
Figure 5

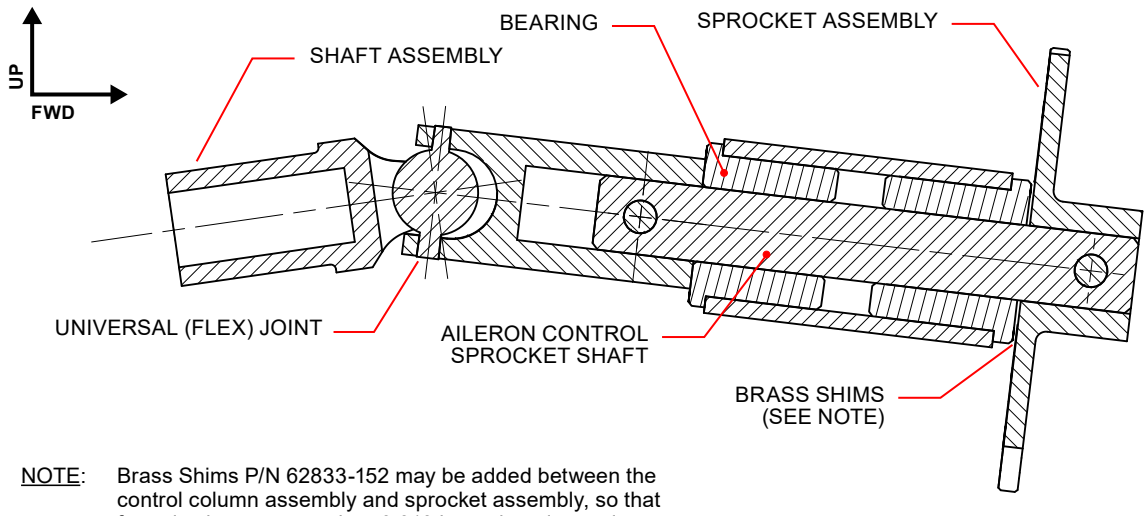


TEE BAR ASSEMBLY

P/N 28F40A003-001 SHOWN (OTHERS ARE SIMILAR)

EXPLODED VIEW (TYPICAL)
SOME COMPONENTS REMOVED FOR CLARITY

Tee Bar Assembly – Weldment Assembly Installation
Figure 6



NOTE: Brass Shims P/N 62833-152 may be added between the control column assembly and sprocket assembly, so that free play is no greater than 0.010 in. and no drag exists.

CONTROL COLUMN ASSEMBLY – TYPICAL

LOOKING OUTBOARD
PILOT SIDE SHOWN (COPILOT SIDE SIMILAR)

Brass Shim P/N 62833-152 Installation Location
Figure 7

C. If replacement of the weldment assembly is required, complete these steps:

(1) Acquire the applicable replacement tee bar weldment assembly, specified in the PA-34-220T Parts Catalog, P/N 761-887.

(2) To replace the tee bar weldment assembly, first remove the control wheels and the control column tee bar assembly from the airplane. See Control Column Assembly, “Removal” on page 27102.

(3) Remove all sprockets, the control wheel columns, and hardware (bolts, nuts, washers) from the tee bar weldment assembly and retain them for reinstallation. Replace any hardware on condition. (See “Figure 6”.)

(4) Assemble the control wheel tee bar assembly: Install the sprockets, control wheel columns, and hardware, removed from the original weldment assembly, onto the replacement tee bar weldment assembly. (See “Figure 6”.)

(5) Reinstall the control column tee bar assembly and control wheel columns into the airplane. See Control Column Assembly, “Installation” on page 27105. If necessary, so that there is no greater than 0.010 in. of free play (relative motion) and no drag exists between the control columns and sprocket assemblies, add Brass Shims, Piper P/N 62833-152, no more than two (2) per side, as shown in Figure 7.

(6) Reinstall the control wheels. See Control Column Assembly, “Installation” on page 27105. If applicable, see also “Control Wheel (S/N’s 3449042 and up.)” on page 27108.

(7) Perform a functional check on any systems that were disturbed during this inspection.

(8) Make a logbook entry documenting compliance with this inspection. Or, if applicable, make a logbook entry documenting the tee bar weldment assembly replacement.

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4. Aileron Control Cables

A. Removal

See “Figure 8” on page 271018.

For the removal of any of the control cables in the fuselage or wings, first remove the floor panel that is located directly aft of the main spar by removing the center seats, seat belt attachments, and the screws securing the panel. Lift the panel and remove from airplane.

(1) To remove either the right or left primary control cables that are located in the fuselage, the following procedure may be used:

(a) Remove the tunnel plate just aft of the tee bar by laying back enough tunnel carpet to remove the plate attachment screws.

(b) Separate the primary control cable at the turnbuckles located in the floor opening aft of the main spar.

(c) Remove the cable pulleys attached to the lower section of the control column tee bar assembly by removing the pulley attachment bolt.

(d) Move the cable guard (See “Figure 8” on page 271018, Sketch B) located under the pulley cluster by removing the cotter pin from the exposed end of the guard and sliding it to the left or right as required.

(e) Remove the cotter pins used as cable guards at the pulley in the forward area of the floor opening aft of the main spar.

(f) Disconnect the cable from the control chain at the control column tee bar assembly by removing the cotter pin, nut, bolt, and bushing that connect the two together. Secure the chains in some manner to prevent them from unwrapping from around the sprockets.

(g) Draw the cable back through the floor tunnel.

(2) The primary control cable in either wing may be removed by the following procedure:

(a) Remove the access plate to the aileron bellcrank located on the underside of the wing forward of the aileron center hinge.

(b) If not previously disconnected, separate the cable at the turnbuckles located in the floor opening aft of the main spar.

(c) Disconnect the cable from the forward end of the aileron bellcrank by removing the cotter pin, nut, washer and bolt.

(d) Draw the cable from the wing.

(3) Either balance cable may be removed by the following procedure:

(a) Separate the balance cable at the turnbuckle in the right side of the floor opening aft of the main spar.

(b) If the left balance cable is to be removed, remove the cotter pin used as a cable guard at the pulley (10) in the center of the floor opening.

(c) Remove the access plate to the aileron bellcrank located on the underside of the wing forward of the aileron center hinge.

(d) Disconnect the cable from the aft end of the aileron bellcrank by removing the cotter pin, nut, washer and bolt.

(e) Draw the cable from the wing.

B. Installation

See “Figure 8” on page 271018.

(1) The installation of either the right or left primary control cable, located in the fuselage, may be accomplished as follows:
   (a) Draw the cable through the fuselage floor tunnel.
   (b) Connect the cable to the end of the control chain and secure using bushing, bolt, nut and cotter pin.
   (c) Place the cable around the pulley that is located in the tunnel. Install cable guard and secure with cotter pin. (See “Figure 8” on page 271018, Sketch B)
   (d) Position cables and install the cable pulleys that attach to the lower section of the tee bar assembly. Secure with bolt, washer and nut. (See “Figure 8” on page 271018.)
   (e) Place the cable around the pulley that is located in the floor opening just aft of the main spar and install cotter pin cable guards.
   (f) If the primary control cable in the wing is installed, connect the control cable ends at the turnbuckle (6) located in the floor opening aft of the main spar.
   (g) Check rigging and adjustment of aileron controls. Per “Rigging and Adjustment” on page 271020 below.
   (h) Position the heat duct and secure with screws.
   (i) Install the tunnel plate aft of tee bar assembly and secure with screws.
   (j) Put the floor carpet in place and secure.
   (k) Place the fuel selector lever on the selector torque tube and secure with pin and cotter pin.
   (l) Install the lower and upper selector covers and secure with screws.

(2) The primary control cable in either wing may be installed by the following procedure:
   (a) Draw the control cable into the wing.
   (b) Connect the cable to the forward end of the aileron bellcrank using a bolt, washer, nut and cotter pin. Allow the cable end to rotate freely on the bellcrank.
   (c) If the primary control cable in the fuselage is installed, connect the ends at the turnbuckle located in the floor opening aft of the main spar.
   (d) Check rigging and adjustment of aileron controls.
   (e) Install the access plate on the underside of the wing.

(3) Either balance cable may be installed by the following procedure:
   (a) Draw the cable into the wing.
   (b) Connect the cable to the aft end of the aileron bellcrank using a bolt, washer, nut and cotter pin. Allow the cable end to rotate freely on the bellcrank.
   (c) Connect the balance cable ends at the turnbuckle in the floor opening aft of the main spar.
   (d) If the left cable was removed, install the cotter pin cable guard at the pulley located in the center of the floor opening.
   (e) Check rigging and adjustment.
   (f) Install the access plate on the underside of the wing.
   (g) Install the floor panel, seat belt attachments and seats.

5. Aileron Bellcrank Assembly

A. Removal

See “Figure 8” on page 271018.

(1) Remove the floor panel located directly aft of the main spar by removing the center seats, seat belt attachments, and the screws securing the floor panel. Lift the panel and remove from the airplane.

(2) Remove the access plate to the aileron bellcrank located on the underside of the wing, forward of the aileron center hinge.

(3) Relieve tension from the aileron control cables by loosening the balance cable turnbuckle located in the floor opening aft of the main spar.

(4) Disconnect the primary and balance control cables from the bellcrank assembly by removing cotter pins, nuts, washers and bolts.

(5) Disconnect the aileron control rod (see “Figure 8” on page 271018, Sketch A) at the aft or forward end as desired.

(6) Remove the nut, pivot bolt (see “Figure 8” on page 271018, Sketch A) and washers that secure the bellcrank. The nut is visible from the underside of the wing.

(7) Remove the bellcrank from within the wing.

B. Installation

See “Figure 8” on page 271018.

(1) Install first the teflon tube then the spacer in the torque tube portion of the bellcrank. (See “Figure 8” on page 271018, Sketch A.)

(2) Place the bellcrank in position in the wing with a washer located between each end of the torque tube and the mounting brackets.

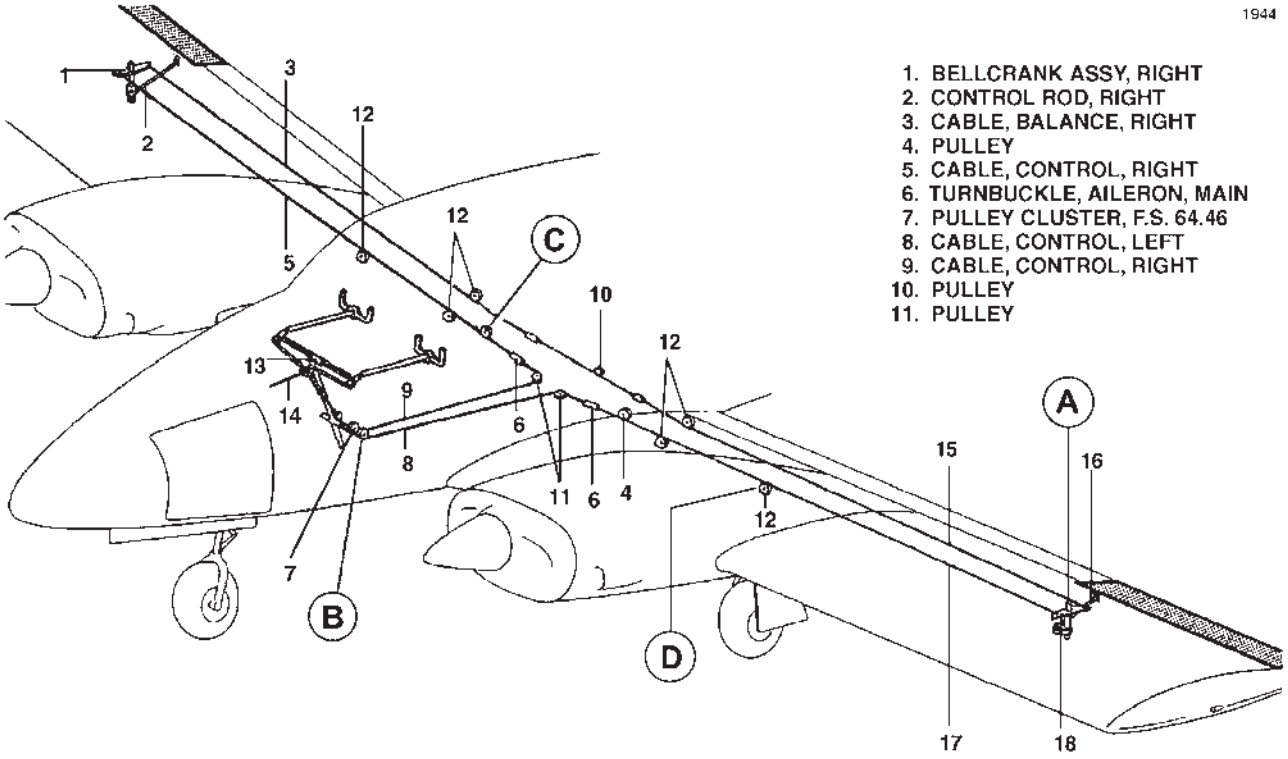
(3) Install the bellcrank pivot bolt (see “Figure 8” on page 271018, Sketch A) with the head up. Install a washer and nut on the bolt and torque nut 20 to 25 inch-pounds. Check that the bellcrank rotates freely with little up-down play. (See “Figure 8” on page 271018, Sketch A, Detail A.)

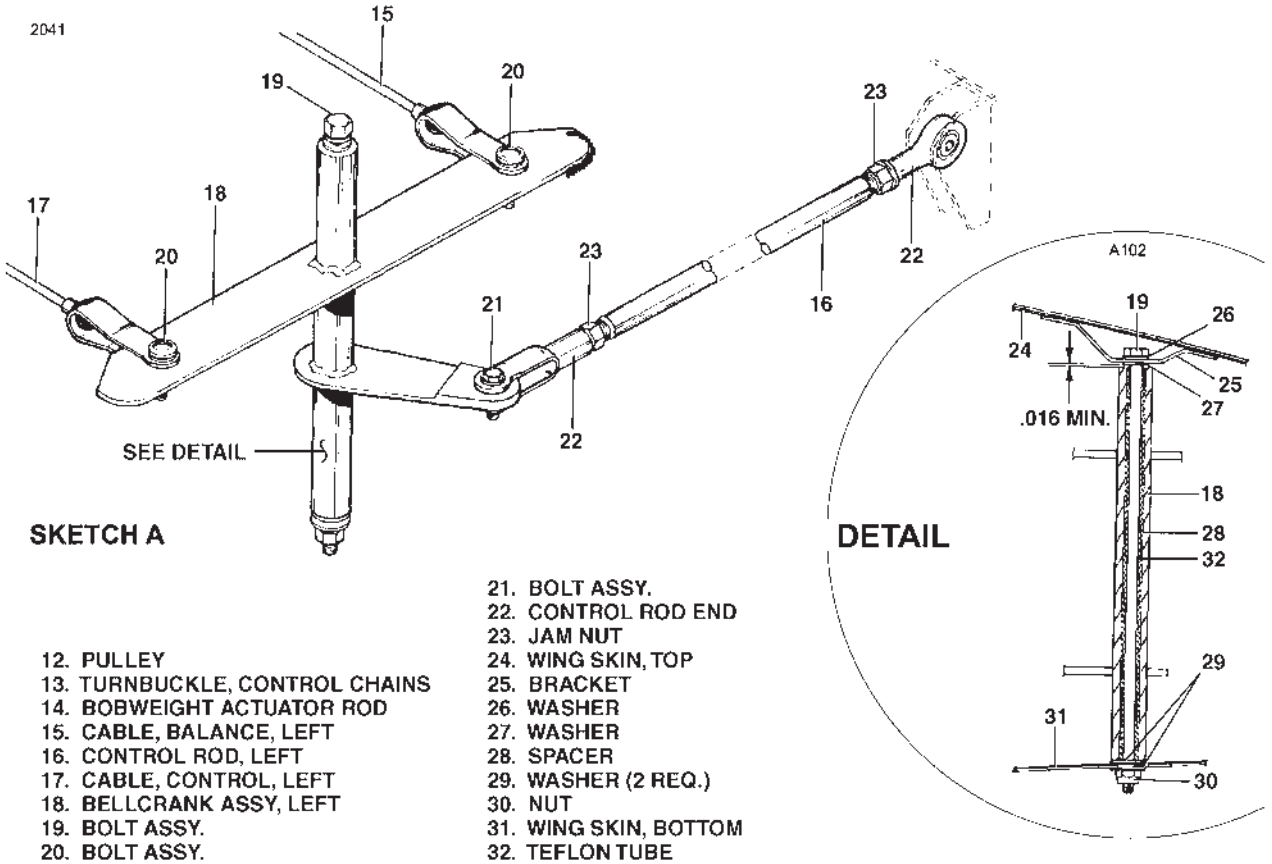
(4) Install and adjust control rod (see “Figure 8” on page 271018, Sketch A) and check aileron travel per Rigging and Adjustment, below.

(5) Connect the ends of the primary and balance control cables to the bellcrank using bolts, washers, nuts and cotter pins. Allow the cable ends to rotate freely on the bellcrank.

(6) Tighten the control cables at the balance cable turnbuckle (6) in the floor opening aft of the main spar. Check cable tension as described in next paragraph.

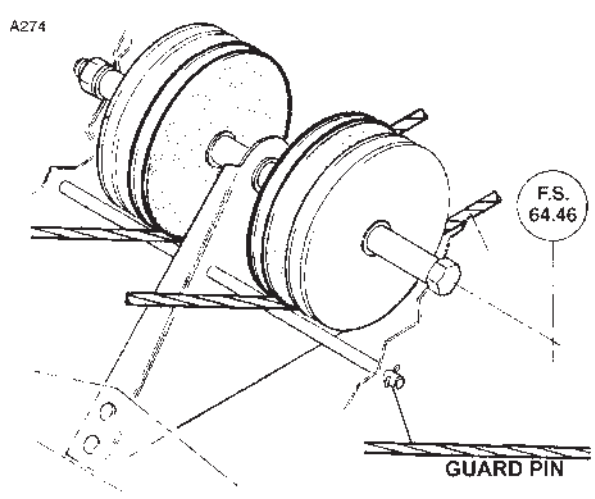
(7) Install the access plate on the underside of the wing, the floor panel aft of the main spar, seat belt attachments and seats.



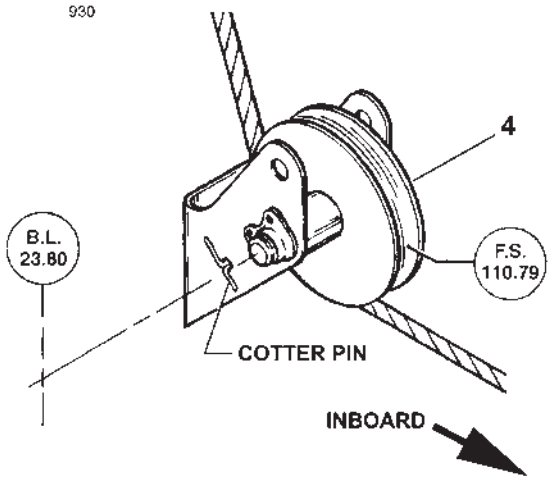


12. PULLEY
13. TURNBUCKLE, CONTROL CHAINS
14. BOBWEIGHT ACTUATOR ROD
15. CABLE, BALANCE, LEFT
16. CONTROL ROD, LEFT
17. CABLE, CONTROL, LEFT
18. BELLCRANK ASSY, LEFT
19. BOLT ASSY.
20. BOLT ASSY.
21. BOLT ASSY.
22. CONTROL ROD END
23. JAM NUT
24. WING SKIN, TOP
25. BRACKET
26. WASHER
27. WASHER
28. SPACER
29. WASHER (2 REQ.)
30. NUT
31. WING SKIN, BOTTOM
32. TEFLON TUBE

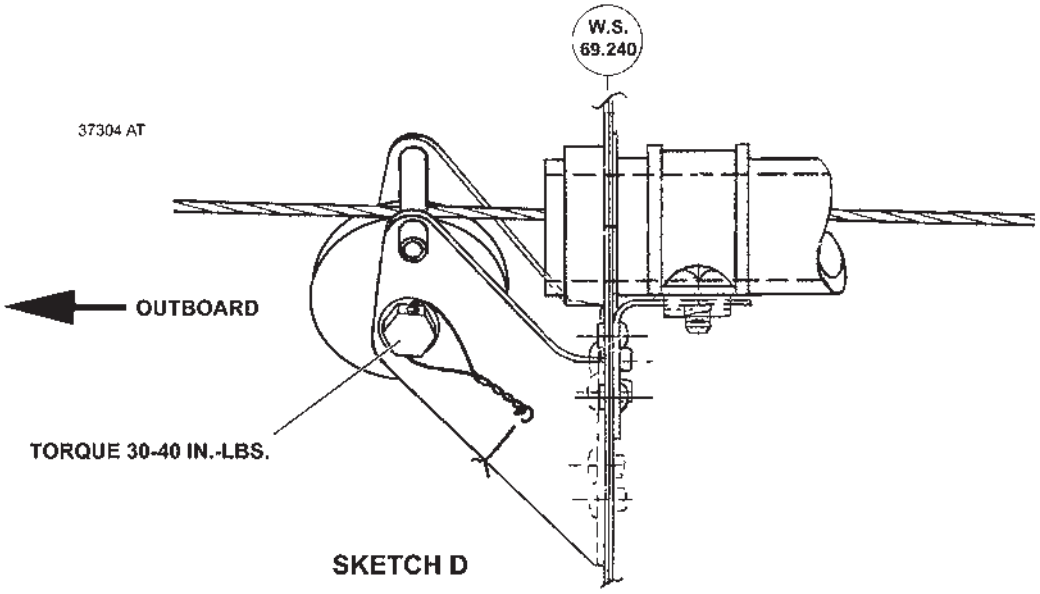
Aileron Controls
Figure 8 (Sheet 1 of 2)



SKETCH B



SKETCH C



SKETCH D

Aileron Controls
Figure 8 (Sheet 2 of 2)

6. Rigging and Adjustment

See “Figure 9” on page 271021, “Figure 10” on page 271022 and “Chart 2” on page 27004.

NOTE: Flap rigging and adjustment must be completed before starting aileron rigging and adjustment. See 27-50-00.

A. Ensure that the control wheels are properly rigged as described previously under Control Column Assembly, “Installation” on page 27105, and lock them in their center of travel.

B. Move the tee bar (control column) to its full forward position, and place weights on the aft side of the stabilator to maintain its position. The stabilator cables must already be at their proper tension.

C. With the aileron bellcrank in its neutral position, install the bellcrank rigging tool. The neutral position of the bellcrank occurs when the center of both of the cable connection holes are at an equal distance from the adjacent inboard wing rib. (See “Figure 10” on page 271022.)

D. Adjust the turnbuckles on the primary and balance cables (located under the floor behind the spar carry through) to the tension specified in “Chart 2” on page 27004. Make sure the bellcranks remain in their neutral positions.

NOTE: Tensions on drive (primary) cables may be slightly less than balance cable tension but must be within specified values. Move the controls after tensioning the cables and recheck as necessary.

E. The aileron’s neutral position is designed to be at a point where the chord line of the aileron forms a 1° 12’ ± 1° down angle with the wing chord, viewed at the inboard end of the aileron. The tool to determine neutral position is shown in “Figure 9” on page 271021 and can be fabricated from dimensions given in “Figure 12” on page 271023. The following procedure should be used for determining aileron neutral position.

(1) Make sure the bellcrank rigging tools fit snug between the bellcranks and their respective ribs.

(2) Place the aileron rigging tool against the underside of the wing and aileron. Position it as close as possible to the center of the aileron without contacting any rivets. It must also be parallel with the wing rib(s). The aft end of the tool should be even with the trailing edge of the aileron.

(3) If not already done, connect the aileron/bellcrank push rod to its mounting points.

(4) Adjust the push rod as necessary such that with the forward surface of the aileron rigging tool and spacer against the wing, the trailing edge of the aileron contacts the aft end of the tool. A slight “up” pressure should be maintained at the center of the aileron trailing edge to remove the slack between the bellcrank and aileron. Be sure to retighten the jamb nuts after adjustment.

F. Remove the bellcrank and control wheel locks.

G. The aileron travel can now be checked. The following procedure is recommended using a bubble protractor and using the neutral position as a datum or reference line.

(1) With one of the ailerons in its neutral position, center the bubble of the protractor over the aileron’s surface and note the reading.

(2) Move the aileron to the extent of its travel and after centering the bubble when full up and full down, record the two readings. The difference of each individual reading from that at neutral, will give the degrees of travel up or down respectively.

(3) The bellcrank stops attached to the rib adjacent to the aileron bellcrank should be adjusted as necessary to allow the appropriate travels. Repeat the procedure with the other aileron.

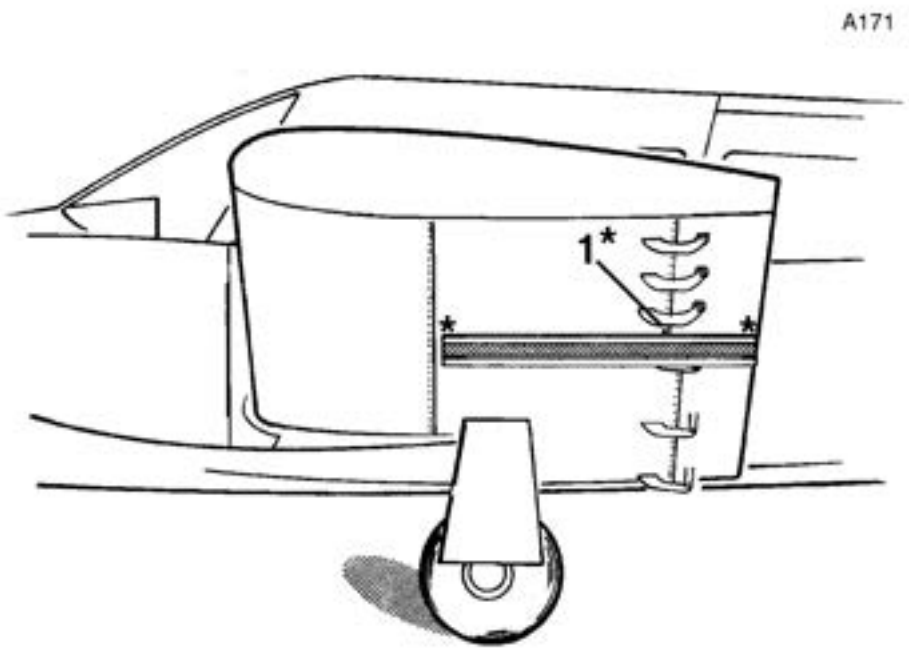
H. If the aileron bellcrank stops are bottomed before the control wheel is turned 90 ± 1 degree from the centered position, lengthen the drive cable and shorten the balance cable an equal amount. Recheck cable tension.

I. Move the pilot's control wheel full travel to ensure freedom of movement. With the control wheel at full travel and the bellcranks on their stops, a "cushion" of .030 to .040 must be maintained between the sprocket stop pin (on the "T" bar) and adjustable stop bolts.

J. Check control operation, bolts and turnbuckles for safety.

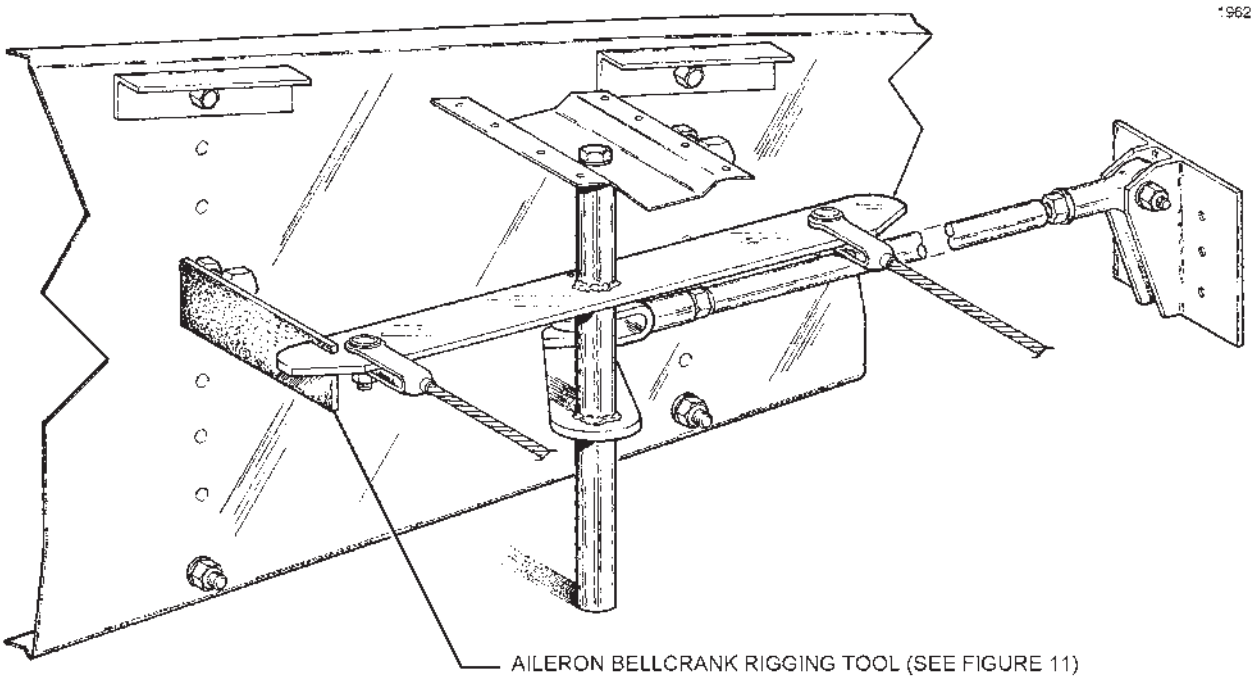
K. Install access plates and panels.

NOTE: When an out-of-trim condition persists, despite all the rigging corrections that can be made, there is a possibility that the trailing edge of the aileron has been used to move the aircraft forward This can result in a slight bulging of the aileron contour at the trailing edge which will cause an out-of-trim condition that is very difficult to correct.

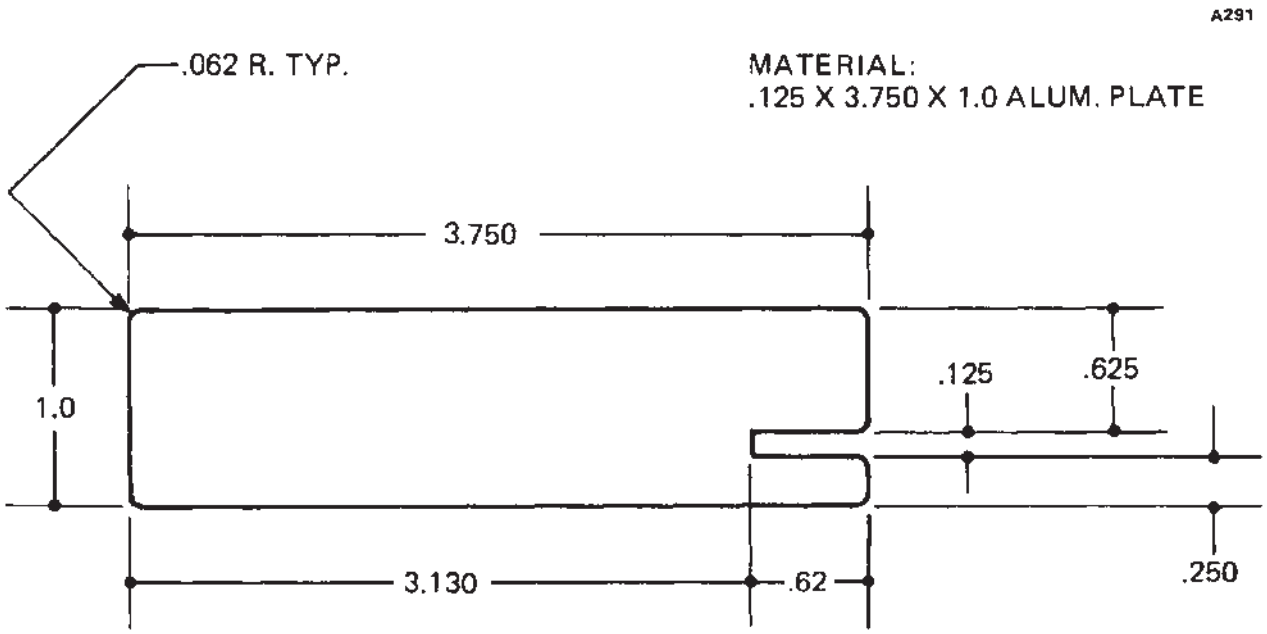


1. PLACE THE .385 HEIGHT SPACER IN LINE WITH THE AFT SPAR RIVET LINE, BUT NOT IN CONTACT WITH ANY RIVETS.
* AILERON NEUTRAL IS ACHIEVED WHEN THESE THREE POINTS MAKE CONTACT WITH WING AND AILERON SKIN SIMULTANEOUSLY.

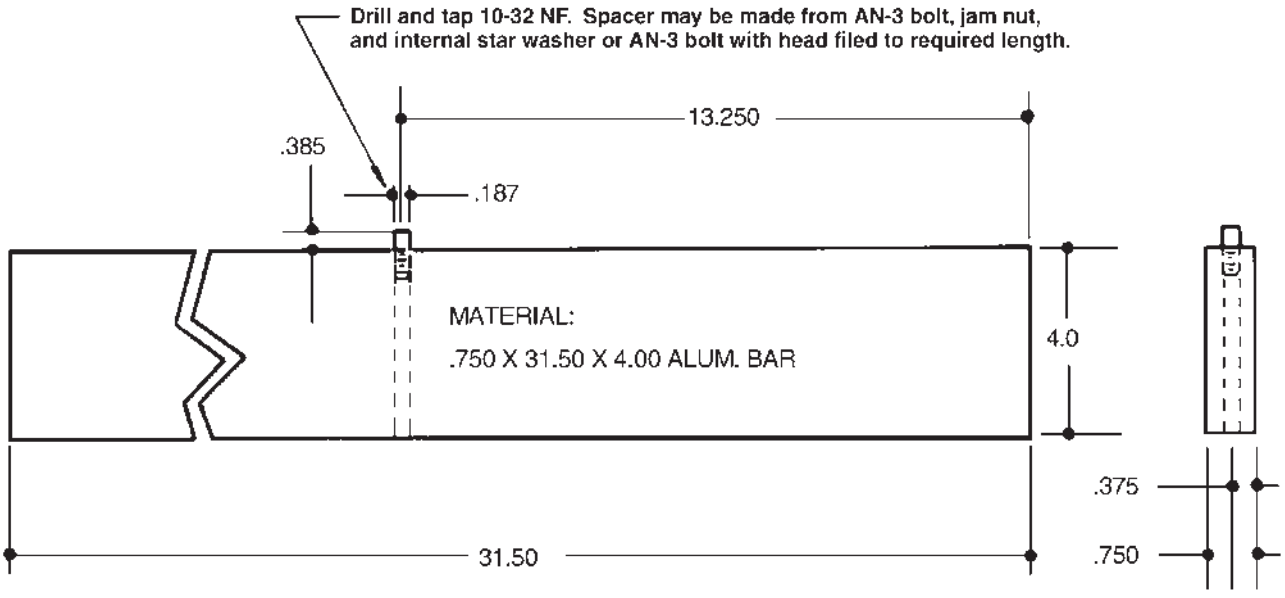
Aileron Rigging
Figure 9



Aileron Bellcrank Rigging
Figure 10



Aileron Bellcrank Rigging Tool
Figure 11



Aileron and Flap Rigging Tool
Figure 12

7. Aileron Post-Rigging Inspection

WARNING: VERIFY FREE AND CORRECT MOVEMENT OF AILERONS. WHILE IT WOULD SEEM SELF-EVIDENT, FIELD EXPERIENCE HAS SHOWN THAT THIS CHECK IS FREQUENTLY MISINTERPRETED OR NOT PERFORMED AT ALL. ACCORDINGLY, UPON COMPLETION OF AILERON RIGGING AND ADJUSTMENT, VISUALLY CONFIRM THAT THE RIGHT AILERON TRAILING EDGE MOVES UP AND THE LEFT AILERON TRAILING EDGE MOVES DOWN WHEN THE CONTROL WHEEL IS TURNED RIGHT; AND THAT THE LEFT AILERON TRAILING EDGE MOVES UP AND THE RIGHT AILERON TRAILING EDGE MOVES DOWN WHEN THE CONTROL WHEEL IS TURNED LEFT.

NOTE: Visual confirmation may require the aid of a second qualified mechanic or pilot.

Upon completion of any aileron control cable replacement, disconnect, or rigging perform the following post-rigging inspection:

A. Rotate the pilot's control wheel CLOCKWISE, as if to bank the airplane to the RIGHT, until reaching the primary stop. Visually confirm the following:

(1) The RIGHT hand wing aileron is deflected UP. The trailing edge of the aileron will be noticeably and significantly ABOVE the trailing edge of the RIGHT wing at the far RIGHT end of the aileron, near the wing tip.

(2) The LEFT hand wing aileron is deflected DOWN. The trailing edge of the aileron has moved noticeably and significantly BELOW the trailing edge of the LEFT wing at the far LEFT end of the aileron, near the wing tip.

B. Rotate the pilot's control wheel COUNTERCLOCKWISE, as if to bank the airplane to the LEFT, until reaching the primary stop. Visually confirm the following:

(1) The LEFT hand wing aileron is deflected UP. The trailing edge of the aileron has moved noticeably and significantly ABOVE the trailing edge of the LEFT wing at the far LEFT end of the aileron, near the wing tip.

(2) The RIGHT hand wing aileron is deflected DOWN. The trailing edge of the aileron has moved noticeably and significantly BELOW the trailing edge of the RIGHT wing at the far RIGHT end of the aileron, near the wing tip.

C. Repeat the above steps for the copilot controls.

D. Make a logbook entry documenting completion of this inspection

RUDDER CONTROLS

1. Troubleshooting

“Chart 1” lists troubles peculiar to rudder control system along with their probable causes and suggested remedies. When troubleshooting the rudder control system, additional reference may be obtained from Chapter 55 on control surface balancing, if required. After the trouble has been corrected, check the entire rudder control system for security and operation.

CHART 1
TROUBLESHOOTING RUDDER CONTROL SYSTEM

Trouble	Cause	Remedy
Lost motion between rudder pedals and rudder.	Cable tension too low.	Adjust cable tension.
	Linkage loose or worn.	Check linkage and tighten or replace.
	Broken pulley.	Replace pulley.
	Bolts attaching rudder to bellcrank are loose.	Tighten bellcrank bolts.
Excessive resistance to rudder pedal movement.	System not lubricated properly.	Lubricate system.
	Rudder pedal torque tube bearing in need of lubrication.	Lubricate torque tube bearings.
	Cable tension too high.	Adjust cable tension.
	Pulleys binding or rubbing.	Replace binding pulleys and/or provide clearance between pulleys and brackets.
	Cables not in place on pulleys.	Install cables correctly. Check cable guards.
	Cables crossed or routed incorrectly.	Check routing of control cables.
Rudder pedals not neutral	Rudder cables incorrectly when rudder is streamlined.	Rerig rudder cables. rigged.
Incorrect rudder travel.	Rudder bellcrank stop incorrectly adjusted.	Rerig bellcrank stops.
	Nose wheel contacts stops before rudder.	Rerig nose wheel stops.
Trim control knob moves	System not lubricated with excessive resistance.	Lubricate system. properly.

2. Rudder Pedal Assembly

A. Inspection

WARNING: ANY CRACKED COMPONENTS DISCOVERED DURING INSPECTION MUST BE REPLACED PRIOR TO NEXT FLIGHT.

In airplanes whose rudder pedal and bar assembly has accumulated 1,200 hours or more time-in-service, conduct the following inspections each 100 hours time-in-service.

NOTE: This inspection incorporates the requirements of Piper Service Bulletin 1242A.

NOTE: Installation of a new rudder pedal and bar assembly and a new rudder bar support assembly relieves this recurring inspection requirement, until the new components accumulate 1,200 hours time-in-service.

NOTE: Use a 10X magnifier, a mirror, and a suitable light source or other equipment capable of providing equal or better resolution.

(1) Gain access to the areas identified in"Figure 1". Remove interior components as required.

(2) Pedal and Bar Assembly Inspection

Visually inspect all welds, with specific emphasis on the locations identified in "Figure 1".

(3) Rudder Bar Support Assembly Inspection

See "Figure 1".

(a) Remove and retain the four (4) bolts that secure the rudder bar support assembly to the cockpit floor.

NOTE: Removing the rudder bar support assembly is not required to accomplish this inspection.

(b) Visually inspect the four (4) bolt holes for cracks and elongation. The diameter of the four bolt holes should measure 0.250 +0.005/-0.001 inches.

(c) Visually inspect the two (2) sheet metal bends common to the base for cracking, the entire length of each bend, both inner and outer surfaces.

(d) Reinstall the four (4) bolts that secure the rudder bar support assembly to the cockpit floor.

(4) Make a logbook entry documenting compliance with this inspection.

B. Removal

See "Figure 2" on page 27205.

(1) Remove the access panel to the aft section of the fuselage.

(2) Relieve rudder and stabilator cable tension by loosening one of the rudder and stabilator cable turnbuckles in the aft section of the fuselage.

(3) Remove the tunnel plate just aft of the tee bar by laying back enough tunnel carpet to remove the plate attachment screws.

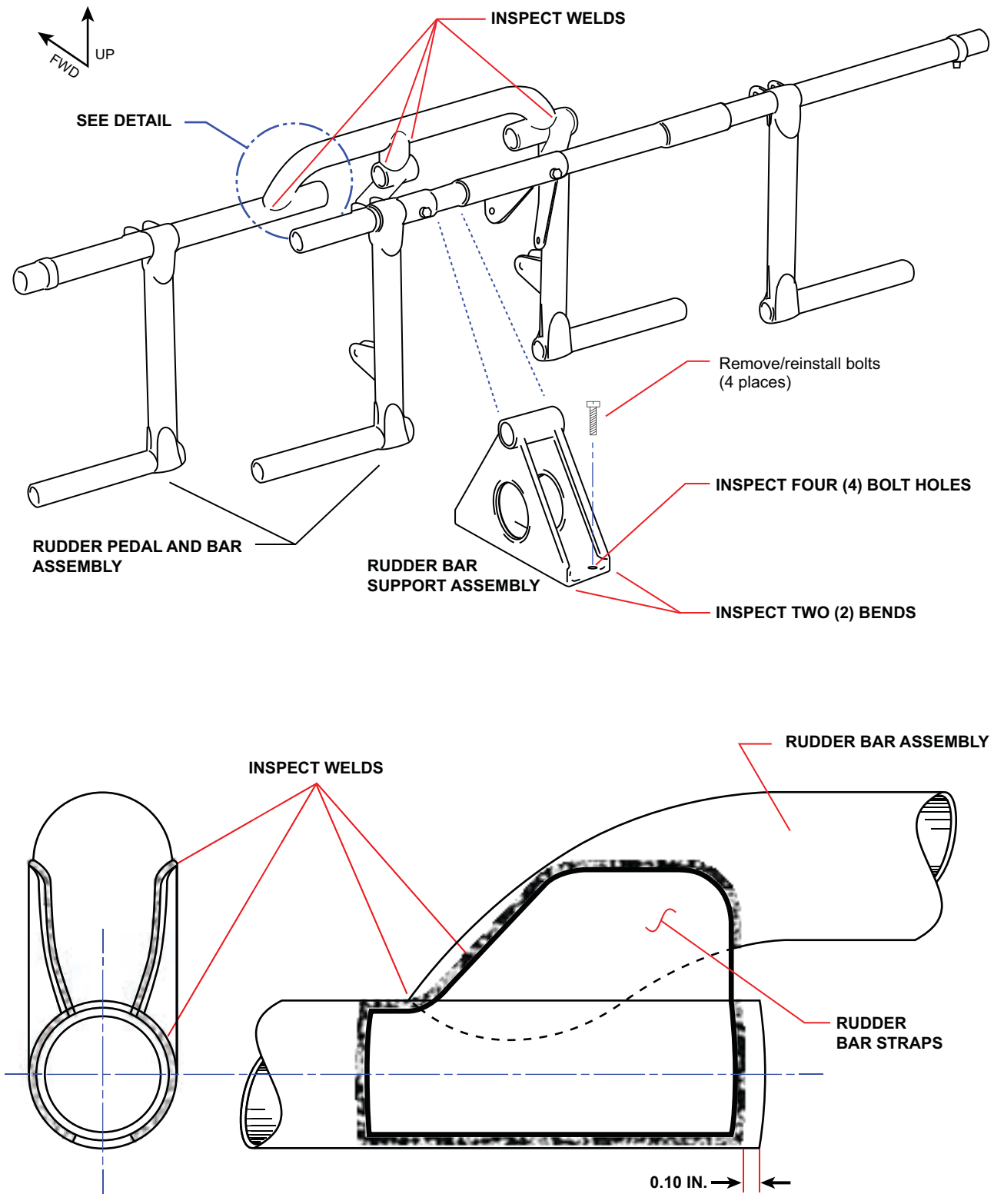
(4) Disconnect the stabilator control cable from the lower end of the tee bar assembly and disconnect the bobweight push rod from the tee bar.

(5) Remove the tee bar attachment bolts with their washers and nuts which are through each side of the floor tunnel. Pull the lower end of the tee bar aft.

(6) Disconnect the control cable ends from the arms on the torque tube by removing the cotter pins, washers, nuts and bolts.

(7) Disconnect the bungee rods at the control arms by removing nuts and bolts.

(8) Disconnect the brake cylinders at the lower end of each cylinder rod by removing the cotter pins and clevis pins.



DETAIL

Rudder Pedal and Bar Assembly Inspection
Figure 1

(9) Disconnect the vee brace(s) from the torque tube by removing nuts. washers and bolts that secure the strap bracket to the vee brace.

(10) Disconnect the torque tube support bracket where it attaches to the floor tunnel by removing its attachment bolts.

(11) Remove the two bolts that extend through the torque tube. They are located at the center of the tube assembly over the floor tunnel. Compress the tubes.

(12) Disconnect the torque tube support blocks from the support brackets on each side of the fuselage by removing the attachment nuts, washers and bolts.

(13) Remove the trim side panels if desired.

(14) Remove the assembly from the airplane. Note the spacer washer on each end and between the support blocks.

C. Installation

See “Figure 2” on page 27205.

(1) Assemble the torque tube assembly as shown in “Figure 2” on page 27205. Do not at this time install the two bolts through the center of the tube assembly.

(2) Place the upper support blocks on the ends of the torque tube assembly. Note that a washer is required on each end of the tube.

(3) Position the support blocks on their mounting brackets at each side of the fuselage and secure with bolts, washers and nuts. Note that a bushing is required in the bolt holes of the upper support block, and a plate on top of the upper block, between the upper and lower blocks and under the block mounting bracket.

(4) Align the bolt holes in the center area of the torque tube assembly; install bolts, washers and nuts. Then tighten nuts.

(5) Position the torque tube support bracket on the floor tunnel and secure with bolts.

(6) Position the vee brace(s) on the torque tube; install the strap bracket around the torque tube and brace. Secure with bolts, washers and nuts.

(7) Connect the ends of the brake cylinder rods. and clevis rods to the idler arms and secure with clevis and cotter pins.

(8) Connect the bungee rods and secure with bolts and nuts. Check steering rod adjustment per Alignment of Nose Gear, 32-20-00.

(9) Connect the rudder trim to the arm of the torque tube and secure with bolt, washer, nut and cotter pin. A thin washer is installed under the nut which is tightened only finger tight.

(10) Connect the ends of the rudder control cables to the arms provided on the torque tube and secure with bolts, washers, nuts and cotter pins. Allow the ends free to rotate.

(11) Swing the tee bar into place and secure with attachment bolts, washers and nuts. Insert the bolts in through each side of the floor tunnel.

(12) Connect the stabilator control cables to the lower end of the tee bar with bolt, washer and nut. Secure with cotter pin. Allow the cable ends to rotate freely. Connect bobweight push rod to tee bar.

(13) Set rudder cable tension and check rigging and adjustment of rudder controls.

(14) Set stabilator cable tension and check rigging and adjustment.

(15) Check aileron cable tension.

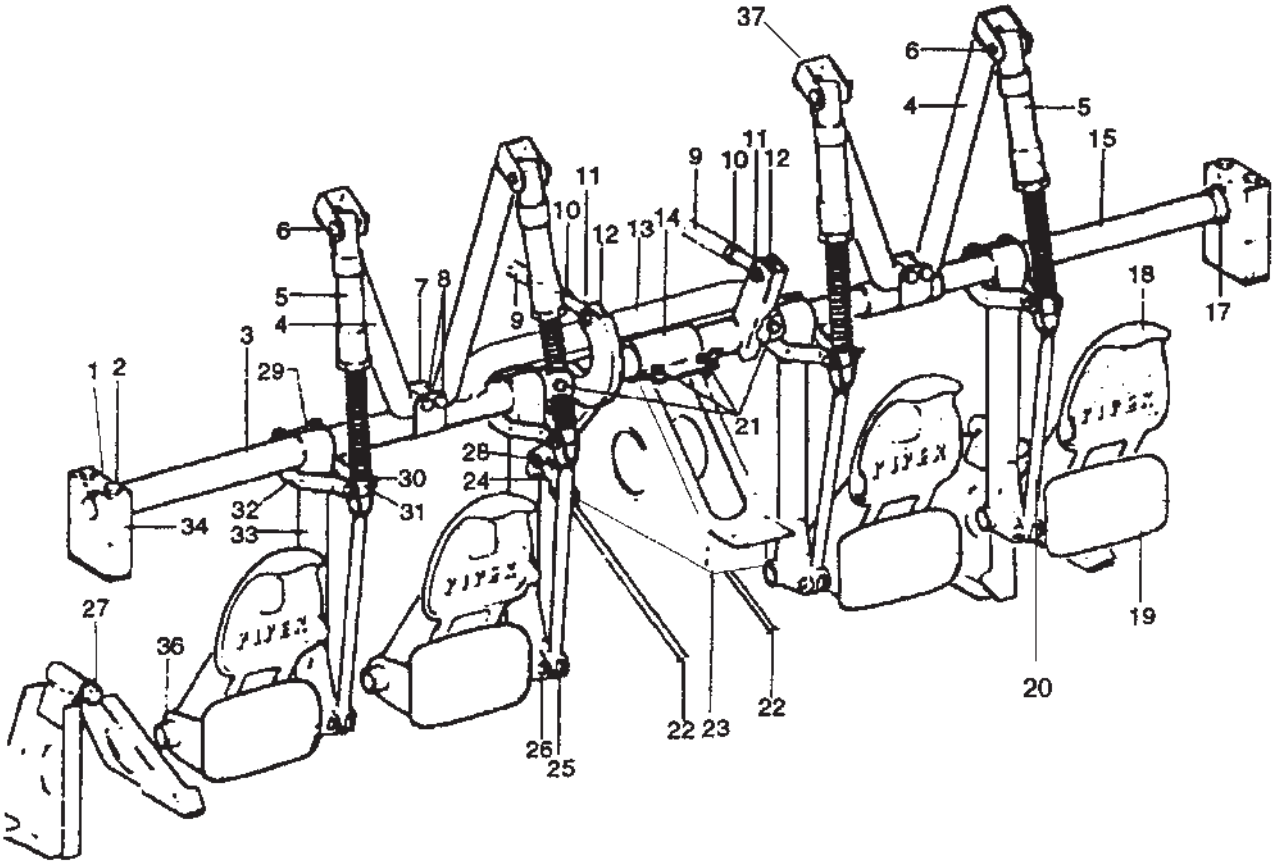
(16) Check safety of bolt and turnbuckles.

(17) Install the floor tunnel plate and secure with screws. Fasten the tunnel carpet in place.

(18) Install the access panel to the aft section of the fuselage.

When installing the rudder pedal assembly, torque bolts common to the following parts in the following order:

14. Bearing clamp, use NAS1149F0332P / NAS1149F0363P washers as required to maintain axial bearing.
34. Bearing assemblies.
3, 13, and 15. Torque tube assemblies.
37. Toe brake brackets.
7. Vee brace brackets.



1. PLATE, LEFT AND RIGHT
2. BOLT & NUT, LEFT AND RIGHT
3. TUBE, LEFT, OUTER
4. VEE BRACE, LEFT AND RIGHT
5. BRAKE CYLINDER
6. CLEVIS PIN & COITER PIN
7. BRACKET, LEFT AND RIGHT
8. BOLT, WASHER AND NUT (4 PLACES)
9. BUNGEE, NOSE WHEEL STEERING
10. JAM NUT
11. BOLT AND NUT
12. ROD END, STEERING
13. TUBE, LEFT, CENTER
14. CLAMP, BEARING
15. TUBE, RIGHT, CENTER
16. NOT USED
17. WASHER, SPACER
18. BRAKE PEDAL
19. RUDDER PEDAL
20. CLEVIS ROD
21. BOLT, WASHER & NUT
22. CONTROL CABLE, RUDDER
23. BRACKET, TUBE END
24. CABLE END
25. ROD END
26. CLEVIS PIN & COTTER PIN
27. STOPS, RUDDER PEDAL
28. BOLT, WASHER, NUT & COITER PIN
29. PIN, WASHER, & COTTER PIN
30. CLEVIS PIN & COITER PIN
31. ROD, BRAKE CYLINDER
32. IDLER ARM
33. TUBE, RUDDER CONTROL
34. BEARING ASSEMBLY, LEFT AND RIGHT
35. NOT USED
36. TUBE, RUDDER CONTROL
37. BOLT, WASHER, & NUT (4 PLACES)

Rudder Pedals Installation
Figure 2

3. Rudder Control Cables

A. Removal

See “Figure 3” on page 27208.

(1) To remove either the forward or aft rudder cables, first remove the access panel to the aft section of the fuselage.

(2) Disconnect the desired cable at the turnbuckles in the aft section of the fuselage.

(3) Either forward rudder cable may be removed by the following procedure:

(a) Remove the tunnel cover in the aft area of the cabin by removing the carpet, heat duct and the cover attachment screws.

(b) Remove the floor panel located directly aft of the main spar by removing the center seats, seat belt attachments and the screws securing the floor panel. Lift and remove the panel from the airplane.

(c) Remove the tunnel plate just aft of the tee bar by removing enough carpet from the tunnel to allow the plate attachment screws and the plate to be removed.

(d) Remove the cable guard plate (See “Figure 3” on page 27208, Sketch C) from the underside of the pulley cluster that is located in the aft area of the floor tunnel, by removing the block attachment screws.

(e) From within the area of the floor opening, remove the cable rub blocks (See “Figure 3” on page 27208, Sketch B) that are attached to the spar housing by removing the block attachment screws.

(f) Remove the cable guard pin (See “Figure 3” on page 27208, Sketch A) located under the pulley cluster by removing the cotter pin from the exposed end and sliding the pin to the left or right as required.

(g) Disconnect the end of the cable from the arm on the rudder pedal torque tube by removing the cotter pin, nut, washer and bolt. (See “Figure 3” on page 27208.)

(h) Draw the cable from the floor tunnel.

(4) The aft rudder control cable may be removed by the following procedure:

(a) Remove the tail cone by removing its attachment screws.

(b) Disconnect the cable from the rudder sector by removing the two cotter pins at the aft center portion of the sector and moving the swaged ball and cable out of the recessed hole in the sector.

(c) Remove the cable guard pins from the pulley brackets at Fuselage Station 280.091.

(d) Draw the cable from the fuselage.

B. Installation

See “Figure 3” on page 27208.

(1) The forward rudder control cables may be installed by the following procedure:

(a) Draw the control cable through the floor tunnel.

(b) Connect the end of the cable to the arm on the rudder pedal torque tube (“Figure 3” on page 27208) by installing bolt, washer, nut and cotter pin, allowing the cable end to rotate freely.

(c) Connect the forward control cable to the aft control cable at the turnbuckles in the aft section of the fuselage. If the aft control cables are not installed, install them at this time per instructions in Step 2. Ascertain that each cable is in the proper pulley groove.

(d) Move the cable guard located in the forward tunnel, under the pulley cluster into position, and secure with cotter pin.

(e) Within the area of the floor opening aft of the main spar, install the cable rub blocks onto the spar housing and secure with screws at the pulley cluster.

(f) Install the cable guard plate under the pulley cluster located in the aft area of the floor tunnel and secure with screws.

(g) Set cable tension as given in “Figure 4” on page 272011 and check rigging and adjustment. Safety the turnbuckle.

(h) Install the forward tunnel plate aft of the tee bar and secure with screws.

(i) Put the floor carpet in place and secure.

(j) Install the floor panel and seat belt attachment aft of the main spar, securing the panel with screws and install the seats.

(k) Install the cover and carpet of the aft floor tunnel.

(2) The aft rudder control cable may be installed by the following procedure:

(a) Position the control cable in the fuselage with the swaged ball next to the rudder sector.

(b) Route the cable ends over the pulleys and install the guard pins in the pulley brackets.

(c) Position the swaged ball of the cable in the recessed hole in the sector and secure in place with two MS24665-283 cotter pins.

(d) Connect the cable ends to the forward control cables at the turnbuckles in the aft section of the fuselage.

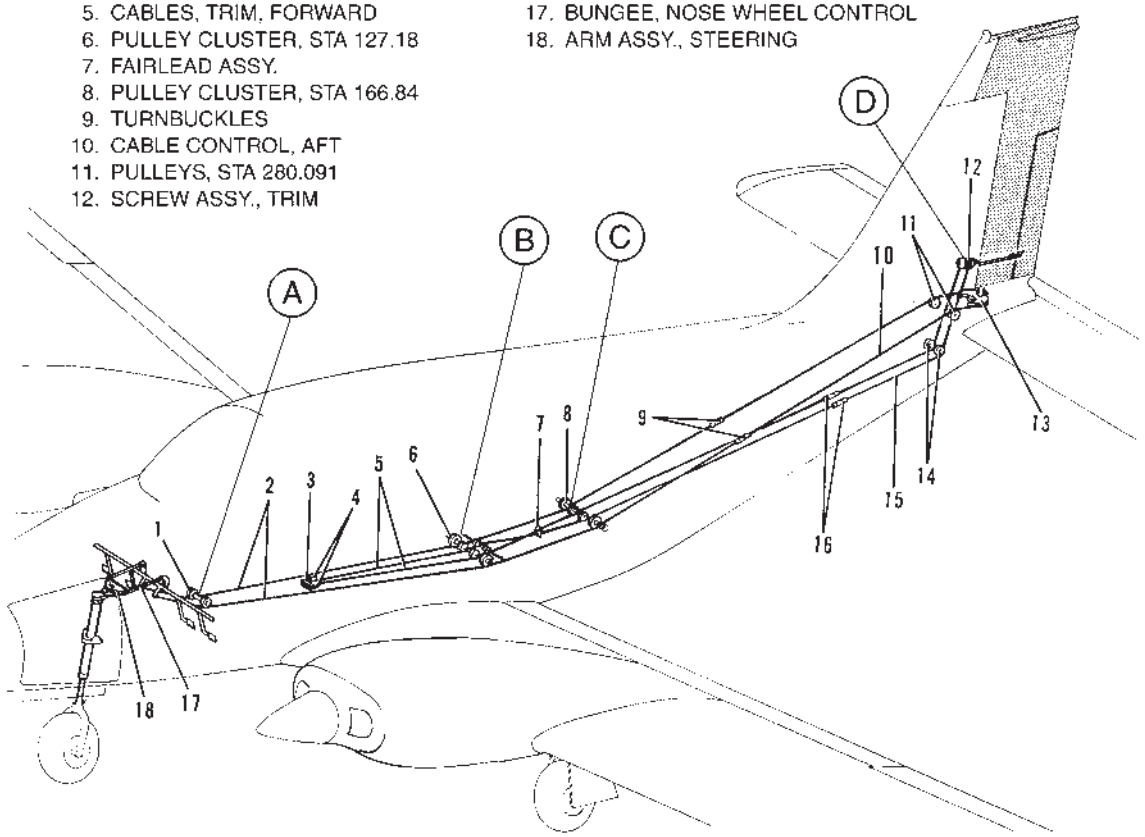
(e) Set cable tension as given in “Figure 4” on page 272011 and check rigging and adjustment of the rudder controls. Safety the turnbuckle.

(f) Install the tail cone and secure with screws.

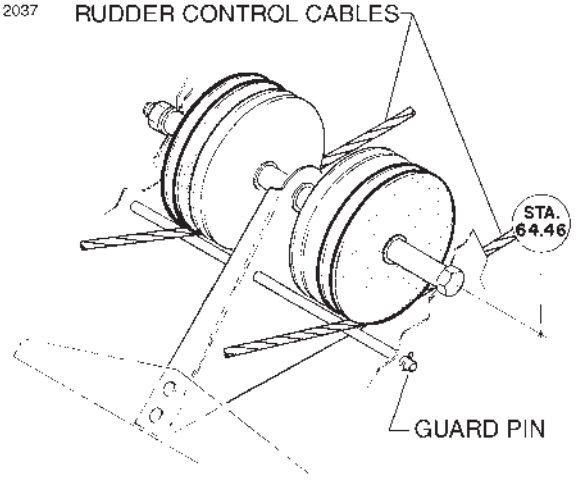
(3) Install the access panel to the aft section of the fuselage.

1963

1. PULLEY CLUSTER, STA 64.46
2. CABLES, CONTROL, FORWARD
3. WHEEL ASSY., TRIM CONTROL
4. PULLEYS, TRIM
5. CABLES, TRIM, FORWARD
6. PULLEY CLUSTER, STA 127.18
7. FAIRLEAD ASSY.
8. PULLEY CLUSTER, STA 166.84
9. TURNBUCKLES
10. CABLE CONTROL, AFT
11. PULLEYS, STA 280.091
12. SCREW ASSY., TRIM
13. SECTOR ASSY., TORQUE TUBE
14. PULLEYS, STA 279.032
15. CABLE, TRIM, AFT
16. TURNBUCKLES
17. BUNGEE, NOSE WHEEL CONTROL
18. ARM ASSY., STEERING

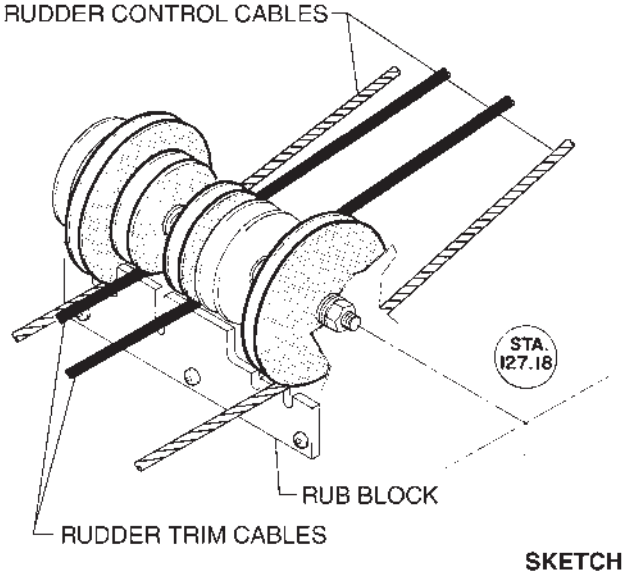


2037



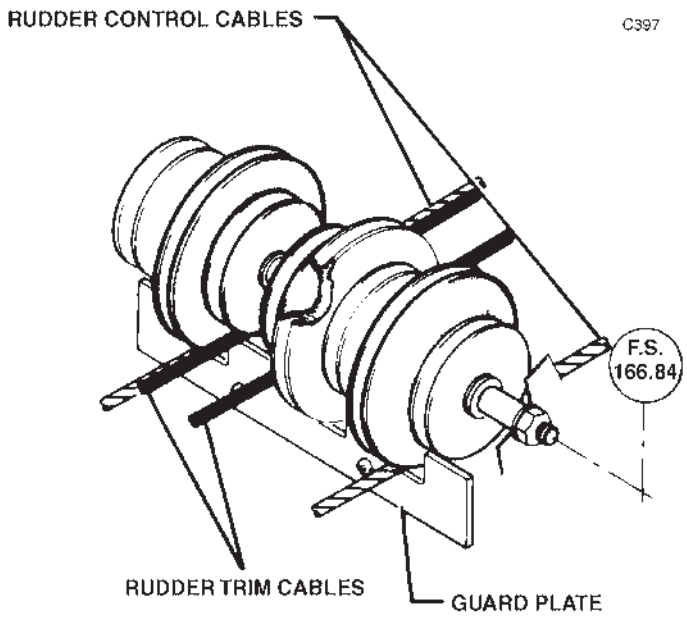
SKETCH A

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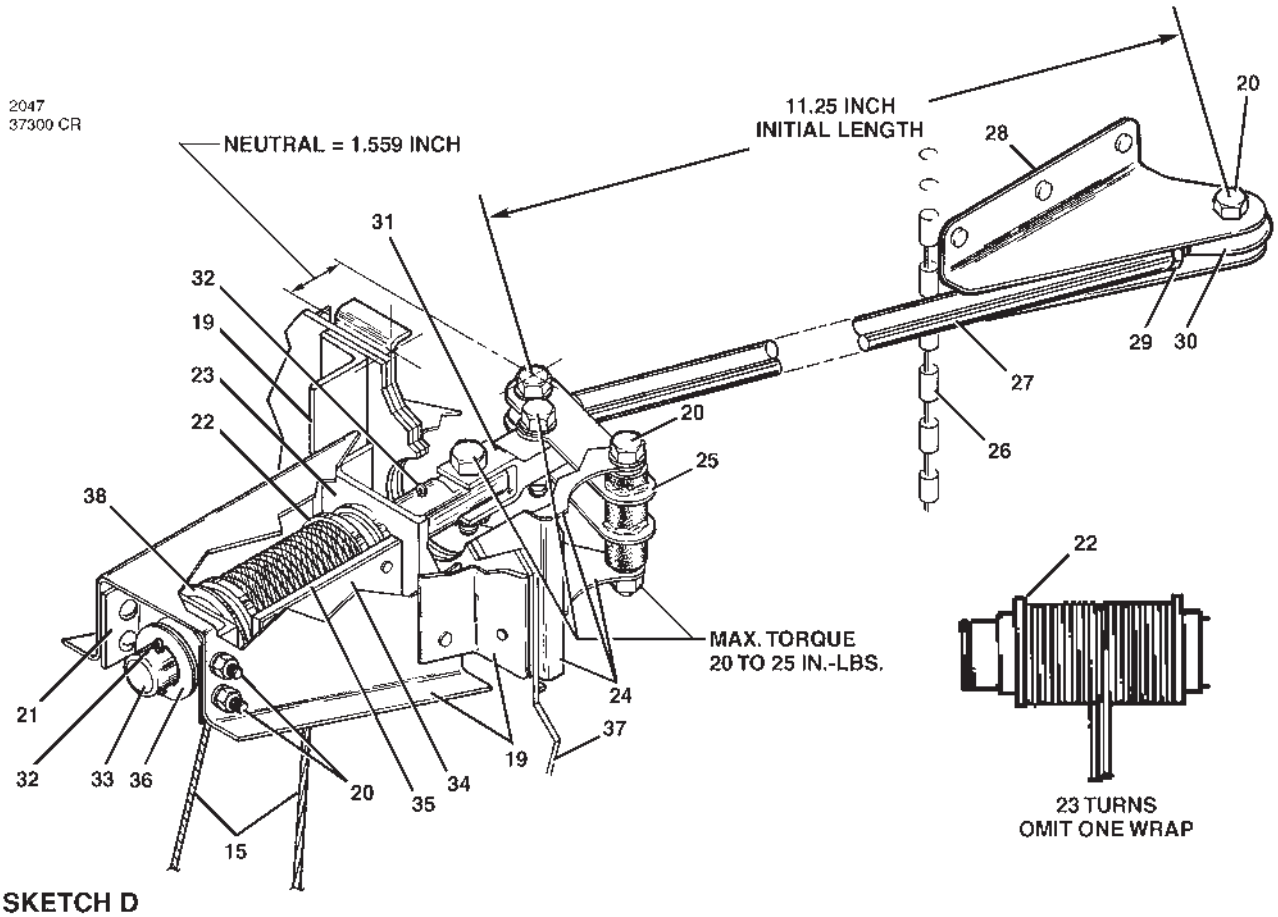
SKETCH B

Rudder Controls Installation
Figure 3 (Sheet 1 of 2)



19. MOUNTING BRACKET, TRIM SCREW
20. BOLT ASSY.
21. SUPPORT ASSY., BARREL MOUNT
22. BARREL, TRIM
23. SUPPORT ASSY., BARREL MOUNT
24. MOUNTING BRACKET, LINK ASSY.
25. ARM ASSY.
26. RUDDER TRIM TAB ASSY.
27. CONTROL ROD
28. ARM ASSY., TRIM TAB
29. JAM NUT
30. ROD END
31. LINK ASSY.
32. COTTER PIN
33. SHAFT ASSY., TRIM SCREW
34. SHIM
35. CABLE GUARD
36. STOP WASHERS
(NAS1149F0863P OR NAS1149F0832P
IN ANY COMBINATION UP TO 6 TOTAL)
37. SPAR
38. SHIM

SKETCH C



SKETCH D

Rudder Controls Installation
Figure 3 (Sheet 2 of 2)

C. Rigging and Adjustment

See “Figure 4”, “Figure 5” on page 272017; 27-00-00, “Chart 2” on page 27004; and “Figure 2” on page 27303.

Measurements for determining rudder travel are initiated from the neutral position (i. e. when the rudder is streamlined with the vertical stabilizer).

WARNING: VERIFY FREE AND CORRECT MOVEMENT OF RUDDER. WHILE IT WOULD SEEM SELF-EVIDENT, FIELD EXPERIENCE HAS SHOWN THAT THIS CHECK IS FREQUENTLYMISINTERPRETEDORNOTPERFORMEDATALL.ACCORDINGLY, UPON COMPLETION OF RUDDER RIGGING AND ADJUSTMENT, VISUALLY CONFIRM THAT THE RUDDER TRAILING EDGE MOVES RIGHT WHEN THE RIGHT PEDAL IS DEPRESSED; AND, THAT THE RUDDER TRAILING EDGE MOVES LEFT WHEN THE LEFT PEDAL IS DEPRESSED.

(1) Using the rudder pedals as necessary, check and set the correct degree of rudder travel as follows:
   (a) Swing the rudder until it contacts the stop and hold it in that position. See “Chart 2” on page 27004 for travels.
   (b) Using the rigging tool as shown in “Figure 4” on page 272011, align the tool at the root against the side of the vertical stabilizer and rudder, making sure to keep it clear of rivets.
   (c) The rigging tool should fit flush against the rudder and stabilizer. If a gap exists at these areas, remove the tail cone and adjust the stop. Use “Figure 4” for reference.
   (d) Swing the rudder in the opposite direction and complete the same procedure.

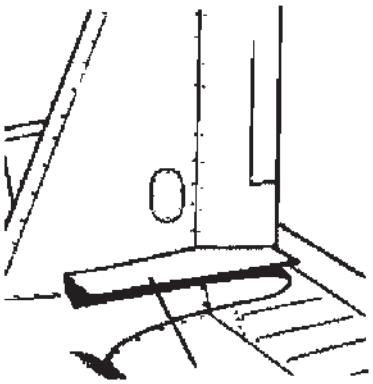
(2) The cable tension may be set as follows:
   (a) Remove the access panels in the rear baggage area to make access to the rear of the fuselage.
   (b) Block the rudder and trim tab such that they are streamlined with the vertical fin.
   (c) Secure the rudder pedals in their neutral position as shown in “Figure 4” on page 272011. If the nose gear steering is found to be out of alignment, see 32-20-00 for adjustment.
   (d) If not already accomplished. the nose wheel must be raised clear of the ground for the remainder of this procedure.
   (e) On the rudder cables in the aft of the fuselage. adjust the turnbuckles to obtain 40 ± 5 pounds tension. Make sure to tighten the cables evenly avoiding uneven strain on aircraft components.
   (f) Unblock the rudder and trim tab.

(3) Apply just enough pressure on the pilot’s left pedal for the rudder to meet the stop. The clearance between the pedal stop and the stop bolt should be .060 to .120 in.

(4) The same procedure and clearance applies to the copilot’s right pedal.

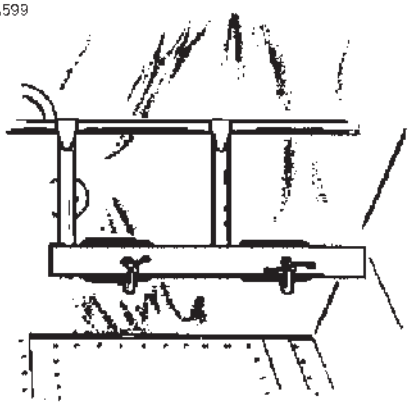
(5) Install tail cone and the access panel.

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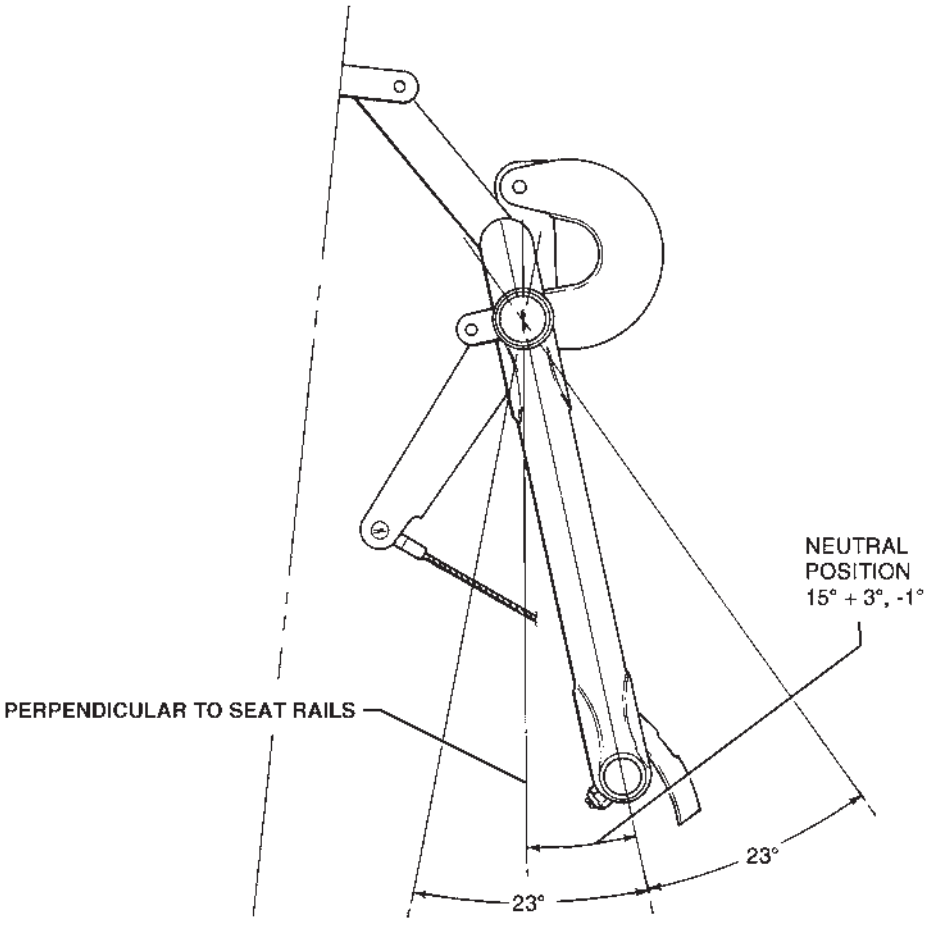
USING RIGGING TOOL

A599



NEUTRALIZING PEDALS

A295



PEDAL TRAVELS

Rigging Rudder and Controls
Figure 4

D. Rudder Post-Rigging Inspection

NOTE: Visual confirmation may require the aid of a second qualified mechanic or pilot.

NOTE: This inspection may need to be performed with the airplane on jacks.

Upon completion of any rudder control cable replacement, disconnect, or rigging perform the following post-rigging inspection:

(1) Depress the pilot's RIGHT rudder pedal as if to yaw the airplane to the RIGHT, until reaching the primary stop. Visually confirm the following:

(2) The trailing edge of the rudder will be noticeably and significantly RIGHT of the trailing edge of the vertical fin.

(3) Depress the pilot's LEFT rudder pedal as if to yaw the airplane to the LEFT, until reaching the primary stop. Visually confirm the following:

(4) The trailing edge of the rudder will be noticeably and significantly LEFT of the trailing edge of the vertical fin.

(5) Repeat the above steps for the copilot controls.

(6) Make a logbook entry documenting completion of this inspection.

4. Rudder Trim Controls

A. Forward Trim Controls

(1) Removal

See "Figure 3" on page 27208.

(a) To remove the trim control wheel assembly and/or trim control cables, first remove the panel to the aft section of the fuselage.

(b) If the aft trim cable is not being removed, block the cables aft to the turnbuckles to prevent the cables from unwrapping at the trim barrel in the fin. (See "Figure 3" on page 27208.)

(c) If the trim control wheel is to be removed, loosen the cables at the turnbuckles and proceed with the following steps:

1) Remove the trim cover assembly by removing the cover attaching screws.

2) Remove the nut, washers and bolt that secures the trim wheel assembly between its mounting bracket. Draw the wheel from the brackets. Use caution not to damage the trim indicator wire.

3) Unwrap the lower cable from the drum.

4) The wheel and drum are joined by three screws. Remove screws and separate these two items and unwrap the upper cable.

5) Tie the cables forward to prevent them from slipping back into the floor tunnel.

(d) If the trim control wheel and forward cables are to be removed, block the aft cables aft of the turnbuckles and proceed with the following steps:

1) Remove the tunnel cover in the aft area of the cabin by removing the carpet and heater duct over the tunnel and the cover attachment screws.

2) Remove the floor panel located directly aft of the main spar by removing the center seats, seat belt attachments and screws securing the panel. Remove the panel from the airplane.

3) Remove the trim cover assembly to gain access to the trim wheel mounting hardware.

4) Disconnect the turnbuckles and remove the guard plate (See "Figure 3" on page 27208, Sketch C) at pulley cluster.

5) Remove the rub block at the pulley cluster.

6) Remove the nut, washers, and bolt securing the rudder trim control wheel and drum assembly to its mounting bracket and remove the complete assembly with cables. Use caution not to damage the indicator wire.

(2) Installation

See “Figure 3” on page 27208.

(a) The trim control wheel with drum may be installed by the following procedure:

1) Wrap the left cable on the trim drum by inserting the swaged ball of the cable in the slot provided in the upper side of the drum which mates with the control wheel. Looking at this side, proceed to wrap three and a half turns of cable in a clockwise direction.

2) Attach the trim control wheel to the cable drum by aligning the long lug of the drum with the long slot of the wheel and securing the two pieces together with three screws.

3) Wrap the right cable on the drum by inserting the swaged ball of the cable into the slot provided in the flanged lower side of the drum. Looking at this side, proceed to wrap three and a half turns of cable in a clockwise direction.

4) Lubricate and install the bushing in the lower side of the drum and the bearing on the upper side of the trim control wheel assembly.

5) Align the trim control cables and position the control wheel assembly between its mounting brackets. Ascertain that the trim indicator wire is positioned in the spiraled slot of the wheel with no binding on the end. Install the retainer bolt from the upper side, along with the washer and secure with washer and nut from below.

6) Install the cover assembly over the trim control wheel and secure with screws unless the control cables have yet to be installed.

(b) The trim control cables may be installed by the following procedure:

1) Draw the cables through the floor tunnel and route them through the pulley clusters at station 127.17 and 166.84. Ascertain that the cables cross at the fairlead between the two pulley clusters.

2) Wrap the cable drum and install the trim control wheel as given in Step 1.

3) Position the cables over the proper pulleys (as shown in “Figure 3” on page 27208, Sketches B and C).

4) Connect the forward cables to the aft cables at the turnbuckles in the aft section of the fuselage. If the aft cable is not installed, proceed with instructions in this section.

5) Remove the blocks securing the aft cables and check that the cables are seated on the pulleys. Install the rub block and guard plate at the appropriate pulley clusters. (See “Figure 3” on page 27208, Sketches B and C.)

6) Set trim cable tension in accordance with specifications given in “Figure 4” on page 272011 and check rigging and adjustment of rudder trim controls. Safety both turnbuckles.

7) Install the tunnel cover on the forward tunnel and secure with screws.

8) Install the carpet over the floor tunnel.

9) Install the cover over the trim control wheels and flap handle and secure with screws.

10) Install the seat belts removed from the top of the floor tunnel and secure with bolt, washer and nut.

11) Install the aft floor tunnel cover, heater duct and carpet.

12) Install the carpet over the aft floor plate.

(c) Install the panel to the aft section of the fuselage and the seats.

B. Aft Trim Controls

(1) Removal

See “Figure 3” on page 27208.

(a) Remove the access panel from the lower side of the fin and the tail cone fairing.

(b) If the forward trim mechanism is not being removed at this time, block the cables forward of the turnbuckles to prevent the cables from unwrapping at the forward trim drum. See “Figure 2” on page 27303.

(c) Secure the trim cables at the aft trim drum barrel.

(d) Disconnect the trim cable turnbuckles in the aft section of the fuselage.

(e) Remove the cable guards from the pulley bracket located at station 279.032.

(f) Disconnect the trim screw link assembly from the screw.

(g) Remove the cotter pin from the aft end of the screw.

(h) Remove the four bolt assemblies securing the forward support to the mounting bracket.

(i) Remove the screw and barrel assembly along with the aft cables from the airplane.

(2) Installation

See “Figure 3” on page 27208.

(a) Insert the complete trim screw and barrel assembly into the fin. Route the trim cable ends around the pulleys at station 279.032.

(b) Insert the trim screw and barrel assembly into the mounting bracket. Place the washer on the forward end of the barrel and install the support assembly in the mounting bracket.

NOTE: Total allowable end play of the barrel in the mounting bracket is .006 to .008 inches. Use 62833-18 laminated shim stock washer as required to achieve the correct end play.

(c) Install the AN960-8 16 and AN-960-816L washers over the forward end of the screw shaft and install the cotter pin. Install the cotter pin in the aft end of the shaft.

(d) Adjust the screw assembly to obtain the neutral position. (See “Figure 3” on page 27208, Sketch D.)

(e) Connect the link assembly to the trim screw.

(f) Connect the aft trim cables to the forward cables with turnbuckles. Check to ensure the cables are properly routed around the pulleys.

(g) Install the cable guards at the pulley bracket in the fuselage at station 279.032.

(h) Remove the clamp securing the forward trim cables and proceed to rig the system.

(i) Lubricate the assembly per instructions in 12-20-00.

(j) Install the access panel and tail cone fairing.

C. Rigging and Adjustment

See “Figure 2” on page 27205, “Figure 4” on page 272011, and “Chart 2” on page 27004.

WARNING: VERIFY FREE AND CORRECT MOVEMENT OF RUDDER TRIM TAB. WHILE IT WOULD SEEM SELF-EVIDENT, FIELD EXPERIENCE HAS SHOWN THAT THIS CHECK IS FREQUENTLY MISINTERPRETED OR NOT PERFORMED AT ALL. ACCORDINGLY, UPON COMPLETION OF RUDDER TRIM RIGGING AND ADJUSTMENT, CENTER THE RUDDER AND VISUALLY CONFIRM THAT THE RUDDER TRIM TAB TRAILING EDGE MOVES RIGHT WHEN THE RUDDER TRIM WHEEL IS TRIMMED LEFT; AND, THAT THE RUDDER TRIM TAB TRAILING EDGE MOVES LEFT WHEN THE RUDDER TRIM WHEEL IS TRIMMED RIGHT.

The rudder and its tab are considered to be in neutral when streamlined with each other and the vertical stabilizer. Travel measurements are taken from the neutral position. To rig the trim controls the following procedure is recommended:

(1) If the assemblies have just been installed the following procedure is recommended. Disregard these instructions if previously accomplished.
   (a) Check the trim drum in the stabilizer and make sure when at neutral there are 23 cable turns on the drum with one omitted in the center.
   (b) Check the tab actuating push rod for its optimum length of 11.25 inches.
   (c) Make sure the nose gear is clear of the ground.

(2) If the cables have been disconnected proceed as follows:
   (a) Make sure the rudder and tab are blocked in the neutral position.
   (b) Center the cockpit rudder trim control, and connect the rudder trim cables. Tighten the cables to 10 ± 2 pounds making sure the system is evenly tensioned to prevent uneven loads on the structure and assemblies.
   (c) Unblock the tab.

(3) With the rudder centered, rotate the trim control wheel in the cockpit to full right trim and then full left trim.

(4) Check that each tab travel is 25 ± 1 degrees (Seneca IV) or 26 ± 1 degrees (Seneca V). Adjust the actuating rod as necessary to obtain correct travel.

(5) If symmetrical travels cannot be reached by adjusting the actuating rod, add or remove stop washers on the trim screw. At least two but no more than five AN-960-8 16 or AN-960-8 16L washers of any combination are allowed.

   NOTE: If the tab cannot be adjusted to the correct travels with the removal or addition of washers along with adjustment of the actuating rod, the cable trim barrel must be repositioned on the actuating trim screw. See the previous subject paragraphs for appropriate instructions.

(6) Remove the blocks securing the rudder in its neutral position.

(7) On Seneca IV models, rudder tab servo travel is determined as follows:

(a) Ascertain that the rudder trim control wheel in the cockpit is in its neutral position.

(b) Push on one of the left rudder pedals until the rudder stop is contacted.

(c) With a bevel protractor or other suitable tool, measure the degree of deflection. Tab travel should be 7 ± 1 degrees left of center line of rudder.

(d) Repeat steps (a) and (b) with the right rudder pedal, checking for 7 ± 1 degrees right of center line of rudder.

(e) If either or both of these readings are off, the trim barrel must be repositioned on the actuating screw along with a readjustment of the actuating rod length. If either of these is adjusted, all rudder and rudder trim rigging and adjustments must be repeated.

(8) On Seneca V models, rudder tab servo travel is determined as follows:

(a) Ascertain that the rudder trim control wheel in the cockpit is in its neutral position.

(b) Push on one of the left rudder pedals until the rudder stop is contacted.

(c) With a bevel protractor or other suitable tool, measure the degree of deflection. Tab travel should be 13 ± 2 degrees right of center line of rudder.

(d) Repeat steps (a) and (b) with the right rudder pedal, checking for 8 ± 2 degrees left of center line of rudder.

(e) If either or both of these readings are off, the trim barrel must be repositioned on the actuating screw along with a readjustment of the actuating rod length. If either of these is adjusted, all rudder and rudder trim rigging and adjustments must be repeated.

(9) Secure the rudder against either stop and measure, if any, the amount of trim tab free play. Measuring at the tab trailing edge, free play must not exceed 0.060 inch.

Alternate Method: Place rudder and rudder tab in neutral position. Utilizing a dial indicator and related holding fixture, determine the amount of free play motion between the rudder trailing edge and rudder tab trailing edge 0.060 max.

WARNING: VERIFY FREE AND CORRECT MOVEMENT OF RUDDER. WHILE IT WOULD SEEM SELF-EVIDENT, FIELD EXPERIENCE HAS SHOWN THAT THIS CHECK IS FREQUENTLY MISINTERPRETED OR NOT PERFORMED AT ALL. ACCORDINGLY, UPON COMPLETION OF RUDDER TRIM RIGGING AND ADJUSTMENT, VISUALLY CONFIRM THAT THE RUDDER TRAILING EDGE STILL MOVES RIGHT WHEN THE RIGHT RUDDER PEDAL IS STEPPED ON; AND, THAT THE RUDDER TRAILING EDGE STILL MOVES LEFT WHEN THE LEFT RUDDER PEDAL IS STEPPED ON.

D. Rudder Trim Post-Rigging Inspection

NOTE: Visual confirmation may require the aid of a second qualified mechanic or pilot.

Upon completion of any rudder trim control cable replacement, disconnect, or rigging perform the following post-rigging inspection:

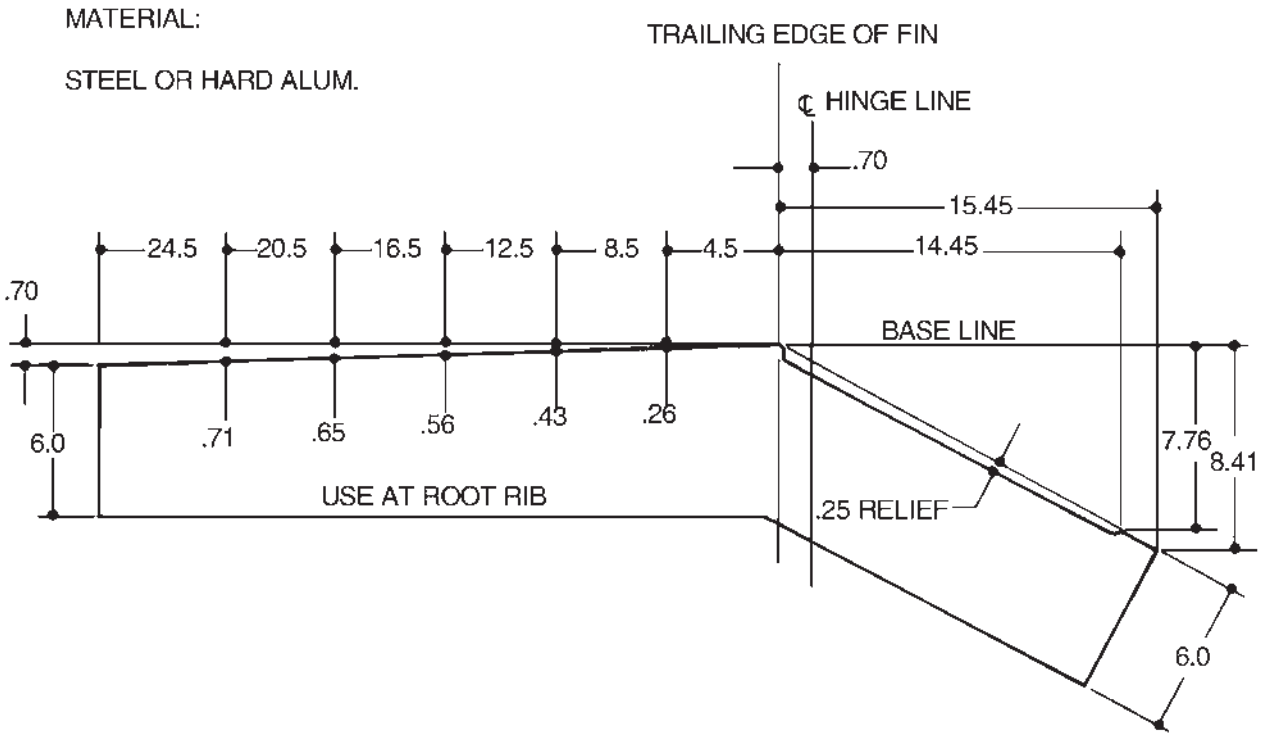
(1) Trim the rudder trim wheel left.

Visually confirm that the rudder tab trailing edge moves right.

(2) Trim the rudder trim wheel right.

Visually confirm that the rudder tab trailing edge moves left.

(3) Make a logbook entry documenting completion of this inspection.



Rudder Rigging Tool
Figure 5

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STABILATOR CONTROLS

1. Troubleshooting

“Chart 1” and “Chart 2” on page 27302 list troubles peculiar to stabilator control system along with their probable causes and suggested remedies. When troubleshooting the stabilator control system, additional reference may be obtained on control surface balancing from Chapter 55. After the trouble has been corrected, check the entire rudder control system for security and operation.

CHART 1
TROUBLESHOOTING STABILATOR CONTROL SYSTEM

Trouble	Cause	Remedy
Lost motion between control wheel and stabilator.	Cable tension too low.	Adjust cable tension.
	Linkage loose or worn.	Check linkage and tighten or replace.
	Broken pulley.	Replace pulley.
	Cables not in place on pulleys.	Install cables correctly.
Resistance to stabilator control movement.	System not lubricated properly.	Lubricate system.
	Cable tension too high.	Adjust cable tension.
	Binding control column.	Adjust and lubricate.
	Pulleys binding or rubbing.	Replace binding pulleys and/or provide clearance between pulleys and brackets.
	Cables not in place on pulleys.	Install cables correctly.
	Cables crossed or routed incorrectly.	Check routing of control cables.
	Bent stabilator hinge.	Repair or replace stabilator hinge.
Incorrect stabilator travel.	Stabilator stops incorrectly adjusted.	Adjust stop screws.
Correct stabilator travel cannot be obtained by adjusting stops.	Stabilator cables incorrectly rigged.	Rerig stabilator cables.

CHART 2
TROUBLESHOOTING STABILATOR MANUAL TRIM CONTROL SYSTEM

Trouble	Cause	Remedy
Lost motion between trim control wheel and trim tab.	Cable tension too low.	Adjust cable tension.
	Cables not in place on pulleys.	Install cables properly.
	Broken pulley.	Replace pulley.
	Linkage loose or worn.	Check linkage and tighten or replace.
Trim control wheel moves with excessive resistance.	System not lubricated properly.	Lubricate system.
	Cable tension too high.	Adjust cable tension.
	Pulleys binding or rubbing.	Replace binding pulleys. Provide clearance between pulleys and brackets.
	Cables not in place on pulleys.	Install cables properly.
	Trim tab hinge binding.	Lubricate hinge. If necessary, replace.
	Cables crossed or routed incorrectly.	Check routing of control cables.
Trim tab fails to reach full travel.	System incorrectly rigged.	Check and / or adjust rigging.
	Trim drum incorrectly wrapped .	Check and/or adjust rigging.
Trim indicator fails to indicate correct trim position.	Trim indicator unit not adjusted properly.	Adjust trim indicator.

2. Stabilator Control Cables

A. Removal

See “Figure 3” on page 27305.

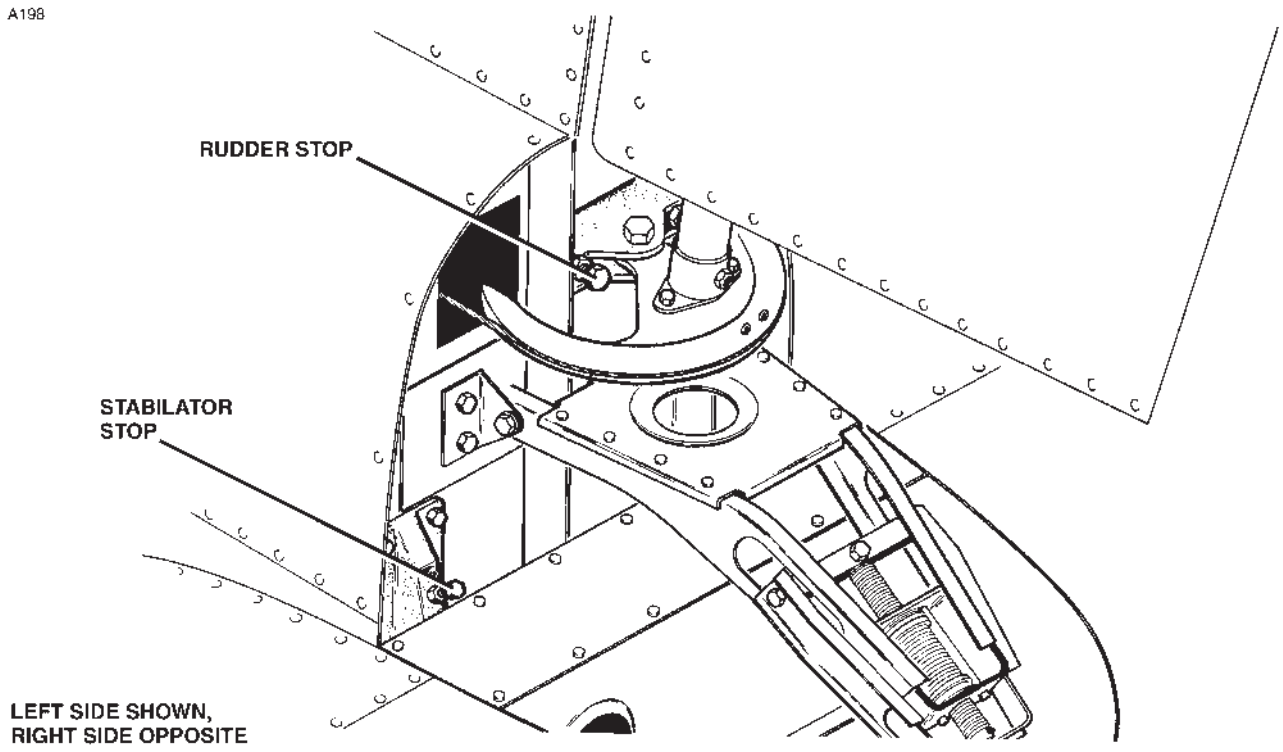
(1) To remove either the forward or aft stabilator cables, first remove the access panel to the aft section of the fuselage.

(2) Relieve cable tension from control system by loosening one of the cable turnbuckles in the aft section of the fuselage.

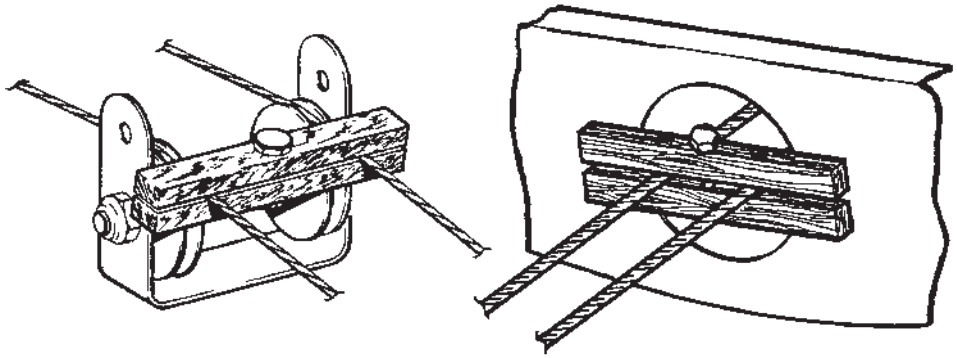
(3) Disconnect the stabilator down springs and clamps from the upper stabilator control cable in the aft section of the fuselage.

(4) Either forward stabilator cable may be removed by the following procedure:

(a) Remove the floor tunnel cover in the aft area of the cabin by removing the carpet and the heater duct over the tunnel and the cover attachment screws.



Rudder and Stabilator Adjustments
Figure 1



Method of Securing Trim Cables
Figure 2

(b) Remove the cable guard plate from the underside of the pulley cluster in the aft area of the tunnel opening by removing the guard attachment screws.

(c) Remove the floor and located directly aft of the main spar by removing the center seats, seat belt attachments, and the screws securing the panel. Lift the panel and remove from the airplane.

(d) Within the floor opening, remove the cable rub blocks that are attached to the spar housing by removing the block attachment screws. Also, remove the cotter pin cable guard at the pulley cluster in the aft area of the opening.

(e) Remove the tunnel plate just aft of the tee bar by removing enough carpet from the tunnel to allow the plate attachment screws and plate to be removed.

(f) If the right (upper) stabilator control cable is to be removed, remove the cotter pin cable guards at the pulley located in the forward area of the tunnel.

(g) Disconnect the cables from the lower end of the tee bar by removing cotter pin, nut, washer and bolt.

(h) Draw the cable aft through the floor tunnel.

(5) Either aft stabilator control cable may be removed by the following procedure:

(a) Disconnect the cable end at the balance arm of the stabilator by removing the cotter pin, nut, washer and bolt.

(b) Remove the cotter pin cable guard at the pulleys located either above or below the balance arm.

(c) Remove the cable from the airplane.

B. Installation

See “Figure 3” on page 27305 and “Figure 4” on page 27307.

(1) The forward stabilator cables may be installed by the following procedure:

(a) Draw the control cable through the floor tunnel. Ascertain that the right (upper) cable is routed around the pulley that is in the forward area of the forward floor tunnel.

(b) Connect the cables to the lower end of the control column tee bar with bolt, washer, nut and cotter pin. Allow the cable to be free to rotate.

(c) If the aft control cable is not installed, install as described in the next step.

(d) Connect the forward control cable to the aft cable at the turnbuckles in the aft section of the fuselage.

(e) For the right control cable, install the cotter pin cable guard at the pulley in the forward area of the tunnel.

(f) Within the forward area of the floor opening aft of the main spar, install the cable rub blocks to the spar housing and secure with screws.

(g) In the area of the floor opening, install the cotter pin cable at the pulley cluster.

(h) Install the cable guard plate under the pulley cluster located in the aft area of the aft floor tunnel and secure with screws.

(i) Set cable tension per “Figure 4” on page 27307 and check rigging and adjustment.

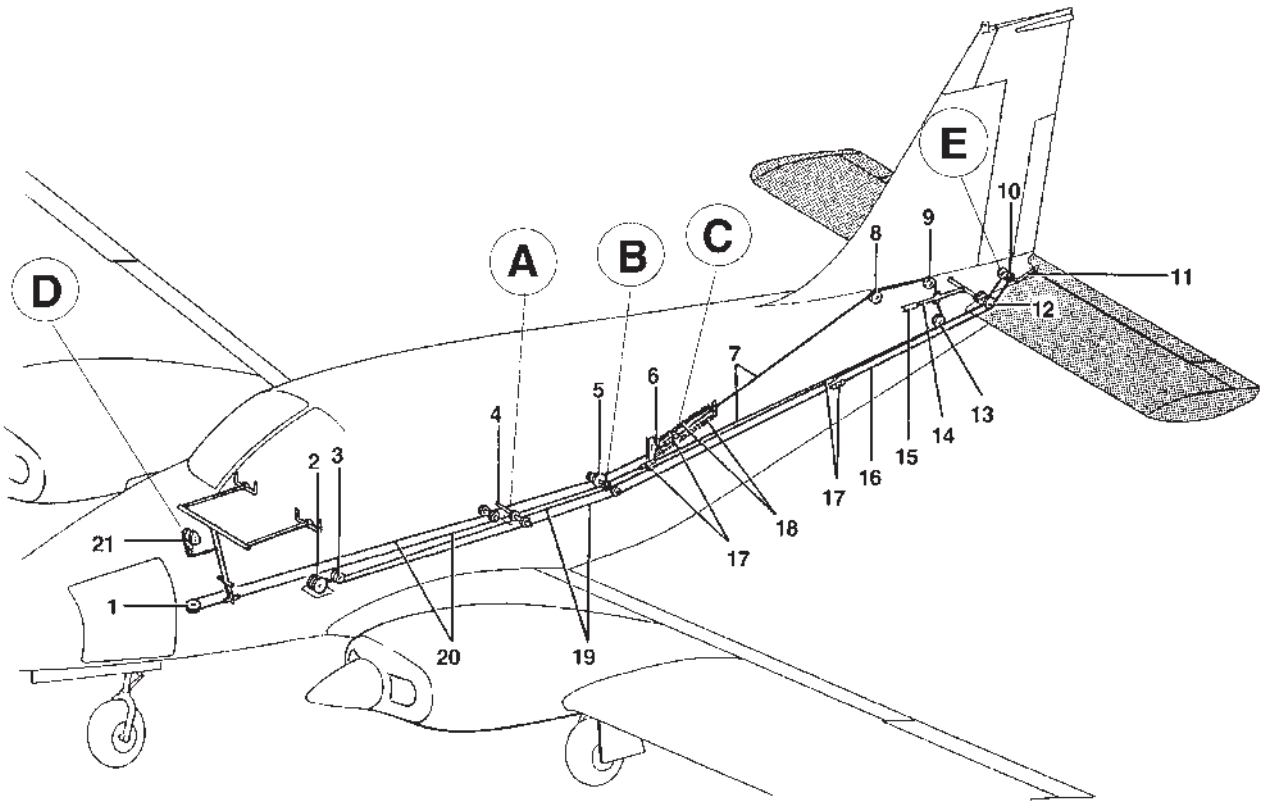
(j) Connect stabilator down springs and clamps to upper aft stabilator control cable. See “Figure 3” on page 27306, Sketch C.

(k) Install the tunnel plate directly aft of the tee bar assembly and secure with screws.

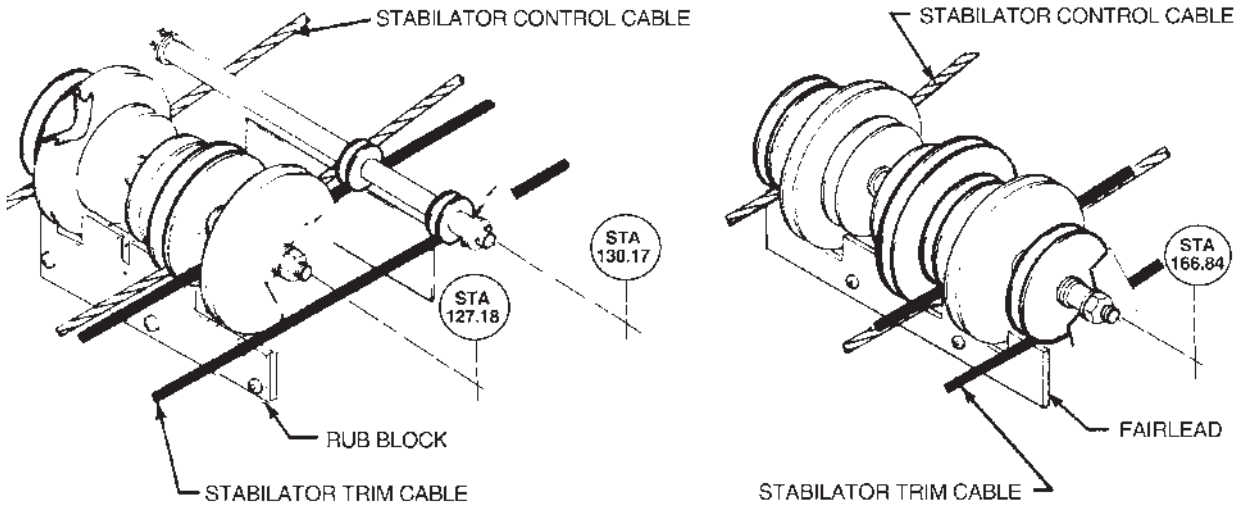
(l) Put the floor carpet in place and secure.

(m) Install the floor panel aft of the main spar and secure with screws. Install the seat belt attachments and seats.

(n) Install the cover, heat duct and carpet over the aft floor tunnel.



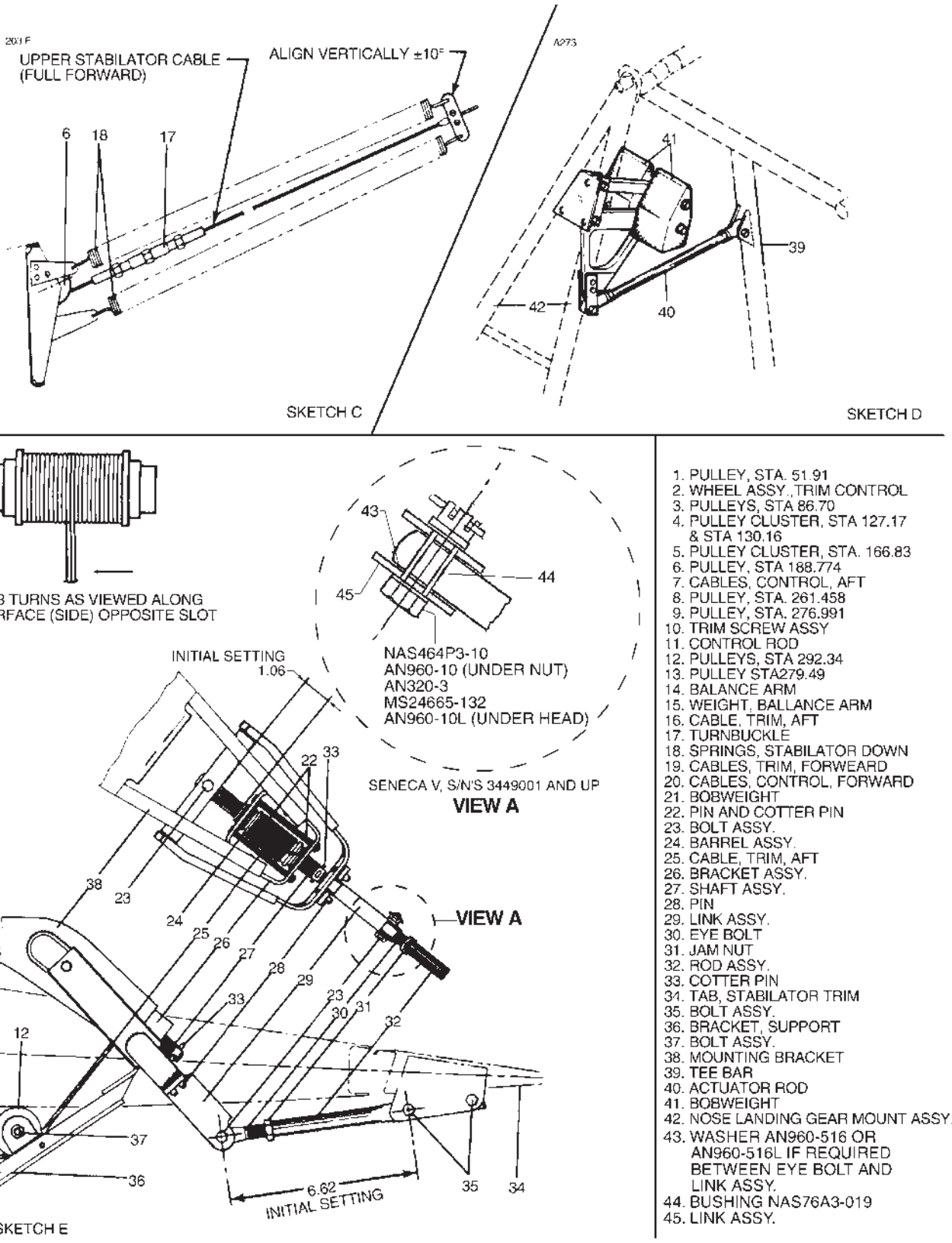
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SKETCH A

SKETCH B

Stabilator Controls Installation
Figure 3 (Sheet 1 of 2)



1. PULLEY, STA. 51.91
2. WHEEL ASSY., TRIM CONTROL
3. PULLEYS, STA 86.70
4. PULLEY CLUSTER, STA 127.17 & STA 130.16
5. PULLEY CLUSTER, STA. 166.83
6. PULLEY, STA 188.774
7. CABLES, CONTROL, AFT
8. PULLEY, STA. 261.458
9. PULLEY, STA. 276.991
10. TRIM SCREW ASSY
11. CONTROL ROD
12. PULLEYS, STA 292.34
13. PULLEY STA279.49
14. BALANCE ARM
15. WEIGHT, BALLANCE ARM
16. CABLE, TRIM, AFT
17. TURNBUCKLE
18. SPRINGS, STABILATOR DOWN
19. CABLES, TRIM, FORWEARD
20. CABLES, CONTROL, FORWARD
21. BOBWEIGHT
22. PIN AND COTTER PIN
23. BOLT ASSY.
24. BARREL ASSY.
25. CABLE, TRIM, AFT
26. BRACKET ASSY.
27. SHAFT ASSY.
28. PIN
29. LINK ASSY.
30. EYE BOLT
31. JAM NUT
32. ROD ASSY.
33. COTTER PIN
34. TAB, STABILATOR TRIM
35. BOLT ASSY.
36. BRACKET, SUPPORT
37. BOLT ASSY.
38. MOUNTING BRACKET
39. TEE BAR
40. ACTUATOR ROD
41. BOBWEIGHT
42. NOSE LANDING GEAR MOUNT ASSY.
43. WASHER AN960-516 OR AN960-516L IF REQUIRED BETWEEN EYE BOLT AND LINK ASSY.
44. BUSHING NAS76A3-019
45. LINK ASSY.

Stabilator Controls Installation
Figure 3 (Sheet 2 of 2)

(2) Either aft stabilator control cable may be installed by the following procedure:

(a) Route the cable around its pulley located either over or under the balance arm of the stabilator.

(b) Connect the cable to the stabilator balance arm and secure with bolt, washer, nut and cotter pin. (Ensure bushing is installed with bolt.)

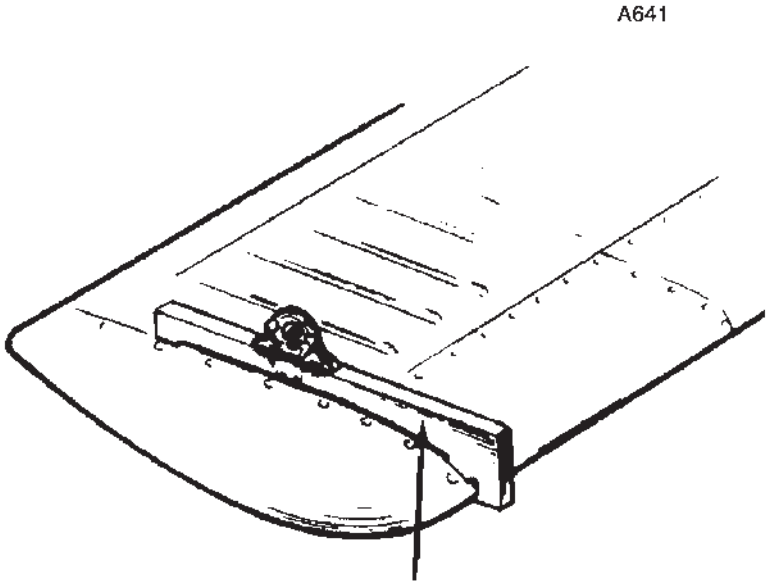
(c) Connect the aft cable to the forward cable at the turnbuckle in the aft section of the fuselage. The upper aft cable connects to the right forward cable and the lower cable to the left cable.

(d) Install the cotter pin cable guard at the pulley where required.

(e) Connect stabilator down spring to the upper aft control cable. (See “Figure 3” on page 27306, Sketch C).

(f) Set cable tension and check rigging and adjustment.

(3) Install the access panel to the aft section of the fuselage.



Stabilator Rigging
Figure 4

C. Rigging and Adjustment

See “Figure 3” on page 27305, “Figure 4” on page 27307 and “Chart 2” on page 27004.

The stabilator control system is designed to have the stabilator in neutral when the stabilator chord line is parallel to that of the front seat tracks. Travels as specified in “Chart 2” on page 27004, are measured from the neutral point.

WARNING: VERIFY FREE AND CORRECT MOVEMENT OF STABILATOR. WHILE IT WOULD SEEM SELF-EVIDENT, FIELD EXPERIENCE HAS SHOWN THAT THIS CHECK IS FREQUENTLY MISINTERPRETED OR NOT PERFORMED AT ALL. ACCORDINGLY, UPON COMPLETION OF STABILATOR RIGGING AND ADJUSTMENT, VISUALLY CONFIRM THAT THE STABILATOR TRAILING EDGE MOVES UP WHEN THE WHEEL IS PULLED BACK; AND, THAT THE STABILATOR TRAILING EDGE MOVES DOWN WHEN THE WHEEL IS PUSHED FORWARD.

(1) Before proceeding with any of the following instructions, level the airplane as described in 8-20-00.

NOTE: When adjusting cables, make sure they are evenly tensioned to prevent uneven strain on aircraft components. After any cable adjustments ensure that there is no interference between turn buckles and pulleys, and that all cables are aligned on their pulleys.

(2) Remove the tail cone, and adjust the stabilator stops to obtain the travels specified in “Chart 2” on page 27004. The stabilator travel can be determined using the rigging tool as shown and is described in “Figure 7” on page 273017. To check stabilator travel proceed as follows:

NOTE: Ensure the stabilator contacts both its stops (See “Figure 1” on page 27303) before the control column (tee bar) contacts its stops.

(a) Align the stabilator rigging tool on the upper surface of the stabilator as shown in “Figure 4” on page 27307.

(b) Position the stabilator in neutral and, using a bubble protractor, set the number of degrees up travel as specified in “Chart 2” on page 27004.

(c) Raise the trailing edge of the stabilator until the elevator reaches its stop. If the bubble is not centered, adjust the elevator stops to obtain the correct travel.

(d) Again place the stabilator in neutral and using the values from 27-00-00, “Chart 2” on page 27004, proceed with measuring the down angle.

(e) Make sure the lock nuts of the elevator stop bolts are secure.

(3) The control column, or tee bar, is designed to be in the neutral position when at a forward angle of 7° from the neutral position (See “Figure 2” on page 27106). Rig the bobweight and control column to the stabilator as follows:

(a) Move the control column to its neutral position and with cable clamps (“Figure 2” on page 27303) or other suitable tool, block the column in this position.

(b) Ascertain that the stabilator is in neutral, and disconnect the feel springs in the rear of the fuselage.

(c) Evenly adjust the cable tensions to those specified in “Chart 2” on page 27004.

(d) Adjust the control column-to-bobweight pushrod as necessary to achieve an angle of 23° +0°, -1° from a line projected from the lower bobweight link below a projected parallel line with the top of the front seat tracks. Use “Figure 2” on page 27106 for reference.

(e) Remove the control column blocking mechanisms.

(4) With the stabilator in neutral and the feel springs still detached proceed as follows to adjust the stabilator tab:

(a) Position the tab control in its neutral position.

(b) As necessary adjust the tab push rod to streamline the tab with the stabilator. This is the neutral position of the tab.

(c) Rotate the trim wheel and cross check the travels with those in “Chart 2” on page 27004.

(d) If adjustments need to be made to alter the tab travels, only adjustment of the rod end fitting on the tab actuating arm and the repositioning of the trim barrel screw are recommended.

(5) Check to ensure the stabilator travel stops are contacted before those of the control column (tee bar).

(6) Connect the stabilator feel springs.

(7) Secure the stabilator against one of its stops and ascertain that total tab free play does not exceed 0.06 inches.

(8) Make sure all cables are aligned in their pulleys and there is no interference throughout the entire stabilator control system.

D. Stabilator Post-Rigging Inspection

NOTE: Visual confirmation may require the aid of a second qualified mechanic or pilot.

Upon completion of any stabilator control cable replacement, disconnect, or rigging perform the following post-rigging inspection.

(1) Pull the pilot’s control wheel FULL AFT, as if to nose the airplane UP, until reaching the primary stop. Visually confirm the following:

The stabilator is deflected leading edge DOWN and trailing edge UP. The trailing edge of the stabilator will be noticeably and significantly ABOVE the leading edge.

(2) Push the pilot’s control wheel FULL FORWARD, as if to nose the airplane DOWN, until reaching the primary stop. Visually confirm the following:

The stabilator is deflected leading edge UP and trailing edge DOWN The trailing edge of the stabilator will be noticeably and significantly BELOW the leading edge.

(3) Repeat the above steps for the copilot controls.

(4) Make a logbook entry documenting completion of this inspection.

3. Stabilator Trim Controls

A. Forward Stabilator Trim Assembly

(1) Removal

See “Figure 3” on page 27305.

(a) To remove the trim control wheel assembly and / or the trim control cables, first remove the panel to the aft section of the airplane.

(b) If the aft trim cable is not to be removed, block the cables at the pulleys in the tail cone to prevent them from unwrapping from the trim drum. (See “Figure 2” on page 27303.)

(c) Loosen the cables if the trim control wheel is to be removed or disconnected if the cables are also to be removed. Do this at the trim cable turnbuckles in the aft section of the fuselage.

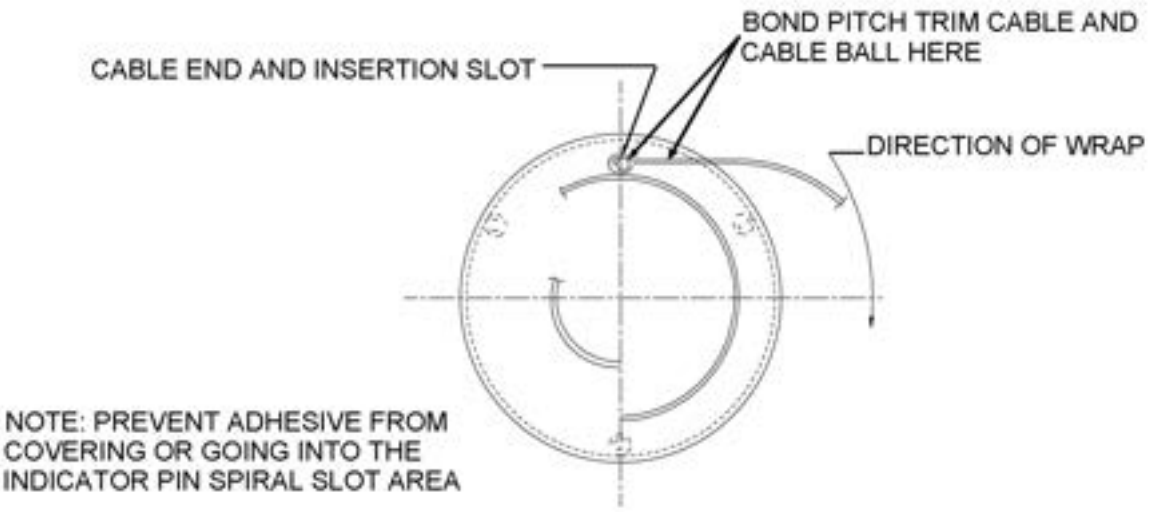
(d) The control wheel with drum may be removed by the following procedure:

1) Remove the control wheel cover by removing the cover attaching screws.

2) The wheel assembly may be removed from its mounting brackets by removing nut, washer, and bolt that secures the wheel between the brackets. Draw the wheel from the brackets. Use caution not to damage trim indicator wire.

3) Unwrap the left cable from the drum.
4) The wheel and drum are joined by three screws. Remove screws and separate these two items with their center bushing and unwrap the right cable.
5) Tie the cables forward to prevent them from slipping back into the floor tunnel.

(e) The trim control cables may be removed by the following procedure:
1) Remove the center seats and the pilot and rear seats if desired.
2) Remove the seat belts attached to the forward floor tunnel by removing attachment nuts, washers and bolts.
3) Unfasten the carpet from the aft portion of the forward floor tunnel and lay it forward.
4) Remove the tunnel cover located between the trim control wheel and the spar cover by removing attachment screws.
5) Remove the cable pulleys located in the forward tunnel by removing the cotter pin, washer and clevis pin.
6) Remove the floor panel aft of the main spar by removing the panel attachment screws and seat belt attachments. Lift the panel and remove from airplane.
7) Remove the cable rub blocks located in the floor opening on the aft side of the main spar by removing the block attachment screws.
8) Remove the carpet and the heater duct over the aft floor tunnel.
9) Remove the cover plate from the top of the aft floor tunnel by removing attachment screws.
10) Remove the cable guard (See “Figure 3” on page 27305, Sketch A) from the underside of the trim cable pulleys located at station 130.167 by removing the cotter pin and withdrawing the roll pin.
11) Remove the cable fairlead (See “Figure 3” on page 27305, Sketch B) from the underside of the pulley cluster located at station 166.837 by removing the plate attachment screws.
12) With the cables disconnected from the trim control wheel, draw the cable(s) through the floor tunnel.



Pitch Trim Cable Installation
Figure 5

(2) Installation

See “Figure 3” on page 27305.

(a) The trim control wheel with drum may be installed by the following procedure:

1) Wrap the right trim cable on the trim drum by inserting the swaged ball of the cable in the slot provided in the side (right side) of the drum that mates with the control wheel, and looking at this side, wrap the drum with three and a half wraps of the cable in a clockwise direction.
2) Attach the control wheel to the cable drum by aligning the long lug of the drum with the long slot of the wheel and securing the two pieces together with three screws.
3) Wrap the left trim cable on the drum by inserting the swaged ball of the cable in the slot provided in the flanged side (left side) of the drum and looking at this side, wrap the drum with three and a half wraps of the cable in a clockwise direction. Bond pitch trim cable and cable ball. See “Figure 5”.
4) Lubricate and install the bushing in the control wheel and drum.
5) Align the control cables and position the control wheel assembly between its mounting brackets. Ascertain that the end of the trim indicator wire is positioned in the spiraled slot of the drum with no bind on the end. Install the retainer bolt and washer from the left side and install washer and nut.
6) Install the cover over the control wheel and secure with screws, unless the control cables have yet to be installed.

(b) The trim control cables may be installed by the following procedure:

1) Draw the cable(s) through the floor tunnel.
2) Wrap the cable drum and install the trim control wheel as given in Step 1.
3) Position the cable pulleys on the mounting bracket and install the clevis pin, washer and cotter pin.
4) Connect the cable to the aft cable at the turnbuckle in the aft section of the fuselage. Install aft cable if not installed.
5) Install the cable fairlead (See “Figure 3” on page 27305, Sketch B) at the underside of the pulley cluster located at station 166.837 and secure with screws.
6) Install the roll pin type cable guard (See “Figure 3” on page 27305, Sketch A) at the underside of the pulleys located in the forward area of the aft floor tunnel and secure it with a cotter pin.
7) Install the cable rub blocks located on the aft side of the main spar housing and secure with screws.
8) Remove the blocks that secure the aft trim cable and check that the cables are seated on the pulleys.
9) Set the cable tension and check rigging and adjustment of the stabilator trim. Safety all turnbuckles.
10) Install the tunnel cover on the forward tunnel and secure with screws.
11) Install the carpet over the floor tunnel.
12) Install the cover over the trim control wheel and secure with screws and special washers.
13) Install the seat belts removed from the top of the floor tunnel and secure with bolt, washer and nut.

14) Install the floor panel and seat belt attachments aft of the main spar and secure panel with screws.

15) Install the aft floor tunnel and secure with screws.

16) Install the heater duct and carpet over the aft floor tunnel.

(c) Install the panel to the aft section of the airplane and the seats.

B. Aft Stabilator Trim Assembly

(1) Removal

See “Figure 3” on page 27305.

(a) Remove the access panel to the aft section of the fuselage.

(b) Block the trim cables at the first set of pulleys forward of the cable turnbuckles in the aft section of the fuselage.

(c) Remove the tail cone attachment screws and tail cone from the airplane.

(d) Block the cable at the trim barrel to prevent it from unwrapping at the barrel.

(e) Disconnect the cables at the turnbuckles.

(f) Remove the cable guard pins at the trim screw and also at the pulleys located below the trim mechanism at station 292.34.

(g) Remove the bolt assembly which connects the forward end of the trim screw with the link assembly.

(h) Unscrew the screw from the trim barrel.

(i) Remove the four machine screws securing the two parts of the bracket assembly to the mounting bracket.

(j) Separate the two parts of the bracket assembly and remove the trim barrel and cable. Note the amount and placement of washers at each end of the barrel to simplify reassembly.

(k) Remove the barrel and cables from the airplane.

(2) Installation

See “Figure 3” on page 27305.

(a) Wrap the trim barrel by first laying the center of the aft trim cable (as measured equally from each end to the center of the cable) in the slot of the barrel. Bring the upper cable through the diagonal slot in the flange at the upper end of the barrel and wrap down in a counterclockwise direction. Bring the lower cable through the diagonal slot in the lower end of the barrel and wrap up in a clockwise direction. Wrap the cable as evenly as possible to obtain 23 wraps on the barrel as viewed from the side opposite the slot and with the cables extending out from the slotted side.

(b) Block both cables by clamping them between two pieces of wood laid next to the wraps to prevent them from unwrapping.

(c) Install the barrel between the two parts of the bracket assembly. Be sure to install the washers at both ends of the barrel before installing it in the brackets.

(d) Secure the barrel and bracket assembly to the mounting bracket with the four machine screws.

(e) Install the screw into the barrel with the drilled bolt hole facing towards the front of the airplane.

(f) Position the stabilator and trim tab in a neutral position as described in the next paragraph, and adjust the trim screw until the bolt hole in the end aligns with the bolt hole in the yoke of the link assembly; then install and secure the bolt assembly.

(g) Route the cable around the pulleys at station 292.34 and forward to the turnbuckles in the fuselage.

(h) Ascertain that the cables are in the pulley grooves; then install the guard pins at the pulleys.

(i) Connect the cables to the turnbuckles and remove the blocking from both the forward and aft cables.

(j) Set cable tension in accordance with “Chart 2” on page 27004 and check rigging and adjustment.

(k) Install the tail cone and secure with screws.

(l) Install the access panel to the aft section of the fuselage.

C. Rigging and Adjustment

WARNING: VERIFY FREE AND CORRECT MOVEMENT OF STABILATOR TRIM TAB. WHILE IT WOULD SEEM SELF-EVIDENT, FIELD EXPERIENCE HAS SHOWN THAT THIS CHECK IS FREQUENTLY MISINTERPRETED OR NOT PERFORMED AT ALL. ACCORDINGLY, UPON COMPLETION OF STABILATOR TRIM RIGGING AND ADJUSTMENT, VISUALLY CONFIRM THAT THE STABILATOR TAB TRAILING EDGE MOVES UP WHEN THE TRIM WHEEL IS TRIMMED DOWN; AND, THAT THE STABILATOR TAB TRAILING EDGE MOVES DOWN WHEN THE TRIM WHEEL IS TRIMMED UP.

See “Figure 3” on page 27305, “Figure 4” on page 27307 and “Chart 2” on page 27004.

(1) Level the airplane.

(2) Remove the tail cone fairing from the fuselage by removing the attaching screws.

(3) Remove the access panel to the aft section of the fuselage.

(4) Secure the stabilator in its neutral position. To find neutral, place a rigging tool on the upper surface of the stabilator as shown in “Figure 4” on page 27307. Zero a bubble protractor on the top of the front seat tracks; then set it on the rigging tool and tilt the stabilator until the bubble is centered.

(5) The following items should be accomplished as a preadjustment check before proceeding with the rigging of the trim tab. If these items were accomplished during the installation, proceed with Step 6.

(a) Ascertain that the cable is wrapped 23 times around the barrel as shown in “Figure 3” on page 27305.

(b) The trim screw is adjusted to an initial length of 1.06 inches as shown in “Figure 3”.

(c) The actuating rod is initially adjusted to 6.62 inches in length as shown in “Figure 3”.

(d) Set the trim cable tension in accordance with 27-00-00, “Chart 2” on page 27004. If the cables were disconnected and replaced, rotate the control wheel several times to allow the cables to seat and then recheck the tension.

(6) Turn the trim control wheel until the trim tab streamlines with the neutral stabilator.

(7) Check the bubble of the protractor over the neutral tab and then check the tab travels as given in “Chart 2” on page 27004. The degree of travel on the protractor is determined by taking the difference between the protractor reading at neutral and up, and neutral and down. The bubble must be centered at each reading with the airplane level.

(8) If correct travels are not obtained, disconnect the actuating push rod from the trim screw and turn the rod end in or out as required.

(9) Reconnect the rod end and secure the jamb nut.

(10) With the trim tab operating at its proper extremes ascertain that there is no cable or turnbuckle interference, as well as binding or chafing.

(11) With the stabilator held at one of its stops, ensure that the trim tab free play does not exceed the limit given in 27-00-00, “Chart 2”.

(12) Reinstall the tail cone.

D. Stabilator Trim Post-Rigging Inspection

NOTE: Visual confirmation may require the aid of a second qualified mechanic or pilot.

Upon completion of any stabilator trim control cable or pushrod replacement, disconnect, or rigging perform the following post-rigging inspection:

(1) Rotate the stabilator trim wheel FULL AFT, as if to nose the airplane UP, until reaching the primary stop. Visually confirm the following:

The stabilator trim tab trailing edge is deflected DOWN. The trailing edge of the stabilator trim tab will be noticeably BELOW the trailing edge of the stabilator .

(2) Rotate the stabilator trim wheel FULL FORWARD, as if to nose the airplane DOWN, until reaching the primary stop. Visually confirm the following:

The stabilator trim tab trailing edge is deflected UP. The trailing edge of the stabilator trim tab will be noticeably ABOVE the trailing edge of the stabilator.

(3) Make a logbook entry documenting completion of this inspection.

4. Stall Warning System

The stall warning system for this aircraft is designed to detect stall conditions through two modes of flight operation. These conditions involve flight of the aircraft with flaps at 0° and 24°, or 25° and 40°. To accomplish this, the system utilizes two lift detectors, two micro switches, a horn, and a 5 amp circuit breaker. (A 15 amp circuit breaker is used when stall heat is installed.)

The two lift detectors are mounted in the left wing outboard of the left nacelle. The detectors are mounted such that as stalling conditions are approached with the flaps at 0° or 24°, the outboard detector gives the indication of stall, while the inboard detector gives the indication with flaps at 25° and 40°.

For the system to function, the lift detectors are interconnected to a microswitch which is operated by a cam on the flap torque tube. The mechanism is located inside the fuselage at the left wing root (See “Figure 6” on page 273015). As the flap torque tube turns, positioning the flap, the cam activates the switch locking out the particular lift detector tab.

To prevent operation of the stall warning system while on the ground, a squat switch, mounted on the left gear trunnion, opens the circuit as the gear is compressed. This switch is in line between the horn and the flap switch.

The electrical circuit is protected by a 5 amp circuit breaker mounted in the circuit breaker panel on the lower right side of the instrument panel. (A 15 amp circuit breaker is used when stall heat is installed.)

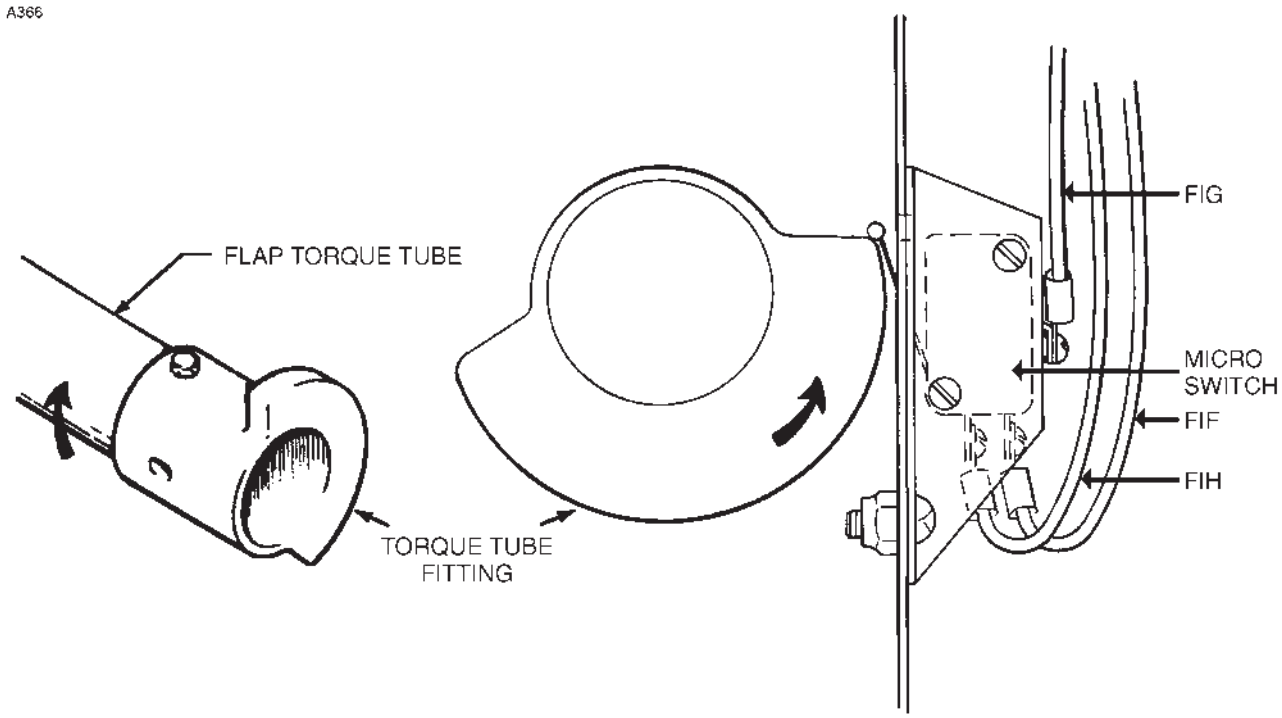
A. Troubleshooting

(1) Insert a wedge or other tool under the leaf of the micro switch (squat switch) on the left main gear. Check wiring for identification. See Chapter 91 for wiring diagrams.

(2) Ensure the flaps are in the full up position and switch on the master switch.

(3) Using light finger pressure, gently raise the sensor blade of the outboard lift detector until the horn comes on. Gently lower the blade and the horn should be deactivated. Move the flap to 10° and check with the same procedure. In the same manner, ensure the inboard detector is also deactivated.

(4) Extend the flaps to 25° and 40° and as previously described, check that the outboard lift detector is deactivated and the inboard lift detector activated.



Stall Warning Flap Microswitch Adjustment
Figure 6

(5) If the stall system fails to operate properly make sure the master switch is off and check the system as follows:
  (a) Using the wiring diagram in Chapter 91, check for proper continuities.
  (b) If the flap micro switch is out of adjustment see 27-50-00 and access the torque tube. With the flaps full up determine that the micro switch actuator is in contact with the fitting on the torque tube (“Figure 6” on page 273015), move the flaps to the 25° down position. Listen for switch activation and adjust the switch as necessary.

(6) Repeat Steps (3) and (4). If system still does not function properly, check continuity at flap switch terminals and throughout system.

B. Lift Detector

(1) Removal
  (a) Remove four screws holding the unit in place and remove the unit from the wing.
  (b) Identify the electrical leads to facilitate reinstallation. Disconnect the electrical leads.

(2) Installation
  (a) Attach electrical leads to the appropriate terminals of the lift detector.
  (b) Position the unit on the wing, determining that the sensor blade of the unit drops down freely, and secure in position with four screws previously removed.
  (c) Perform “Flight Test Procedure” on page 273016.

C. Flight Test Procedure

(PIR-FTP2001-12, Rev. J / FTP2001-12-1, Rev. E / FTP2010-3, Rev. D/ FTP2013-2, Rev. A, FTP2014-1, Rev. B.)

NOTE: The following procedure is typically performed after installation of either lift detector or installation of a replacement wing.

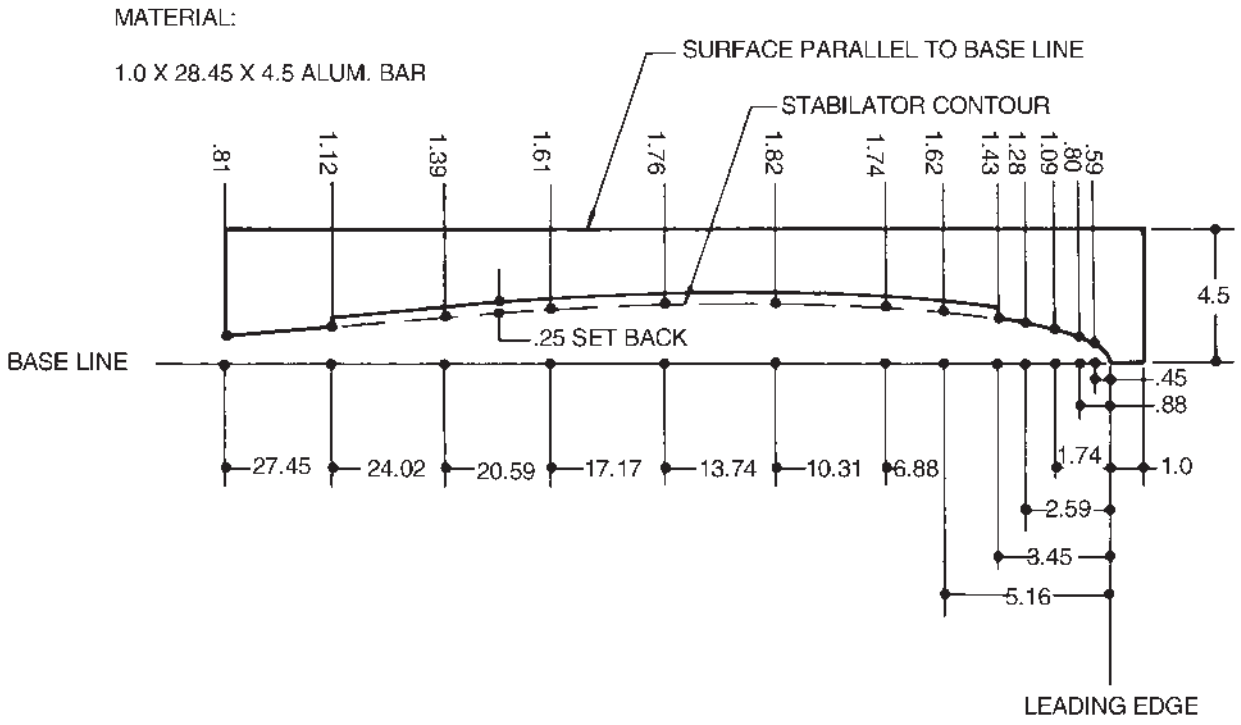
Fly airplane, and at a rate of 1 KT/SEC maximum, record the following warning and stall speeds:

	TRIM KIAS	RECORD:	WARNING KIAS	STALL KIAS
0° FLAPS =	100		_____	_____
FULL FLAPS =	94		_____	_____

(1) Verify that the stall warning system performs within the following limits:
   (a) Maximum: stall warning begins at 5 to 10 KIAS prior to stall and continues until stall occurs.
   (b) Desired: stall warning begins at 7 to 9 KIAS prior to stall and continues until stall occurs.

(2) If the stall warning system fails to perform within the limits above, adjust the system and refly the test.

(3) If not able to adjust the stall warning system sufficient to achieve the limits above, recheck any newly installed component(s) (lift detector(s) or wing) to ensure it (they) is (are) installed and rigged correctly.



Stabilator Rigging Tool
Figure 7

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FLAP CONTROLS

1. Troubleshooting

   "Chart 1" lists troubles peculiar to flap control system along with their probable causes and suggested remedies. When troubleshooting the flap control system, additional reference may be obtained from Chapter 57 on control surface balancing, if required. After the trouble has been corrected, check the entire flap control system for security and operation.

2. Wing Flaps

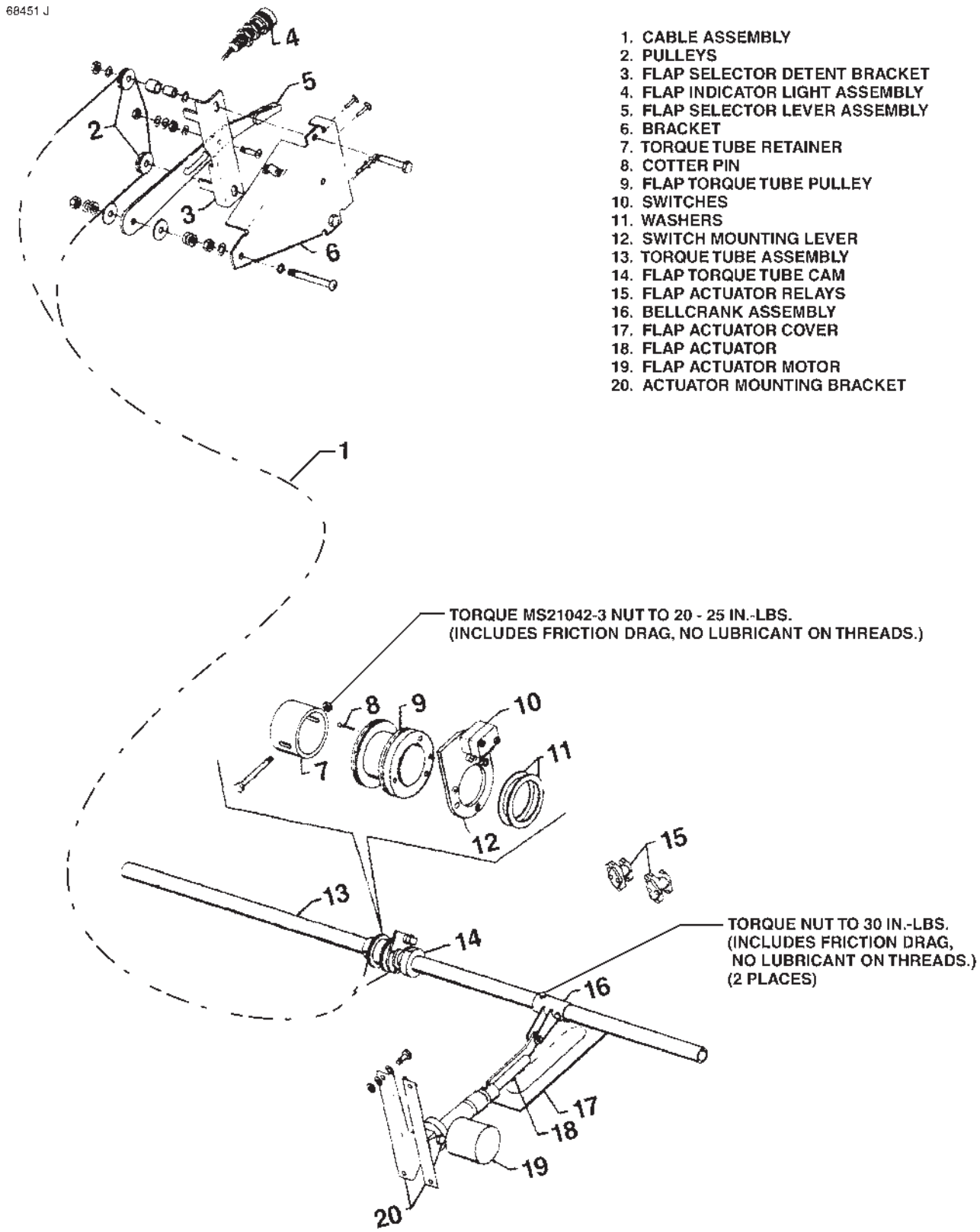
   A. Removal

      See "Figure 1" on page 27502.

      (1) To remove flap torque tube assembly:
         (a) Extend flaps to 40° position.
         (b) Remove floor panel located aft of main spar by removing center seats, seat belt attachments and screws securing panel. Lift panel and remove from airplane.
         (c) Remove access plate located between underside of aft section of each wing and fuselage by removing attaching screws.
         (d) Disconnect left and right flap control tubes (rods) either:
            1) At the flaps by removing nuts, washers and bolts at the torque tube cranks (arms) or by;
            2) Removing bolts and washers from inner side of each crank. Remove bolt through a hole in the fuselage side skin located over torque tube.
         (e) Disconnect electrical connections from limit switches mounted to torque tube switch plate.
         (f) Disconnect cable ends from torque tube pulley assembly by removing the cotter pins.
         (g) Disconnect jack screw actuator from torque tube bellcrank by removing nut, washers and bolt.
         (h) Remove tube support bearing blocks by removing block attachment bolts.
         (i) Remove nuts, washers and bolts securing right and left cranks, and stop fittings on torque tube.
         (j) From between each wing and fuselage, remove cranks from torque tube.
         (k) Disconnect one bearing block from its mounting brackets by removing nuts, washers and bolts.
         (l) Slide tube from bearing block still attached to its brackets. Raise end and lift it from floor opening.

CHART 1
TROUBLESHOOTING FLAP CONTROL SYSTEM

Trouble	Cause	Remedy
Flaps fail to extend or retract.	Motor or actuator inoperative	No electrical power to motor. Check circuit breaker.
Flaps not synchronized or fail to move evenly when retracted.	Incorrect rigging of system.	Adjust flaps.



Flap Control System Installation
Figure 1

(2) The flap control cable may be removed by the following procedure:
   (a) If the center seats and floor panel have not been removed, remove the seats and the screws securing the floor panel.
   (b) Remove the aft heat deflectors on each forward floor tunnel by sliding them far enough to release the spring fasteners.
   (c) Lift the aft section of the tunnel carpet far enough to remove the screws securing the tunnel cover and remove the cover.
   (d) If not previously accomplished, remove the cotter pins securing the cable ends to the pulley assembly on the torque tube; and the clamps securing the cable housings to the support bracket.
   (e) Disconnect the selector lever and cable from the selector lever support bracket mounted on the aft side of the instrument panel.
   (f) Remove the cable assembly from the tunnel.

(3) The jack screw and motor assembly may be removed by the following procedure.
   (a) Remove the center seats and floor panels.
   (b) Disconnect the electrical leads to the motor.
   (c) If not previously accomplished, remove the nut, washers and bolt securing the screw jack actuator to the torque tube bellcrank.
   (d) Remove the nut, washers and bolt securing the jack screw to its mounting bracket. Do not drop the bushing in the jack screw mounting end.

B. Installation

See “Figure 1” on page 27502.

(1) To install flap actuator jack screw and motor assembly:
   (a) Position flap actuator jack screw and motor assembly through center floor opening. Do not drop bushing in jack screw mounting end.
   (b) Install nut. washers and bolt securing flap actuator jack screw to its mounting bracket.
   (c) Install nut, washers and bolt securing flap actuator jack screw to torque tube bellcrank.
   (d) Connect electrical leads to flap actuator motor.

(2) To install flap control cable:
   (a) Position cable assembly in tunnel.
   (b) Connect cable to flap selector lever and flap selector lever support bracket mounted on aft side of instrument panel.
   (c) Attach cable ends to pulley assembly on torque tube by installing cotter pins .
   (d) Install clamps securing cable housings to support bracket.
   (e) Install aft section tunnel cover and secure with screws.
   (f) Install tunnel carpet and heat deflectors,
   (g) Install center seats.

(3) To install flap torque tube assembly:
   (a) Check that one end bearing block fitting is installed between its attachment brackets.
   (b) Slide the other end bearing block over its respective end of torque tube.
   (c) Position torque tube by placing end with bearing block attached between appropriate mounting bracket Slide other end into previously attached end bearing block. Secure with bolts, washers and nuts.
   (d) Between each wing and fuselage, attach cranks to torque tube.

(e) Install nuts, washers and bolts securing right and left cranks, and stop fittings on torque tube.

(f) Install tube support bearing blocks. Secure by installing block attachment bolts.

(g) Connect jack screw actuator to torque tube bellcrank and secure with nut, washers and bolt.

(h) Connect cable ends to torque tube pulley assembly and secure with cotter pins.

(i) Connect electrical connections to limit switches mounted to torque tube switch plate.

(j) Connect left and right flap control tubes (rods) either:

1) At flaps by installing nuts, washers and bolts at torque tube cranks (arms) or by;

2) Installing bolts and washers to inner side of each crank. Install bolt through a hole in the fuselage side skin located over torque tube.

(k) Install access plate located between underside of aft section of each wing and fuselage by installing attaching screws.

(l) Install floor panel located aft of main spar and secure with screws

(m) Install center seats and seat belt attachments.

(n) Retract flaps.

C. Control Cable Rigging

WARNING: VERIFY FREE AND CORRECT MOVEMENT OF FLAPS. WHILE IT WOULD SEEM SELF-EVIDENT, FIELD EXPERIENCE HAS SHOWN THAT THIS CHECK IS FREQUENTLY MISINTERPRETED OR NOT PERFORMED AT ALL. ACCORDINGLY, UPON COMPLETION OF FLAP RIGGING AND ADJUSTMENT, VISUALLY CONFIRM THAT THE FLAP TRAILING EDGES MOVE UP WHEN THE SELECTOR LEVER IS UP; AND, THAT THE FLAP TRAILING EDGES MOVE DOWN WHEN THE SELECTOR LEVER IS DOWN.

(1) Loosen lever cable clamp nut so that cable can move freely through cable clamp. Secure lever in the full down position. (See “Figure 2” on page 27505, Sketch B.)

(2) Position the swash plate assembly on torque tube and secure in place. (See “Figure 3” on page 27505.)

(3) Loosen cable housing clamps at pulley support channel. Adjust cable tension so that a 5 ± 0.5 pound pull midway between cable housing clamps and swash plate assembly will deflect the cable 0.38 inch from relaxed position. Tighten cable housing clamp. (See “Figure 3” on page 27505.)

(4) Tighten lever cable lamp nut so that cable is compressed to 1/2 its full diameter. (See “Figure 2” on page 27505, Sketch B.)

D. Cam Adjustment

(1) Apply power to the aircraft buss.

(2) Move the flap selector lever, then pull the electric flap circuit breaker so that there is clearance between the torque tube and the actuator pushrod. (See “Figure 2” on page 27505, Sketch A.)

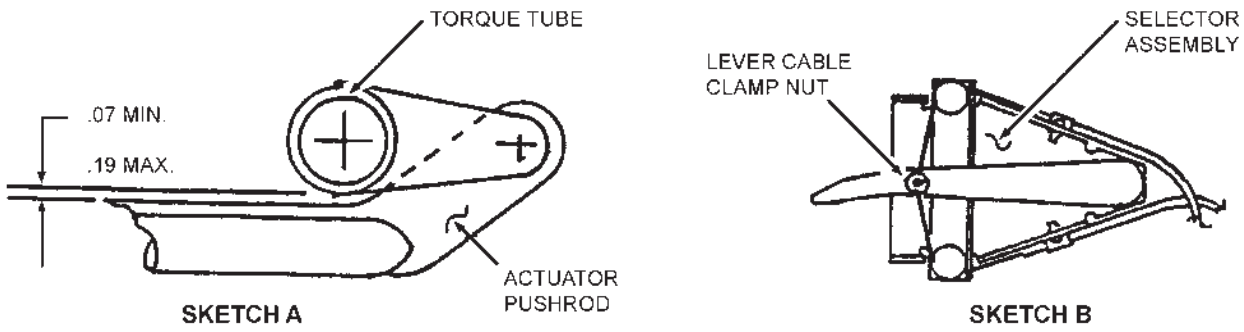
(3) Loosen the set screw in the cam and rotate until the flap in-transit light is out. (See “Figure 3” on page 27505, Sketch 1.)

NOTE: Ensure that the switch rollers are in the cam “null point” as shown in “Figure 3” on page 27505, Sketch 2 and not 180 degrees off.

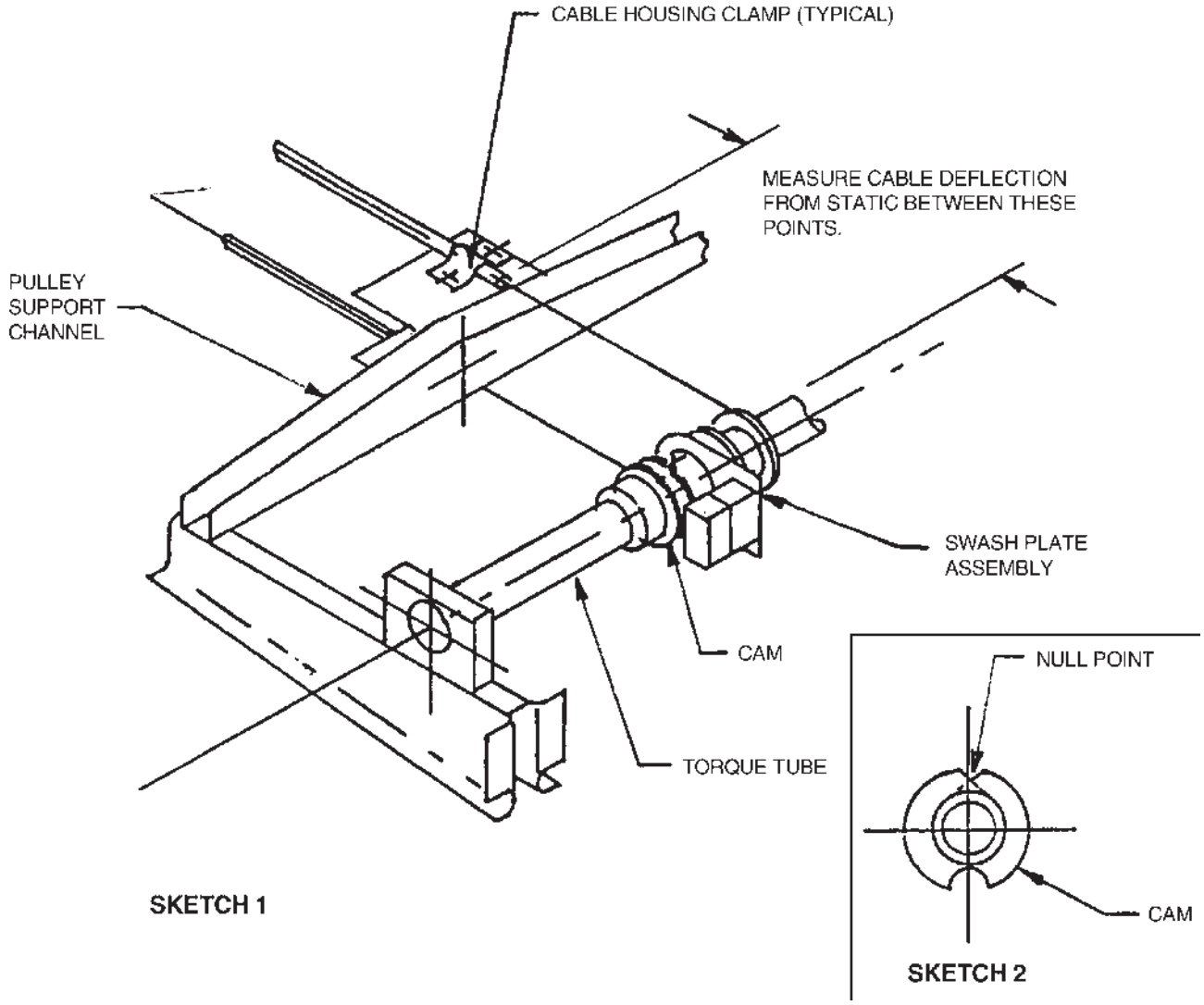
(4) Tighten the cam set screws.

(5) Push in the electric flap circuit breaker and verify that the flap motor does not run.

(6) Move the selector lever to the full “down” position and verify that the actuator retracts and stops about 0.4 inch short of bottoming out.



Rigging of Electric Flap Controls
Figure 2



Rigging of Flap Cam-Cable Assembly
Figure 3

(7) Move the selector lever to the full UP position and verify that clearance between the actuator pushrod and torque tube is 0.07 inches minimum, 0.19 inches maximum. See “Figure 2” on page 27505, Sketch A.

(8) Adjust the left and the right stop screws so that contact is just made with the stop. Tighten the jam nuts. (See “Figure 4” on page 27507.)

E. Flap Angle Setting

NOTE: While making this adjustment, maintain a slight “up” pressure on the underside of the flap sufficient to take the slack out of the linkage.

With the flap selected UP, adjust each flap pushrod so that the chord line of the flap forms a 0° ± 1° angle with the wing chord at the outboard end of the flap.

F. Flap Travel Check

While maintaining a light “up” pressure on the underside of the flap, check that the flap travel is:

(1) 0° ± 1° in full UP position.

(2) 10° ± 2° at the FIRST stop.

(3) 25° ± 2° at the SECOND stop.

(4) 40° ± 2° at the full DOWN position.

G. Flap Indicator Calibration (Seneca V only)

WARNING: EXCEEDING THE TRAVEL LIMIT WILL DESTROY THE POTENTIOMETER.

NOTE: Steps (4) and (5) are not applicable to G1000 equipped aircraft. To verify flap proper position indication on G1000/G1000 NXi aircraft, complete steps (1), (2) and (3), disconnect the Ohmmeter and plug the assembly into the harness. Then see Flaps Calibration in 34-25-01 or 34-25-02 as appropriate.

After the flaps have been rigged completely, ensure that external power is CONNECTED AND ONLINE. Indicator CANNOT be adjusted and WILL NOT indicate correctly without 28 volts on the system. A regulated external supply of 28 to 28.5 Volts must be used.

Calibrate the flap indicator as follows:

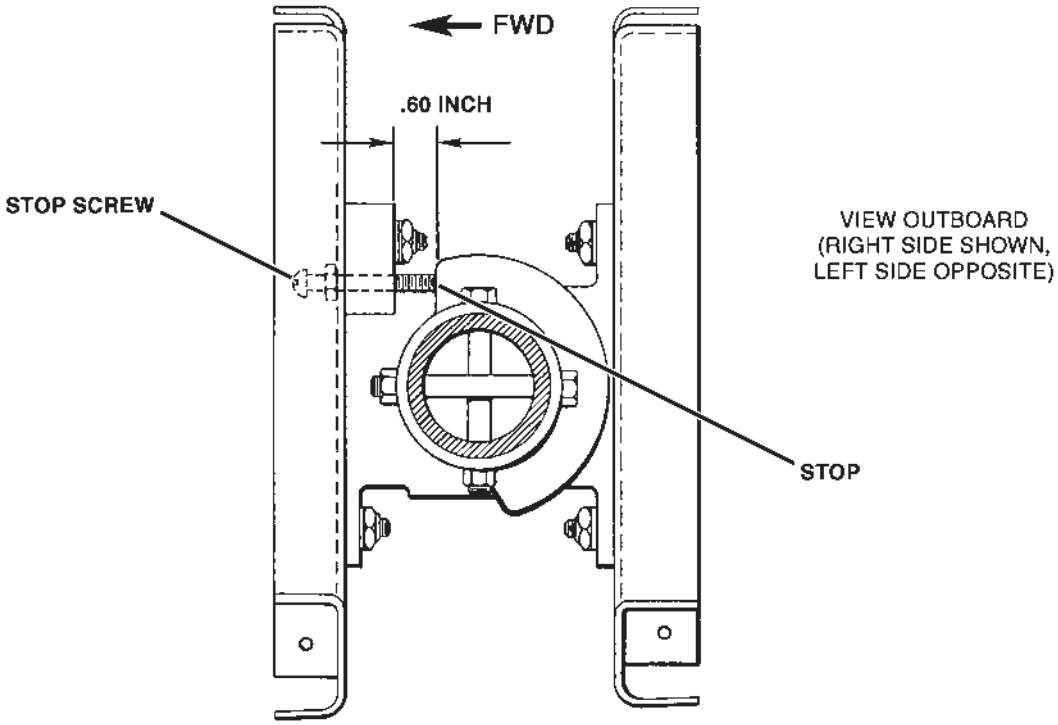
(1) With the aircraft power off, loosen the adjustment lock screw. Loosen the flap potentiometer (pot) mounting nut and tighten to finger tight. (See “Figure 5” on page 27507.)

(2) Unplug the flap pot assembly from the harness and connect a digital ohmmeter to the leads. Ensure the 28.0–28.5 Volt external power unit is connected and on line. Power up the aircraft and set the flaps to 40°. Read the resistance of the pot. If it does NOT read 97.4 ohms, move the adjustment plate to achieve 97.4 ohms (maximum) and tighten the adjustment lock screw. Tighten the pot mounting nut and verify reading. Repeat adjustment as necessary.

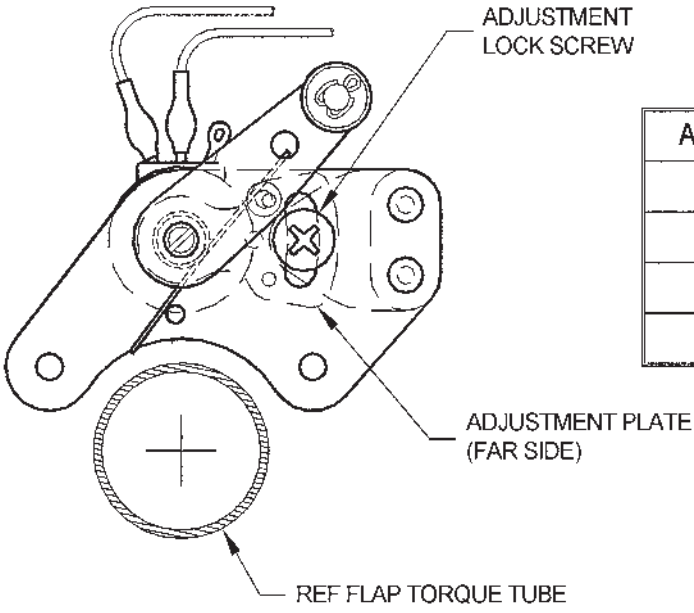
(3) Move the flap lever to 25°,10°,and 0° verify ohm readings for each (See “Figure 5” on page 27507). If readings are NOT with in limits, repeat steps (1) and (2). This will ensure that the pot travel limit is never exceeded. Exceeding the travel limit will destroy the pot. Power down the aircraft.

(4) Disconnect the ohmmeter and plug the assembly into the harness. Ensure regulated external supply is connected and online. Power up the aircraft and verify that the indicator reads correctly for all positions.

(5) The needle must be inside of the degree markings at 0°, 10°, 25°, and 40°. If the voltage is correct and the indicator is not within the marks and the resistances are to “Figure 5” on page 27507, the flap indicator is out of tolerance and must be replaced.



Stop Screws Adjustment
Figure 4



ANGLE	RESISTANCE
0°	LESS THAN 13 OHMS
10°	49 ± 2 OHMS
25°	75.6 ± 2 OHMS
40°	97.4 +0/-2 OHMS

Flap Potentiometer Adjustment
Figure 5

H. Flaps Post-Rigging Inspection

NOTE: Visual confirmation may require the aid of a second qualified mechanic or pilot.

Upon completion of any flap actuator, cable, pushrod, or torque tube replacement, disconnect, or rigging, perform the following post-rigging inspection.

(1) Position the flap control lever in the flaps DOWN position. Visually confirm the following:

Both flaps move fully DOWN when electrical power is applied and remain ALIGNED. That is, the trailing edge of each flap is significantly LOWER than when streamlined with the wing.

(2) Position the flap control handle to the flaps UP position. Visually confirm the following:

Both flaps move fully UP when electrical power is applied and remain ALIGNED. That is, the trailing edge of each flap is streamlined with the wing.

(3) Make a logbook entry documenting completion of this inspection.

3. Torque Tube/Push Rod Distortion Inspection

If flaps have been extended at or above V_{FE}, inspect the flap torque tube arms and pushrods for evidence of distortion.

A. If the paint is cracked or peeling anywhere along the torque tube arm or pushrod, torsional movement has occurred.

B. Remove the paint and inspect for cracks:

(1) In the welds at the arm on the torque tube end.

(2) In the rod ends and pushrod tube.

(3) Use a dye penetrant method of inspection.

C. If cracks are not found, repaint the part(s) and reinstall.

D. If cracked, replace the affected part(s).

CHAPTER

28

FUEL

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CHAPTER 28

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GENERAL

This chapter provides information for the maintenance of repairable components of the fuel system. A troubleshooting section is included to assist in isolating and correcting troubles which may occur.

1. Description and Operation

The fuel system is actually made up of two individual systems, one for each engine. Crossfeed lines are included, providing a source of fuel from one side to the other when needed and selected.

The standard system consists of two aluminum fuel tanks and a bladder cell in each wing. The aluminum fuel tanks form an integral part of the wing surface. The two aluminum fuel tanks and bladder cell in each wing are interconnected providing each wing with a capacity of 64 U. S. gallons, giving a total capacity of 128 U. S. gallons.

Fuel quantity transmitters are mounted only in the inboard and outboard aluminum tanks. The transmitters are interconnected on their respective side to a single fuel quantity gauge with dual (LEFT and RIGHT) indications. The gauge senses the total resistances sent by the transmitters and thereby indicates the level of fuel.

With the tanks interconnected, fuel for the respective engine is taken off the inboard tank. Fuel, drawn by one of the engines or electric pumps, leaves the inboard tank through a finger strainer to the fuel selector valve. The fuel is directed by the valve either to crossfeed or to its specific engine. From the selector valve, fuel goes through a filter to the electric pump and into the engine driven pump which supplies fuel pressure to the metering unit.

In Seneca IV models, a dual fuel flow gauge showing both left and right flow rates is located on the pilot's side of the instrument panel. The gauge is connected to a vent line from the intake manifold and to the fuel manifold valve.

In Seneca V models, two fuel flow gauges are located on the pilot's side of the instrument panel. The gauges are connected electrically to a flow meter mounted between the engine driven fuel pump and the fuel injector.

Each engine has an engine driven fuel pump that is part of the fuel injection system. An auxiliary fuel system is provided to supply fuel to the engine in case of engine driven fuel pump failure or malfunction, for ground and inflight starting, and for vapor suppression. On Seneca IV models, the two auxiliary fuel pump switches are the three position, rocker type (LO, HI and center OFF), and are located on the lower left instrument panel. The LO auxiliary fuel pressure is selected by pushing the top of the switch. The Hi auxiliary fuel pressure is selected by pushing the bottom of the switch. To prevent accidental activation of the Hi position, a switch guard must be unlatched before the switch is placed in the Hi position. When the Hi auxiliary fuel pump is activated, an amber light near the annunciator panel is illuminated for each pump. These lights dim whenever the pump pressure reduces automatically due to the manifold pressure dropping below approximately 21 inches. On Seneca V models, the two auxiliary fuel pump switches are the two position (ON-OFF), rocker type, and are located on the overhead switch panel.

In Seneca IV models, in case of a failed engine driven fuel pump, auxiliary fuel pressure may be selected. Adequate pressure and fuel now will be supplied for up to approximately 75% power. Manual leaning to correct fuel flow will be required at altitudes above 15,000 feet and for rpm's less than 2300. An absolute pressure switch automatically selects a lower fuel pressure when the throttle is reduced below 21 in. Hg manifold pressure and the Hi auxiliary fuel pump is on.

In Seneca V models, the auxiliary fuel system will supply fuel for up to 100% power. No manual leaning is required. There is no absolute pressure switch.

CAUTION: (SENECA IV MODELS) EXCESSIVE FUEL PRESSURE AND VERY RICH FUEL/AIR MIXTURES WILL OCCUR IF THE HI POSITION IS ENERGIZED WHEN THE ENGINE FUEL INJECTION SYSTEM IS FUNCTIONING NORMALLY.

In Seneca IV models, low auxiliary fuel pressure is available and may be used during normal engine operation both on the ground and in flight for vapor suppression should it be necessary as evidenced by unstable engine operation during idle or at high altitudes.

Primer operation is accomplished with two separate spring-loaded button type switches, located adjacent to the starter switches. These switches are used to select the HI auxiliary fuel pump operation for priming, irrespective of other switch positions. These primer buttons may be used for both hot or cold engine starts.

2. Troubleshooting

Troubles peculiar to the fuel system are listed in “Chart 1” on page 28003 along with their probable causes and suggested remedies. When troubleshooting, check from the power supply to the items affected. If no trouble is found by this method, the trouble probably exists inside individual pieces of equipment: they may be removed from the airplane and an identical unit or units, tested and known to be good, installed in their place.

NOTE: Refer to 71-00-00, Troubleshooting, for additional fuel system troubleshooting.

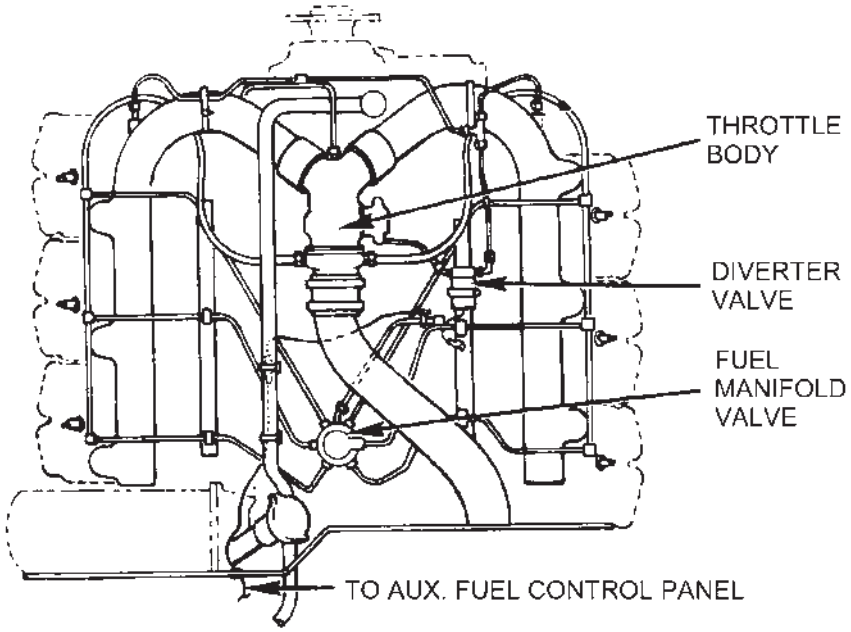
3. Inspection of Fuel System

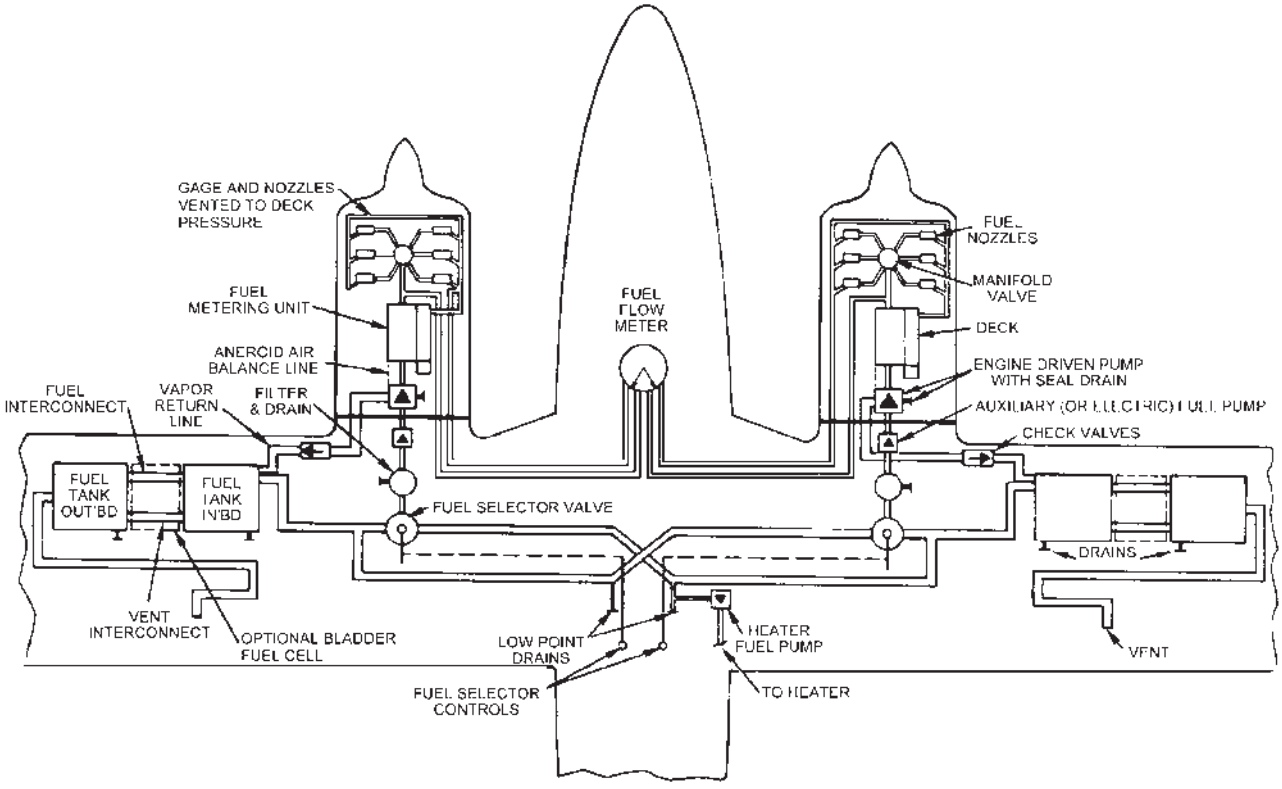
Fill tanks with fuel. Inspect tanks and fuel line connections for leaks. If fuel tanks leak, follow instructions given in “Repair” on page 28102 for aluminum fuel tanks or “Repair” on page 28105 for bladder type fuel cells. If fuel line connections leak, tighten clamps or replace hose connections after first draining tanks.

See also “Fuel System Leakage Inspection” on page 28109.

CHART 1
TROUBLESHOOTING FUEL SYSTEM

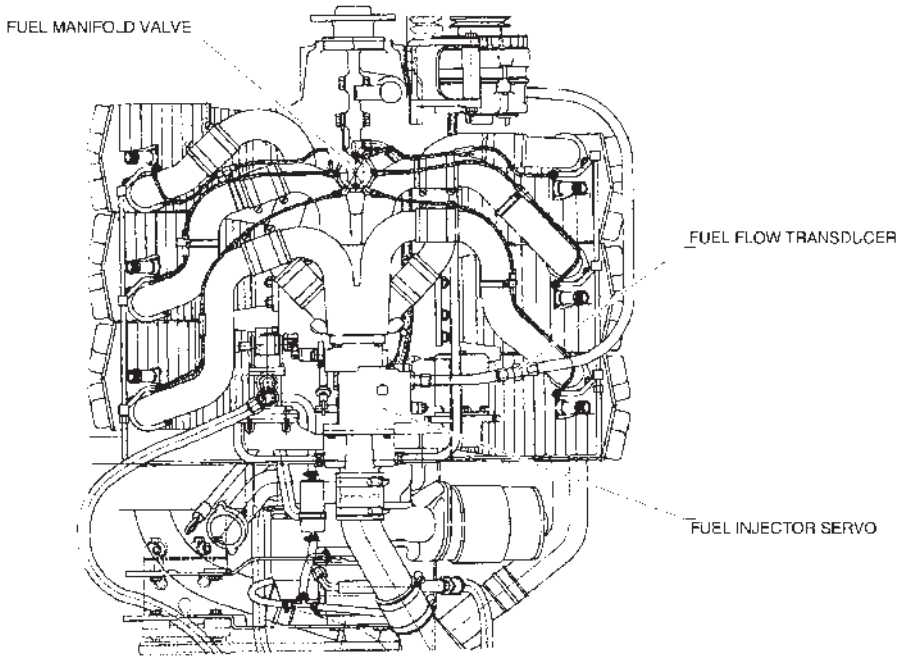
Trouble	Cause	Remedy
Failure of fuel to flow.	Fuel line blocked.	Flush fuel system.
	Fuel vent cap blocked.	Check and clean vent hole in cap.
	Mechanical or electrical fuel pump failure.	Check and replace if necessary.
	Fuel selector valve in improper position.	Reposition as required.
		Check for obstructions in the fuel selector leverage mechanism.
		Check fuel selector cable for freedom of movement.
	Damaged fuel selector valve.	Replace fuel selector valve.
Fuel quantity gauge fails to operate.	Broken wire.	Check and repair.
	Gauge inoperative.	Replace gauge.
	Fuel sender float partially or completely filled with fuel.	Replace sender.
	Circuit breaker open.	Check and reset.
	Float and arm assembly of fuel sender sticking.	Check.
	Bad ground.	Check for good contact at ground lip or rear of gauge.
Low pressure or pressure surges.	Obstruction in inlet side of pump.	Trace lines and locate obstruction.
	Air in line to pressure gauge.	Bleed line.

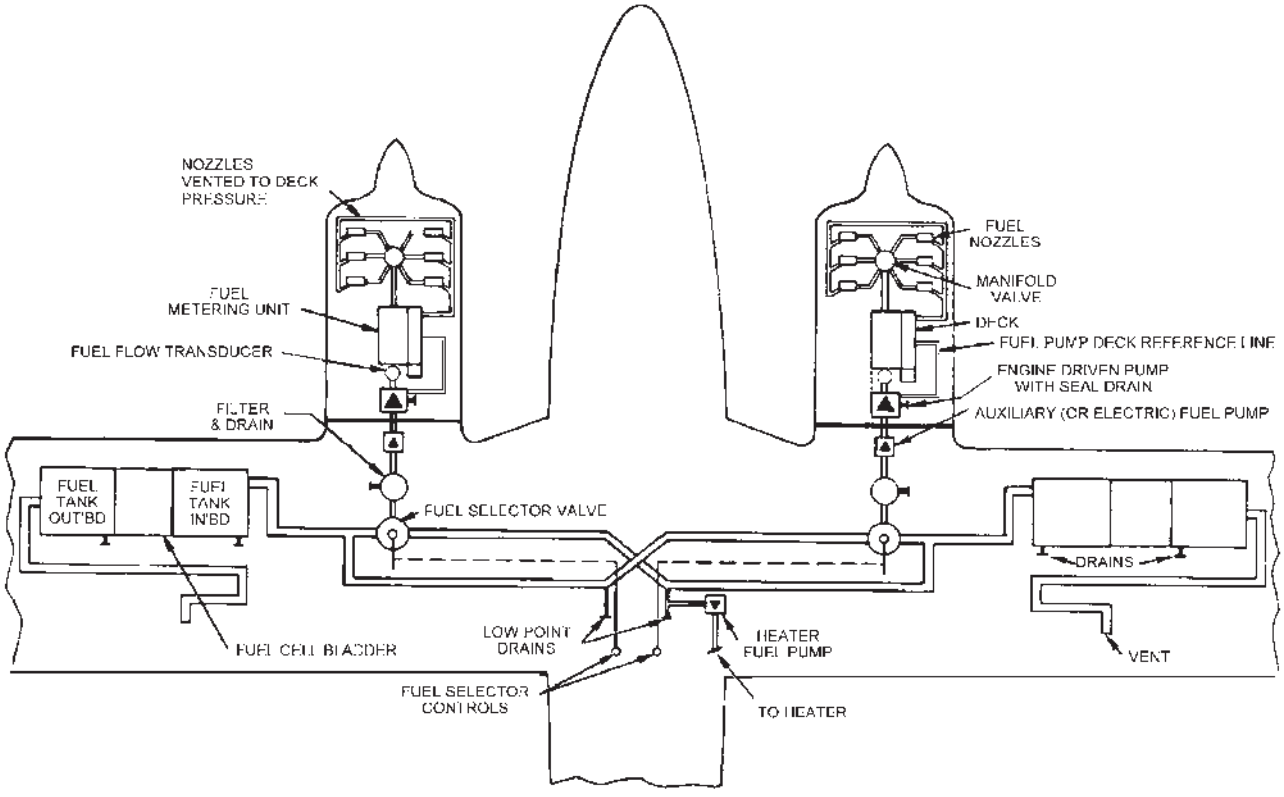




Effectivity
Seneca IV

Fuel System Schematic
Figure 1 (Sheet 1 of 2)





Fuel System Schematic
Figure 1 (Sheet 2 of 2)

Effectivity
Seneca V

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STORAGE

1. Fuel Tanks

WARNING: OBSERVE ALL FUEL SYSTEM FIRE HAZARD PRECAUTIONS THROUGHOUT ALL REMOVAL AND INSPECTION PROCEDURES. USE VAPOR-PROOF LIGHT FOR INSPECTION.

WARNING: IF DRAIN VALVES ARE REMOVED TO DRAIN TANKS, APPLY PARKER HANNIFIN THREAD LUBE, PIPER P/N 913-224, TO MALE PIPE THREADS BEFORE INSTALLING. DO NOT ALLOW LUBRICANT TO ENTER FUEL SYSTEM.

Two aluminum fuel tanks and a bladder type fuel cell are installed in each wing. The aluminum fuel tanks form an integral part of the wing surface. The two aluminum fuel tanks and bladder cell in each wing are interconnected.

Remove bladder type fuel cell first and install last when removing and installing the aluminum fuel tanks.

A. Inboard Fuel Tank

See “Figure 1” on page 28004 and “Figure 1” on page 28103.

(1) Removal

NOTE: Fuel tank removal is rare. Accordingly, each time a fuel tank is removed and reinstalled or replaced, take the opportunity to replace the associated flexible fuel hoses, interconnects, and/or vent lines.

(a) Remove bladder type fuel cell per “Removal” on page 28102.
(b) With fuel tank completely drained, loosen clamps at hose connections, and slide hose connections away from fuel tank.
(c) Remove screws from around perimeter of the tank.
(d) Carefully pull tank away from the wing far enough to gain access to remove sender wire and fuel line.
(e) The tank is now free to be removed.

(2) Installation

(a) Position fuel tank in its recess in the wing.
(b) Connect fuel line and fuel sender wires.
(c) Slide tank completely into position and secure with screws around its perimeter.
(d) Install bladder type fuel cell per “Installation” on page 28104.
(e) Fill fuel tanks and check for leaks, unrestricted fuel flow, and accurate sender indications on fuel quantity gauge.

B. Outboard Fuel Tank

See “Figure 1” on page 28004 and “Figure 1” on page 28103.

(1) Removal

NOTE: Fuel tank removal is rare. Accordingly, each time a fuel tank is removed and reinstalled or replaced, take the opportunity to replace the associated flexible fuel hoses, interconnects, and/or vent lines.

(a) Remove bladder type fuel cell per “Removal” on page 28102.
(b) With fuel completely drained from the tank, loosen clamps at hose connections, and slide hose connections away from fuel tank.
(c) Remove screws from around the perimeter of the tank.

(d) Carefully pull tank away from the wing far enough to gain access to remove sender wire and fuel vent line.

(e) The tank is now free to be removed.

(2) Installation

(a) Position fuel tank in its recess in the wing.

(b) Connect vent line on outboard side, and fuel sender wires.

(c) Slide tank completely into position and secure with screws around its perimeter.

(d) Install bladder type fuel cell per “Installation” on page 28104.

(e) Fill fuel tanks and check for leaks, unrestricted fuel flow, and accurate sender indications on fuel quantity gauge.

C. Repair

WARNING: SLOSHING OF FUEL TANKS IS PROHIBITED.

Completely drain fuel tanks. (See 12-10-00, Fuel Systems, Draining Entirely.) Inspect each tank for signs of leaks as indicated by telltale stains. If a fuel leak is detected, remove fuel tank and repair as follows:

(1) If tank has previously been sloshed, use a mirror and inspection light inserted through the filler neck to inspect tank interior for signs of peeling or chipping sealer. If peeling and/or chipping has occurred, and separated material is found, sloshing material must be completely removed or tank replaced.

(2) Seal leaks with Products Research Corporation PR 1422A series or PR1433G series sealant. For example: PR1422A1.

2. Bladder Type Fuel Cells

Two aluminum fuel tanks and a bladder type fuel cell are installed in each wing. The aluminum fuel tanks form an integral part of the wing surface. The two aluminum fuel tanks and bladder cell in each wing are interconnected.

See Fuel System Schematic, “Figure 1” on page 28004 and Fuel Cell Installation, “Figure 1” on page 28103.

A. Removal

NOTE: Fuel cell removal is rare. Accordingly, each time a fuel cell is removed and reinstalled or replaced, take the opportunity to replace the associated flexible fuel hoses, interconnects, and/or vent lines.

(1) Drain fuel tanks and remove fuel cell access panel located on top side of wing between Wing Station 138.00 and Wing Station 161.00.

(2) Reaching through fuel cell opening, remove hose clamp securing fuel cell vent nipple to vent tube of inboard and outboard fuel tanks.

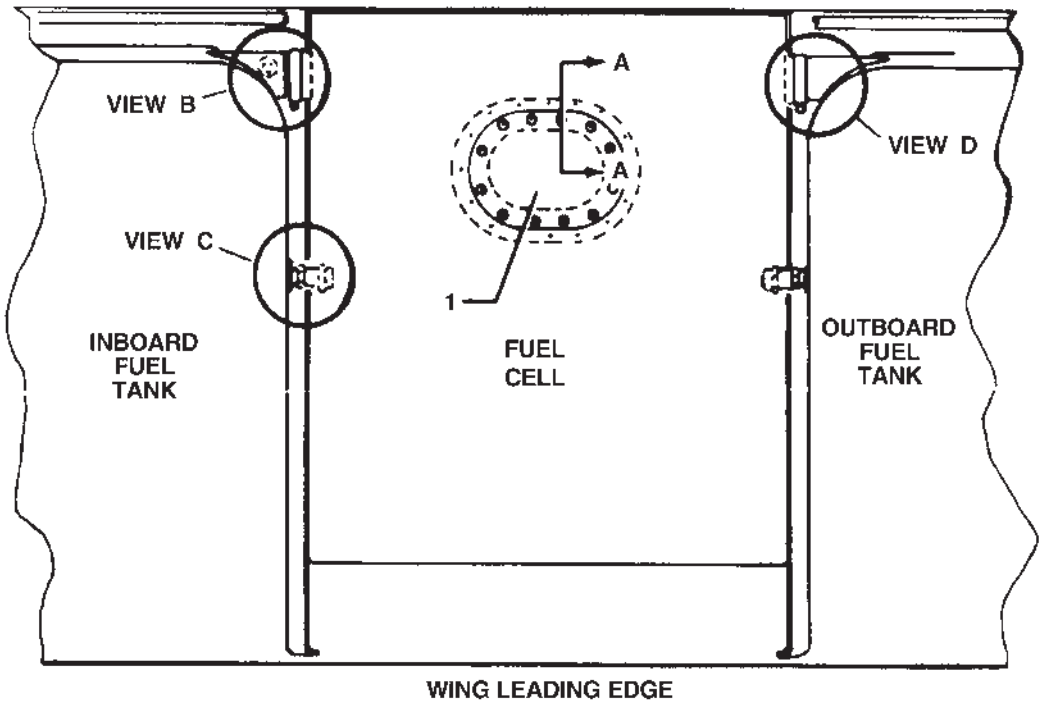
(3) Remove wing plugs from underside of wing at Wing Stations 138.00 and 161.00 and, using a common screwdriver, loosen clamp securing fuel cell interconnect nipple to inboard and outboard fuel tanks.

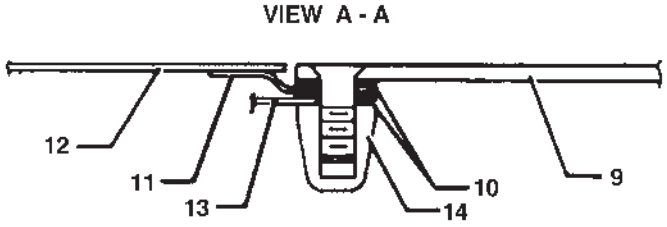
(4) Reaching through fuel cell access hole, gently separate Velcro fasteners holding fuel cell to surrounding structure.

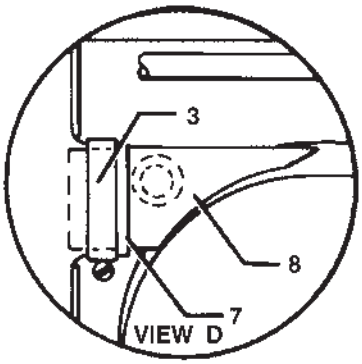
(5) Separate fuel cell vent nipple and fuel cell interconnect nipple from inboard and outboard fuel tanks.

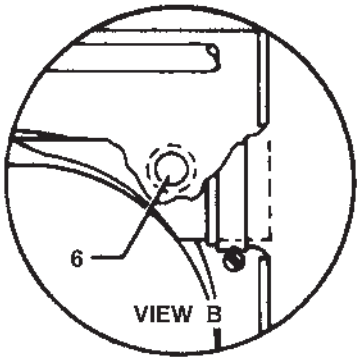
CAUTION: PAD EDGES OF ACCESS HOLE TO PREVENT DAMAGE TO FUEL CELL.

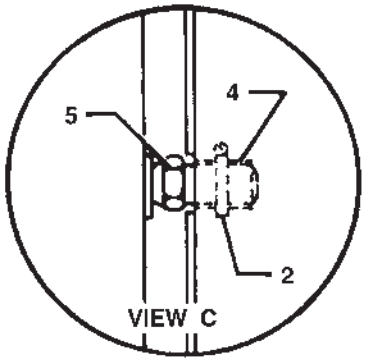
(6) Carefully fold fuel cell and remove through fuel cell access hole.











1. Access Panel
2. Hose Clamp
3. Hose Clamp
4. Fuel Vent Nipple
5. Fuel Vent Tube
6. Wing Plug
7. Fuel Interconnect Nipple
8. Interconnect Fitting
9. Access Panel
10. Gaskets
11. Doubler
12. Skin
13. Nut Flange
14. Fuel Cell

Fuel Cell Installation
Figure 1

B. Installation

(1) Before installing fuel cell, inspect airframe cavity for cleanliness.

CAUTION: PAD EDGES OF ACCESS HOLE TO PREVENT DAMAGE TO FUEL CELL.

(2) Place fuel cell into airframe cavity through access opening. Make sure no wrinkles exist in fuel cell upon installation.

(3) Install clamp on fuel cell interconnect nipple and tighten finger tight.

NOTE: Position so that screw on clamp will be facing plug hole in underside of wing.

(4) Reaching into fuel cell, work fuel cell interconnect nipple onto interconnect fitting of inboard and outboard fuel tanks.

(5) Using a common screwdriver and working through plug hole in underside of wing, tighten clamps. Torque should be 30 to 35 inch-pounds.

(6) Press fuel cell vent nipple onto fuel tank vent fitting of inboard and outboard fuel tanks. Position clamp on nipple fitting so that when tightened the screw body does not contact top of fuel cell. Torque to 15 inch-pounds.

(7) Press outward firmly on sides and top of fuel cell to engage cell with Velcro tape.

(8) Position gaskets as shown in “Figure 1” on page 28103. Place access panel over opening and secure with screws. Torque to 25 in.-lbs.

(9) Insert wing plugs in openings on underside of wing.

(10) Service fuel tanks and inspect for leaks.

C. Maintenance

WARNING: OBSERVE ALL FUEL SYSTEM FIRE HAZARD PRECAUTIONS THROUGHOUT ALL REMOVAL AND INSPECTION PROCEDURES. USE VAPOR-PROOF LIGHT FOR INSPECTION.

(1) Cleaning

(a) New Cells: It should not be necessary to clean new cells upon removing them from their containers, if they are installed in the airframe cavities promptly. If for any reason the cells are not installed immediately, and become dirty, they should be cleaned with soap and warm water to remove foreign material prior to installation in a clean cavity.

(b) Used Cells: Prior to removal, the cells are to be drained of fuel, purged with fresh air and swabbed out to remove all traces of fuel. Following removal, the cells are to be cleaned inside and out with soap and warm water.

(2) Inspection

(a) New Cells: Inspect the cell surface inside and outside for cuts, abraded (scuffed) areas and accessory damage. Also, inspect the fitting seals for nicks, scratches and foreign material.

(b) Used Cells: Cells removed from the airframe cavity for inspection and repair or cells being returned to service from storage. should be inspected as outlined above

(c) Cells installed in the airframe cavity may be inspected for possible repairs by reaching through the fuel cell access plate and taking a section of cell between the thumb and forefinger. Wipe the ridge created by this action with MEK. If fine cracks are evident, the fuel cell is not repairable.

(3) Fuel Cell Compartment

(a) Thoroughly clear the cell compartment of all fittings, trimmings. Loose washers, bolts or nuts.

(b) Round off all sharp edges of the fuel cell compartment.

(c) Inspect the fuel cell compartment just prior to fuel cell installation.

(d) Tape over all sharp edges and all rough rivets.

(4) Handling and Storage

(a) Prevent needless damage by exercising common sense care in all handling of the cells. Folding or collapsing of cells is necessary to place them in containers for storage, install in airframe cavities and carry from place to place. Protect cell from tools, hot lights, etc, when working around them. Avoid stepping on folds or creases of cells. Do not carry cells by fittings. Maintain original cell contours or folds when refolding for boxing, rolling to insert in airframe cavities or handling in the repair area. The cells to be repaired should be placed on a well-lighted table. Maintain natural contours, if possible while repairing. Prevent contact with sharp edges, corners, dirty floors or other surfaces. Repair area must be well-ventilated. Do not stack cells. Inspect cavities and ensure cleanliness prior to installing any cell.

(b) When storing cells, observe the following rules:

1) Fold cells smoothly and lightly as possible with a minimum number of folds. Place protective wadding between folds.

2) Wrap cell in moisture-proof paper and place it in a suitable container. Do not crowd cell in container, use wadding to prevent movement.

3) Stack boxed cells to allow access to oldest cells first. Do not allow stacks to crush bottom boxes. Leave cells in boxes until used.

4) Storage area must be dry, 70°F, and free of exposure to sunlight, dirt and damage.

5) Used cells must be cleaned with soap and warm water prior to storage. Dry, and box as outlined above.

D. Repair

The following is the repair procedure recommended for field repair of fuel cells constructed of Goodyear Vithane material. There are two methods by which these repairs may be accomplished. One method is by heat cure, the other is air cure. The end result of either repair is a neat, permanent repair. The heat repair allows the cell to be cured and ready for reinstallation in two hours while the air cure method required that the cell not be moved for 72 hours during the air cure period. See “Chart 1” on page 28106 for a list of repair materials and equipment.

NOTE: Air cure repairs to be made at room temperature at approximately 75°F. For each 10° drop in temperature add 20 hours cure time. For instance if room temperature reads 65°F, air cure for 92 hours instead of 72 hours.

(1) Limitations

Repair limitations are as follows:

(a) FT-192 repair fabric is for repair of simple contours only. Patches referred to in this text are of this material.

(b) Inside patches are to lap defect edges a minimum of 1.0 inch in each direction.

(c) Outside patches are to lap defect edges .25 to .50 of an inch larger than inside patches.

(d) Outside patches are to be applied and cured prior to applying an inside patch.

(e) Blisters between inner liner and fabric, larger than .25 of an inch in diameter require an outside and an inside patch.

(f) Separations between layers or plies larger than .50 of an inch in diameter require an outside and inside patch. Holes and punctures require an outside and inside patch.

(g) Slits or tears up to 6.0 inches maximum length require an outside and inside patch.

(h) External abraded or scuffed areas without fabric damage require an outside patch only.

(i) A loose edge may be trimmed, provided that .50 of an inch minimum lap or seam is maintained.

CHART 1
FUEL CELL REPAIR MATERIALS AND EQUIPMENT

Repair Kit, Goodyear Drawing No. 2F1-3-37813

80C27 Repair Cement	8	Pint cans, 320 gms in each
80C28 Cross-Linker	8	4 oz. bottles, 81 cc in each
Methyl Ethyl Ketone	2	Pint cans
FT-192 Repair Fabric	2	Sheet 12” x 12”

Group I Materials

The following equipment is necessary to perform the repair.

Group II equipment will be furnished at additional cost, if ordered by customer.

Foam Rubber Cloth Back Sheet, 1/4” x 12” x 12”	2
Paint Brush, 1 inch wide	2
Aluminum Plates, 1/4” x 6” x 6’	4
Measuring Cup (250 ml)	1
Cellophane (Sheet 12” x 24”)	2

Torques For Specific Nipple Fitting Sizes

Fitting Size (I.D.)	Clamp Torques (in.-lb.)
1/4” - 1/2”	12 - 16
3/4” - 1”	15 - 20
1 1/2”	25 - 30
2”	30 - 35
3”	35 - 40

NOTE: Accessories - order per individual cell equipment. Phenol plates, phenol plate assemblies and phenol test equipment can be ordered as required from cell manufacturer. Cure Iron (set 240°F) optional.

CAUTION: FOR EACH 10°F (5.6°C) DROP IN TEMPERATURE FROM 75°F (24°C), ADD 20 HOURS CURE TIME. FOR EXAMPLE, AT 65°F (18°C), CURE FOR 92 HOURS.

(j) Air cure repair patches are to remain clamped and undisturbed for 72 hours at room temperature of approximately 75°F (24°C).

(k) All heat cured patches are ready for use when cool.

(l) Fitting repairs are confined to loose flange edges, seal surfaces rework and coat stock.

NOTE: Any damage not covered by the above should be returned to Meggitt (Rockmart), Inc, in Rockmart, Georgia, for repair.

(2) Handling of Repair Materials

(a) All materials are to be protected from dirt contamination, sunlight, and excessive heat or cold while in storage. Containers are to be tightly capped and stored at a temperature of 70°F.

CAUTION: 80C27 REPAIR CEMENT REQUIRES THOROUGH MIXING TO OBTAIN FULL ADHESIVE VALUES.

(b) The repair cement code 80C27 referred to in this text is prepared immediately prior to use by mixing repair cement 80C27 (pint can with 320 gms) with cross-linker 80C28 (4 oz. bottle with 81 cc).

CAUTION: ALL CONTAINERS FOR CEMENTS AND SOLVENTS SHOULD BE PROPERLY IDENTIFIED.

(c) Repair cement has a pot life of 20 minutes after mixing. The unmixed 80C27 and 80C28 have a shelf life of six months from date of packaging.

(3) Procedures

CAUTION: THE REPAIR OF GOODYEAR VITHANE FUEL CELLS IS RESTRICTED TO AUTHORIZED PERSONNEL. AUTHORIZED PERSONNEL ARE THOSE WHO HAVE BEEN CERTIFIED AND TRAINED BY GOODYEAR REPRESENTATIVES, OR THOSE WHO HAVE RECEIVED THEIR TRAINING FROM PERSONS WHO HAVE BEEN CERTIFIED AND TRAINED BY GOODYEAR REPRESENTATIVES.

(a) Patch Repair (Heat Cure Method)

1) Prepare exterior cell wall and exterior patch first. Cut repair patch from FT-192 material to size required to ensure proper lap over injury in all directions. (See “Limitations” on page 28105) (Hold shears at an angle to produce a beveled edge (feather) on patch.) Round corners of patch. (Dull side or gum contact face of repair patch should be the largest surface after beveling.)

2) Wash one square foot of cell wall surrounding injury and repair patch contact side with a clean cloth soaked with Methyl Ethyl Ketone solvent.

3) Abrade cell wall surface about the injury and contact side of patch with fine emery cloth to remove shine.

4) Repeat Methyl Ethyl Ketone washings two more times. A total of three washings each surface.

5) Tape a 8” x 8” piece of cellophane inside cell over injury.

6) When all the above preparatory work has been done and cell has been positioned for patch application on repair table, mix the 80C27 cement (320 gms) with the crosslinker 80C28 (81cc), and stir mixture thoroughly for five minutes.

NOTE: Cement must be at a minimum of 70°F before mixing. Keep away from water and excessive heat.

7) Brush one even coat of mixed repair cement on the cell wall around injury and on the contact side of repair patch. Allow to dry for fifteen minutes.

CAUTION: DO NOT USE FIRST CAN OF MIXED CEMENT FOR THIS COAT.

8) Repeat a second mixing of repair cement and brush a second coat.

CAUTION: MAKE SURE CELLOPHANE INSIDE CELL OVER INJURY REMAINS IN PLACE AS ANY CEMENT WILL STICK CELL WALL TOGETHER WITHOUT THE CELLOPHANE AS A SEPARATOR.

9) Allow cement to dry approximately five minutes and then center patch over injury. Lay repair patch by rolling down on surface from center to edge without trapping air. Hold the unrolled portion of repair patch off the cemented surface until roller contact ensures an air-free union. At this time repair patch may be moved by hand on wet surface to improve lap. Do not lift repair patch, slide it.

10) Cover one smooth surface each of two aluminum plates (plates must be larger than patch), with fabric-backed airfoam fabric side out. Tape airfoam in place. Foam must cover edges of plate for protection. Use a cellophane separator to prevent the cement from sticking in the wrong place.

CAUTION: MAKE SURE THAT CELL FOLD IS NOT CLAMPED BETWEEN PLATES. THIS WOULD CAUSE A HARD PERMANENT CREASE. ALSO MAKE SURE THAT PATCH DOES NOT MOVE WHEN CLAMP IS TIGHTENED.

11) Center a repair iron 2F1-3-25721-1 on the plate over the repair patch. Secure the assembly with a C-clamp. Tighten by hand. Check cement flow to determine pressure.

12) Connect repair iron into 110-volt electrical outlet and cure repair for two hours. After two hours cure, unplug electric and allow repair iron to cool to touch. Then remove C-clamp. Wet cellophane to remove from repair.

CAUTION: SUCCESS OF APPLYING BOTH AN OUTSIDE AND INSIDE REPAIR PATCH SIMULTANEOUSLY IS DOUBTFUL AND NOT RECOMMENDED.

13) Inside patch is applied same as above procedure except for size of repair patch (see limitations) after outside patch has been cured.

(b) Patch Repair (Air Cure Method)

Follow procedure for heat cure method, except omit repair iron and cure each patch per air cure limitations (minimum 72 hours), undisturbed, at 75°F.

(c) Defect Repairs

1) Blisters: Remove loose material by trimming. Apply an outside and inside repair patch.

2) Holes, Punctures, Cuts, Tears and Deep Abraded Areas: Trim away any ragged material and apply an outside and inside repair patch.

3) Loose Seams: Buff loose edge and contact surface with emery cloth. Wash three times with Methyl Ethyl Ketone. Apply 80C27 mixed cement two coats as with repair patch. Clamp and cure. Either method may be used. See repair patch. Loose seams may be trimmed if minimum lap remains.

E. Testing

Either of the following test procedures may be used to detect leaks in the bladder cells.

NOTE: The chemical test is the more sensitive and the preferred test.

(1) Soap Suds Test

(a) Attach test plates to all fittings.

(b) Inflate the cell with air to a pressure of 1/4 psi MAXIMUM.

(c) Apply a soap and water solution to all repaired areas and any areas suspected of leakage. Bubbles will appear at any point where leakage occurs.

(d) After test, remove all plates and wipe soap residue from the exterior of the cell.

(2) Chemical Test

(a) Attach test plates to all fitting openings except one.

(b) Make up a phenolphthalein solution as follows: Add 40 grams phenolphthalein crystals in 1/2 gallon of ethyl alcohol, mix, then add 1/2 gallon of water.

(c) Pour ammonia on an absorbent cloth in the ratio of 3 ml per cubic foot of cell capacity. Place the saturated cloth inside the cell and install remaining test plate.

(d) Inflate the cell with air to a pressure of 1/4 psi MAXIMUM, and maintain pressure for fifteen minutes.

(e) Soak a large white cloth in the phenolphthalein solution, wring it out thoroughly, and spread it smoothly on the outer surface of the cell. Press the cloth down to ensure detection of minute leaks.

(f) Check the cloth for red spots which will indicate a leak. Mark any leaks found and move the cloth to a new location. Repeat this procedure until the entire exterior surface of the cell has been covered. If red spots appear on the cloth, they may be removed by resoaking the cloth in the solution.

(g) The solution and test cloth are satisfactory only as long as they remain clean. Indicator solution that is not in immediate use should be stored in a closed rustproof container to prevent evaporation and deterioration.

(h) After the test, remove all plates and test equipment. Allow the cell to air out.

(i) In conducting either test outlined above, the cell need not be confined by a cage or jig, providing the 1/4 psi pressure is not exceeded.

3. Fuel System Leakage Inspection

(PIR-PPS60049-4, Rev. C.)

CAUTION: THE FUEL CELLS AND/OR TANKS MAY BE DAMAGED IF MORE THAN 1.5 PSIG PRESSURE IS INTRODUCED INTO THE SYSTEM.

The following procedures are provided for troubleshooting purposes only.

A. Setup

(1) Ensure the fuel lines and components of the fuel system are readily accessible.

(2) Defuel the airplane prior to performing any of the following tests.

(3) A test fixture containing the following features is required to perform this test.

See “Figure 2” on page 281011 to fabricate.

- Two air filters
- Pressure regulator
- Two fuel filler test caps each incorporating a 5 psig gauge and a shutoff valve
- Main pressure gauge (0 - 5 psig; marked red above 1.5 psig)
- Two main air supply shutoff valves
- Main air supply needle valve
- Manometer (with “Aircraft” and “Reference” sides labeled)
- Manometer needle valve
- Flow meter (0.0 - 0.5 SCFH)
- Two flow meter shutoff valves

(4) Ensure that appropriate caps are available for closing open lines and vents.

(5) Apply bubble testing liquid (P/N 279-246) to surfaces being checked for leaks.

B. Fuel Tank Leak Test

The following procedure, as written, tests both fuel tanks and the fuel cell as a unit when installed in the airplane.

NOTE: If the fuel cell is removed from the airplane, test per “Testing” on page 28108.

NOTE: The metal fuel tanks can be tested individually by capping all outlets from the tank being tested. (Testing either inboard tank will require fabrication of an adapter incorporating a 5 psig gauge and a shutoff valve which can be substituted for the fuel filler test cap and attached to the tank interconnect fitting.)

(1) Cap all fuel vent outlets on the wings.

(2) Verify that the fuel sump drain valves (underneath the LH and RH wings) are in the closed position.

(3) Cap all the fuel lines exiting from the tank.

(4) On the test fixture, check that the following valves are in the OFF position (See “Figure 2” on page 281011):

- Main air supply shutoff valves (#1 and #2)
- Main air supply needle valve (#4)
- Shutoff valve on each fuel filler test cap

(5) Verify that the manometer needle valve (#3) is open.

CAUTION: DO NOT OVERTIGHTEN THE FUEL CAP.

(6) Install the test unit fuel cap at the fuel filler port in the wing(s) being tested. Ensure that the test unit fuel cap is snug.

(7) Connect shop air to the test fixture.

CAUTION: DO NOT EXCEED THE 1.5 PSIG LIMIT OR FUEL CELL DAMAGE MAY OCCUR.

(8) Select the Test Fixture “Valve 2” to the ON position and, using “Valve 1”, pressurize the line to a minimum of 1.0 PSIG but no greater than 1.5 PSIG.

NOTE: Monitor pressure on main pressure gauge. Do not let air into fuel tanks if pressure reads above 1.5 psig.

(9) Open the test cap shutoff valve(s), slowly open “Valve 4”, and monitor pressure on the main pressure gauge. Adjust the pressure to between 1.0 psig minimum and 1.5 psig maximum.

(10) Select Test Fixture “Valve 1” to the OFF position, then “Valve 2” to OFF. Observe the position of the test fluid inside the manometer. The “AIRCRAFT” and “REFERENCE” sides should be the same.

(11) Disconnect the shop air from the test fixture and monitor the leakage rate.

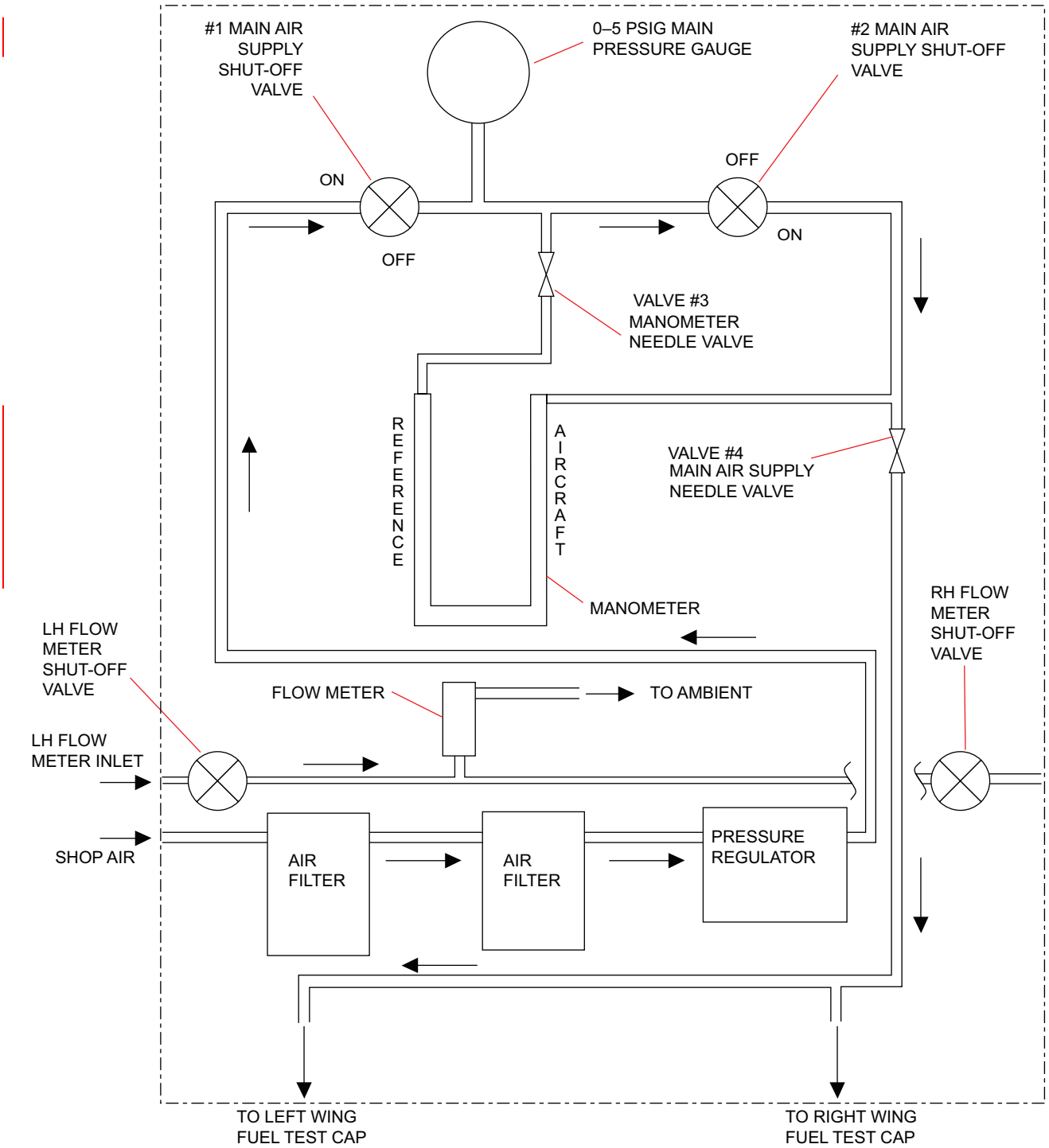
NOTE: No leakage should occur for a period of five (5) minutes.

If the “AIRCRAFT” side of the manometer moves upwards during a five-minute hold period, then a leak exists at the connections. If a leak exists, then apply bubble testing liquid to all fuel system connections during the hold period, and inspect each connection for a bubble formation.

If the “REFERENCE” side of the manometer moves upwards, then there is a temperature/pressure differential. Releasing half of the air in the system and refilling it may solve this issue.

(12) If leaks are found, repair as follows:

- Metal tank skins: repair leaks per “Repair” on page 28102.
- Fuel cells (bladder type): repair per “Repair” on page 28105.
- For other areas, if inspection and tightening do not stop the leak, then component replacement is required.



Test Fixture Schematic
Figure 2

(13) Slowly remove fuel vent cap(s) (on left and right wings) and verify that remaining fuel cell pressure bleeds off through each vent.

(14) Remove all test equipment and return the airplane fuel system to operational configuration.

C. Fuel System Leak Test

See “Figure 3” on page 281013 for the location of fuel system components.

(1) Cap all the fuel vent outlets on the left and right wings. (See “Figure 1” on page 28004.)

(2) Verify that fuel sump drain valves (located underneath the left and right wings) are in the closed position.

(3) On the test fixture, check that the following valves are in the OFF position (see “Figure 2” on page 281011):

- Main air supply shutoff valves (#1 and #2)
- Main air supply needle valve (#4)
- Shutoff valve on each fuel filler test cap
- Both flow meter shutoff valves

(4) Verify that the manometer needle valve (#3) is open.

(5) Disconnect the main fuel lines forward of the firewall.

(6) Install the test unit fuel caps at the fuel filler port on each wing.

(7) Connect the test fixture’s left and right side flow meter hoses to the main fuel lines at the firewall, of the appropriate engine.

(8) Select the fuel selector valves both to ON.

(9) Connect the shop air to test fixture.

CAUTION: DO NOT EXCEED THE 1.5 PSIG LIMIT OR FUEL CELL DAMAGE MAY OCCUR.

(10) Select Test Fixture “Valve 2” to the ON position and, using “Valve 1”, slowly pressurize to a minimum of 1.0 psig and a maximum of 1.5 psig.

NOTE: Monitor the pressure on the main pressure gauge. Do not let air into the fuel tanks if pressure exceeds 1.5 psig.

(11) Open both test cap shutoff valves, slowly open “Valve 4” and monitor pressure on main pressure gauge. If pressure exceeds 1.5 psig, close “Valve 1” IMMEDIATELY, then open the flow meter valves to bleed excess pressure.

(12) Monitor pressure levels on the fuel test cap gauges and allow aircraft fuel system to pressurize to a minimum 1.0 psig but no greater than 1.5 psig.

NOTE: PA-34 aircraft take approximately 15 minutes to fill.

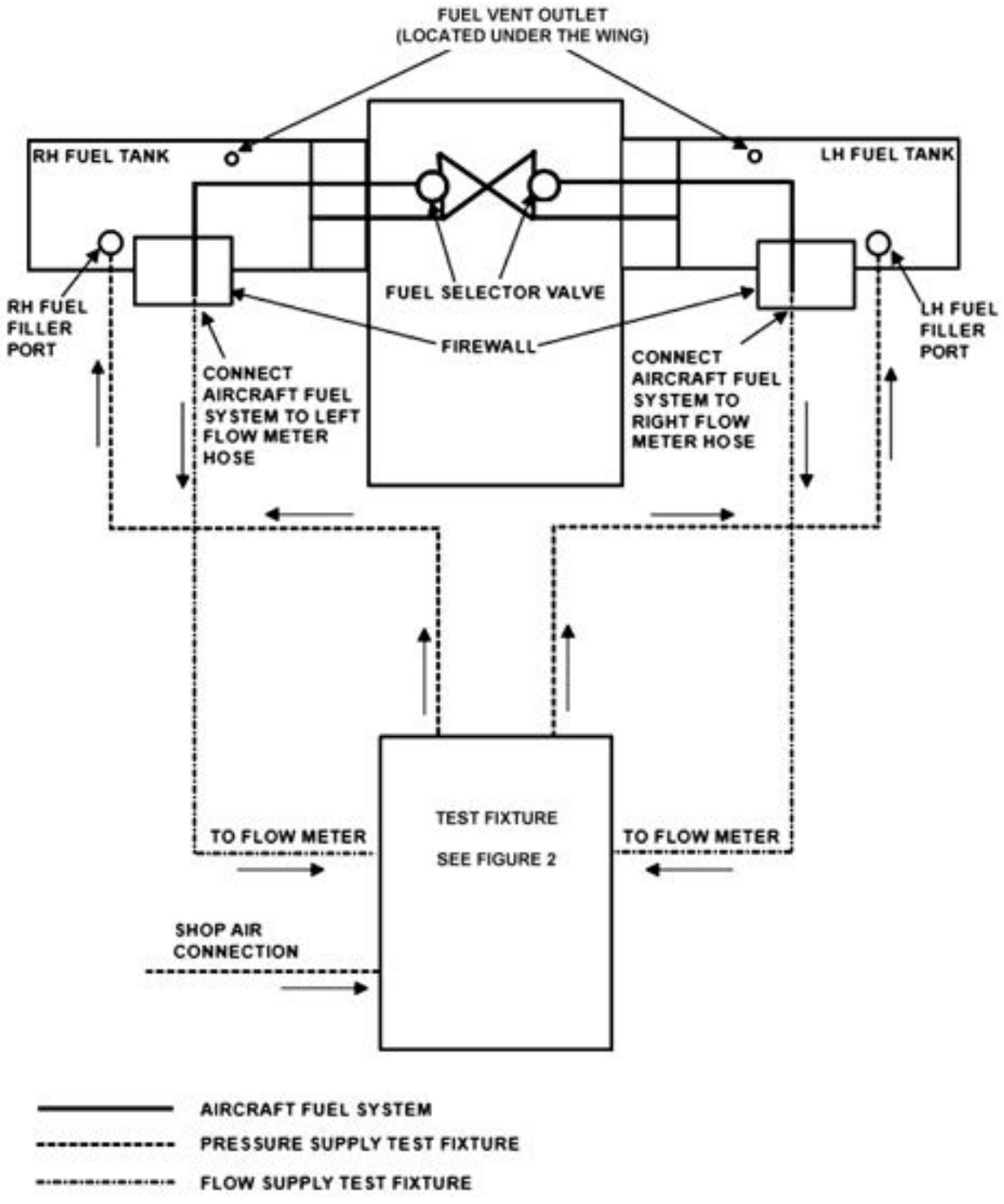
(13) Select Test Fixture “Valve 1” to OFF position, then select “Valve 2” to OFF position. Observe the position of the test fluid inside the manometer. The “AIRCRAFT” and “REFERENCE” sides should be the same.

(14) Disconnect the shop air from the test fixture and monitor leakage rate.

NOTE: The system should not leak for a period of 5 minutes.

If the “AIRCRAFT” side of the manometer moves upwards during a five-minute hold period, then a leak exists at the connections. If a leak exists, then apply bubble testing liquid to all fuel system connections during the hold period, and inspect each connection for a bubble formation.

If the “REFERENCE” side of the manometer moves upwards, then there is a temperature/pressure differential. Releasing half of the air in the system and refilling it may solve this issue.



Fuel System Leakage Test Schematic
Figure 3

(15) If leaks are found, then repair as follows:

- Metal tank skins: repair leaks per “Repair” on page 28102.
- Fuel cells (bladder type): repair per “Repair” on page 28105.
- For other areas, if inspection and tightening do not stop the leak, then component replacement is required.

(16) Connect shop air to test fixture and proceed to steps D through F, below, to complete fuel flow tests, as required. If no flow tests are required, then proceed to step G, “Return to Service”.

NOTE: After all of the following flow tests (D, E and F) are completed, continue to step G, “Return to Service”.

D. Flow Test

(1) With both fuel selector valves in the ON position (i.e., levers full forward), move the shutoff valves on all fuel filler test caps to the OFF position (i.e., closed).

(2) Move the flow meter valve connected to the LEFT firewall to the ON position. Monitor the flow meter and confirm that it reads at least 0.5 SCFH. Confirm that the left tank pressure gauge drops and the right tank is unchanged. Move the same flow meter valve to the OFF position.

(3) Move the flow meter valve connected to the RIGHT firewall to the ON position. Monitor the flow meter and confirm that it reads at least 0.5 SCFH. Confirm that the right tank pressure gauge drops and the left tank is unchanged. Move the same flow meter valve to the OFF position.

(4) Depress the left half of the fuel drain valve lever (located in the right side of the spar carry-through) and verify that the pressure in the left tank is decreasing.

(5) Depress the right half of the fuel drain valve lever and verify that the pressure in the right tank is decreasing.

E. Cross Flow Test

(1) Set the left and right fuel selector valves to the CROSS FEED position (i.e., levers full aft).

(2) Move the flow meter valve connected to the LEFT firewall to ON position. Monitor that the flow meter reads at least 0.5 SCFH. Confirm that the right tank pressure gauge drops and the left tank is unchanged. Move the same flow meter valve to the OFF position.

(3) Move the flow meter valve connected to the RIGHT firewall to ON position. Monitor that the flow meter reads at least 0.5 SCFH. Confirm that the left tank pressure gauge drops and the right tank is unchanged. Move the same flow meter valve to the OFF position.

F. Off-Flow Test

(1) Set the left and right fuel selectors to the OFF position (i.e., levers at the center).

(2) Move the flow meter valve connected to LEFT firewall to ON position and verify that the flow meter reads 0.0 SCFH.

(3) Move the flow meter valve connected to RIGHT firewall to ON position and verify that flow meter reads at least 0.0 SCFH.

NOTE: For steps (2) and (3), the flow meter may momentarily rise due to air trapped in the lines downstream of the fuel selector valves during the Cross Flow Test. However, it should quickly level off at 0 SCFH.

(4) Push open each fuel sump drain valve (located underneath the LH and RH wings) and verify that air flows outside the fuel tanks. Close the fuel sump drain valves.

G. Return to Service

(1) Disconnect shop air from test fixture.

(2) Slowly remove fuel vent caps (left and right wing) and verify that remaining fuel cell pressure bleeds off through each vent.

(3) Remove all test equipment and return the airplane fuel system to operational configuration.

4. Fuel Caps

A. Flush Fuel Cap (Old Style)

See “Figure 4” on page 281016.

NOTE: These instructions do not apply to new style fuel caps (P/N 654-556 Locking, and 654-557 Non-Locking) installed in 2006 and later.

The flush fuel cap is designed to afford the aircraft a cleaner surface and a reduction in drag. Anytime the cap does not close tightly or gas leakage is evident the cap should either be replaced, or repaired. The cap consists of three basic assemblies; the handle/plate assembly, gasket spring assembly, and lock assembly.

(1) Disassembly

The cap can be disassembled as follows:

(a) Remove the screws that hold the cap assembly together and make note of the relative position of the lock to the up plate.

(b) Separate the assembly and replace the spring and gaskets as necessary.

(2) Assembly

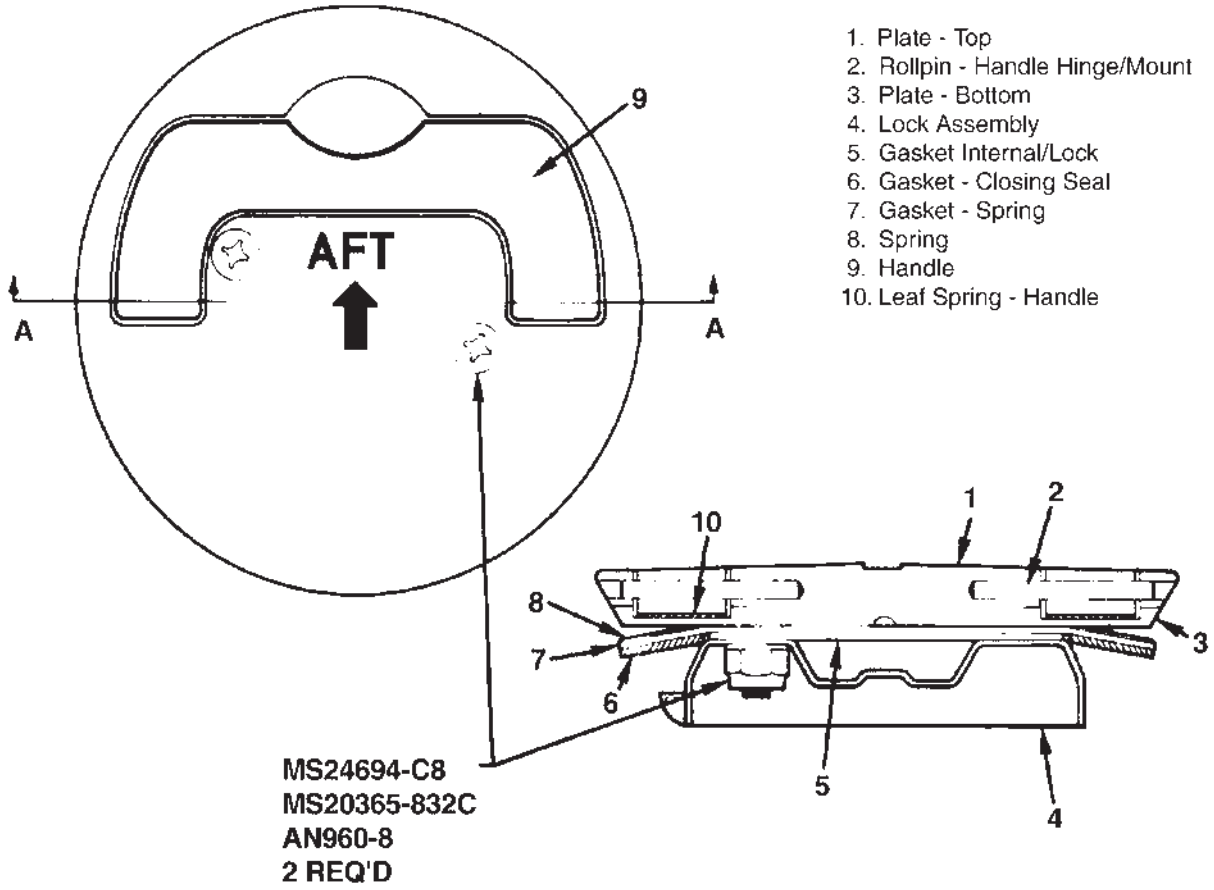
The cap should be assembled as follows:

(a) Install the gaskets on the spring assembly if necessary and align the gasket and spring on the handle plate with the spring concave towards the lock.

(b) Align the lock assembly against the gasket and spring assembly in the same position as noted at removal.

(c) Coat the threads of the bolts with Tite-Seal Gasket and Joint Sealing Compound (medium weight).

(d) Install bolts.



Effectivity
Old Style
Circa 2006 and earlier

Flush Fuel Cap Assembly
Figure 4

B. Locking Fuel Cap (Old Style)

Refer to “Figure 5” on page 281018.

NOTE: These instructions do not apply to “new style” fuel caps (P/N 654-556 Locking, and 654-557 Non-Locking) installed in 2006 and later.

(1) Disassembly

(a) Remove the two screws from the top of the fuel cap.

(b) Remove the screw and lock washer which secures the pawl to the bottom of the key lock. Remove pawl.

(c) Remove the nut which secures the key lock to the cover.

(d) Slide lock, gaskets and spring over the back of the key lock.

(e) The key lock may be removed by removing the o-ring and pushing the key lock through the cover. Ensure that the teflon lock gasket is not lost.

(2) Assembly

(a) Insert the key lock through the cover; making sure that the teflon lock gasket is installed under the head of the key lock.

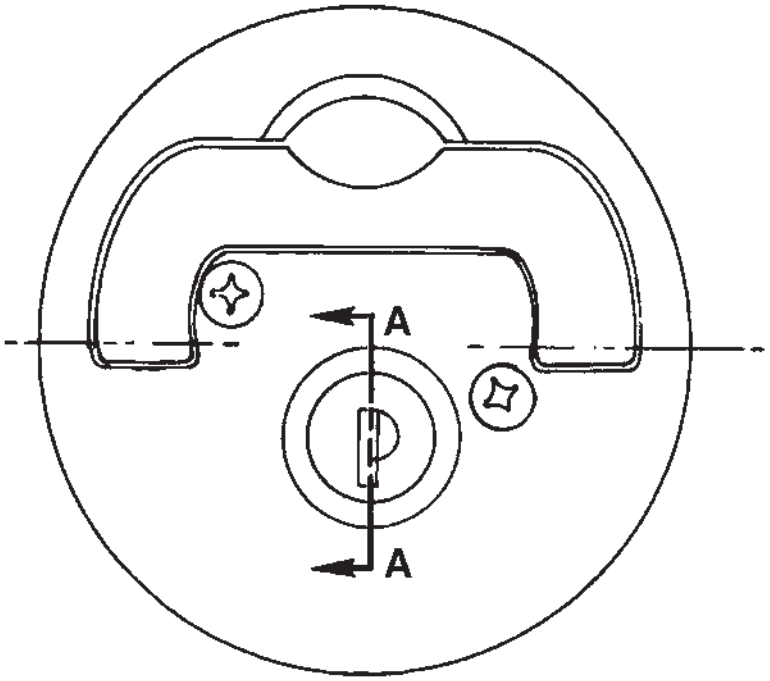
(b) Insert the o-ring in the groove on top of the cover.

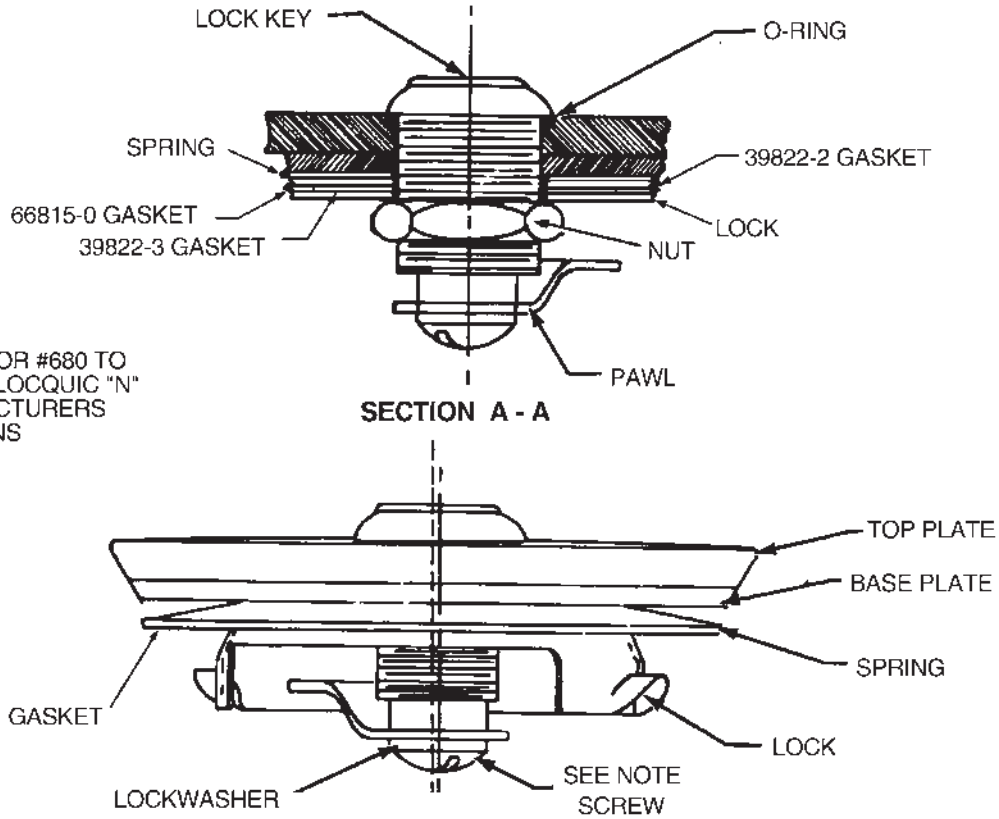
(c) Slide spring, gaskets and lock over the back of the key lock.

(d) Reinstall nut which secures the key lock to the cover.

(e) Attach the pawl to the back of the lock assembly with the screw and lock washers.

(f) Reinstall the two screws and lockwashers on top of the fuel cap.





Effectivity
Old Style
Circa 2006 and earlier

Locking Fuel Cap
Figure 5

DISTRIBUTION

1. Fuel Vent and Vapor Return Systems

   A. Vapor Return System (Seneca IV only)

      Both fuel systems utilize vapor return and vent systems. Although it is not difficult to maintain the systems, it is recommended that they be inspected routinely with the following information in mind.

      The fuel system on each engine is a “continuous flow” type and makes use of a vapor return line to provide a tank return for fuel vaporized in the pump swirl chamber. The vapor return line for each engine is routed from the forward elbow of the engine driven fuel pump back over the top of the engine to an elbow mounted to the baffle on the right side of the engine. At this point the installations differ slightly in that on the left engine installation the hose, connected to the rear of the elbow, is routed to the left side of the firewall just outboard of the engine mount attachments, while the right engine has the related hose routed to the right side of the firewall but also outboard of the engine mount attachment. Both installations however use an elbow on the firewall to connect the hose from the engine and tubing from the tank.

      It is important to note that the check valve included in the line from the firewall to the tank used to prevent reverse flow, is installed in a specific manner in order for it to operate properly. On the barrel of the valve is an arrow showing fuel direction, and on one of the flats the HINGE. When installed the arrow must be pointed in the direction of the tank and the flat with HINGE on it facing up (refer to “Figure 1” on page 28202). Access to the valve can be made by removing the leading edge fairing at the outboard root of the wing and nacelle.

   B. Vapor Return System

      The two fuel systems also use their own vent system. Each system is similar and designed to vent through the outboard tank with interconnects between the other tanks cells. The vents are located under their respective wing and behind the main spar near Wing Station 119.10. A line made up of a series of hoses and tubing connects the outboard tank and vents. The vent line is connected to the outboard tank through a limited check valve which functions to vent ambient pressure into the tank while preventing fuel from escaping. The vent underneath the wing should therefore be checked periodically for fuel stains or other indications of significant fuel leakage to make sure the limited check valve is functioning properly.

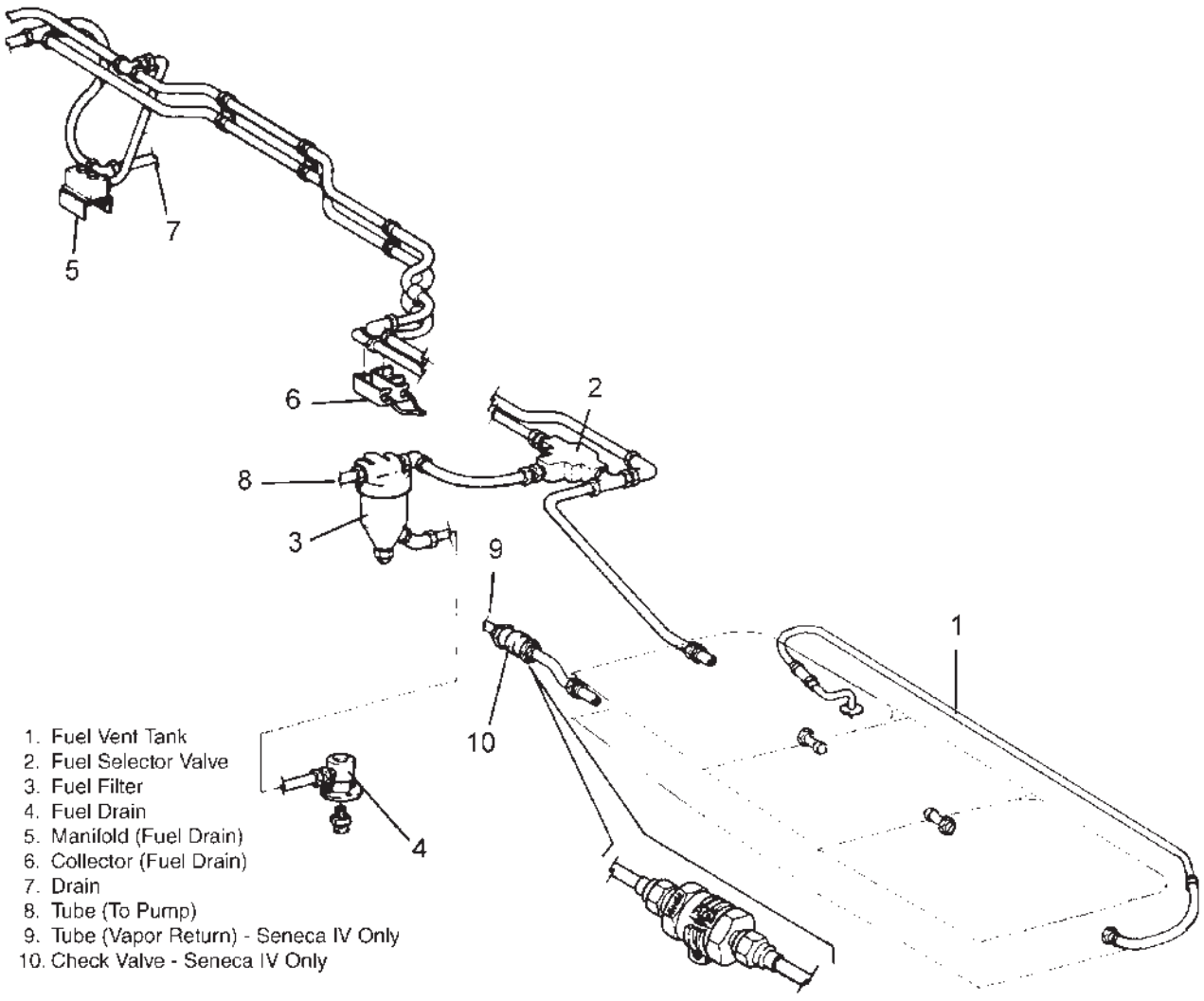
2. Cleaning the Fuel System

   A. To flush the fuel tank and selector valve, disconnect the fuel line from the injector.

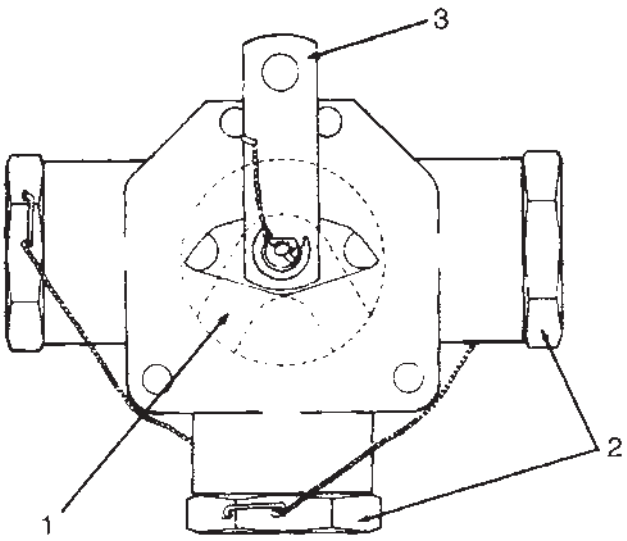
   B. Select a tank, turn on the electric fuel pump and flush fuel through the system until the tank is empty. Agitation of the fuel within the tank during this operation will help pick up and remove dirt and other foreign matter from the fuel tank and selector valve.

   C. Repeat this procedure for each fuel tank.

   D. When all the tanks are flushed, clean the filter and fuel tank finger screens.



Fuel Vent System
Figure 1



1. VALVE BALL
2. END FITTING
3. SELECTOR ARM

Fuel Selector Valve
Figure 2

3. Fuel Selector Valve

CAUTION: NO FIELD DISASSEMBLY OR REPAIR OF FUEL SELECTOR VALVES IS AUTHORIZED. MAINTENANCE IS LIMITED TO REMOVAL AND REPLACEMENT OF THE WHOLE UNIT.

A. Removal

(1) Remove the access plate located forward of the main spar on the under side of the wing and outboard of the nacelle.

(2) Drain the appropriate fuel tank (refer to Draining Fuel Tank, Chapter 12).

(3) Disconnect the control cable from the valve selector arm. Disconnect fuel lines and mounting hardware. Remove fuel selector valve.

B. Leak Test

(1) Connect the inlet port of the valve assembly to a 50 psi air source.

(2) Plug the right hand port and close the left hand port by moving the handle counterclockwise (as viewed with the lever facing you) until the stop is reached.

(3) Apply pressure to 50 psi. There should be no evidence of leakage either through the port or around the fitting and lever when the selector valve is submerged in kerosene or a similar petroleum based fluid for 30 seconds.

(4) Depressurize. Remove the plug from the right hand port and place it in the left hand port. Close the right port by moving the handle clockwise (as viewed with the lever facing you) until the stop is reached.

(5) Repeat Step (3).

(6) Disconnect and wipe fluid from the valve exterior.

C. Installation

(1) Position the selector valve in the wing with the lever facing down and the center port facing forward. Secure to the mounting bracket with four screws and nuts.

(2) Connect the fuel lines and control cable to the valve.

(3) Refer to the following paragraph for rigging of the selector valve.

(4) Install access panel.

D. Rigging

(1) Remove the access panel located on the underside of the wing, forward of the main spar, outboard of the nacelle.

(2) Ensure that the selector valve is connected to the control cable and the selector valve arm is in its center detent position.

(3) Ensure that the control cable is disconnected where it attaches to the cockpit lever.

(4) Place the fuel selector handle in the cockpit in its OFF position (the levers centered on the OFF position of the cover placard). Adjust and connect the cable end to the cockpit lever.

(5) Actuate the selector to ascertain that the valve moves into its three detent positions and that the control levers have a positive clearance between the lever and cover assembly.

(6) Reinstall the access panel.

4. Electric Fuel Pump

NOTE: Electric Fuel Pump replacement is on condition / as required.

A. Removal and Installation of Electric Fuel Pump

There is one electric rotary vane type fuel pump for each engine. The pump is mounted in a bracket on the aft side of the firewall. To remove pump, proceed as follows:

(a) Remove rectangular hatch assembly located on the top of the nacelle, aft of the firewall.

(b) Remove fuel lines from the pump and disconnect the electrical leads.

(c) Remove straps holding pump in position and withdraw pump through hatch opening.

(d) Do not attempt to disassemble or repair the fuel pump. If fuel pump proves to be defective, it should be replaced.

(e) Reinstall pump in reverse order of removal.

B. Auxiliary Fuel System Adjustment (Seneca IV only)

Adjustment of the auxiliary fuel system if installed is accomplished as follows for each engine:

(a) Remove the access panels from the top of each engine nacelle to gain access to the slider resistor mounted on the nacelle bulkhead.

(b) Install a calibrated pressure gauge (31 to 37 psi) in the fuel line forward of the firewall.

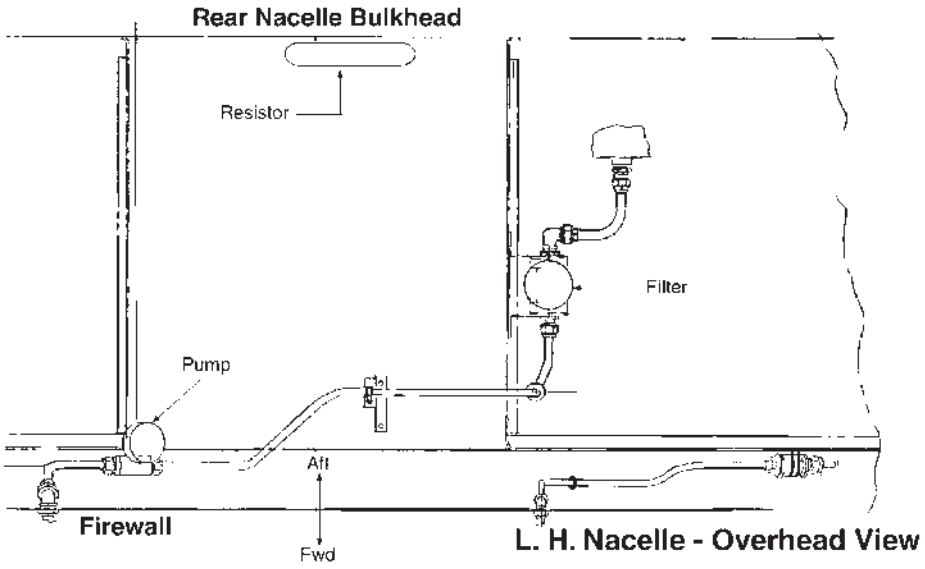
(c) Pull the circuit protector (for the auxiliary fuel pump which is to be adjusted) to the off position and ensure that the aircraft master switch is in the off position also.

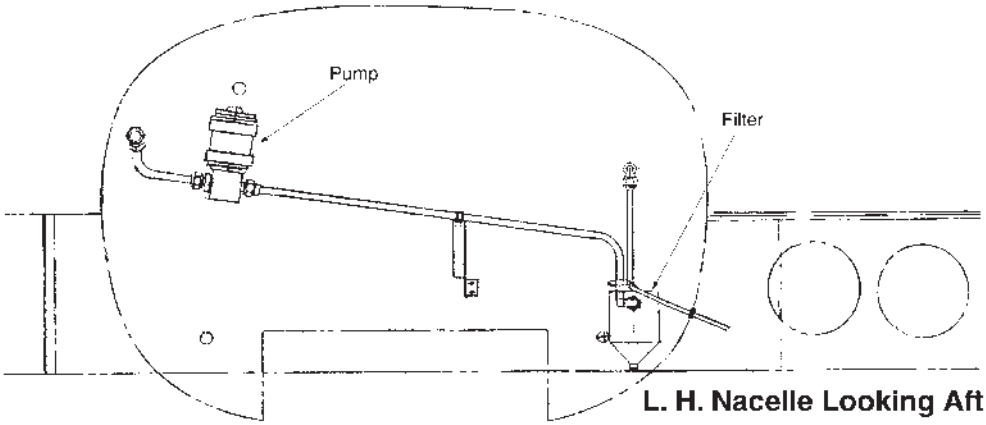
(d) Connect the negative lead from an external DC power source to ground on the aircraft and the positive lead to the slider resistor high position. (Refer to “Figure 3” on page 28205.)

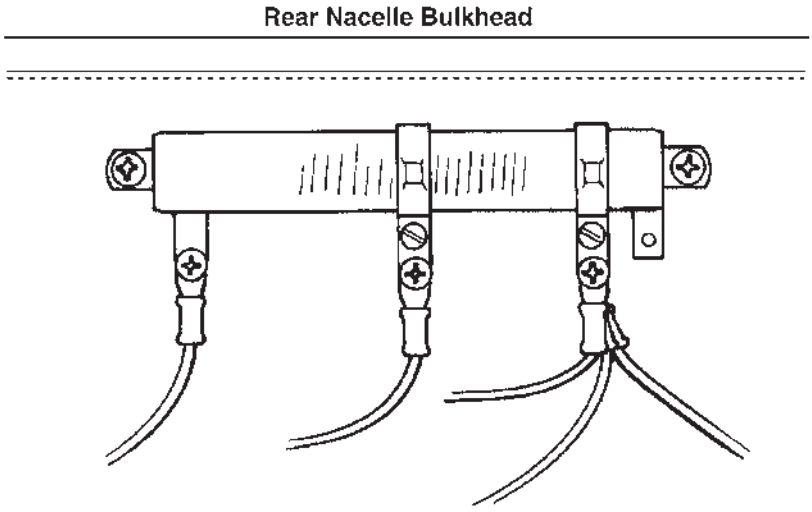
(e) Using a calibrated voltmeter, adjust the external power source to indicate 12.0 to 12.5 volts DC at the auxiliary fuel pump. Note the voltage reading on the external power source voltmeter.

(f) The calibrated pressure gauge should indicate 31 to 37 psi.

(g) Connect the positive lead from the external power source to the slider resistor low position. (Refer to “Figure 3” on page 28205.) Adjust the power supply voltage level to the same voltage obtained in Step (e).







Fuel Pump, Slide Resistor and Filter Installation
Figure 3

Effectivity
Seneca IV

(h) Adjust the slider on the variable resistor to obtain a pump pressure of 8 to 10 psi. Readjust the power supply and slider to ensure a pump pressure of 8 to 10 psi, at the power supply voltage noted in Step (e), then secure the slider in position on the resistor.

(i) Disconnect the manifold pressure switch located on the firewall, and connect the positive lead from the power supply to the slider resistor medium position.

(j) Adjust the power supply voltage level to the same voltage obtained in Step (e).

(k) Adjust the slider on the variable resistor to obtain a pump pressure of 23.5 to 24.5 psi. Readjust the power supply and slider to ensure a pump pressure of 23.5 to 24.5 psi, at the power supply voltage noted in Step (e), then secure the slider in position on the resistor and reconnect the manifold pressure switch.

(l) If the aircraft is equipped with an optional fuel diverter valve, operate the primer switch and ensure that the diverter valve is being energized and that pump is in high boost. Release the primer switch and operate the fuel pump switch in the Hi-Boost position and ensure that the fuel pump operates and that the diverter valve does not.

(m) Perform Steps (a) thru (n) on the opposite engine, then reinstall the access panels.

(n) Refer to Chapter 72, Engine Setup Procedures, for additional adjustments relating to the power plant fuel control system.

C. Auxiliary Fuel System Operational Check (Seneca IV only)

(1) Disconnect the external power source from the aircraft, and disconnect the electrical leads from the manifold pressure switch located on the firewall.

(2) Check to ensure that all cockpit controllable electrical equipment switches are in the off position, except as directed in the following steps.

(3) Place the master switch in its on position.

(4) Place the electric fuel pump switch in its low position. The calibrated fuel pressure gauge should indicate a pressure increase (not exceeding 10 psi), which would show that the pump is operating.

(5) Place the electric fuel pump switch in its high position. The calibrated fuel pressure gauge should indicate a pressure above 10 psi but not more than 24.5 psi.

(6) Place the aircraft master switch in the off position then reconnect the leads to the manifold pressure switch.

(7) Keep the fuel pump switch in its high position and return the master switch to its on position. The calibrated pressure gauge should indicate a pressure above 24.5 psi but not more than 37 psi.

(8) Place the fuel pump switch in its off position and depress the prime switch. The calibrated pressure gauge should indicate a pressure above 24.5 psi but not more than 37 psi.

5. Fuel Filters

A. Removal of Fuel Filter

Refer to “Figure 4” on page 28207.

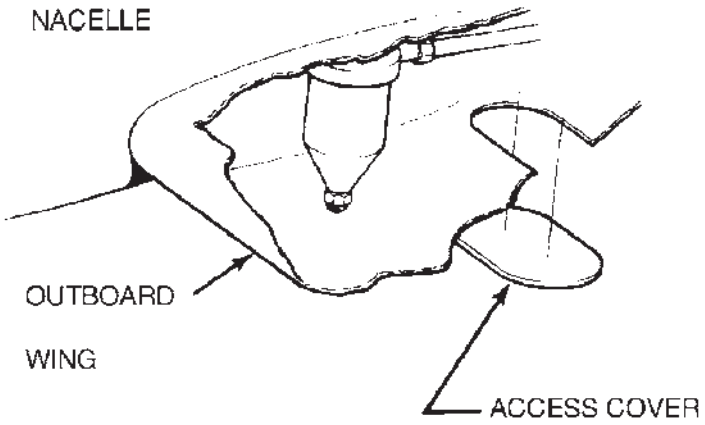
The instructions given are for the removal of the complete filter from the airplane. For cleaning and servicing purposes, only Steps (1) and (2) of this paragraph are necessary; then proceed to the next paragraph.

(1) Position the fuel selector valve to the OFF position.

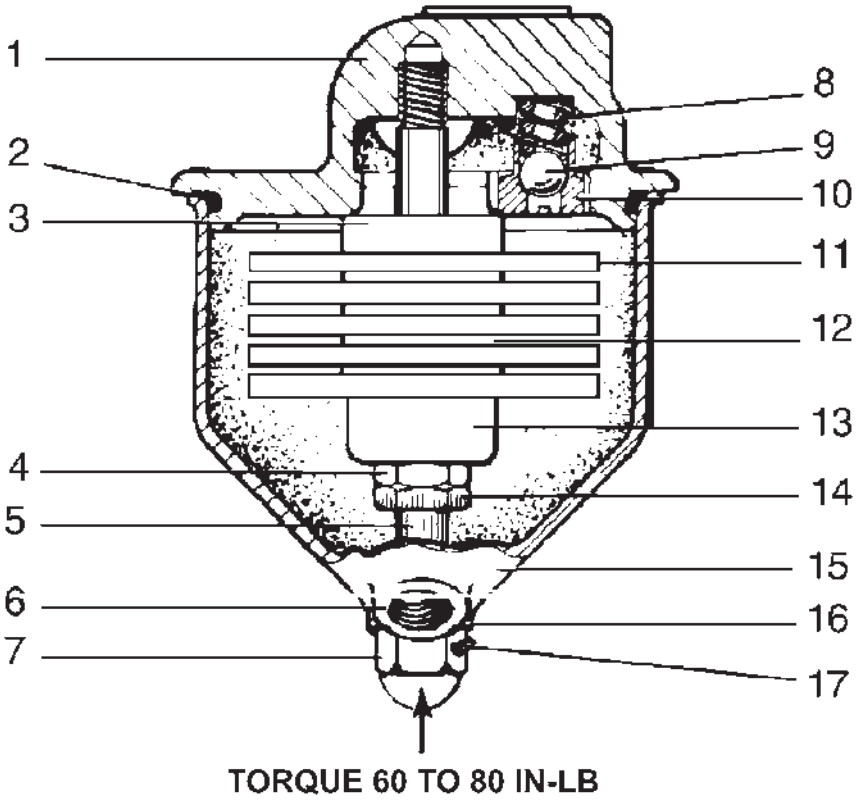
(2) Remove the access panel forward of the main spar, at Wing Station 91.00, on the bottom of the wing panel.

(3) Disconnect the filter drain line and fuel lines from the filter assembly. Cap the line ends to prevent contamination.

(4) Remove the bolts that secure the filter to its mounting bracket and remove the filter from the aircraft.



1. Body, Filter
2. O-Ring Seal
3. Tube, Outer
4. Nut
5. Stud
6. Drain, Filter, Fitting
7. Nut
8. Spring
9. Ball, Relief
10. Seat, Relief
11. Discs, Filter
12. Washers
13. Cup, Retainer
14. Nut, Check
15. Bowl, Filter
16. Washer
17. Safety Wire



Fuel Filter Assembly
Figure 4

B. Disassembly of Fuel Filter

Refer to “Figure 3” on page 28205 and “Figure 4” on page 28207.

(1) Cut safety wire and remove cap nut from the bottom of the filter bowl.

(2) Remove the bowl from the filter body.

(3) The O-ring seal may be removed from the body.

(4) Loosen and remove both the check nut and nut from the stud that holds the filter cartridge subassembly.

(5) Slide the filter cartridge from the stud. The filter discs and washers need not be separated from the filter cartridge.

(6) If necessary to disassemble the filter cartridge, remove the retainer cup from the outer tube and slide discs and washers from the outer tube. Do not use a screwdriver or sharp tool that may damage the discs.

(7) The filter bypass assembly may be removed by using the proper size screwdriver and turning out the relief seat. Remove relief ball and spring element outer tube for normal cleaning.

C. Cleaning, Inspection and Repair of Fuel Filter

(1) Carefully remove the filter pack from the housing and remove all O-rings, valves, springs, etc. Do not disassemble the filter pack from the center tube at this time.

(2) Plug the open ends of the filter and immerse in oil solvent such as Stoddard solvent. Let soak for 30 to 60 minutes.

(3) Metal valve parts may also be soaked in this cleaner.

(4) Remove filter and parts (if any) from the cleaner and rinse thoroughly in clean hot flowing tap water.

(5) Drain or blow off with filtered low pressure air to dry.

(6) Inspect the filter discs for damage and/or broken screens.

(7) Check condition of bowl O-ring seal and washer.

(8) Check for corrosion of filter parts.

(9) Check movement of bypass valve.

(10) Normal repairs necessary for the filter are replacement of bowl gaskets and damaged filter discs.

D. Assembly of Fuel Filter

Refer to “Figure 4” on page 28207.

(1) If removed, install bypass valve spring, relief ball and seat.

(2) Place the filter pack (assembled) on the housing stud. Ascertain that the end of the outer tube has positioned itself in the filter body.

(3) Secure the filter pack with nut. Torque nut 10 to 15 inch-pounds. Torque check nut against nut 40 to 60 inch-pounds.

(4) Place the O-ring packing on the housing and install bowl, washer and cap nut. Torque cap nut 60 to 80 inch-pounds and safety.

(5) Install the filter in the aircraft. If the filter was not removed, proceed to Step 3 of the next paragraph.

E. Installation of Fuel Filter

(1) Position the filter assembly in the wing. Ascertain that it is positioned properly and secure to its mounting bracket with the two bolts.

(2) Connect the drain line to the filter bowl.

(3) Connect the fuel lines to the filter assembly.

(4) Turn the fuel selector to the ON position and check for any fuel leaks.

(5) Install the access plate.

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INDICATING

NOTE: The fuel quantity calibration procedure for the Seneca V fuel quantity gauge is found in this section. Additional information on the fuel quantity gauge installed in Seneca V models is found in 77-40-00.

1. Seneca IV

A. Fuel Quantity Sender Units

Each fuel cell contains a sender unit which is interconnected with the other units of its particular system to provide the gauge with a combined, calibrated resistance.

The resistances of these units (1.0 ohm maximum at the empty position, and 45 ± 2 ohms at the full position) should be checked before installation. If any unit is assumed to be faulty check for proper travel, and resistance. Full travel of the units should be 95° ± 2°.

B. Installed Fuel Quantity Sender/Gauge Check

(1) Level airplane laterally and longitudinally, ± 1° (See 8-20-00).

(2) Ensure both wing tanks are completely empty. (See Fuel System, Draining Entirely, 12-10-00.)

(3) Place battery switch in OFF position.

(4) Connect external power supply unit to airplane's external power connector.

(5) Adjust power supply to provide 24 to 28 Vdc.

(6) With tanks completely dry, position fuel selector to left or right.

(7) Add 2 1/2 U. S. gallons of fuel to each tank. Check that needle of each (left and right) gauge points to "0". If not, adjust appropriate NULL trim potentiometer until needle centers on 0, +0, –1/2 needle width (Refer to "Figure 1" on page 28402.)

(8) Completely fill fuel tanks. Check that needle of each gauge points to "F". If not, adjust respective GAIN trim potentiometer until needle centers on F. (Refer to "Figure 1" on page 28402.)

(9) If gauge system fails to pass test specified in items 1 thru 6:

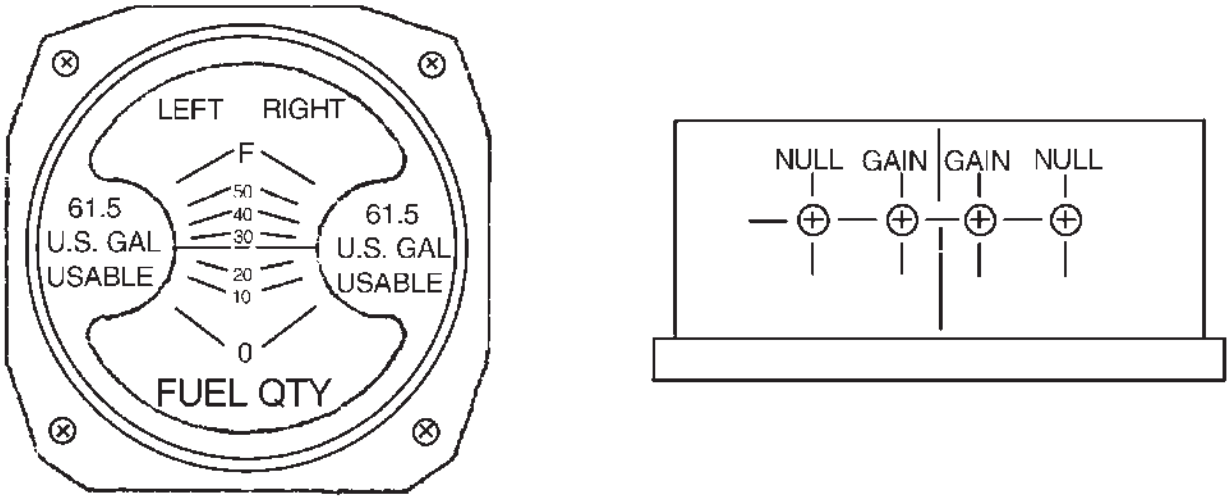
NOTE: Measure all fuel drained or added with a suitable device

(a) With tanks full, drain fuel from each tank per "Chart 1" on page 28402.

(b) After measured amount has been drained, vibrate tank by bumping its lower surface. Vibrate gauge by tapping gauge glass with fingers.

(c) The fuel quantity reading at each increment, after fuel has been drained, shall be within the tolerance specified in "Chart 1" on page 28402.

(10) Replace appropriate gauge and/or sender of any system that fails to meet the accuracy requirements specified above.



Seneca IV Fuel Quantity Indicators
Figure 1

CHART 1
FUEL QUANTITY SENDER GAUGE TOLERANCES (SENECA IV)

Additional Fuel Drained (U. S. Gallons)	***Total*** Fuel Remaining Within Tanks (U.S. Gallons)	Gauge Reading (U.S. Gallons)	Resistance Both Senders (Ohms)
0	64	F (± 1 1/2)	90.0
11 1/2	52 1/2	50 (± 1 1/2)	73.5
10	42 1/2	40 (± 1 1/2)	62.5
10	32 1/2	30 (± 1 1/2)	53.5
10	22 1/2	20 (± 1)	38.5
10	12 1/2	10 (± 3/4)	26.5
10	2 1/2	0 (+0, –1)	6.5

NOTE: Values in parentheses are needle-width tolerances.

NOTE: If starting test with empty tanks, add 2 1/2 U. S. gallons to each tank and start at bottom of table and progress upward.

2. Seneca V

A. DDMP Fuel Quantity Calibration

CAUTION: REPLACEMENT OF THE FUEL QUANTITY INDICATOR REQUIRES THAT THE NEW INDICATOR BE CALIBRATED PER THE PROCEDURE BELOW.

NOTE: Calibration is not required if the existing instrument is simply removed and reinstalled.

(1) Prepare the aircraft for testing as follows:

(a) Level the aircraft ± 1 degree laterally and longitudinally. (See 8-20-00.)

(b) Ensure both wing tanks are completely empty. (See Fuel System, Draining Entirely, 12-10-00.)

(c) Place the battery switch in the OFF position.

(d) Connect the external power supply unit to the aircraft electrical system using the APU connector.

(e) Adjust the power supply to provide 28.0 ± 1 VDC.

(f) Record test results in logbook.

(2) Initial Calibration Procedure

NOTE: If power failure occurs during calibration, the calibration must be started over.

(a) Enter the maintenance mode of the Digital Display Monitor Panel (DDMP) by turning the rotary switch to “electrical mode”, and pressing the key sequence up, down, up, up, “SEL”. If entered incorrectly, the DDMP will show an error message.

(b) The next screen allows you to select between maintenance mode, initiate self test, or set the clock. Press the “SEL” key to select maintenance mode.

(c) Press the “SEL” key to start the password sequence. Enter the password using the up/down arrow to change the character, and the “SEL” key to enter that character and move to the next. The password is currently “A130”. (See “Figure 2” on page 28405 for menu flow chart.)

(d) Select “fuel” using the up/down arrow to move the cursor, then press the “SEL” button to enter the selection.

(e) Select “left” tank, then press the “SEL” key.

(f) When “left cal complete re-cal” appears move cursor to “yes” and press “SEL”.

(g) Choose “yes” to delete all left info and press “SEL” key. (This will delete non-factory calibration program).

(h) When “left calibration data deleted” menu appears press “SEL” key.

(i) The next menu to appear will be “fuel cal; left; right; cancel,” move cursor to “left” and press “SEL” key.

(j) The DDMP will now query the instrument to find out what calibration step it is at. The instrument knows which quantities have been calibrated, and automatically goes to the next quantity to be calibrated. This requirement forces the instrument to be calibrated sequentially at “E”, 10, 20, 30, 40, 50, and “F”. Once the DDMP has queried the instrument, it displays the quantity to be calibrated, and gives the user three choices: “start the calibration”, “clear all”, or “cancel”. “Cancel” takes you out of the menu, with nothing being modified, “start calibration” begins the sequence and “clear all” uncalibrates the instrument. In other words, the clear command will delete all auxiliary calibration data, and revert to the factory calibration.

(k) Add 3.0 gallons of fuel (unusable) to left wing tank. Select “calibration” from menu in DDMP. Use up/down buttons to move cursor to selection noted above and enter by pressing the “SEL” key. After selection has been made “o.o” will appear in upper part of right window for the first quantity to be calibrated. Wait one minute for the fuel to stop sloshing. Press “SEL” key and in middle of right window a message “working” will flash. In lower portion of left window, the digits will change until the number is within + or - (0.1) of the number in upper portion of right window. When this occurs, the message changes from “working” to “pass”. If “error” message appears, the resistance of the tank senders is out of range and one or both senders may have to be replaced. Also, during this step, a horn will sound indicating a low fuel level (5 + or - 1 gallon) condition. Depress “SEL” key and the horn will be canceled.

(l) Once the “SEL” key is depressed after the message “pass” has appeared, the menu will display allowing a choice between “right” or “left” tanks. Select “left” tank and depress “SEL” key. (10) will appear in the upper portion of the right screen. Add 10 gallons to left tank and wait one minute for fuel to stop sloshing. Press “SEL” key and a “working” message will appear. When the digits in the lower portion of the left window agree to within + or - (0.1) of the (10) indicated in upper portion of right window, the “working” message will change to “pass” and the “SEL” key can be depressed.

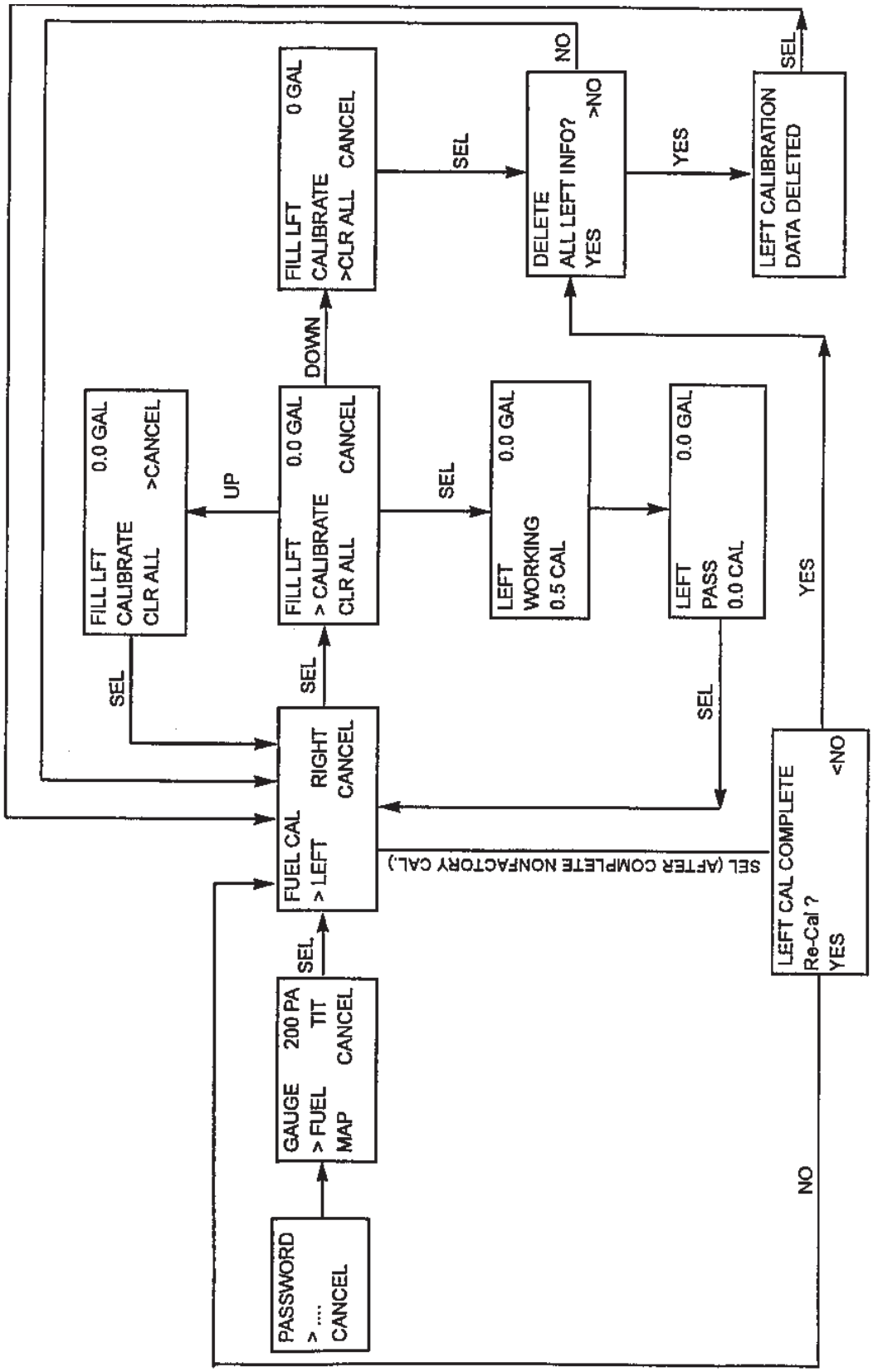
(m) Repeat step (12) for 20, 30, 40, 50, and (F) calibration points.

NOTE: The calibration must be done sequentially; otherwise, a possible error could be introduced into the program.

(n) When “complete” message is displayed, depress “SEL” key and use cursor to select right tank and hit “SEL” to accept.

(o) Repeat steps (j), (k), (l), (m), and (n), for right tank.

(p) When right tank calibration is complete, turn rotary switch to “Inst” mode. This will save the calibration.



Fuel Calibration Menu Flow Chart (Seneca V)
Figure 2

B. Avidyne Fuel Quantity Calibration

From Engine Setup Page

CAUTION REPLACEMENT OF ANY OF THE FUEL QUANTITY SENDERS OR REPLACEMENT OF THE MULTI-FUNCTION DISPLAY REQUIRES RECALIBRATION.

(1) Prepare the aircraft for testing

(a) Level the aircraft ± 1 degree laterally and longitudinally. (See 08-20-00.)

(b) Ensure both wing tanks are completely empty. (See Fuel System, Draining Entirely, 12-10-00.)

(c) Place the battery switch in the OFF position.

(d) Connect the external power supply unit to the aircraft electrical system using the APU connector.

(e) Adjust the power supply to provide 28.0 ± 1 VDC.

(2) Initial Calibration Procedure.

(a) From the maintenance page, depress line select key (L4) to access the Engine setup Page Depress the Fuel Cal button (L1) to enter the fuel calibration page

(b) Depress “Begin Cal” button (R3) and follow the on-screen calibration procedure for left & right tank zero fuel (3.0 +.1-0 gal. unusable) and full fuel calibration (61 +0;-1gal. usable) points.

(c) Use the Left Knob to select the current calibration point. The selected calibration point is highlighted and the value displayed is the current reported fuel quantity from the DAU. A message at the bottom of the screen prompts the operator to add the appropriate amount of fuel and then to select “Accept Value” once the value reported form the DAU has stabilized.

(d) If the DAU reported value is not within 2.5 gallons of the test point value, a message “DAU Reported Fuel Quantity Out of Tolerance” will be presented and the value will not be accepted.

(e) Once all points have been calibrated, the operator presses “Calibration Complete” to cause the calibration factors to be computed and applied to the DAU reported fuel quantity.

(f) Other options from the Calibration Underway state are to “Restore Last Cal” and “Clear Cal”.

4) Pressing Restore Last Cal” causes the calibration values from the last completed calibration to be restored and the state to change to Calibrated.

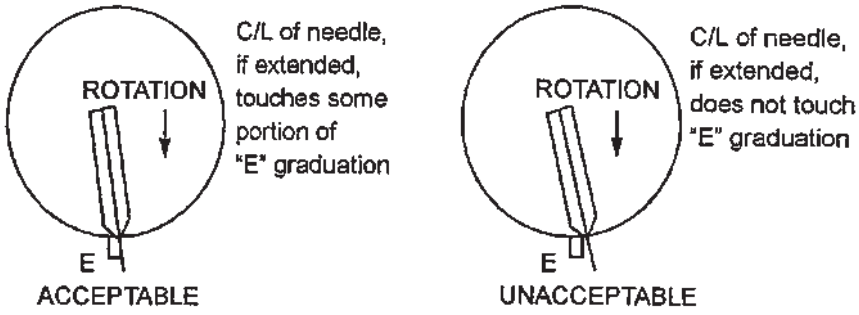
5) Pressing “Clear Cal” causes all calibration values to be cleared and the state to change to “Not Calibrated”. An “Are Your Sure?” prompt will give the operator a chance to reconsider the decision to either “Restore Last Cal” or “Clear Cal”.

(g) When the calibration procedure has been completed, press the Save button. If you decide not to save the changes, pressing the Cancel button from the Underway state causes the current calibration session to be aborted with any unsaved interim calibration values being discarded. Changes will not take effect until the MFD has been restarted.

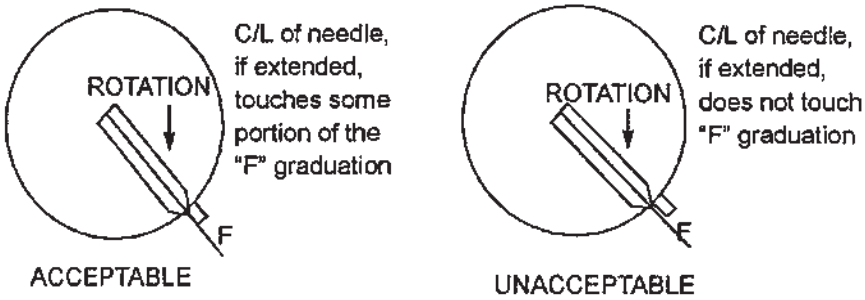
NOTE: The “empty” and “full” fuel readings on this set-up page will NOT display zero(0) or 61 gallons usable fuel for the respective empty and full calibration points. However, the appropriate zero and full (61 U.S. gallons) will be displayed on the MFD in normal operation mode (not maintenance set-up).

C. Full Gauge Range Check

Defuel both fuel tanks. Add 3 gallons of fuel to each tank (unusable). Verify that fuel gauges indicate “O” or “E”. See “Figure 3” for analog needle display limits at empty. Add 10 gallons of fuel to each tank and again observe analog and digital gauge readings. Repeat this procedure for 20, 30, 40, 50, and “F” gallons for left and right tanks. For analog needle display limits at full, See “Figure 4” on page 28407. Allow ±2 gallon tolerance at each increment except for “E”, 3 gallons (unusable) and “F”. “E” and 3 gallons (unusable) have ±0 tolerance and “F” (full) tolerance is +0/-1 gallons. Record data in a xerox copy of “Chart 2” on page 28408.



Analog Needle Display Limits at Empty (Seneca V)
Figure 3



Analog Needle Display Limits at Full (Seneca V)
Figure 4

CHART 2
FUEL CALIBRATION (SENECA V)

FUEL CALIBRATION				
TANK CAPACITY (EITHER SIDE) (GALLONS)	RESISTANCE [1] (OHMS) (± 1)	GAUGE READINGS [2] ANALOG	GAUGE READINGS [2] DIGITAL	TOLERANCE (GALS) (ON GAUGE READING)
0.0 (E)	0.5			± 0
(3) Gal. Unusable	0.5			± 0
10	24.0			± 2
20	38.0			± 2
30	52.0			± 2
40	64.0			± 2
50	75.0			± 2
61 (F)	93.0			+ 0, - 1

1 - Reference only - not a required measurement.

2 - Allowed 1 gallon spread between analog and digital readings except at “0”, “(3) gal unusable” and “Full”. Digital and analog must read the same at “0”, “(3) gal unusable” and “Full”.

D. Garmin G1000/G1000 NXi Fuel Quantity Indication

(1) Description

The fuel quantity indication is incorporated into the Garmin G1000/G1000 NXi Multi-Function Display (MFD). The raw data from the 4 fuel senders (2 per side) is provided to the GEA's which in turn output fuel quantity and diagnostic information to the GIA's. The data then is fed through the respective PFDs and displayed on the MFD.

If a sender or wire failure were to occur, the GEA sends an invalid data bit to the Garmin G1000/G1000 NXi system which in-turn will apply a RED-X across the fuel indicator on the affected tank side.

(2) Calibration

(PIR-PPS55032, Rev. D. / 107867, Rev. C./34G34A020, Rev. New)

CAUTION: ENTER "MAINTENANCE MODE" WHEN ACTIVATING THE GARMIN G1000 SYSTEM ON THE GROUND TO CONDUCT TESTS, INSPECTIONS, ETC. SEE MAINTENANCE MODE UNDER INTEGRATED AVIONICS SYSTEM - GARMIN G1000 IN 34-25-01 OR G1000 NXi IN 34-25-02.

(a) Setup

1) Level the airplane per 8-20-00.
2) Drain the fuel tanks per 12-10-00, Fuel System, Draining Fuel System - Complete.
3) Pull OUT the #1 PFD, #2 PFD, and MFD circuit breakers.

(b) Procedure

NOTE: If the squat switch is deactivated due to jacking of aircraft, pull the hydraulic pump circuit breaker prior to performing the fuel tank calibration.

NOTE: The large FMS Knob changes Groups and the small FMS knob changes Pages within a Group.

1) Turn on power to the aircraft.
2) Start the #1 PFD, MFD, and #2 PFD in the configuration mode by holding the ENT key on the PFDs, and the far right softkey on the MFD while applying power (i.e., pushing IN the #1 PFD, #2 PFD, and MFD circuit breakers).

NOTE: The G1000/G1000 NXi System must be powered for three minutes before proceeding. If aircraft is "IN-AIR CONFIGURATION LOCKOUT", add a jumper wire to the terminals of one of the squat switches before proceeding to remove the lockout.

3) Add unusable fuel to both wings (3 US gal. per side).
4) After 5 minutes, select the 'CAL' group, then select 'FUEL CALIBRATION' page on 'PFD #1'.
5) Unlock the page by pressing softkeys 12 (far right softkey), 11, 10, 9 in sequence.
6) Verify the 'CALIBRATION VALUE' indication for the Left Tank is stable before proceeding.
7) Press the 'RAW DATA' softkey to de-select the scale function.
8) Press the 'EMPTY' softkey. Press the 'ENT' key to acknowledge the prompt after the 'EMPTY' softkey is pressed.
9) Verify the 'ACTUAL QUANTITY' shown in the "CALIBRATION TABLE" in the lower right corner of the screen shows 0 gallons.
10) Press the 'TNK SEL' softkey.
11) Rotate the small FMS knob to generate a pick list and select the Right Tank.
12) Press the 'ENT' key.

13) Repeat items (6) thru (9) for the right tank.

NOTE: For actual fuel quantity entries, the large FMS knob moves the cursor. The small FMS knob changes the digits.

14) Add 15 gal fuel to each wing.
15) After at least 5 minutes, Verify the 'CALIBRATION VALUE' indication for the Right Tank is stable before proceeding.
16) Press the 'FUEL QTY' softkey.
17) Enter 15 gal for 'ACTUAL FUEL QUANTITY'. Press the 'ENT' key to accept.
18) Press 'ENT' Key again to calibrate.
19) Verify the 'ACTUAL QUANTITY' shown in the "CALIBRATION TABLE" in the lower right corner of the screen shows 15 gallons.
20) Press the 'TNK SEL' softkey.
21) Rotate the small FMS knob to generate a pick list and select the Left Tank.
22) Press the 'ENT' key.
23) Repeat items (15) thru (19) for the Left Tank.
24) Add 15 gal fuel to each wing.
25) After at least 5 minutes, Verify the 'CALIBRATION VALUE' indication for the Left Tank is stable before proceeding.
26) Press the 'FUEL QTY' softkey.
27) Enter 30 gal for 'ACTUAL FUEL QUANTITY'. Press the 'ENT' key to accept.
28) Press 'ENT' Key again to calibrate.
29) Verify the 'ACTUAL QUANTITY' shown in the "CALIBRATION TABLE" in the lower right corner of the screen shows 30 gallons.
30) Press the 'TNK SEL' softkey.
31) Rotate the small FMS knob to generate a pick list and select the Right Tank.
32) Press the 'ENT' key.
33) Repeat items (25) thru (29) for the Right Tank.
34) Add 15 gal fuel to each wing.
35) After at least 5 minutes, Verify the 'CALIBRATION VALUE' indication for the Right Tank is stable before proceeding.
36) Press the 'FUEL QTY' softkey.
37) Enter 45 gal for 'ACTUAL FUEL QUANTITY'. Press the 'ENT' key to accept.
38) Press 'ENT' Key again to calibrate.
39) Verify the 'ACTUAL QUANTITY' shown in the "CALIBRATION TABLE" in the lower right corner of the screen shows 45 gallons.
40) Press the 'TNK SEL' softkey.
41) Rotate the small FMS knob to generate a pick list and select the Left Tank.
42) Press the 'ENT' key.
43) Repeat items (35) thru (39) for the Left Tank.
44) Add 16 gal more fuel to each wing.

NOTE: If 16 gallons cannot be entered, note the fuel quantity added to the tank until full to the filler neck.

45) After at least 5 minutes, Verify the 'CALIBRATION VALUE' indication for the Left Tank is stable before proceeding.

46) Press the ‘FULL’ softkey. Press the ‘ENT’ key to acknowledge the prompt after the ‘FULL’ softkey is pressed.

47) Verify the ‘ACTUAL QUANTITY’ shown in the “CALIBRATION TABLE” in the lower right corner of the screen shows 61 gallons.

48) Press the ‘TNK SEL’ softkey.

49) Rotate the small FMS knob to generate a pick list and select the Right Tank.

50) Press the ‘ENT’ key.

51) Repeat items (45) thru (47) for the Right Tank.

(c) Full Range Accuracy Check

NOTE: If an aircraft fails the fuel accuracy check, troubleshoot to determine cause and corrective action and repeat the fuel calibration and accuracy check, if required.

1) Level the airplane per 8-20-00.

2) Drain the fuel tanks per 12-10-00, Fuel System, Draining Fuel System - Complete.

3) Remove power from aircraft. Turn on power to the aircraft. Allow PFD and MFD to start in normal mode.

4) At MFD splash screen press softkeys in the following sequence: 1, 2, 3, 1. Then press the far right softkey.

5) Verify Maintenance Mode CAS advisory on PFD.

6) Defeat both squat switches.

7) Add unusuable fuel to both wings (3 US gal per side) and allow to stabilize (5 min.).

8) Verify the Garmin fuel indicator shows left and right fuel quantities at 0 gal.

9) Verify fuel quantity gauge digits are RED and “L FUEL QTY” and “R FUEL QTY” warning CAS messages are displayed.

10) Add 10 gal of fuel to both wings and verify the corresponding reading on the Garmin system is within +/- 4 gal per side (+/- 8 gal total).

11) Verify that the fuel quantity gauge digits turn from RED to YELLOW and “L FUEL QTY” and “R FUEL QTY” warning CAS messages are extinguished when fuel quantity is greater than 5 gal of fuel per side. “L FUEL QTY” and “R FUEL QTY” warning CAS messages should appear after 10 seconds.

12) Continue adding fuel in 10 gal increments (total of 5 increments) and verify the corresponding reading on the Garmin system is within +/- 4 gal per side (+/- 8 gal total).

13) Verify fuel quantity gauge digits turn GREEN and “L FUEL QTY” and “R FUEL QTY” warning CAS messages are extinguished within 10 seconds upon reaching greater than 10 gallons of fuel per side.

14) If quantity readings at all intervals are within limits, then fuel accuracy check is complete. Turn aircraft power OFF.

(d) Calibration Table Delete/Edit

This procedure can be used to edit or delete an existing calibraiton point within the calibration table.

1) Press the CAL TBL softkey.
2) Scroll down to the entry to edited or deleted.
3) To edit the entry press the EDIT softkey. Fill the selected tank to the level of the selected entry. Allow the CALIBRATION VALUE to stabilize.
4) Press enter and confirm you want to overwrite the existing calibration point.
5) To delete the entry press the DELETE softkey.
6) Press and confirm you want to delete the calibration point.

CHAPTER

29

HYDRAULIC POWER

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CHAPTER 29

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
29-List of Effective Pages	1	Oct 10/23			
	2	Sep 30/18			
29-Table of Contents	1	Oct 10/23			
	2	Sep 30/18			
29-00-00	1	Sep 30/18			
	2	Sep 30/18			
	3	Sep 30/18			
	4	Sep 30/18			
	5	Sep 30/18			
	6	Sep 30/18			
	7	Sep 30/18			
	8	Sep 30/18			
29-10-00	1	Sep 30/18			
	2	Sep 30/18			
	3	Sep 30/18			
	4	Sep 30/18			
	5	Sep 30/18			
	6	Sep 30/18			
	7	Sep 30/18			
	8	Oct 10/23			

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CHAPTER 29 - HYDRAULIC POWER

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GENERAL

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

WARNING: USE ONLY GENUINE PIPER AIRCRAFT PARTS OR PIPER AIRCRAFT APPROVED PARTS OBTAINED FROM PIPER APPROVED SOURCES, IN CONNECTION WITH THE MAINTENANCE AND REPAIR OF PIPER AIRPLANES.

CAUTION: PRIOR TO STARTING ANY INVESTIGATION OF THE HYDRAULIC SYSTEM, PLACE THE AIRPLANE ON JACKS (SEE JACKING, 07-10-00.)

NOTE: The appropriate source for identifying the correct part number of a component used on these airplanes is the applicable Piper parts catalog.

The hydraulic system components covered in this section consist of the combination hydraulic pump and reservoir, hydraulic pressure switch, free-fall valve assembly, actuating cylinders and hydraulic lines. The brake system, although hydraulically operated, is not included in this section as it has its own hydraulic system independent of the gear retraction system. The brake system along with landing gear and components is covered in 32-00-00.

This section provides instructions for remedying difficulties which may arise in the operation of the hydraulic system. The instructions are organized so that the mechanic can refer to: Description. for a basic understanding of the system; Troubleshooting, for a methodical approach in locating difficulty; Corrective Maintenance, for the removal, repair and installation of components; and; Adjustments and Checks, for the operation of the repaired system.

1. Description

See “Chart 2” on page 29005.

Hydraulic fluid to the landing gear actuating cylinders is supplied by an electrically powered reversible pump located in the right forward area of the fuselage nose section. A reservoir is an integral part of the pump. The pump is controlled by a selector handle on the instrument panel, to the left of the control quadrant. As the handle is placed in either the up or down position, the pump directs fluid through the particular pressure line to each individual actuating cylinder. As fluid pressure increases at one side of a cylinder piston, fluid at the other side is directed back through the other line to the pump. Both lines serve either as pressure or return passages depending on the rotation of the pump to retract or extend the gear.

A pressure switch is mounted on the pressure line in the right aft side of the nose cone. This switch opens the electrical circuit to the pump solenoid when the gear fully retracts and pressure in the system increases to 1800 + 100 psi. The switch will continue to hold the circuit open until pressure in the system drops 200 to 400 psi; when at that time the pump will again operate to build up pressure as long as the gear selector is in the up position. The down position of the selector handle does not affect the pressure switch.

The Oildyne hydraulic pump is a gear type unit driven by a 24 volt reversible motor designed to operate in a pressure range of 2,400 ± 200 psi. To prevent excessive buildup of pressure in the hydraulic system due to expansion, a primary thermal relief valve is incorporated in the pump body which will open at 3,000 +300/-200 psi. Other valves in the pump, channel fluid to the proper outlet during retraction or extension of the gear. A shuttle valve located in the base of the pump allows fluid displaced by the cylinder pistons to return to the reservoir without back pressure. This shuttle valve has a delivery pressure of 400-800 psi during the extension cycle.

A bypass free-fall valve assembly is incorporated in the system to permit extension of the landing gear should a malfunction in the system occur. This valve is manually operated by means of an emergency gear knob located on the instrument panel. This knob must be fully extended to permit emergency extension. Restrictions in the system prevent the gear from extending too fast.

CHART 1
HYDRAULIC SYSTEM LEADING PARTICULARS (WITH OILDYNE PUMP)

Hydraulic Pump: High Pressure	2,400± 200 psi
Low Pressure	600 ± 200 psi
Flow Rate @ 1000 psi	60 cu. in. per minute
High Pressure Control	2,400± 200 psi
Hydraulic Fluid	MIL-H-5606
Thermal Relief Valve	3,000+ 300/-200 psi
Shuttle Value Delivered Pressure	400 to 800 psi
Pressure Switch	
Open (OFF) Pressure	1,800 ± 100 psi
Close (ON) Pressure	Pressure decreasing 200 to 400 psi

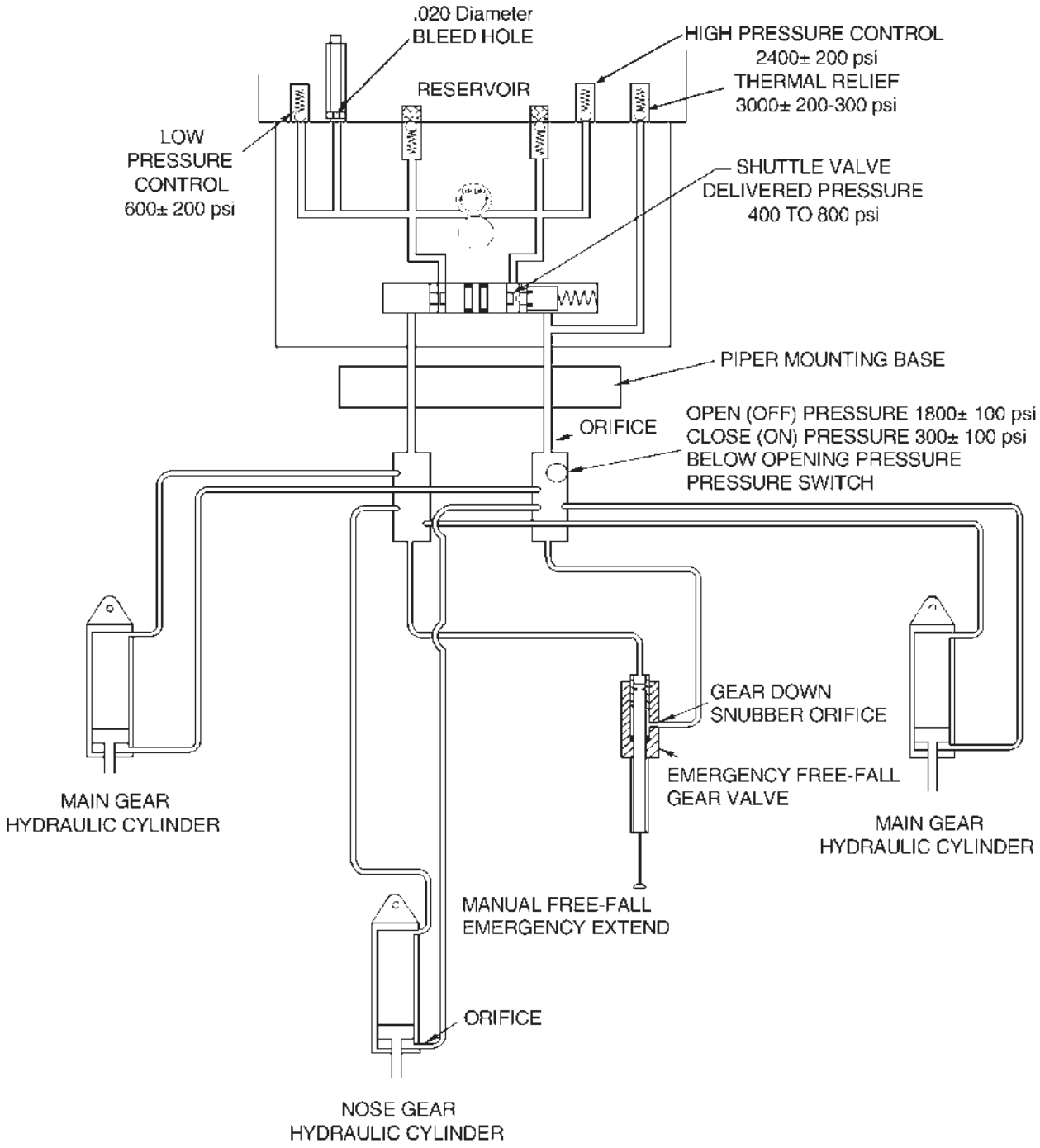
For a description of the landing gear and electrical switches, see 32-00-00.

2. Troubleshooting

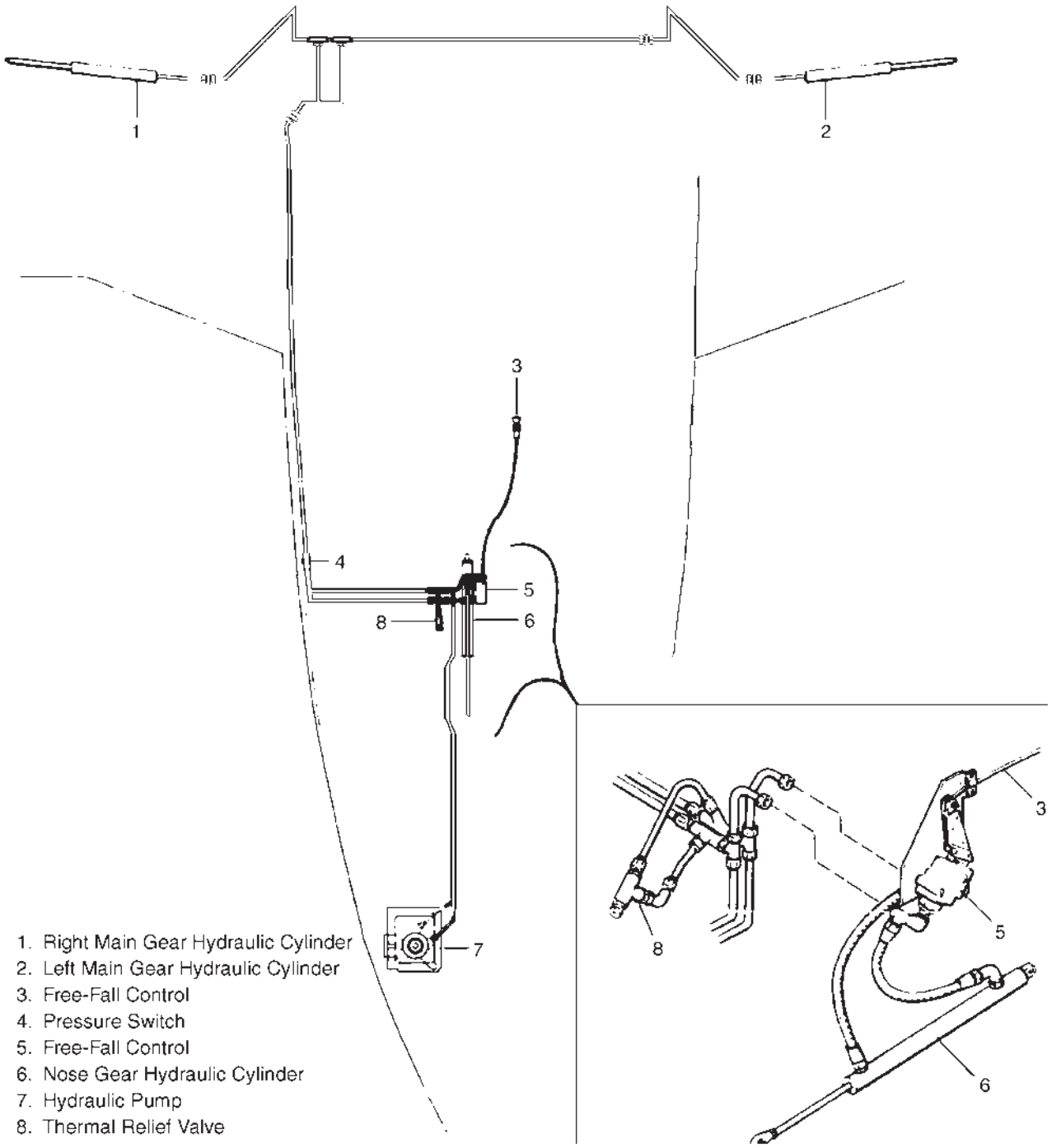
CAUTION: PRIOR TO STARTING ANY INVESTIGATION OF THE HYDRAULIC SYSTEM, PLACE THE AIRPLANE ON JACKS. WITH THE AIRPLANE ON JACKS, PULL THE FREE-FALL VALVE KNOB FULL OUT, THUS PREVENTING THE BUILDUP OF UNNECESSARY PRESSURE ON THE ACTUATING CYLINDERS AND CONNECTING HYDRAULIC LINES WHEN THE GEAR IS RAISED OR LOWERED MANUALLY. FAILURE TO COMPLY WITH THESE INSTRUCTIONS COULD RESULT IN THE BUILDUP OF SUFFICIENT PRESSURE TO UNLOCK THE DOWN LOCK MECHANISM ALLOWING THE GEAR TO COLLAPSE WHEN THE WING JACKS ARE REMOVED. PRIOR TO REMOVING THE AIRPLANE FROM JACKS, PUSH THE FREE-FALL VALVE KNOB IN, TURN ON THE MASTER SWITCH AND SELECT GEAR DOWN, OBSERVE THAT ALL THREE GREEN LIGHTS INDICATING THE LANDING GEAR IS DOWN AND LOCKED ARE ENERGIZED. TURN MASTER SWITCH OFF.

Malfunctions in the hydraulic system will result in failure of the landing gear to operate properly. When trouble develops, jack up the airplane (see Jacking, 07-00-00) and then proceed to determine the extent of the trouble. Generally, hydraulic system troubles fall into two types; trouble involving the hydraulic supplying system and troubles in the landing gear hydraulic system. “Chart 2” on page 29005 lists the troubles which may be encountered and their probable cause, and suggests a remedy for the trouble involved. A hydraulic system operational check may be conducted using “Figure 1” on page 29003 and “Figure 2” on page 29004. When the trouble has been recognized, the first step in troubleshooting is isolating the cause. Hydraulic system troubles are not always traceable to one cause. It is possible that a malfunction may be the result of more than one difficulty within the system. Starting first with the most obvious and most probable reasons to the trouble, check each possibility and, in turn, by process of elimination, isolate the troubles.

NOTE: Field service of the Oildyne hydraulic pump is limited to motor replacement and removal, cleaning, and inspecting the hydraulic fluid reservoir. Should pump malfunction, replace pump, or return pump to Piper via your local Piper distributor for servicing or repairs.



Hydraulic System Schematic Diagram (Oildyne)
Figure 1



Hydraulic System Installation
Figure 2

CHART 2 (Sheet 1 of 4)
TROUBLESHOOTING HYDRAULIC SYSTEM

Trouble	Cause	Remedy
Landing gear retraction system fails to operate.	Landing gear actuator circuit breaker open.	Reset circuit breaker and determine cause for open circuit breaker.
	Landing gear selector circuit breaker open.	Reset circuit breaker and determine cause for open circuit breaker.
	Landing gear actuator circuit wires broken.	Check wiring.
	Landing gear selector circuit wires broken.	Check wiring.
	Safety (squat) switch out of adjustment.	Readjust switch. (Refer to Adjustment of Safety Switch, Chapter 32.)
	Squat switch inoperative.	Replace switch.
	Pressure switch inoperative.	Replace switch.
	Pump retraction solenoid in-operative (upper solenoid).	Replace solenoid.
NOTE: If the retracting solenoid of the pump can be heard to actuate when operating the gear selector switch, it may be assumed that the gear control circuit is operating properly and the actuator circuit should be further checked.		
	Gear selector switch ground incomplete.	Check ground.
	Gear selector switch inoperative.	Replace switch.
	Hydraulic pump ground incomplete.	Check ground.
	Hydraulic pump inoperative operative.	Replace or overhaul pump.
	Hydraulic fluid in reservoir below operating level.	Fill reservoir with hydraulic fluid.
	Battery low or dead.	Check condition of battery.
	Check for internal leakage of free-fall valve.	Replace valve.
	Check for internal leakage of gear up check valve in pump.	Replace or overhaul pump.

CHART 2 (Sheet 2 of 4)
TROUBLESHOOTING HYDRAULIC SYSTEM

Trouble	Cause	Remedy
Landing gear extension system fails to operate.	Landing gear actuator circuit breaker open.	Reset circuit breaker and and determine cause for open circuit breaker.
	Landing gear selector circuit breaker open.	Reset circuit breaker and determine cause for open circuit breaker.
	Landing gear actuator circuit wires broken.	Check wiring.
	Landing gear selector circuit wires broken.	Check wiring.
	Pump extension solenoid inoperative (lower solenoid).	Replace solenoid.
NOTE: If the extension solenoid of the pump can be heard to actuate when operating the gear control circuit is operating properly and the actuator circuit should be further checked.		
	Gear selector switch ground incomplete.	Check ground.
	Gear selector switch inoperative.	Replace switch.
	Hydraulic pump ground incomplete.	Check ground.
	Hydraulic pump inoperative.	Replace or overhaul pump.
	Hydraulic fluid in reservoir below operating level.	Fill reservoir with hydraulic fluid.
	Low or dead battery.	Check condition of battery.
Landing gear retraction extremely slow.	Hydraulic fluid in reservoir below operating level.	Fill reservoir with hydraulic fluid.
	Restriction in hydraulic lines.	Isolate and check hydraulic lines.
Pump stops during gear retraction.	Landing gear actuator circuit breaker opens.	Reset circuit breaker and determine cause for overload.
	Landing gear selector circuit breaker opens.	Reset circuit breaker and determine cause for overload.
	Pressure switch out of adjustment.	Remove and readjust or replace switch.
	Mechanical restriction or obstruction in hydraulic system to allow pressure to build up and shut off pump before gear has retracted.	Place airplane on jacks and run retraction check. Isolate and determine cause.

CHART 2 (Sheet 3 of 4)
TROUBLESHOOTING HYDRAULIC SYSTEM

Trouble	Cause	Remedy
Pump stops during gear extension	Landing gear actuator circuit breaker opens	Reset circuit breaker and determine cause for overload.
	Landing gear selector circuit breaker opens.	Reset circuit breaker and determine cause for overload.
Pump fails to shut off though gear has fully retracted.	Pressure switch in-operative.	Replace switch.
	Pressure switch out of adjustment.	Replace switch.
	Pump retraction solenoid sticking (inboard solenoid).	Replace solenoid.
	Internal leakage of system.	Check gear actuating cylinders and free-fall valve for internal leakage.
		Check for internal damage to hydraulic pump.
	External leakage of system.	Check gear actuating cylinders for external leakage.
		Check for broken or damaged hydraulic lines or hoses.
	Pump relief valve out of adjustment.	Replace pump.
Pump fails to shut off though the gear has fully extended.	Pump extension solenoid sticking (outboard solenoid.)	Replace solenoid.
	Nose gear down limit switch actuator out of adjustment.	Adjust switch actuator. (Ref. Nose Gear Down Limit Switch, Chapter 32.)
	Nose gear down limit switch failed.	Replace switch.
	Main gear down limit switch out of adjustment.	Adjust switch. (Refer to Adjustment of Main Gear Down Limit Switch, Chapter 32.)
	Main gear down limit switch failed.	Replace switch.

NOTE: The out of adjustment or failed switch may be determined by noting which down light is not lit.

CHART 2 (Sheet 4 of 4)
TROUBLESHOOTING HYDRAULIC SYSTEM

Trouble	Cause	Remedy
Pump running intermittently after gear has retracted.	Leakage of high pressure check valve.	Remove pump and replace check valve.
	Internal leakage of system.	Check free-fall valve for internal leakage.
		Check gear actuating cylinders for internal leakage.
	External leakage of system.	Check gear actuating cylinders for external leakage.
		Check for broken or damaged hydraulic lines.
Gear stops part way up, but pump continues to run	Pump high pressure relief valve out of adjustment.	Replace pump.
	Internal leakage of system.	Check gear actuating cylinders and free-fall valve for internal leakage.
		Check for broken or damaged hydraulic lines.
	Hydraulic fluid in reservoir below operating level.	Fill reservoir with hydraulic fluid.
All gears fail to free-fall.	Free-fall valve fails to open.	Check valve and replace.

MAIN

WARNING: USE ONLY GENUINE PIPER AIRCRAFT PARTS OR PIPER AIRCRAFT APPROVED PARTS OBTAINED FROM PIPER APPROVED SOURCES, IN CONNECTION WITH THE MAINTENANCE AND REPAIR OF PIPER AIRPLANES.

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

1. Hydraulic Pump

NOTE: No field repair of Oildyne hydraulic pumps is authorized. Any faulty hydraulic pump must be replaced, or returned to Piper via the local Piper distributor for repairs.

NOTE: The appropriate source for identifying the correct part number of a component used on these airplanes is the applicable Piper parts catalog.

A. Removal

See “Figure 1” on page 29102.

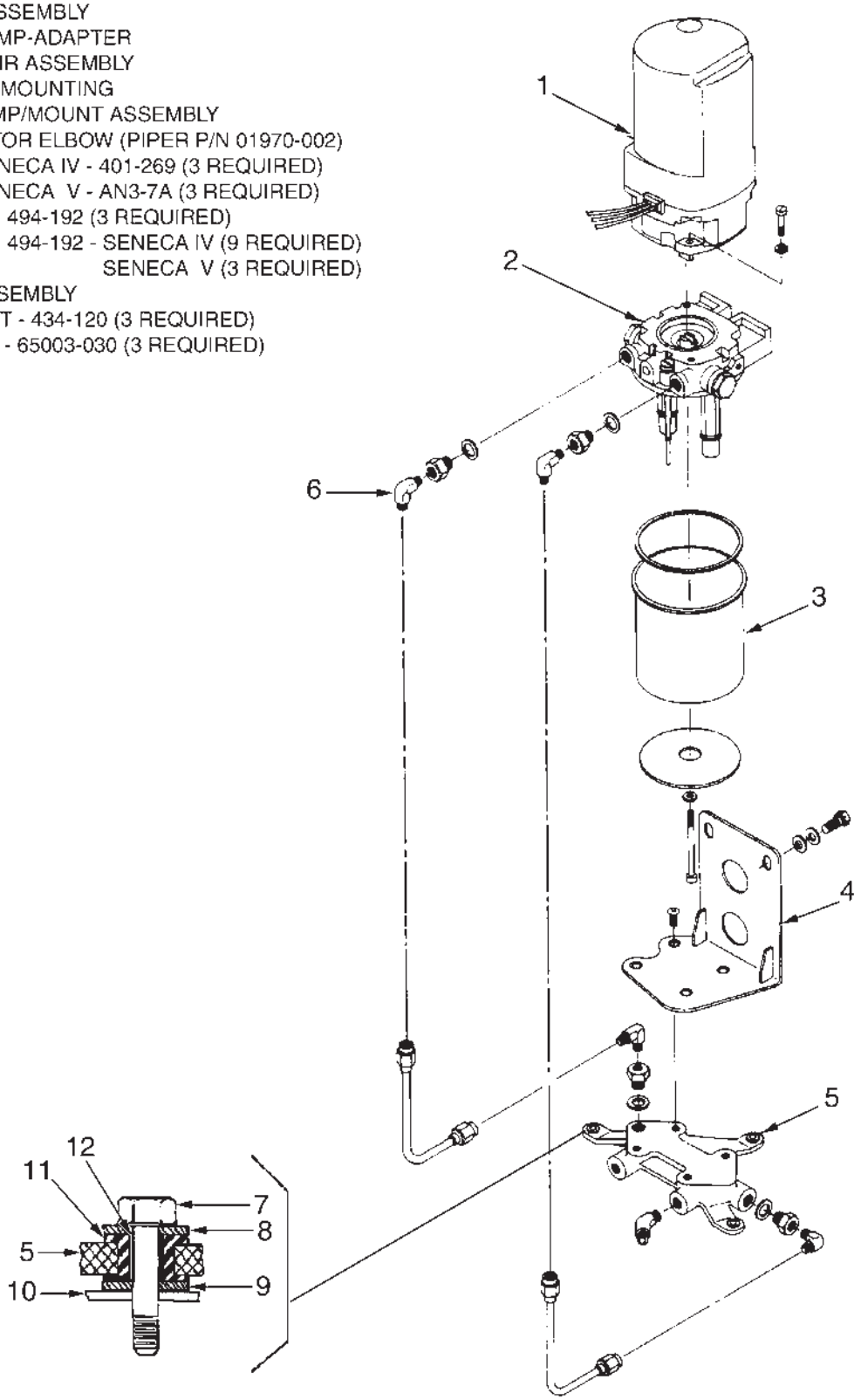
The Oildyne hydraulic pump with reservoir incorporated is located in the nose section of the fuselage. Access to the pump is through the access panel in the nose baggage compartment.

(1) Remove the ABS nose gear cover.
(2) Remove anti-splash cover (Piper P/N 96374-0) by removing the four attaching screws.
(3) Disconnect the electrical harness:
  (a) in early airplanes (1995–2015), three knife connectors that attach the black, blue, and green forward and reverse harness wires.
  (b) in later airplanes (2015 and up), a quick disconnect connector.
(4) If installed, disconnect the grounding strap.
(5) Disconnect and plug the “up” and “down” pressure hydraulic lines from pump mount. Cap the lines.
(6) Remove the three each mounting bolts and washers securing pump mount to deck.
(7) Lift assembly from airplane.

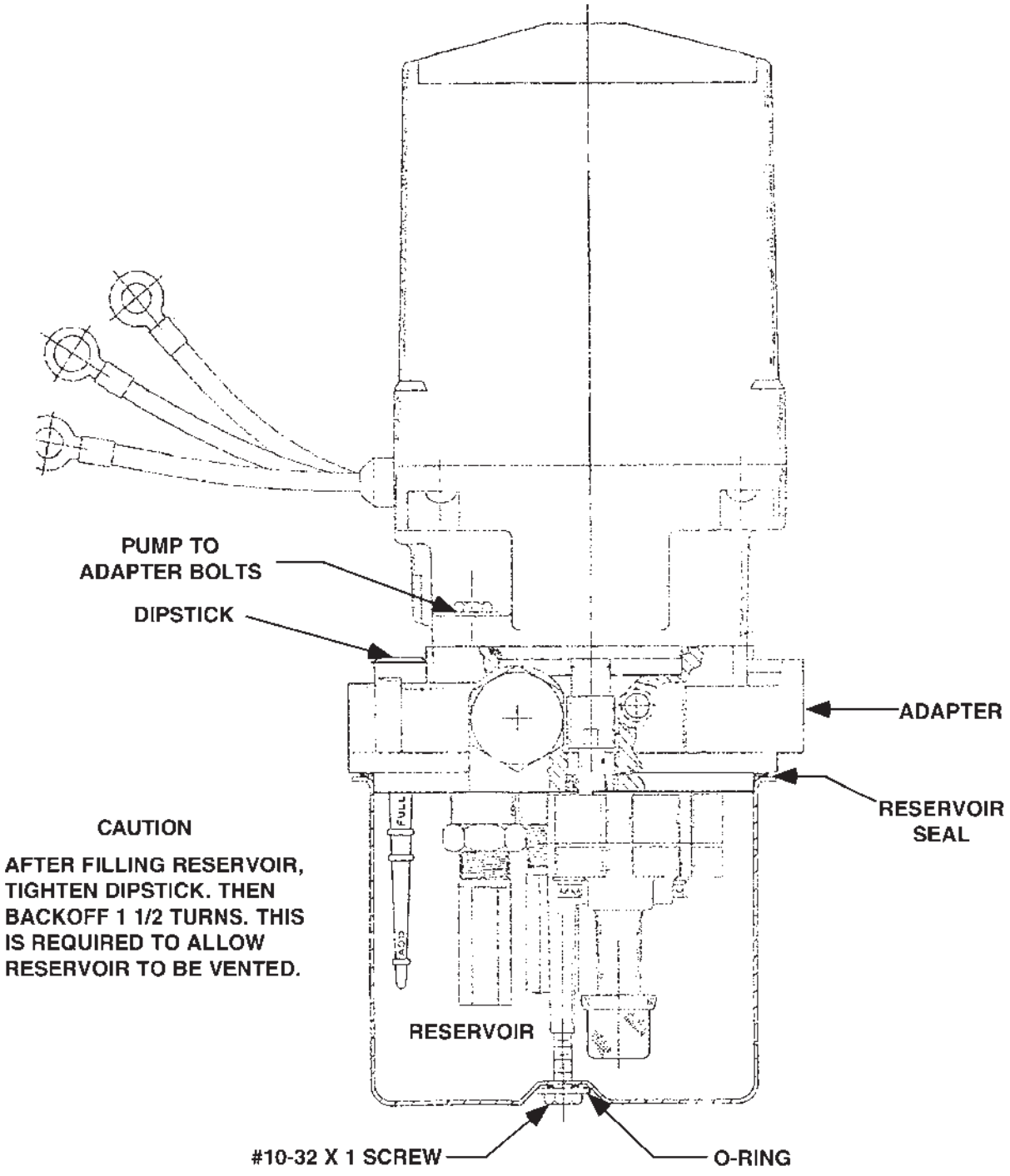
CHART 1
CHARACTERISTICS, HYDRAULIC PUMP - OILDYNE

Electrical Characteristics:	
Voltage	24 DC
Rotation	Reversible
Polarity	Negative ground
Operating Current	25 amps, max. at 24 volts (both rotations)
Operating Time	Continuous with circuit breaker protection.
Overload Protection	Thermal circuit breaker 25 amp
Mechanical Characteristics:	
NOTE: Since the Oildyne hydraulic pump has a sealed motor assembly, with disassembly not approved, there are no mechanical characteristics.	

1. MOTOR ASSEMBLY
2. BASIC PUMP-ADAPTER
3. RESERVOIR ASSEMBLY
4. BRACKET-MOUNTING
5. BASE-PUMP/MOUNT ASSEMBLY
6. RESTRICTOR ELBOW (PIPER P/N 01970-002)
7. BOLT - SENECA IV - 401-269 (3 REQUIRED)
   SENECA V - AN3-7A (3 REQUIRED)
8. WASHER - 494-192 (3 REQUIRED)
9. WASHER - 494-192 - SENECA IV (9 REQUIRED)
   SENECA V (3 REQUIRED)
10. DECK ASSEMBLY
11. GROMMET - 434-120 (3 REQUIRED)
12. BUSHING - 65003-030 (3 REQUIRED)



Oildyne Hydraulic Pump
Figure 1 (Sheet 1 of 2)



Oildyne Hydraulic Pump
Figure 1 (Sheet 2 of 2)

B. Installation

See “Figure 1” on page 29102.

(1) Position assembled pump, bracket, and pump mount on pump deck in airplane.

(2) Secure pump assembly to deck by installing the three AN3-10 bolts with one MS35489-64 washer under each bolt head and three No. 5712-45 (Piper P/N 494 192) washers between Piper mount assembly and pump deck.

NOTE: Before positioning the complete hydraulic pump assembly on the mounting bracket, ensure that the cushion pad is secured in place on the reservoir base.

(3) Install the “up” and “down” pressure hydraulic lines to pump mount.

(4) If equipped, install the grounding strap.

(5) Connect the electrical harness connector(s).

(6) Install anti-splash cover (Piper P/N 96374-0) by installing the four attaching screws,

(7) Install the ABS nose gear cover.

C. Field Service

NOTE: No field repair of Oildyne hydraulic pumps is authorized. Any faulty hydraulic pump must be replaced, or returned to Piper via the local Piper distributor for repairs.

Field service of Oildyne hydraulic pump is limited to removal, cleaning, and inspecting the hydraulic fluid reservoir. Should pump malfunction, either replace pump, or return pump to Piper, via the local Piper distributor, for servicing or repairs.

D. Disassembly

(1) Oildyne Hydraulic Pump from Piper Bracket

See “Figure 1” on page 29102.

(a) Remove safety wire securing two bolts that attach bracket to pump.

(b) Remove the two bolts and washers.

(c) Separate pump assembly from bracket.

(2) Oildyne Motor Assembly from Oildyne Pump/Adapter Assembly

See “Figure 1” on page 29102.

(a) Remove two each mounting bolts on flange of motor assembly and separate the motor assembly from the pump.

(b) Remove coupling and O-ring and discard.

NOTE: New O-ring and coupling are included in replacement motor assembly.

(3) Oildyne Reservoir Assembly from Oildyne Pump/Adapter Assembly

See “Figure 1” on page 29102.

CAUTION: DO NOT DISASSEMBLE PUMP ASSEMBLY FROM ADAPTER ASSEMBLY. DAMAGE TO VALVES AND PRESSURE SETTINGS, WHICH ARE NON-ADJUSTABLE, WILL OCCUR.

(a) Remove screw and O-ring securing the reservoir to the adapter assembly.

(b) Remove reservoir and reservoir seal.

(c) When replacing reservoir, remove the cushion pad. It will have to be bonded to the new reservoir base using Scotch Grip 2210 or Contact Adhesive B-10161 rubber cement.

E. Assembly

(1) Oildyne Reservoir Assembly to Oildyne Pump/Adapter Assembly

See “Figure 1” on page 29102.

(a) Locate the cushion pad and bond it to the bottom surface of the reservoir using Scotch Grip 2210, or Contact Adhesive B-10161 rubber cement.

(b) Position the reservoir seal between the reservoir and the adapter assembly.

(c) Locate the O-ring and bolt that secures the reservoir to the pump-adapter assembly and apply a light coating of Loctite No. 567 PST (Piper Code Number 179-771) in back of first two bolt threads.

(d) Position the O-ring on the bolt, and install it through the reservoir and into the pump-adapter securing the reservoir.

(e) Tighten this bolt to a torque value of 40 - 50 inch pounds.

(2) Oildyne Motor Assembly to Oildyne Pump/Adapter Assembly

See “Figure 1” on page 29102.

(a) Locate the replacement O-ring and coupling.

(b) Place the coupling and O-ring into position between the motor assembly and pump-adapter assembly.

(c) Apply light coating of Loctite No. 567 PST (Piper Code Number 179-771) in back of the first two threads of mounting bolts.

(d) Positioning the two units in place, install two each mounting bolts through the flange of the motor assembly and into the pump-assembly housing.

(e) Torque bolts to 15 - 20 inch pounds.

(3) Oildyne Hydraulic Pump to Piper Bracket

See “Figure 1” on page 29102.

(a) If bracket was removed from pump mount, install bracket to Piper pump mount with four MS24693-S298 screws.

(b) Position pump assembly on bracket so that tapped holes in oildyne adapter align with bolt holes on bracket.

(c) Install two AN960-616 washers and two MS20074-06-05 bolts to secure pump assembly to bracket.

(d) Safety bolts with MS20995-C41 wire.

2. Landing Gear Free Fall Valve Assembly

A. Inspection and Repair

This valve is located directly above the nose wheel actuating cylinder. Inspection is limited to determining if any signs of hydraulic fluid leakage are evident around the seam between the end fitting and valve body, and around the periphery of the piston assembly shaft. If leaks appear, the valve assembly should be replaced since it is impractical to repair the valve.

B. Removal

See “Figure 2” on page 29107.

In the event it becomes necessary to replace the free-fall valve assembly, proceed as follows:

(1) Loosen three screws and clamp securing cable in position and withdraw cable.

(2) Disconnect hydraulic lines connected to the valve. Place a rag in position to absorb any hydraulic fluid spillage that may result. Cap the lines to avoid contamination.

(3) Remove the hex head bolts securing the valve and bracket to the frame and remove the assembly from the airplane.

(4) Remove rivet and nut securing link to piston shaft. Note position of elbow and tee fittings to assure their being replaced in the same position at reassembly. Remove fittings and two bolts securing the valve to the bracket.

C. Installation

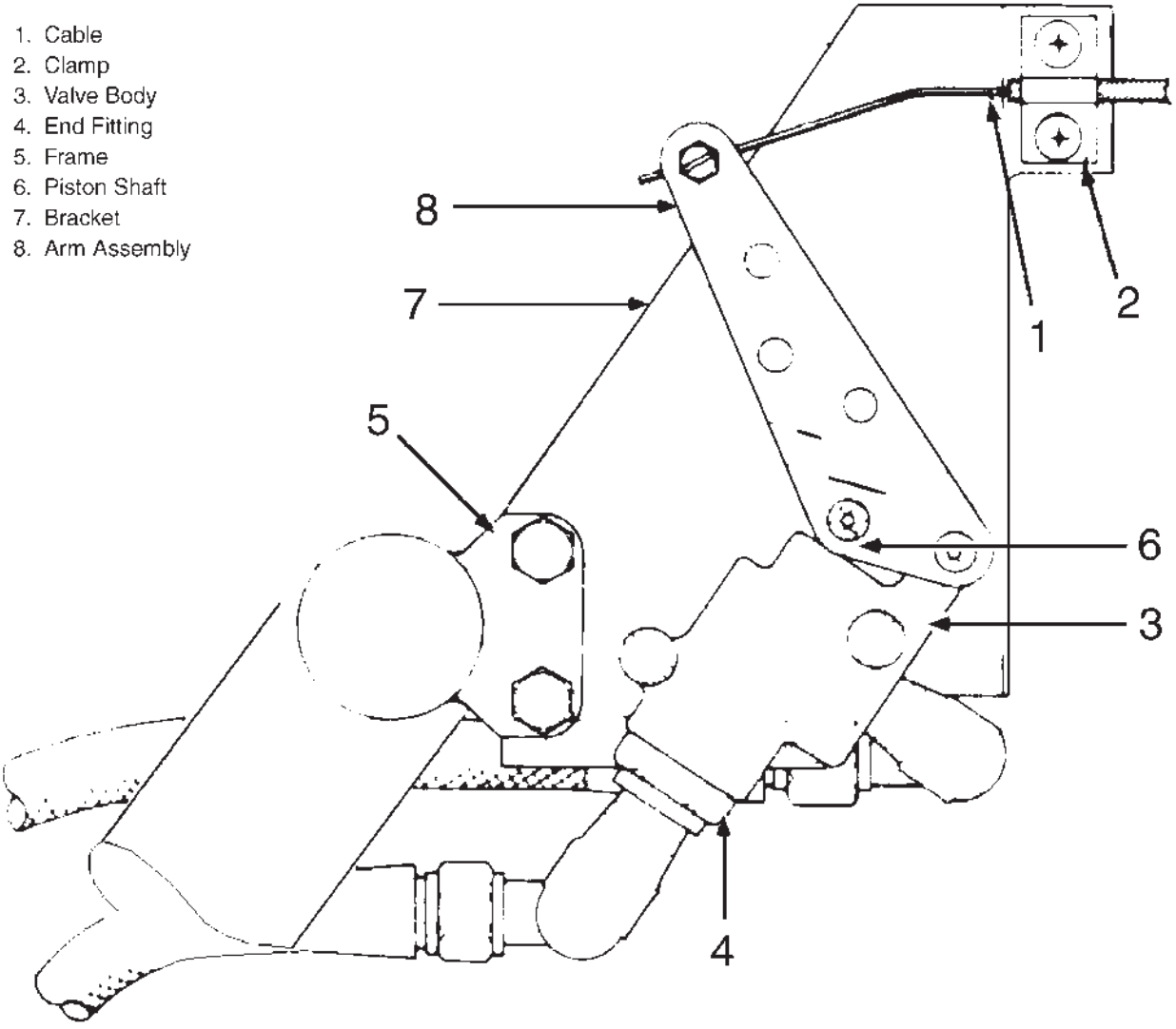
See “Figure 2” on page 29107.

(1) Apply Loctite No. 567 PST (Piper Code Number 179-771) in back of first two MALE threads of elbows and tees and insert fittings in valve. Loctite should be applied sparingly to prevent it entering the hydraulic system.

(2) Install valve on bracket and secure in position. Push piston shaft into the valve until it bottoms. Align hole in link with hole in piston shaft and insert rivet. Attach nut to rivet.

(3) Position bracket with valve on frame. Apply Loctite No. 567 PST (Piper Code Number 179-771) in back of first two MALE threads of tees and connect hydraulic lines. Loctite should be applied sparingly to prevent it entering the hydraulic system.

(4) Push arm assembly fully forward. Pull cable full forward. Place clamp over reinforced portion of cable and tighten screws. Insert loose end of cable through the hole in the bushing of the arm assembly. Tighten lock screw on cable.



Free-Fall Valve Assembly
Figure 2

3. Hydraulic Lines

Removal and Installation

Remove damaged hydraulic lines by disconnecting fittings at both ends and disconnecting where secured by brackets. See Figure 1, 6-20-00, as an aid in locating attaching brackets and bends in lines. Provide a small, clean container for draining the lines. Install a new or repaired line in reverse. Operate the pump to purge air from the system. Check fluid level in the reservoir.

4. Testing Hydraulic System

CAUTION: TURN MASTER SWITCH OFF BEFORE INSERTING OR REMOVING EXTERNAL POWER SUPPLY PLUG.

The hydraulic system should be tested to determine that it functions properly after performing any service or repairs. It is suggested that the airplane be connected to an outside power source in order to conserve the battery. (See External Power Receptacle, 12-00-00 and 24-00-00.)

A. Place airplane on jacks. (See Jacking, 07-00-00.)

B. With gear down, master switch ON, and circuit breaker closed, place landing gear selector handle in the UP position. The pump should immediately start operating and the gear retract. The red gear unsafe light on the instrument panel should light up until the gear is fully retracted. The hydraulic pump should stop operating after full retraction of the gear.

C. Place gear selector handle in DOWN position. The gear should extend and lock in position. Gear down lights on the instrument panel will light up when all three gears are locked in position. Inspect hydraulic system for leakage of hydraulic fluid.

D. Recycle the landing gear to determine that it functions properly.

CAUTION: PRIOR TO REMOVING THE AIRPLANE FROM JACKS, TURN MASTER SWITCH ON AND DETERMINE THAT ALL THREE GREEN LIGHTS ARE ENERGIZED. THIS WILL INDICATE THE LANDING GEAR IS DOWN AND LOCKED.

E. To check operation of the free-fall valve assembly, retract the landing gear and turn the master switch off. Pull the free-fall valve knob full out. The landing gear should extend and lock in position.

5. Servicing Hydraulic Pump/Reservoir

The Oildyne pump incorporates a dipstick to check the quantity of hydraulic fluid in the reservoir. Check the fluid level every 50 hours time-in-service. Replenish only with MIL-H-5606 petroleum base hydraulic fluid. (See Figure 1.)

CHAPTER

30

ICE AND RAIN PROTECTION

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CHAPTER 30

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CHAPTER 30 - ICE AND RAIN PROTECTION

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GENERAL

The material contained in this chapter provides information for general maintenance characteristic of the ice protection system. If further information is necessary contact the product manufacturer or Piper Service Representative.

Description and Operation

The ice protection system is actually made up of five separate systems: a pneumatic deice system (“Figure 1” on page 30103); an electrical prop deice system (“Figure 1” on page 30607); an electrically heated windshield panel (“Figure 1” on page 30402); heated pitot/ stall warning system((“Figure 1” on page 30302); and an “Ice Detection Light” (see 33-40-00) installation. These systems can be installed individually or in any combination. The switches for all installed ice protection systems are in a common group installed directly in the instrument panel directly above the throttle quadrant.

The pneumatic system, which utilizes boots to displace ice from the leading edges of the flight surfaces, necessitates, on Seneca IV models, replacing the standard air pumps with pumps of greater capacity. On Seneca V models, the pumps of greater capacity are standard. Along with the boots and related plumbing, the system also utilizes two control valves, two check valves, a deflate valve, and a timing module. A control and check valve are used on each side of the system and, being mounted behind their respective firewall, are accessible upon removal of the appropriate nacelle hatch cover. The deflate valve and time module are mounted under the floor on the left side of the fuselage as shown in “Figure 1” on page 30103.

Boots are attached to the leading edge of the wings, vertical stabilizer, and stabilator. The boots are of a fabric reinforced rubber containing built-in span wise inflation tubes. A ply of conductive neoprene is cured to each boot surface to dissipate static electric charges and prevent damage to the boots from those charges, as well as preventing a fire hazard after each flight. Attached to the flight surfaces with cement, the boots are connected to the plumbing, through the skin by flexible and/or aluminum air connections.

Operation of the pneumatic deice system is controlled by a momentary, single pole, single throw switch on the control panel. During normal operation, vacuum, provided continuously from the pump inlets to the vacuum system, is also directed to the boots system through the deflate valve to hold the boots down in flight. The control valves, closed during normal operation, allow pressure air from the pumps to be dumped overboard. Activation of the momentary switch initiates power to the aforementioned units causes: (1) the deflate valve to close the system to vacuum and outside pressure; and, (2) activation of the timer in the timing module.

When the switch is activated, pressure begins to build in the system. Upon reaching 8 psi, a pressure switch on the deflate valve activates an indicator light on the control panel. With the system still operating, pressure continues to build until a pressure activated switch (also on the deflate valve) senses 17 psi, or the timer reaches 6 seconds, whichever comes first. At this point power is removed from the control valves and deflate valve causing: (1) the control valves to close, routing pump pressure overboard; and, (2) the deflate valve to dump system pressure overboard. As pressure decreases, the 8 psi switch is deactivated extinguishing the light. With the pressure dumped from the system, the deflate valve again directs vacuum to the boots. The system, now operating in its normal condition, can be reactivated if necessary, through the momentary switch on the control panel. The vacuum gauge may fluctuate momentarily upon deice cycling.

The propeller deice system, which can be installed by itself, or included in a package with others of the ice protection system, is designed for both the two and three blade propeller installations. Each propeller deice system consists of: (1) an electrically heated boot bonded to each blade; (2) (on Seneca IV models) a slip ring assembly connected to the hub of each propeller or (on Seneca V models) a slip ring assembly connected to the engine crankshaft flange; (3) modular brush assemblies; (4) a timer; (9) a circuit breaker and (7) a control switch located on the control panel; (5) an ammeter; and, (6) a shunt installation located on the left side of the fuselage on a longeron behind the side panel on the left side of the cockpit.

On Seneca IV models, dual element deicers are utilized on the two blade propeller installation. Each deicer has two separate heaters: one for the outer half and one for the inner half. By heating all outer or inner heaters on only one propeller at a time, rotational balance is held during deicing. Current is drawn from the airplane electrical system through the switch, ammeter and timer. The timer delivers current via the slip ring and modular brush arrangement to (phase l) the outer heaters on the right propeller, (phase 2) the inner heaters on the same propeller, (Phase 3) the outer heaters of the left propeller, and (phase 4) the inner heaters on the left propeller. The timer energizes each of these four phases in turn for about 34 seconds and then repeats the cycle as long as the control switch is on. The cycling sequence given is vital so that outboard heaters on each propeller operate before the inboard heaters. See cycle sequence. (See “Figure 2” on page 306010) The system may be used continuously in flight if needed. To conserve electrical power, current is cycled to the deicer heaters at timed intervals rather than continuously.

NOTE: Heating may begin at any phase in the cycle depending on the timer position when the switch was turned off from previous use.

On Seneca V models, single element deicers are utilized on the two blade propeller installation. See “Figure 2” on page 306010, Sheet 3.

On Seneca IV and Seneca V models, the three blade propeller installation utilizes single element deicers. When the switch is turned on power is directed through the brush block and slip ring to all the heating elements on one propeller for approximately 34 seconds. The timer then directs the power to the other propeller for approximately 34 seconds. This cycle continues until the switch is turned off.

A heated windshield panel can also be installed as a separate item or with any combination of systems. The installation utilizes a glass panel, imbedded with wire filaments, mounted to a metal frame just outside of the windshield on the pilot’s side, and secured to the fuselage by two screws. The panel is controlled by a switch on the deice control panel. The frame is hinged at its base to facilitate cleaning of the windshield and panel. When not required, the panel can be removed by removing the two attaching screws and harness from the fuselage.

The pitot and stall warning heat systems are operated as a single system. It should be noted that, although the pitot heat system can be installed in the aircraft by itself, the stall warning heat combination can only be included if the pitot heat system is also installed. These system(s) utilize a switch located among the deice switches installed directly in the instrument panel above the throttle quadrant. Other parts of the system(s) include: (1) a heated pitot head; (2) heated lift detectors (inner and outer); and (3) two circuit breakers (one for each part of the system).

An ice detection light can also be included with the systems for seeing and detecting ice at night. When installed, it is mounted to the outboard side of the left nacelle. It is controlled by a push ON–push OFF switch located among the deice switches installed directly in the instrument panel above the throttle quadrant, see 33-40-00.

NOTE: For wiring diagrams (schematics) see 91-30-60.

AIRFOILS

Airfoil deicing is performed by the pneumatic deicing system. The system utilizes inflatable boots to clear ice off the leading edges of the wings, vertical stabilizer, and stabilator.

The boots are of a fabric reinforced rubber construction containing built-in span wise inflation tubes. Attached to the leading edges of the flight surfaces with cement, they are connected through the skin by aluminum and/or flexible rubber air connection stems.

A ply of conductive neoprene is provided on the surface to dissipate static electric charges. These charges, if allowed to accumulate, would eventually discharge through the boot to the metal skin beneath creating static interference with radio equipment, and possible punctures in the rubber. Also such static charges would constitute a temporary fire hazard after each flight.

1. Description and Operation

The pneumatic system, which utilizes boots to displace ice from the leading edges of the flight surfaces, necessitates, on Seneca IV models, replacing the standard air pumps with pumps of greater capacity. On Seneca V models, the pumps of greater capacity are standard. Along with the boots and related plumbing, the system also utilizes two control valves, two check valves, a deflate valve, and a timing module. A control and check valve are used on each side of the system and, being mounted behind their respective firewall, are accessible upon removal of the appropriate nacelle hatch cover. The deflate valve and time module are mounted under the floor on the left side of the fuselage as shown in “Figure 1” on page 30103.

Boots are attached to the leading edge of the wings, vertical stabilizer, and stabilator. The boots are of a fabric reinforced rubber containing built-in span wise inflation tubes. A ply of conductive neoprene is cured to each boot surface to dissipate static electric charges and prevent damage to the boots from those charges, as well as preventing a fire hazard after each flight. Attached to the flight surfaces with cement, the boots are connected to the plumbing, through the skin by flexible and/or aluminum air connections.

Operation of the pneumatic deice system is controlled by a momentary, single pole, single throw switch on the control panel. During normal operation, vacuum, provided continuously from the pump inlets to the vacuum system, is also directed to the boots system through the deflate valve to hold the boots down in flight. The control valves, closed during normal operation, allow pressure air from the pumps to be dumped overboard. Activation of the momentary switch initiates power to the aforementioned units causes: (1) the deflate valve to close the system to vacuum and outside pressure; and, (2) activation of the timer in the timing module.

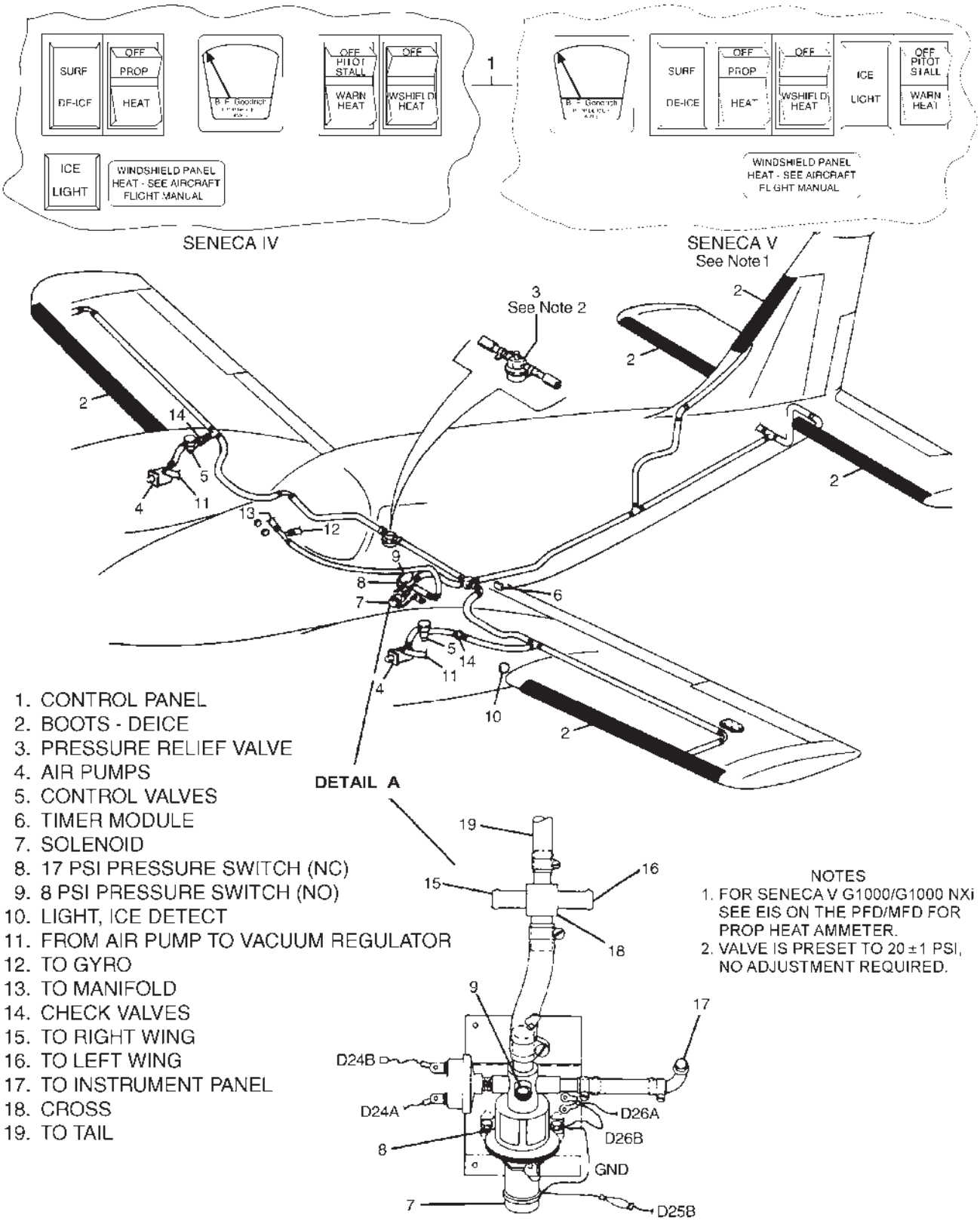
When the switch is activated, pressure begins to build in the system. Upon reaching 8 psi, a pressure switch on the deflate valve activates an indicator light on the control panel. With the system still operating, pressure continues to build until a pressure activated switch (also on the deflate valve) senses 17 psi, or the timer reaches 6 seconds, whichever comes first. At this point power is removed from the control valves and deflate valve causing: (1) the control valves to close, routing pump pressure overboard; and, (2) the deflate valve to dump system pressure overboard. As pressure decreases, the 8 psi switch is deactivated extinguishing the light. With the pressure dumped from the system, the deflate valve again directs vacuum to the boots. The system, now operating in its normal condition, can be reactivated if necessary, through the momentary switch on the control panel. The vacuum gauge may fluctuate momentarily upon deice cycling.

2. Troubleshooting

The troubleshooting chart contained herein is based on the premise, except as specified, that the engine driven pneumatic pumps and the electrical system are operating properly. It is further assumed that the system components were installed properly. See “Chart 1” on page 30102.

CHART 1
TROUBLESHOOTING PNEUMATIC DEICE SYSTEM

Trouble	Cause	Remedy
Deicers do not inflate. Both engines operating at minimum cruise rpm or either engine at 2575 rpm.	Open circuit breaker.	Push circuit breaker to set.
	System connection loose or wire broken.	Tighten or repair as required.
	Timer not functioning.	Test or replace as required.
	Control valves not functioning.	Make electrical test. Check for sticking poppet. Clean. Ensure that both vent ports on solenoid are open.
	Lines blocked or not connected.	Blow out lines and inspect inspect connections. Make air leakage test.
Deicers INFLATE slowly (inflation time - 6 seconds).	Lines partially blocked or not connected securely.	Blow out lines and inspect connections. Perform air leakage test.
	Deflate valve not functioning properly.	Ensure that both vent ports on solenoid are open.
	System pressure not being reached.	Check performance to manufacturers specifications.
	Deicer puncture.	Repair per specification or replace.
Deicers DEFLATE slowly.	Lines partially blocked.	Inspect and blow out lines.
	Deflate valve not functioning properly.	Ensure that both vent ports on solenoid are open.
Deicers inflate, indicator light does not function. (Ascertain that deicer boot switch is ON.	Indicator lamp burned out	Replace lamp.
	System pressure not being reached.	Check “Deicers Inflate Slowly” above.
	Pressure switch not functioning.	Make electrical test and replace if required.
	Wires loose or broken. Poor grounding of pressure switch.	Make electrical test. Repair or replace broken wires. Check for proper ground.
Deicer boots do not hold their form in flight or vacuum to the system inadequate.	Deflate valve not functioning properly by not moving to full position.	Remove and troubleshoot valve. Replace if necessary.
	Vacuum line restricted.	Disconnect line from instruments and deflate valve, and blowout line.
	Broken line.	Inspect system and repair.



Pneumatic Deice System Installation
Figure 1

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3. Inspection

A ground check of the entire deicer system should be made at least every 100 hours time-in-service.

Before checking the system, all deicers should be inspected for damaged areas and repaired according to the procedure in this section outlining cold patch or vulcanized repairs. In order to check the system, See “Chart 2” and the paragraph “Final Test and Adjustment” on page 301021 for operating pressure and check procedures.

A. Ground Procedure

After the test pressure range is established, connect an external source of air providing this pressure and a pressure gauge to the pneumatic deice line at the manifold assembly. Disconnect the deice line from the manifold to accomplish the test. The deicer system should be within one psi of the recommended operating pressure with each inflation cycle.

If deicers do not reach the operating pressure, check the inflation time to ascertain that the solenoid valves are open the specified length of time (six seconds). If this is not the cause of trouble or if the boots deflate slowly. the lines or valves may be plugged; then the lines should be disconnected and blown clear.

Check the timing of the system through several complete cycles. Boots ON six seconds, then OFF. The wing and empennage boots operate simultaneously. If cycle time is off the specified time, determine and correct the difficulty.

Inflation must be rapid to provide efficient deicing. Deflation should be complete before the next inflation cycle of the boots.

B. 100 Hour Inspection

At each 100 hour inspection of the airplane, inspect and operate the deicer boots. Make checks as follows:

CAUTION: IN COLD WEATHER, EXTREME CARE MUST BE TAKEN TO SEE THAT ENGINE OIL DOES NOT COLLECT IN CRITICAL PARTS OF THE SYSTEM AND CONGEAL. CONGEALED OIL WILL CAUSE STICKING OF THE CONTROL VALVES AND DEFLATE VALVE. IF STICKING OF THESE PARTS IS ENCOUNTERED, REMOVE FROM AIRPLANE, CLEAN OUT AND REPLACE.

CAUTION: OIL WHICH REACHES THE DEICERS WILL CAUSE RAPID DETERIORATION OF THE RUBBER.

(1) Carefully inspect the deicers for evidence of damage or deterioration, and repair or replace damaged boots.

(2) Resurface boots which show signs of considerable wear or deterioration.

CHART 2
OPERATING PRESSURES

Recommended Operating Pressure psi	Test Pressure in psi	
	MIN.	MAX.
15	13	17
18	16	20

(3) Inspect all hose connections which form a part of the pneumatic deicing system. Replace deteriorated sections on non-kink hose.

(4) Check the operation of the boots and the operating pressure of the system as outlined in this section.

(5) If new or replacement boots have been installed, check the tube inflation to make sure that the air connection stems have been properly connected.

(6) Disconnect all drain lines in the system and check for proper drainage.

(7) Check the on-off control switch for freedom of action. Check associated electric wiring.

(8) Clean or replace the air filters.

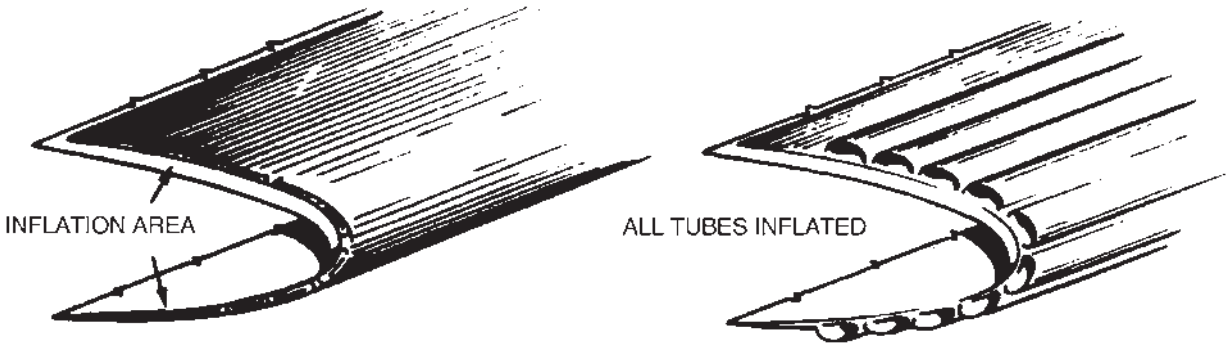
C. Operation Check

See “Figure 1” on page 30103 and “Figure 2”.

The pneumatic deicing system should be checked at least every 100 hours. This check can be done on the ground. A visual inspection should be performed to determine the condition of the deicer boots, and any areas in need of repair should be taken care of before continuing with the operation check of the system.

CAUTION: DO NOT MANUALLY HOLD SURFACE DEICE SWITCH IN ON POSITION. THE SWITCH IS SPRING LOADED AND HOLDING SWITCH IN ON POSITION COULD INDUCE SYSTEM FAILURE.

With one engine operating, activate the deicing system switch (place switch in ON position and release). Observe the operation of the deicers carefully for evidence of malfunctioning. Look for tubes which leak or fail to inflate and deflate properly. Repeat the procedure for the other engine.



Pneumatic Deicer Boots Operation
Figure 2

4. Electrical Tests

A. Procedures

(1) With engines OFF, Turn ON-Master Switch.

CAUTION: DO NOT HOLD SURFACE DEICE SWITCH IN ON POSITION.

(2) TIMER: Activate the Deice System Switch. The switch is momentary push type.

(a) Check Timer operates immediately. (See “Figure 1” on page 30103.)

(b) If Timer does not indicate operation, check aircraft power from Circuit Breaker thru switch, to Timer- BLUE and GREEN leads. Also, check BLACK lead for proper ground.

(c) If no electrical faults are present - replace Timer.

(3) Left and RIGHT CONTROL VALVE SOLENOIDS: Activate the De-Ice System Switch and run engines.

(a) Check both Control Valve Solenoids actuate. (An audible “CLICK” can be heard or the action can be felt by holding a hand against the unit.)

(b) Check system pressure begins to build in the boot system.

(c) If pressure does not build (check LIGHT on control panel after 6 seconds) Control Valves are suspect.

(d) Disconnect electrical leads from Control Valve Solenoids and measure power is available when system is activated. Also confirm ground is good on BOTH SOLENOIDS.

(e) If problem is not electrical - perform “TEST FOR PRESSURE LEAKS.”

(4) DEFLATE VALVE ASSEMBLY: Activate De-Ice System Switch.

(a) Check LIGHT glows on control panel after system pressure builds to 8 psi. Continue to monitor system as pressure continues to build to 17 psi system pressure, or 6 seconds, whichever occurs first.

(b) Control Valves should actuate and dump pressure overboard.

(c) As pressure decreases below 8 psi, LIGHT should extinguish as cycle is completed.

(d) Check power applied to Deflate Valve and confirm grounds are both good. If no electrical faults exist, replace Deflate Valve Assembly.

(5) If Step (3) shows Control Valves to be operating correctly, but Step (4) shows system cycle to be faulty, and system “Test For Pressure Leaks” shows no leaks; check DEFLATE VALVE ASSEMBLY.

(6) DEFLATE VALVE ASSEMBLY:

Check power applied to Deflate Valve and confirm grounds are both good. If no electrical faults exist, replace Deflate Valve Assembly.

5. Pressure Leak Test

A. This test can be performed in either the left or right nacelles.

B. Cap the overboard ports of the control valve.

C. Connect a source of clean air to the inlet port of the control valve. It is necessary that the inlet pressure be a minimum of 18-20 psi pressure to the system. By means of a hand operated valve, trap the pressure in the deicer system. Observe the system for leakage. The leakage rate should not exceed a pressure drop of 3.0 psi per minute.

CAUTION: INSTALL ALL VACUUM LINES DRY.

NOTE: Seal all pneumatic pipe threads with Loctite No. 567 PST (Piper P/N 179-771). Lubricate all pneumatic male hose connections and threaded fasteners with LPS heavy duty silicone lubricant. Allow LPS lubricant to dry before assembly.

D. Remove test equipment, lubricate all threads, and replace all system components.

6. Component Maintenance and Replacement

NOTE: Do not attempt internal repairs of the timer, ammeter or switch. If inoperative, these components must be replaced. For any other repair or maintenance problems not covered herein, inquire at De-Icing & Specialty Systems Division of the Goodrich Corporation.

A. Filters

Air supply for the system is supplied through the vacuum system. See 37-10-00 for additional information.

B. Control Valves

After each 100 hours of engine operation, the valve poppet and internal lining of the control valve can become coated with a film of dried oil causing the valve to stick. Perform electrical test to determine if valve poppet is sticking. If solenoid checks satisfactory, remove valve poppet and clean control valve bore and poppet.

To clean:

(a) Remove nacelle hatch cover to gain access to the valve.

CAUTION: DO NOT LOSE STEEL HEX ACTUATOR PIN.

(b) Remove electrical connector. Unscrew solenoid.

(c) Remove valve poppet. It may be necessary to apply slim nose pliers to pin projection to pull poppet from valve.

(d) Thoroughly clean valve bore and poppet with commercial hydrocarbon type solvent.

(e) Assemble valve and solenoid.

C. Timer

No field maintenance is recommended. For repair or replacement, contact your B. F. Goodrich dealer or distributor.

7. Pneumatic Boots

WARNING: CEMENTS AND SOLVENTS USED TO REMOVE AND INSTALL DEICERS ARE EXTREMELY FLAMMABLE AND TOXIC. EXTINGUISH OPEN FLAMES. AVOID SPARKS. USE IN WELL-VENTILATED AREA. AVOID SKIN CONTACT AND/OR PROLONGED BREATHING OF VAPORS. CONSULT MSDS FOR ADDITIONAL SAFETY INFO.

CAUTION: DISPOSE OF UNUSED MEK AND OTHER CHEMICALS AND SOLVENTS IN A MANNER CONSISTENT WITH LOCAL LAWS AND/OR ENVIRONMENTAL PROTECTION AGENCY REGULATIONS.

CAUTION: IF INSTALLED, DO NOT USE MEK ON DEICER BOOT INSTALLATIONS AS MEK WILL ATTACK ESTANE® MATERIAL. USE TOLUENE INSTEAD.

NOTE: Goodrich Black Standard Pneumatic De-Icer Installation, Maintenance & Repair Manual, ATA Report No. 30-10-31, provides approved, alternate procedures for removing and installing deice boots. Reference to it is highly recommended. It can be obtained online at https://customers.collinsaerospace.com.

A. Removal

NOTE: Disconnect line fittings from boot fittings.

(1) Starting at one corner of the upper trailing edge of the deicer, apply a minimum amount of toluene, naphtha or MEK (methyl ethyl ketone) to the seam line while tension is applied to peel back the corner of the deicer.

(2) Using a pressure handle squirt can filled with solvent, separate the deicer boot from the surface for a distance of four (4) inches all the way along the upper trailing edge. The area between the boot and the wing which has been separated will act as a reservoir for the solvent.

(a) If the deicer boot is to be reinstalled, continue to use solvent to soften the adhesion line and pull down and toward the lower trailing edge with uniform tension.

(b) If the boot is to be scrapped, you may remove it by stripping it in sections parallel to the tubes. It is recommended that the stretchable surface material in the tube area be removed first by carefully slitting around the edges and down the thread lines. Remove the remainder of the boot by stripping it in sections.

(3) Remove any remaining installation cement from the exposed wing area, and the boot if it is being reinstalled, using solvent. Then, clean the wing area thoroughly with MEK.

CHART 3
FILLER EPOXIES

Supplier	Supplier P/N	Mix Ratio	Piper P/N
Lord	305-1 305-2	(305-1:305-2) By weight: 1.2:1 By volume: 1:1	279-162 279-163
Loctite	Hysol EA 960F (Kit includes parts A & B)	(Part A:Part B) By weight: 100:50 By volume: 3:2	279-135
	Hysol EA 9309NA (Kit includes parts A & B)	(Part A:Part B) By weight: 100:23	279-035

CHART 4
MATERIALS AND EQUIPMENT

Description	Piper Part No.
Masking Tape 1 inch, 3M Scotch No. 2308 or equivalent	N/A
Abrasive Pad, Type A very fine pads, 3M Scotch-Brite 7447 or equivalent	279-235
Goodrich A-851-B Adhesive	279-137
3M EC-1300L Adhesive	179-929
Toluene	179-451
Methyl Ethy Ketone (MEK)	193-106
VM&P Naphtha	193-105
Roller	N/A
Goodrich A-56-B Conductive Cement	279-142
Type 1 Class B or Type 8 Sealant	see Consumable Materials, 91-10-00

B. Installation

(PIR-PPS50052, Rev. AB.)

NOTE: The following procedures assume the aircraft is set up with the provisions for the necessary connections and hardware.

NOTE: Do not install deicer boots with more than 60 months elapsed time from date of cure.

NOTE: See “Chart 4” on page 301010 for required materials and equipment.

(1) Preparation of Leading Edges

NOTE: Deicers can be installed on alodined or anodized surfaces. Satisfactory adhesion is also possible on epoxy surfaces if they are sanded lightly to remove the gloss.

NOTE: Boots may be bonded to bare aluminum areas having chemical film conversion coatings per MIL-DTL-5541 Class 3.

NOTE: Any epoxy listed in “Chart 3” on page 301010 may be applied under deicer boots for cosmetic filling purposes. Edges must be masked prior to body work. Epoxy must be fully cured per manufacturer’s directions before installing boots.

(a) Dry fit the deicer boot to the leading edge. Make sure that the air connection holes in the leading edge and cutouts align properly. Use the boot as a template to mask the installation area with a 1/2-inch margin from the edge of the boot using one-inch (1-in.) masking tape. Mark the deicer centerline on the masking tape at each end.

NOTE: Masking accurately will eliminate the need to clean off excess cement.

(b) Remove the paint and primer in masked area with stripper or sand painted/primed area to roughen. Use clean pressurized air or dry cloth to remove sanding particles before proceeding with deicer installation.

(c) Clean the metal surfaces thoroughly, at least twice, with acetone.

1) Moisten a cloth with appropriate solvent. The solvent shall be poured onto the cloth. Cloths shall not be dipped into solvent containers unless a special dipping type container is used.

2) Wipe the surface with the moistened cloth in one direction only, followed by wiping with a dry cloth in the same direction. The dry wipe shall occur before the solvent has had time to dry.

3) After wiping dry, check the cloth for any signs of dirt, grease, or other contaminants. Repeat surface cleaning steps as necessary until all contaminants are removed as evidenced by a clean cloth after wiping dry.

(d) In rivet areas or panel seams of fuel tanks under the deicer boots, apply a one-inch (1-in.) wide strip of fuel barrier cement as follows:

1) Stir cement A-851-B (P/N 279-137) thoroughly to blend solids.

2) Apply one even coat and let it dry for 30 minutes.

3) Apply a second coat and let it dry for two (2) hours.

(e) Mark a straight line along the centerline of the leading edge of the airfoil section where the boot will be mounted.

(2) Preparation of Deicer Boot

CAUTION: DO NOT SATURATE THE BACK SURFACE TOO HEAVILY WITH SOLVENT OR SCRUB IT REPEATEDLY. ALLOW THE DEICER TO DRY THOROUGHLY BEFORE CEMENTING.

CAUTION: ACETONE AND MEK ATTACK ESTANE.

(a) Moisten a clean cloth with acetone and carefully clean the rough surface on the back of the boot at least twice.

(b) Change cloths frequently to avoid recontamination of the cleaned areas.

(c) Fill gaps of skin splices that lead under deicers with sealing compound EC-801.

(d) Remove the sump plugs from the air connection grommets. In some cases, it will be necessary to remove sections of doped fabric used to cover the air connection holes. Draw out the ends of the non-kink hose section so that they protrude through the connection holes in the leading edge. If hose is cracked or deteriorated, replace with new hose.

(3) Mounting Boot on Leading Edge

WARNING: THE CEMENTS AND SOLVENTS USED FOR INSTALLATION ARE FLAMMABLE AND THEIR FUMES SLIGHTLY TOXIC. THEREFORE, ALL WORK SHOULD BE DONE IN A WELL VENTILATED AREA AWAY FROM ANY SPARKS OR FLAMES. (THE USE OF SOLVENT RESISTANT TYPE GLOVES IS RECOMMENDED.)

NOTE: The ambient temperature for installation should be between 50°F (10°C) and 110°F (43°C) and relative humidity should not exceed 90%.

NOTE: The deicer boot and leading edge may have cement applied a maximum of 48 hours before installation, if the cemented parts are covered and kept clean.

(a) Thoroughly mix EC-1300L (P/N 179-929) cement before using.

(b) Apply one even brush coat to the cleaned back surface of the boot and to the cleaned metal surface.

(c) Allow the cement to air dry for a minimum of one hour.

(d) Apply a second coat to both surfaces and allow it to air dry a minimum of one hour.

(e) Snap a chalk line along the leading edge of the airfoil section.

(f) Intensify the chalk line on the leading edge and the white reference line on the boot with a ballpoint pen.

CAUTION: TIGHTEN EACH CLAMP WITH A PAIR OF SLIP JOINT PLIERS. DO NOT SQUEEZE THE CLAMP SO TIGHT THAT THE HOSE IS DAMAGED.

NOTE: Most boots are made with an excess of material at the inboard and outboard edges for final trimming after installation.

(g) Hold the backside of the boot close to the leading edge and fasten the end of each non-kink hose to the corresponding air connection stem. Tighten securely but do not damage any components. Use Tinnerman or other suitable non-kink hose clamps.

(h) Obtain sufficient personnel to hold boot steady during installation. (Limit handling cemented side of the boot with fingers.)

(i) Position the deicer boot centerline to coincide with leading edge centerline. Hold the boot in this position and reactivate about three inches around the connections and the corresponding holes in the leading edge, using a clean, lint-free cloth moistened with naphtha or MEK.

(j) Insert connections in the leading edge holes when the cement has dried to a tacky state, then rubber roll the boot to leading edge in the tacky area.

CAUTION: AVOID EXCESSIVE SOAKING OR RUBBING OF THE CEMENT WHICH COULD REMOVE THE CEMENT FROM THE SURFACE.

NOTE: When installing a longer boot (approximately 6 ft. (1.83 m) or longer) the cement may not remain tacky to roll down the entire length of a 6-inch (15.24-cm) width. Therefore, roll the boot down part way, leaving an open angle to permit easy access for reactivating the cement.

(k) Continue installation by reactivating the cement along the centerline leading edge surface and boot in spanwise strips approximately six inches wide.

CAUTION: AVOID TWISTING OR SHARP BENDING OF THE DEICER.

(l) Using a rubber roller, press the boot firmly against the wing leading edge, in the activated area, without trapping air under the boot. Always roll parallel to the boot's inflatable tubes.

(m) If the boot should attach incorrectly, use naphtha or MEK to remove and reposition properly.

(n) Rubber roll the boot, applying pressure over its entire surface.

NOTE: All rolling should be done parallel to the inflatable tubes. Roll trailing edges with a narrow stitcher-roller.

(o) Trim the ends of the deicer boots as required. Do not trim the deicer boot closer than one (1) inch from the tubes or cutout area.

(p) After bonding, inspect for unbonded areas at the edges of the boot, and for apparent solvent blisters or entrapped air bubbles on the exterior surface of the boot. Neither condition is acceptable and, if present, must be remedied.

(4) Cleanup and Finishing

(a) After mounting boots, remove all masking tape. Clean the surfaces carefully with naphtha, MEK or acetone, so that no solvent will run under the deicer boot edges. Leave a minimum of 0.25 inch of bare metal between the edges of the adhesive and the paint.

(b) Apply semi-conductive corrosion protection to the bare metal strip. Alodine, EC-1300L and Corrosion Resistant Epoxy Primer (Piper P/N 279-240) are acceptable semi-conductive bases under the edge filler. A-851-B cement and top coat paint are not semi-conductive, and therefore the 0.25-inch gap must be free of these materials.

NOTE: Apply semi-conductive corrosion protection that meets the requirements of MIL-DTL-5541 Class 3.

(c) Apply edge filler AC665B-1/2 (Piper P/N 279-065) to fair full thickness deicer edges to adjacent A/C structure along upper wing de-icer, around cutouts, recessed edges, and/or between two deicers which are installed close together.

(d) Apply two lines of masking tape bordering the edges of the area to be filled. Apply one line to deicer surface 0.0625 to 0.125 inch from the edge. Apply second line of masking tape on the leading edge 0.25 to 0.5 inch from the deicer edge. (The width between masking tape lines may vary depending on the area to be filled; use narrowest width possible.)

(e) Mix and apply conductive cement A-56-B (Piper P/N 279-142) edge sealer in the area between the masking tape lines to obtain a smooth, even coating with 0.004 inch minimum final thickness. Allow for 10 to 15 minutes drying time between coats. Remove the masking tape before the final coat dries.

(f) Normal cure time is 48 hours before operating the boot system. Only a system ground check and flight check may be conducted after 24 hours. However, the system should not be operated in icing conditions prior to normal cure time.

(5) Adhesion Test

Using excess boot material trimmed from the ends of any wing and empennage deicers, prepare one test specimen for each deicer installed. This specimen should be a 1 x 8 inch full thickness strip of boot material cemented to the wing skin adjacent to installed boot following the identical procedure used for installation. Leave one inch of the strip uncemented to attach a clamp. Four hours or more after the installation, attach a spring scale to the uncemented end of each strip and measure the force required to remove strip at the rate of one inch per minute. The pull should be applied 180° to the surface. (Strip doubled back on itself.)

A minimum of five pounds tension (pull) shall be required to remove the test strip. If less than five pounds is required, then acceptability of the boot adhesion shall be based on the following tests:

(a) Carefully lift one corner of boot in question sufficiently to attach a spring clamp.

(b) Attach a spring scale to this clamp and pull with force 180° to the surface and in such a direction that the boot tends to be removed on the diagonal.

(c) If a force of five pounds per inch of width can be exerted under these conditions, the installation shall be considered satisfactory. Remember, the width increases as the corner peels back.

(d) Re-cement corner following previous procedure.

(e) Failure to meet this requirement shall result in reinstallation of the boot.

NOTE: Possible reasons for failure are: dirty surfaces, cement not reactivated properly, cement not mixed thoroughly. Corrosion of the metal skin may occur if good adhesion is not attained, especially around rivet heads and metal skin splices.

If these adhesion requirements are met, the airplane may be flown immediately. Do not inflate deicers within 12 hours of installation or until adhesion strength of 8 to 10 pounds is obtained.

(6) When installing deicers on stabilator leading edges, balance stabilator per 55-20-00 upon completion of the deicer installation.

C. Repairs

Deicer repairs are classified as cold repairs (temporary), which are made with the boot installed on the airplane, and vulcanized repairs, which are made on the demounted boot in the shop. (See the appropriate paragraph for vulcanized repairs.)

(1) Cold Repair

The materials and supplies for making cold repairs are listed in “Chart 5” on page 301016.

(a) Scuff Damage

This type of damage will be most commonly encountered and, fortunately, it is not necessary in most cases to make a repair. On those rare occasions when the scuff is severe and has caused the removal of the entire thickness of surface ply in spots (the brown natural rubber underneath is exposed), repair the damage using Part No. 74-451-16 and proceed as follows:

1) Clean the area around the damage with a cloth dampened slightly with solvent. Buff the area around the damage with steel wool so that it is moderately but completely roughened. Wipe the buffed area with a clean cloth slightly dampened with solvent to remove all loose particles.

2) Select a patch of ample size to cover the damaged area. Apply one even thorough coat of cement, Part No. 74-451-20, to the patch and the corresponding damaged area. Allow cement to set a couple of minutes until tacky.

3) Apply the patch to the deicer with an edge, or the center adhering first. Work down the remainder of the patch carefully to avoid trapping air pockets. Thoroughly roll the patch with stitcher-roller, Part No. 74-451-73, and allow to set for 10 to 15 minutes.

4) Wipe the patch and surrounding area from the center outward with a cloth slightly dampened with solvent. Apply one light coat of A-56-B conductive cement, Part No. 74-451-11, to the patched area.

5) Satisfactory adhesion of patch to deicer will be reached in four hours. Deicer may be inflated for checking repair in a minimum of 20 minutes. (Shop check only, do not place in service.)

CHART 5
MATERIAL AND SUPPLIES FOR COLD REPAIR

Part No.	Quantity	Description
74-451-C (FSN1650-856-7939)	1	Cold Patch Repair Kit (Goodrich)
74-451-11	1/1 pt. can	A-56-B Conductive Cement
74-451-16	30 pcs.	Small Oval Patch 1-1/4 x 2-1/2 in.
74-451-17	30 pcs.	Medium Oval Patch 2-1/2 x 5 in.
74-451-18	10 pcs.	Large Oval Patch 5 x 10 in.
74-451-19	3 pcs.	Patch 5 x 19 in.
74-451-20	(2) 1/2 pt.	* No. 4 Cement (patching only)
74-451-70	2	Cement Brush 1/2 in.
74-451-73	1	1/8 in. Steel Stitcher
74-451-75	6	Emery Buffing Sticks
74-451-87	1	Buffing Shield
	* This cement will give best results with the patches in this kit.	

The following items may be procured from the B. F. Goodrich Co., Akron, Ohio, or other manufacturer, as required:

74-451-21	6 ft. roll x 6 in. wide	Type 21 or 22 Fillet
74-451-22	15 ft. roll x 2 in. wide	Neoprene Coated Splicing Tape
74-451-23	4 ft. roll x 8 in. wide	Neoprene Surface Ply
74-451-24 (FSN80628-4199 and/ or FSN8040-514-1880)	1 quart	§ EC-1403 Cement and/or EC-1300 L
74-451-74	1	2-1/2 in. Sponge Rubber Roller

§ Minnesota Mining and Manufacturing Company, Adhesives Division
3M Center; St. Paul, MN 55144 (612-733-1110)

The following materials may be obtained from local supply:

	Toulene
As required	Clean, Lint-Free Cloths (preferably cheese cloth)
Rolls	1 in. Masking Tape
1	Sharp Knife
6 ft. long	Steel Measuring Tape
1	Fine Sharpening Stone
As required	Steel Wool Pads
As required	Hypodermic needles (22 gauge or smaller)

NOTE: Methylethylketone (MEK) can be used instead of Toulene, however MEK causes very rapid drying and provides only 10 seconds working time compared with 40 seconds for Toulene.

(b) Tube Area Damage.

CAUTION: THESE PATCHES ARE MANUFACTURED SO THAT THEY WILL STRETCH IN ONE DIRECTION ONLY. BE SURE TO CUT AND APPLY THE PATCH SELECTED SO THAT STRETCH IS IN THE WIDTHWISE DIRECTION OF THE INFLATABLE TUBES.

CAUTION: DO NOT TRAP AIR BETWEEN PATCH AND DEICER SURFACE

CAUTION: ALLOW A MINIMUM OF FOUR HOURS BEFORE INFLATING A REPAIRED DEICER

Repair cuts, tears, or ruptures to the tube area shall be repaired with fabric reinforced patches, Part No. 74-451-16) depending on size of damage.

1) Select a patch of ample size to cover the damage and to extend to at least 5/8 inch beyond the ends and edges of the cut or tear. If none of the patches is of proper size, cut one to the size desired from one of the larger patches. If this is done, bevel the edges by cutting with the shears at an angle.
2) Buff the area around the damage with buffing stick, Part No. 74-451-75. so that the surface is thoroughly roughened.
3) Apply the patch to the deicer with the stretch in the widthwise direction of the inflatable tubes. Stick edge of patch in place. Work remainder down with slight pulling action so the injury is closed.

(c) Loose Surface Ply in Dead Area

Non-inflatable area.

Peel and trim the loose surface ply to the point where the adhesion of surface ply to the deicer is good.

1) Scrub (roughen) area in which surface ply is removed with steel wool. Scrubbing motion must be parallel to cut edge of surface ply to prevent loosening it. Scrub with steel wool and Toulene directly over all edges, but parallel to edges or surface ply to taper them down to the tan rubber ply.
2) Cut a piece of surface ply material. Part No. 74-451-23, to cover the damaged area and extend at least one inch beyond in all directions.
3) Mask off the damaged boot area 1/2 inch larger in length and width than the size of surface ply patch. Apply one coat of cement, Part No. 74-451-11, to damaged area and one coat to patch. Allow cement to set until tacky. Roll the surface ply to the deicer with 2 inch rubber roller, Part No. 74-451-74. Roll edges with stitcher- roller, Part No. 74-451-73. Apply just enough tension on the surface ply when rolling to prevent wrinkling and be careful to prevent trapping air. If air blisters appear after surface ply is applied, remove them with a hypodermic needle.
4) Clean excess cement from deicer with solvent.

(d) Loose Surface Ply in Tube Area.

Loose surface ply in tube area is usually an indication of the deicer starting to flex fail. This type of failure is more easily detected in the form of a blister under the surface ply when deicer is pressurized. If this type of damage (or void) is detected while still a small blister (about 1/4 or 3/8 inch diameter) and patched immediately, the service life of the deicer will be appreciably extended. Apply repair patch as outlined in Paragraph “Loose Surface Ply in Dead Area” on page 301017.

(e) Damage to Fabric Back Ply of Deicer During Removal of cement has pulled loose from the wing skin and adhered to the back surface of the deicer, remove it with steel wool and MEK. In those spots where the coating has pulled off the fabric, leaving bare fabric exposed, apply at least two additional coats of cement, Part No. 74-451-24. Allow each coat to dry thoroughly.

(2) Vulcanized Repairs

Due to the variety of boot damage possible, it is recommended that the B. F. Goodrich Company be contacted so they can determine the extent of damage and whether it is repairable by the vulcanized method or not. The overall condition of the deicer boot must be given careful consideration before deciding on any repairs. Damages can vary from minor punctures which may be easily repaired, to extensive ripping of the tube or stretch areas which may make repairs exceedingly difficult or actually impossible. The determination of just where this division between repairable and unrepairable damage exists will depend upon the careful judgment of the inspector. For this reason, we recommend contacting the B. F. Goodrich Company at Akron, Ohio.

D. Cleaning Deicer Boots

CAUTION: AVOID THE USE OF PETROLEUM PRODUCTS AS CLEANING AGENTS.

NOTE: The temperature of the soap solution and rinse water should by exceed 140°F (60°C).

NOTE: If cleaning compound MIL-C-25769 is used to clean the airplane, thoroughly rinse off the deicers with clean water.

(1) Wash deicers with a mild soap and water solution.

(2) Rinse with clean water.

(3) In cold weather, wash the boots with the airplane inside a warm hangar if possible. If the cleaning is to be done outdoors, heat the soap and water solution before taking it out to the airplane.

(4) If difficulty is encountered with the water freezing on the boots, direct a blast of warm air along the region being cleaned, using a portable type ground heater.

(5) Limited use of mineral spirits or non-leaded gasoline is not harmful in cleaning the deicers if the cloth is dampened (not dripping) with solvent and a clean, dry cloth is used to wipe the deicer before the solvent has time to soak into the rubber.

E. Icex Application

WARNING: USE ONLY WITH ADEQUATE VENTILATION. AVOID PROLONGED OR REPEATED BREATHING OF VAPOR. AVOID PROLONGED OR REPEATED CONTACT WITH SKIN. KEEP COMPOUND AWAY FROM OPEN FLAMES AND ELECTRIC HEATERS; DECOMPOSITION PRODUCTS MAY BE HARMFUL.

CAUTION: AVOID BOOT EDGE SEALER/FILLER WHEN APPLYING ANY PRESERVATIVE / COSMETIC COATING OR ICE INHIBITOR TO DEICE BOOTS. SUCH COATINGS MAY LESSEN THE ADHESIVE PROPERTIES OF THE EDGE SEALER/FILLER.

(1) Before applying Icex, thoroughly clean deicer surfaces with a rag dampened with non-leaded gasoline. Follow by a scrub wash of mild soap and water. Allow time for surfaces to dry.

(2) Apply Icex sparingly to the dry deicer in a strippling fashion with a felt pad or small, soft cloth. Avoid a heavy, sticky application of “fly paper” consistency.

(3) Follow the application by rubbing the entire surface with a soft, dry cloth until a smooth gloss is achieved.

(4) Reapply Icex every 150 hours.

(5) Icex is not a cure-all for icing problems. Icex will not prevent or remove ice formations. Its only function is to keep ice from initially getting a strong foothold, thus making removal easier.

(6) One quart can of Icex will cover approximately 500 square feet.

F. Resurfacing Conductive Cement

WARNING: CEMENTS AND SOLVENTS USED FOR RESURFACING ARE FLAMMABLE AND THEIR FUMES SLIGHTLY TOXIC. THEREFORE, ALL WORK SHOULD BE DONE IN A WELL VENTILATED AREA AWAY FROM ANY SPARKS OR FLAMES.

(1) The following materials are required to remove and replace the old, damaged coating:

(a) Fine grit sandpaper.

(b) Two inch paint brush.

(c) One inch masking tape.

(d) Conductive neoprene cement, No. A-56-B, B.F. Goodrich Company.

(e) Isopropyl Acetate, Federal Specification TT-1-721, as cleaning or thinning solvent.

(f) Alternate solvent (Toluene may be used as an alternate for Isopropyl Acetate).

During cold weather, place the airplane in a warm hangar and locate so that the boots are in line with one or more blast heaters. Do resurfacing before any other work on the airplane to allow as much time as possible for the new coat to cure.

NOTE: If, for some reason the resurfacing cannot be done indoors, it may be deferred at the discretion of the inspector until a warm, clear day permits the work to be satisfactorily accomplished outdoors. However, if the deicers are in such condition that immediate resurfacing is required, remove them from the airplane and resurface in a shop.

(2) Clean deicer thoroughly with Isopropyl Acetate.

(a) Roughen entire surface of boot, using a fine grit sandpaper.

(b) Clean surface again with clean, lint-free cloth moistened with cleaning solvent.

(c) Apply masking tape beyond upper and lower trailing edges, leaving a 1/4 inch gap of bare metal.

(d) Mask off any legible deicer brands.

(e) Apply one brush coat of A-56-B cement to deicer and allow to dry at least one hour. Then apply second coat and allow to dry at least four hours before operating deicers. Plane may be flown as soon as cement is dry.

NOTE: If A-56-B cement has aged 3 months or over, it may be necessary to dilute the cement with Isopropyl Acetate to obtain proper brushing consistency. Mix thoroughly, approximately 5 parts cement to one part Isopropyl Acetate.

8. Final Test and Adjustment

(PIR-PPS60054-2, Rev. F.)

A. Remove the hatch covers on each nacelle and disconnect the line between the control valve and check valve.

B. On each installation install a pressure gauge having a capacity of 0 to 25 psi in between the two valves. Set them up such that they can be read in the cabin.

C. Perform the following procedure with just the left engine operating and then just the right engine operating.

(1) Start the specific engine. Allow the engine to warm up and bring up the power to 1800 – 2000 rpm.

(2) Check the vacuum regulator for 5.0 ± 0.2 in. of mercury (Hg). If the reading is not correct, check system for fouled filters, or old lines. Should the system check out OK, adjust the vacuum regulator on the left side of the respective firewall.

(3) Run the engine at 2400 rpm.

(4) Cycle the deicer boot system. (The vacuum gauge may momentarily drop as much as 1.5 in-Hg during cycling).

(5) On non-G1000 equipped aircraft: While the boots are inflating, check that the blue/green wing-tail indicator light on the deice panel comes on (when the valve pressure reads from 8 to 13 psi, and remains on above that pressure).

(6) On G1000/G1000 NXi-equipped aircraft only: Check that the SFC DEICE ON Crew Alerting System (CAS) advisory message illuminates (when the valve pressure reads greater than 8 psi).

(7) Full inflation pressure should read 17 ± 1 psi, and should occur within 2.0 to 3.0 seconds at sea level. All cells should fully inflate.

(8) Upon full inflation, deflation and boot hold down shall occur within 15 seconds. (Hold down vacuum is the same as gyro vacuum).

NOTE: Several cycles may be required to complete the above. Allow 30 seconds between cycle activations to assure complete cycling.

(9) Continually hold the pneumatic de-ice switch ON for 20 seconds. Full inflation pressure must not exceed 23 psi.

(10) Shut down the left engine, and repeat the above with the right engine.

(11) Repeat the above with both engines running at 2400 rpm. (Cycle times may decrease.)

(12) With both engines running at 2400 rpm, place the propeller de-ice switch in the ON position.

(a) For airplanes equipped with two blade Hartzell propellers, the ammeter should show a surge of current to the inboard, then outboard sections of each propeller in turn. During current surges the ammeter should indicate within its green arc.

(b) For airplanes equipped with three blade McCauley propellers, the ammeter should show a surge of current to one propeller, followed by a surge of current to the opposite propeller. Time between surges should not exceed ninety (90) seconds. During current surges the ammeter should indicate within its green arc.

(13) Return the propeller de-ice switch to the OFF position.

CAUTION: DO NOT OPERATE PROPELLER DE-ICE EQUIPMENT WITH ENGINES NOT OPERATING.

NOTE: On G1000/G1000 NXi-equipped aircraft, the ammeter is displayed on the EIS page.

(14) Shut down engines. Place magneto and aircraft battery-alternator switches in the OFF position.

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PITOT AND STATIC

Both a heated pitot and heated stall warning system are available. It is significant to note that although they are separate systems in themselves, they are installed as an individual system controlled by a single switch located among the deice switches installed directly in the instrument panel above the throttle quadrant. However, if both systems are not installed, the pitot system can be installed by itself using the same spot for the switch as previously mentioned.

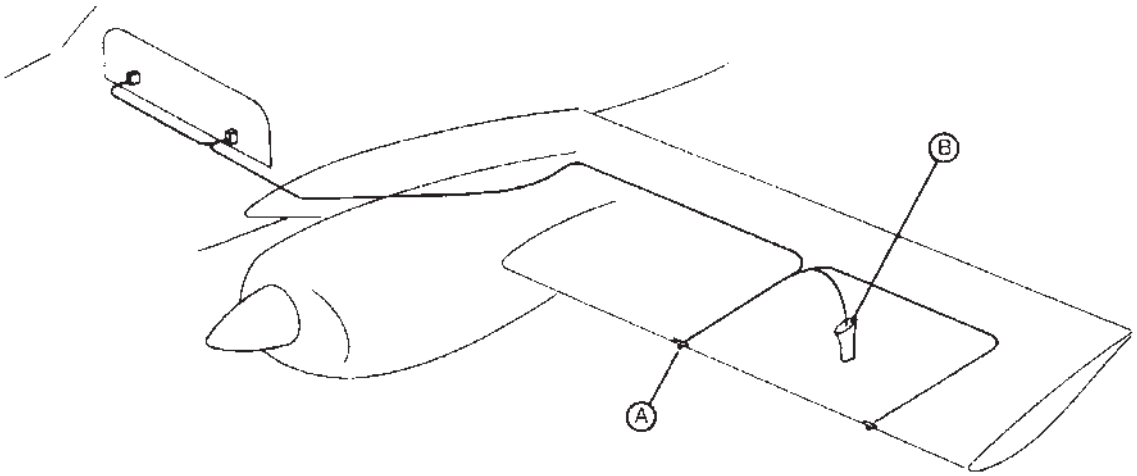
These systems are quite simple in that they contain a heated pitot head, heated lift detectors (inner and outer); and two circuit breakers (one for each part of the system). The units for these installations are installed on the left wing (See “Figure 1” on page 30302). See 27-30-00 and 34-10-00 for removal and installation procedures. For wiring diagrams (schematics) see Chapter 91.

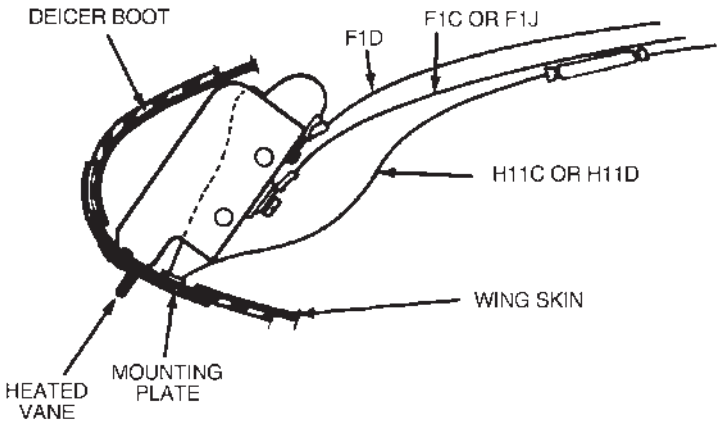
Pitot Heat Test

A. Remove the pitot head from mast assembly and the wing. (See 34-10-00.)

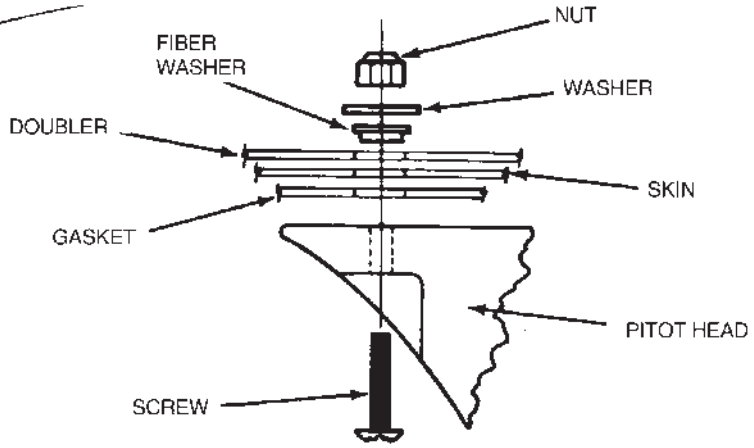
B. With one of the test leads connected to one of the terminals of the head, ground the other lead to the metal body of the head assembly. There should be no continuity.

C. Reinstall the pitot head into the mast assembly and the wing.





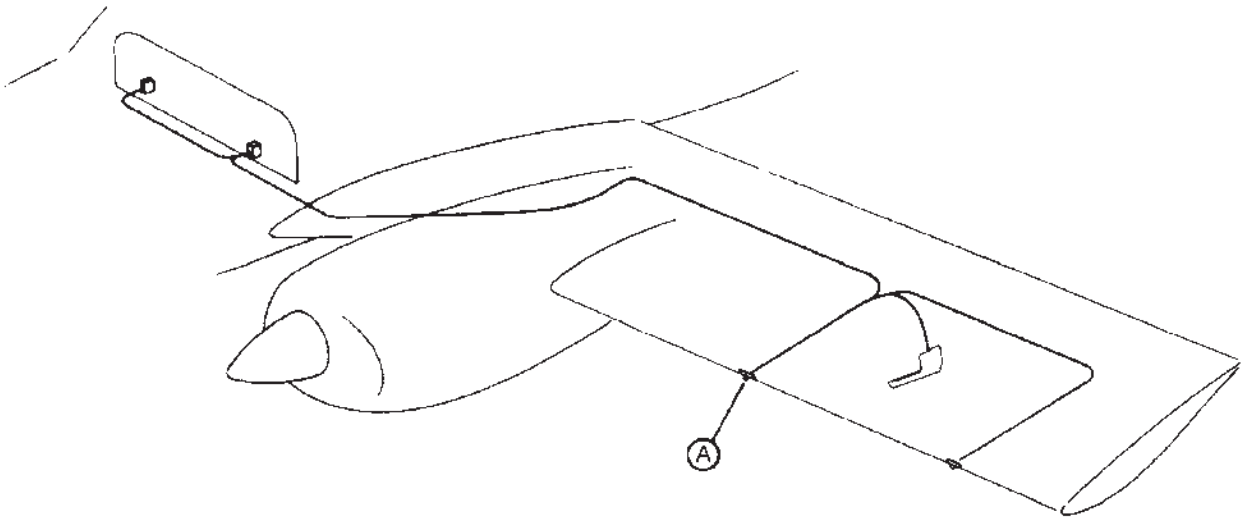
DETAIL A

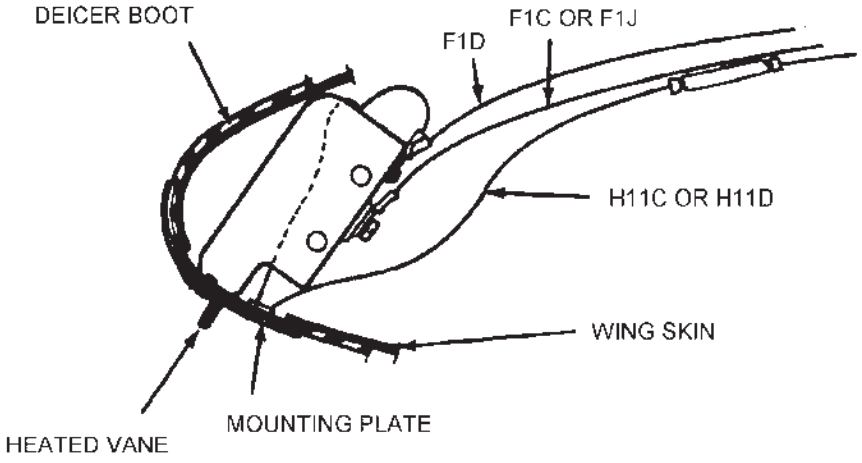


DETAIL B PITOT HEAD INSTALLATION DETAIL

Effectivity
Seneca IV

Heated Pitot & Stall Warning System Installation
Figure 1 (Sheet 1 of 2)





DETAIL A

Heated Pitot & Stall Warning System Installation
Figure 1 (Sheet 2 of 2)

Effectivity
Seneca V

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WINDOWS AND WINDSHIELD

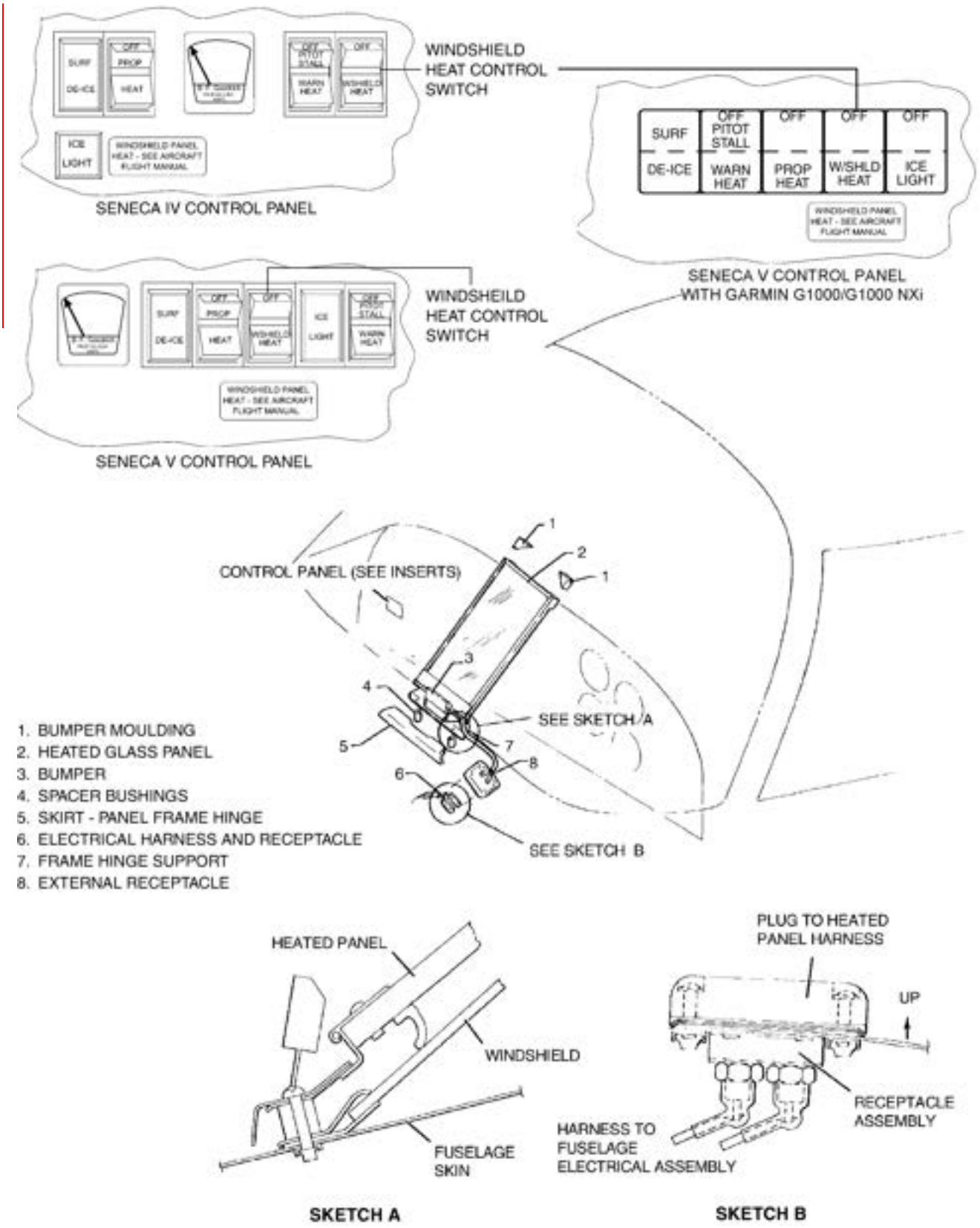
1. Heated Windshield Panel

   The heated windshield panel is a rectangular glass panel containing electrically heated wires imbedded in the glass which is mounted in a metal frame. The assembly is mounted on the exterior side of the pilot's windshield and is hinged at its base to facilitate windshield cleaning. The heated panel is operated by a single pole, single throw rocker type switch located among the deice switches installed directly in the instrument panel above the throttle quadrant. It is protected by the WSHLD HEAT circuit breaker installed in the circuit breaker panel.

2. Heated Panel

   Removal and Installation

   (1) Disconnect the electrical connector located next to the heated panel on the exterior side of the windshield, by removing the two screws and pulling the plug out of the receptacle.
   (2) Remove the two screws that attach the panel assembly to the windshield collar. Remove the panel from the airplane.
   (3) If the airplane is to be flown with the heated panel removed, rotate the receptacle plate 180° and replace it to cover the holes in the fuselage skin. Also replace the windshield collar screws.
   (4) Installation of the heated windshield panel is accomplished in the reverse order of removal.



Windshield Heat
Figure 1

PROPELLERS

The deice system for the propeller uses electrically heated rubber boots, glued to the inner portion of each blade. The boots which contain special heater wires, are designed to be protected on the air side by pieces of rubber impregnated fabric resistant to abrasion and oil. When installing the boots, ensure the side with the dull finish is glued to the propeller, with the glossy finish towards the air.

1. Description and Operation

The propeller deice system, which can be installed by itself, or included in a package with others of the ice protection system, is designed for both the two and three blade propeller installations. Each propeller deice system consists of:

A. Electrically heated boot bonded to each blade

On Seneca IV models, two bladed propellers have two heating elements incorporated in each boot, while three bladed propellers use boots incorporating a single heating element. On Seneca V models, two bladed propellers and three bladed propellers both use boots incorporating a single heating element.

(1) The dual element deicers (Seneca IV only) have a separate lead for the inboard and outboard heater and a third lead which is a common ground. These leads are so marked. An unmarked ground can be identified by using an ohmmeter across the three possible pairs of leads. One pair will show twice the resistance of the other pairs. The latter are the “hot” leads and the lead excluded from the pair that shows twice the resistance of the other pairs is the ground lead.

Current is drawn from the airplane electrical system through the switch, ammeter and timer. The timer delivers current via the slip ring and modular brush arrangement to (Phase 1) the outer heaters on the right propeller, (Phase 2) the inner heaters on the same propeller, (Phase 3) the outer heaters of the left propeller, and (Phase 4) the inner heaters on the left propeller. The timer energizes each of these four phases in turn for about 34 seconds and then repeats the cycle as long as the control switch is on. The cycling sequence given is vital so that outboard heaters on each propeller operate before the inboard heaters. See cycle sequence (“Figure 4” on page 306015). The system may be used continuously in flight if needed. To conserve electrical power, current is cycled to the deicer heaters at timed intervals rather than continuously.

(2) The single element deicers have only two leads; one input and one ground.

When the switch is turned on power is directed through the brush block and slip ring to all the heating elements on one propeller for approximately 34 seconds. The timer then directs the power to the other propeller for approximately 34 seconds. This cycle continues until the switch is turned off.

NOTE: Heating may begin at any phase in the cycle depending on the timer position when the switch was turned off from previous use.

B. Slip ring assembly - connected to the hub of each propeller (Seneca IV); or, connected to the engine crankshaft flange (Seneca V)

C. Modular Brush Assemblies

D. Timer

E. Ammeter

The ammeter is located among the deice switches installed directly in the instrument panel above the throttle quadrant. If replacement is necessary, take care to ensure the correct replacement ammeter (by part number) is used. It is important to note that during periods of low battery voltage, which can occur when an engine is shut down, the ammeter will indicate lower than at full voltage. When operating at full voltage, the needle should be within the shaded range, if current flow to the deicers is normal.

CHART 1 (Sheet 1 of 4)
TROUBLESHOOTING PROPELLER DEICE SYSTEM

Trouble	Cause	Remedy
Ammeter shows zero current (All 4 phases of the 2 minute cycle – 2 Bladed Propeller.)	Tripped circuit breaker.	Locate and correct short before setting circuit breaker.
(Both phases of the 1 minute cycle – 3 Bladed Propeller.)	No power from airplane.	If no voltage into switch, locate and correct open.
	Circuit breaker switch faulty.	If no voltage at switch output with voltage at switch input, replace the switch. If voltage is satisfactory at switch output, go to next step.
	Ammeter faulty. (If some or all deicers heat with ammeter at zero, replace the ammeter.)	Test for voltage up to and out of ammeter. If low or zero output and input satisfactory, replace ammeter. If no voltage to ammeter, locate and fix open between switch and ammeter.
	Open ammeter to timer.	Disconnect harness at timer and check voltage at Pin B (of harness) to ground. If none, locate and correct open.
Ammeter shows normal current part of cycle, zero current rest of cycle.	Open in wiring between timer and brush block assembly.	Use heat test to find deicers not heating & test for voltage on that contact of wire harness plug. (At brush block assembly.) If zero over 2 minutes (2 bladed propeller),or 1 minute (3 bladed propeller) locate and fix open in wiring from timer to wire harness plug.
	Open between brush block assembly and deicer lead straps.	If there is voltage to brush block wire harness plug, try voltage at junction to deicer lead and slip ring lead. If no voltage, find and correct open in wiring within brush block or no contact of brush to slip ring.
	No ground circuit, one engine.	If voltage is found at deicer leads, locate and fix open from deicer to ground.

CHART 1 (Sheet 2 of 4)
TROUBLESHOOTING PROPELLER DEICE SYSTEM

Trouble	Cause	Remedy
Ammeter shows normal current part of cycle, low current rest of cycle.	Inner and outer deicers heating same phase (2 bladed Propeller).	Locate and repair incorrect connections.
	Open in deicer or slip ring leads.	Disconnect deicer harness to check heater resistance. If satisfactory, locate and fix open in slip ring leads.
	High resistance in circuit with low current.	If not in contact of brush to slip ring (including ground brush), trace wiring to deicer and to timer to fix partially broken wire, loose or corroded connection.
Ammeter shows low current over entire cycle.	Aircraft voltage low.	Check voltage into switch.
	Ammeter faulty.	Test for voltage up to and out of ammeter. If low or zero output and input satisfactory, replace ammeter. If no voltage to ammeter, locate and fix open between switch and ammeter.
	High resistance up to timer.	Check for partially broken wire, loose or corroded connection in wiring from aircraft supply to timer input.
Ammeter shows excess current over entire cycle.	Ammeter faulty.	Test for voltage up to and out of ammeter. If low or zero output and input satisfactory, replace ammeter. If no voltage to ammeter, locate and fix open between switch and ammeter.
	Ground between ammeter and timer.	Disconnect harness at timer and with ohmmeter check from Pin B (of harness) to ground. If ground is indicated, locate and correct.

CHART 1 (Sheet 3 of 4)
TROUBLESHOOTING PROPELLER DEICE SYSTEM

Trouble	Cause	Remedy
Ammeter shows normal current part of cycle, excess current rest of cycle.	Ground between timer and brush block.	Disconnect leads at brush block and with ohmmeter check from power leads to ground. If ground indicated, locate and correct.
	Ground between brush block and deicers. (Excluding ground brush circuit.)	If no short exists at brush slip ring contact, check for ground from slip ring lead to propeller assembly while flexing slip ring and deicer leads. If a ground is indicated locate and correct.
	Short between two adjacent circuits.	Check for cuts or low resistance between circuits. If any, locate and correct.
	Timer faulty.	Test timer.
Ammeter does not “flick” approximately every 34 seconds.	Timer ground open.	Disconnect harness at timer and check with ohmmeter from Pin G (of harness) to ground. If no circuit, fix open per schematic diagram.
	Timer contacts are welded (caused by short circuit in system).	Test timer. If timer does not cycle with voltage at Pin B, replace timer but be sure short causing original failure has been located and corrected.
Ammeter flicks between 34 second phase periods.	Loose connection between aircraft power supply and timer input.	If trouble occurs over entire cycle, trace wiring from power source to timer input to locate and tighten loose connection.
	Loose or poor connection timer to deicers.	If trouble occurs in part of cycle, find which deicers are affected and check for rough or dirty slip rings causing brush to “skip.” If not this, trace circuits to locate and fix loose or poor connection. (If all deicers on one propeller are affected, check the ground circuit.) Flex deicer straps for break in deicer straps.

CHART 1 (Sheet 4 of 4)
TROUBLESHOOTING PROPELLER DEICE SYSTEM

Trouble	Cause	Remedy
Ammeter flicks between 34 second phase periods. (cont.)	Timer cycles erratically.	Test timer.
Radio noise or interference with deicers on.	Brushes “arcing.”	Check brush alignment Look for rough or dirty slip rings. If this is the cause, clean, machine, or replace slip ring assembly, as required. Check slip ring alignment.
	Loose connection.	Refer to “Ammeter flicks between 34 second phase period.”
	Switch faulty.	Try jumper wire across switch. If radio noise disappears, replace the switch.
	Wiring located within 8 inches of radio equipment wiring.	Relocate at least 8 inches away from input wiring to radio equipment.
Cycling sequence not correct.	Crossed connections.	Check system wiring circuit diagram for improper connections. (Refer to “Figure 2” on page 30106.)
Rapid brush wear or frequent breakage.	Brush block out of alignment.	Check brush alignment.
	Slip ring wobbles.	Check slip ring alignment with dial indicator. (Refer to “Figure 3” on page 306015.)

F. Shunt installation located on the left side of the fuselage on a longeron behind the side panel on the left side of the cockpit

G. Control switch located on the control panel

Control of the propeller deice system is through a switch located on the ice protection control panel.

H. Circuit breaker

The complete circuit, along with its component parts, are protected by a circuit breaker located on the main bus circuit breaker panel. The ampere value of the circuit breaker varies depending on whether the system is installed on two bladed or three bladed propellers.

2. Troubleshooting

See “Chart 1” on page 30602.

3. Using the Ammeter

Whether in flight or during ground testing, the ammeter can be used to indicate the general nature of most electrical problems. The troubleshooting chart is primarily based on the use of the ammeter and assumes that the user does understand all normal operating modes of the system as described in the beginning of this section.

NOTE: When troubleshooting, first use the “ammeter test” and “heat test” to determine which circuits are involved. Use electrical schematic, 91-30-60 for assistance to check voltages or continuity.

Helpful Tips

(1) (Seneca IV, two blade prop) If the ammeter reading drops to one-third normal current, this indicates that one heater circuit is open or, on the dual element deicer, possibly improper connections are allowing both inboard and outboard units to heat at the same time.

(2) Excess current reading on the ammeter always indicates a power lead is shorted to ground. Thus, when trouble of this nature is found, it is vital that the grounded power lead be located and corrected.

(3) A considerable number of timers that have been returned for repair proved to be fully workable when tested. Accomplish the test described in “Timer Test” before concluding that the timer is defective.

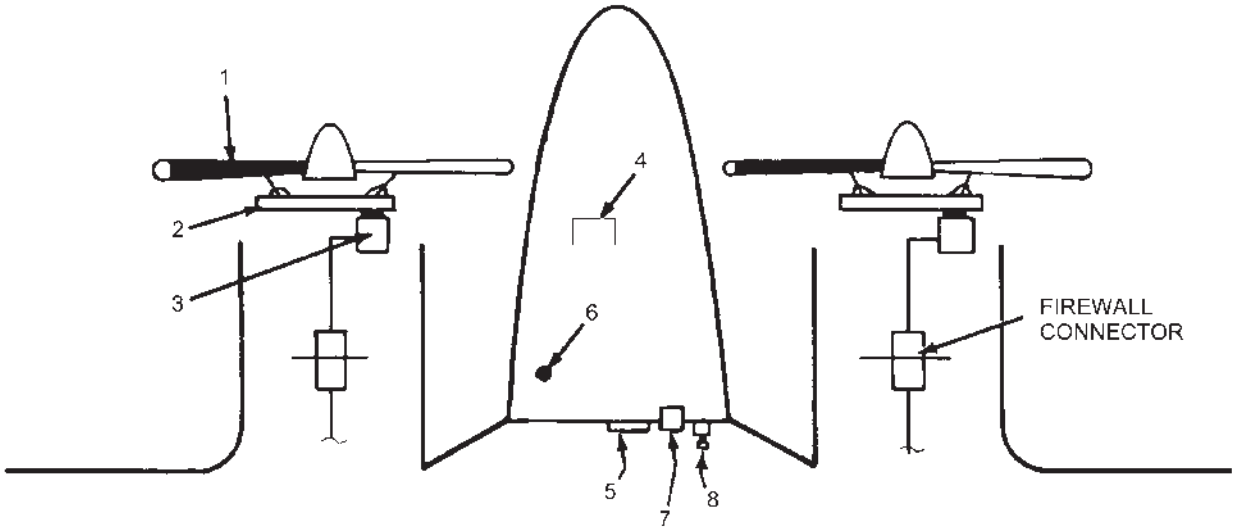
4. Inspection

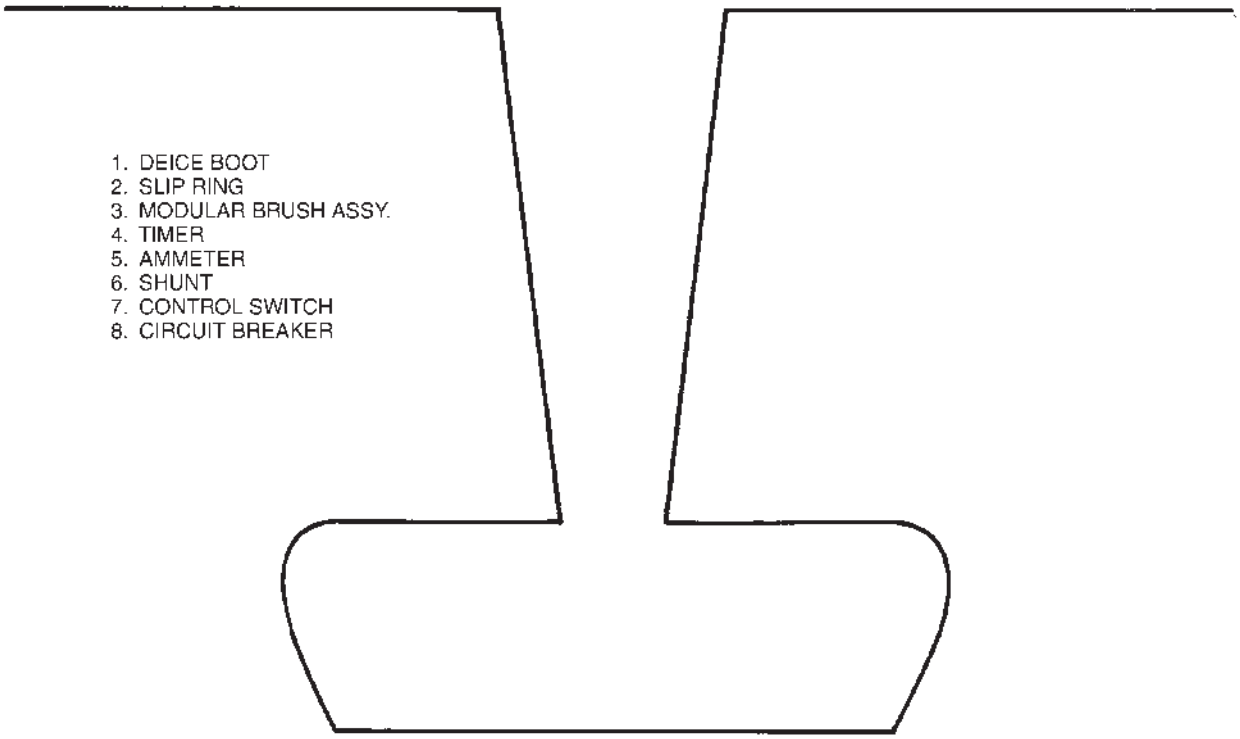
A. 50 Hour Inspection

(1) Lock brakes and operate engines at near takeoff power. Turn deicer system switch ON and observe deicer ammeter for at least two minutes. Ammeter needle must rest within the shaded band, except for a “flicker”, approximately every 34 seconds, as the step switch of the timer operates. If not, refer to the appropriate entry of the troubleshooting chart.

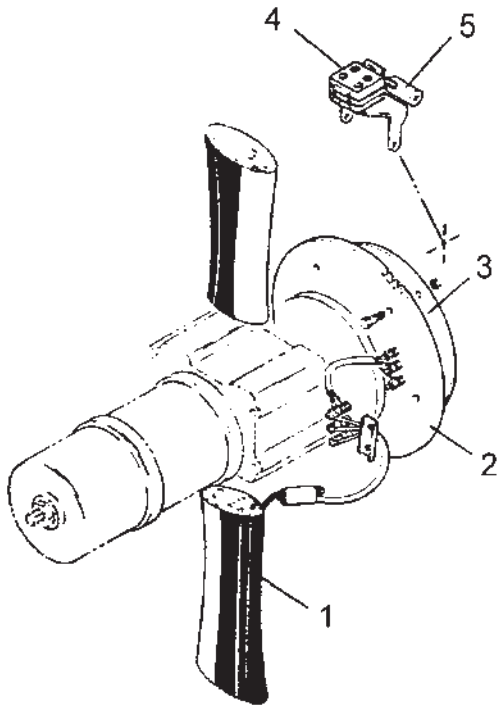
(2) (Seneca IV, two blade prop) With engines stopped, turn deicer switch ON and feel deicers on propellers for proper sequence of heater operation. The starting point is not important but sequence is vital and must be: Right Outboard, Right Inboard, Left Outboard and Left Inboard Heaters, in that order. Temperature rise should be noticeable and each heater should warm for about 34 seconds. Local hot spots indicate surface damage of deicer heaters; inspect and repair as directed in the boots section of this chapter.

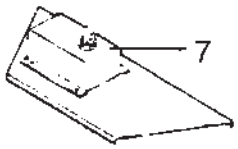
(3) Remove spinner dome and engine cowling. With assistant observing deicer ammeter and with deicer switch ON, flex all accessible wiring, particularly the deicer lead straps, leads from slip ring assembly, and the firewall electrical connectors and their wiring. Any movement of the ammeter needle other than the “34 second flicker” of cycling indicates a short or open that must be located and corrected.



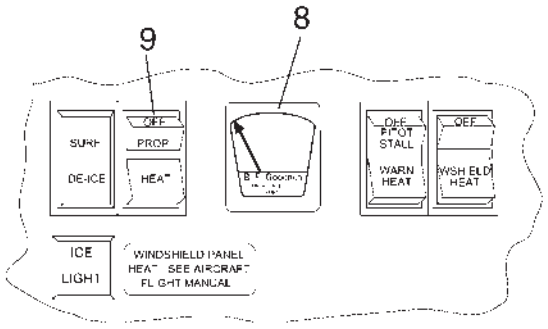


Propeller Deice Installation
Figure 1 (Sheet 1 of 3)



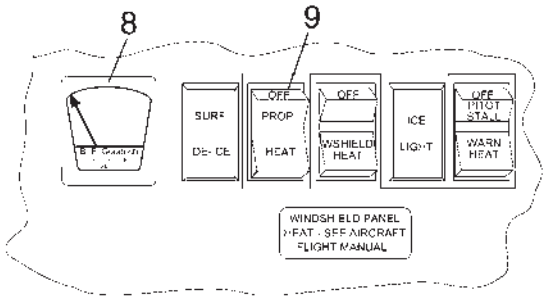






SENECA IV

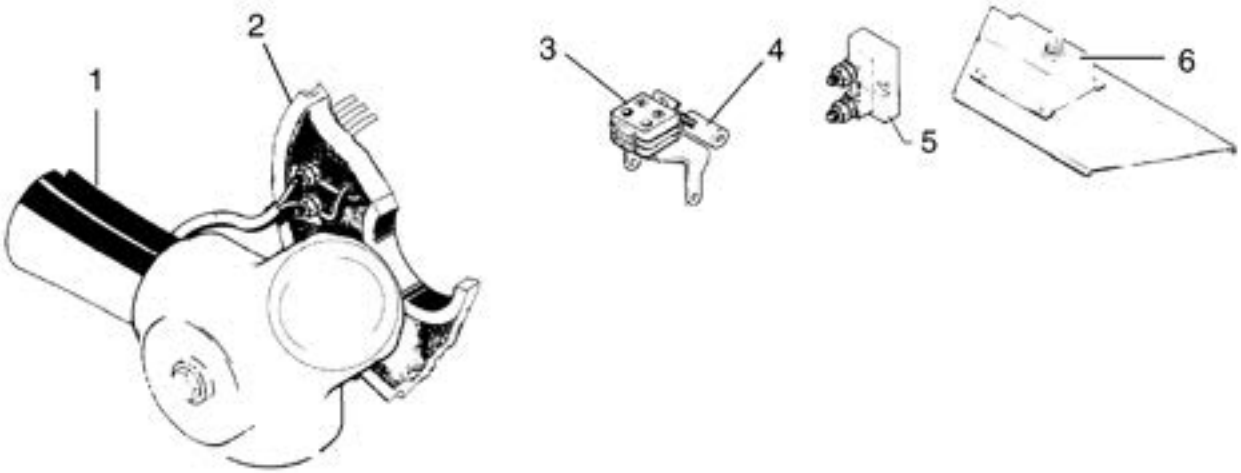
1. PROPELLER DEICER BOOT
2. SLIP RING
3. MOUNTING PLATE
4. MODULAR BRUSH ASSEMBLY
5. MODULAR BRUSH ADJUSTMENT
6. SHUNT (LOCATED ON LONGERON) BEHIND PILOT'S SIDE PANEL
7. TIMER
8. AMMETER
9. PROP DEICE SWITCH

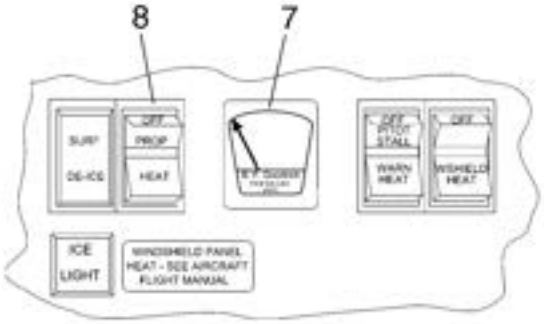


SENECA V

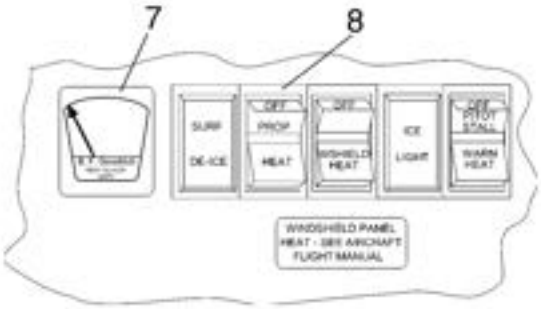
TWO BLADE INSTALLATION

Propeller Deice Installation
Figure 1 (Sheet 2 of 3)



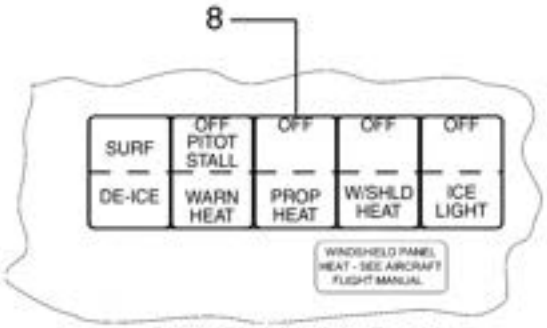


SENECA IV



SENECA V

1. PROPELLER DEICER BOOT
2. ALTERNATOR DRIVE SHEAVE
3. MODULAR BRUSH ASSEMBLY
4. MODULAR BRUSH ADJUSTMENT (SENECA IV ONLY)
5. SHUNT (LOCATED ON LONGERON)
6. TIMER
7. AMMETER
8. PROP DEICE SWITCH



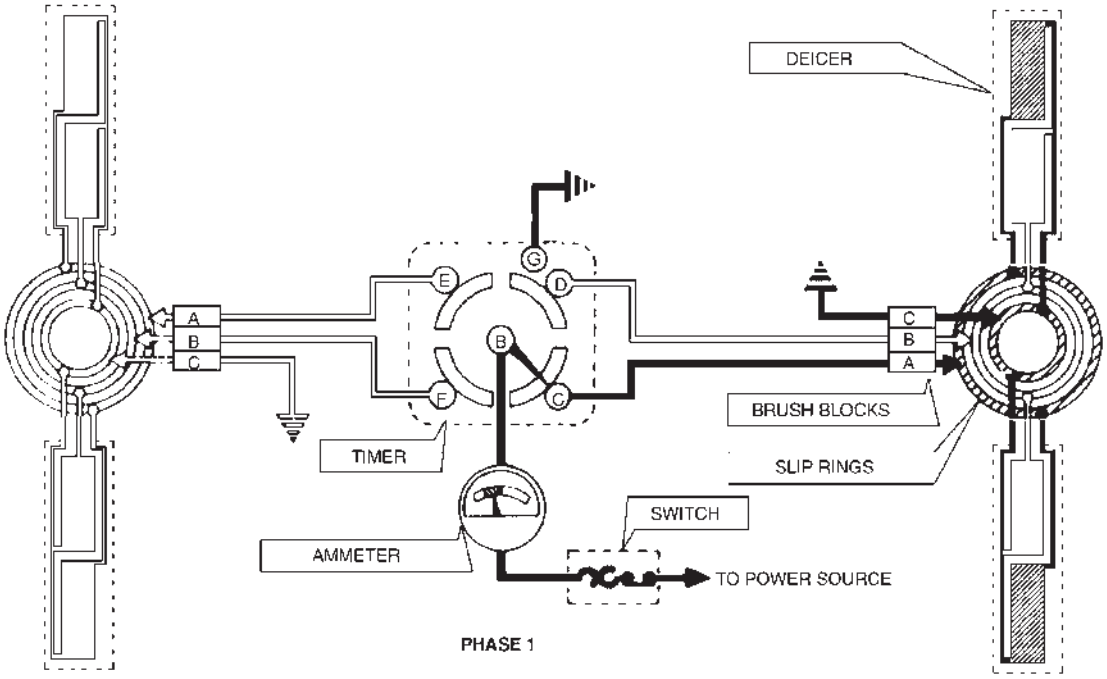
SENECA V WITH G1000
CONTROL PANEL

NOTE: FOR SENECA V G1000/G1000 NXi SEE EIS ON THE PFD/MFD FOR PROP HEAT AMMETER.

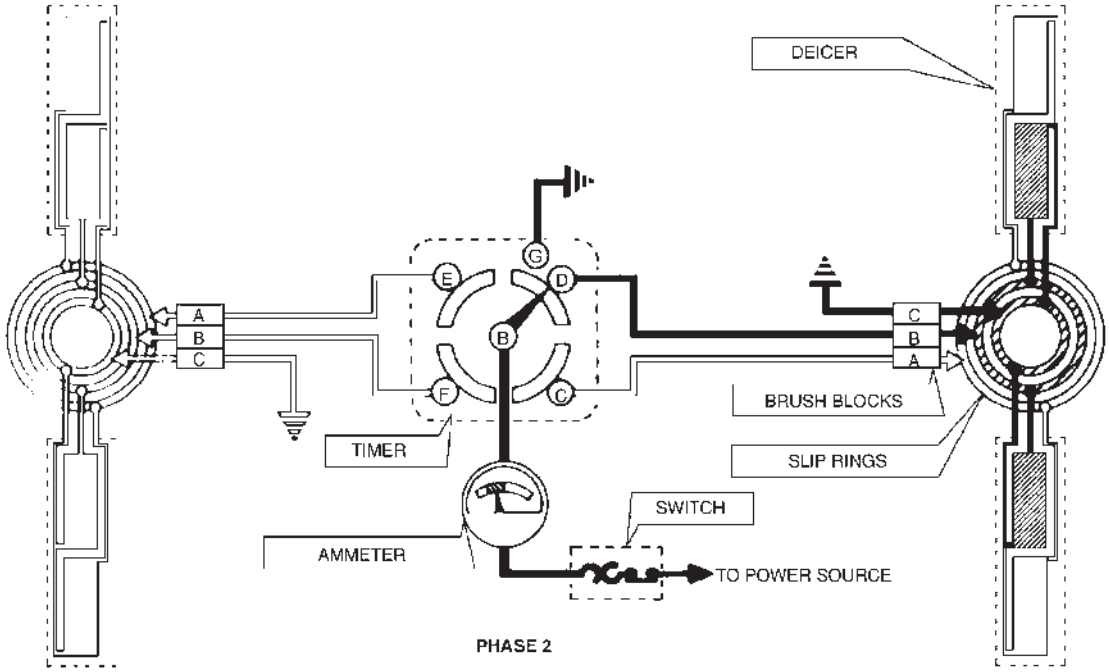
THREE BLADE INSTALLATION

Propeller Deice Installation
Figure 1 (Sheet 3 of 3)

ELECTRICAL DIAGRAM
SHOWING CYCLE SEQUENCE



PHASE 1

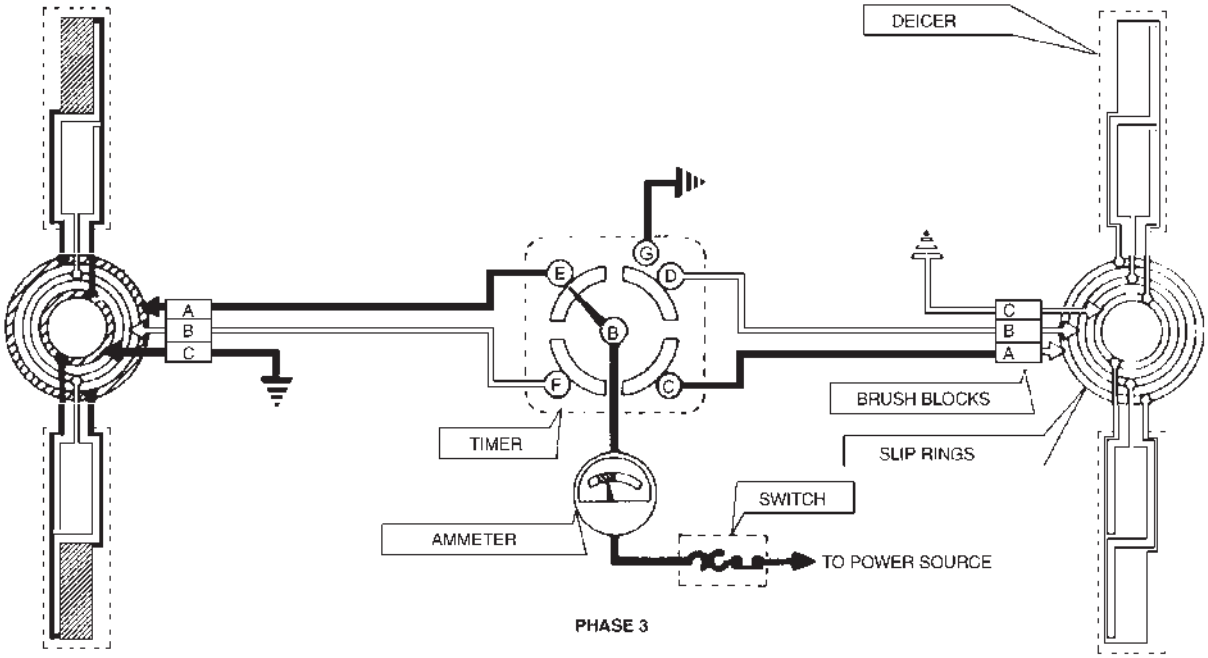


PHASE 2

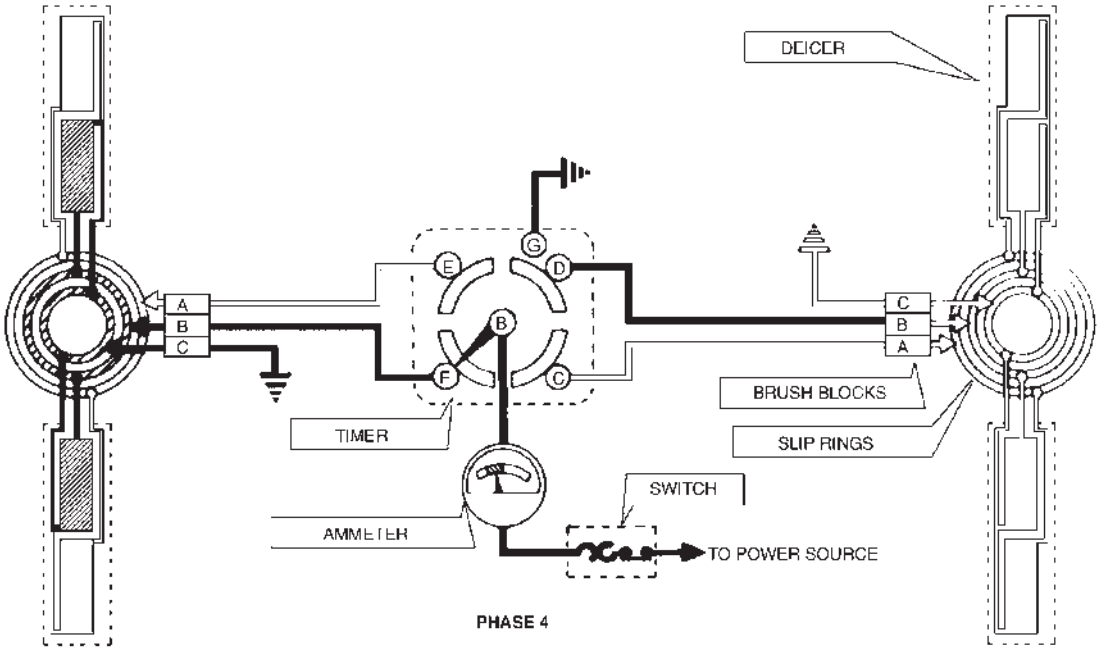
Effectivity
Seneca IV - Two Blade

Deicer Cycling Sequence
Figure 2 (Sheet 1 of 4)

ELECTRICAL DIAGRAM
SHOWING CYCLE SEQUENCE



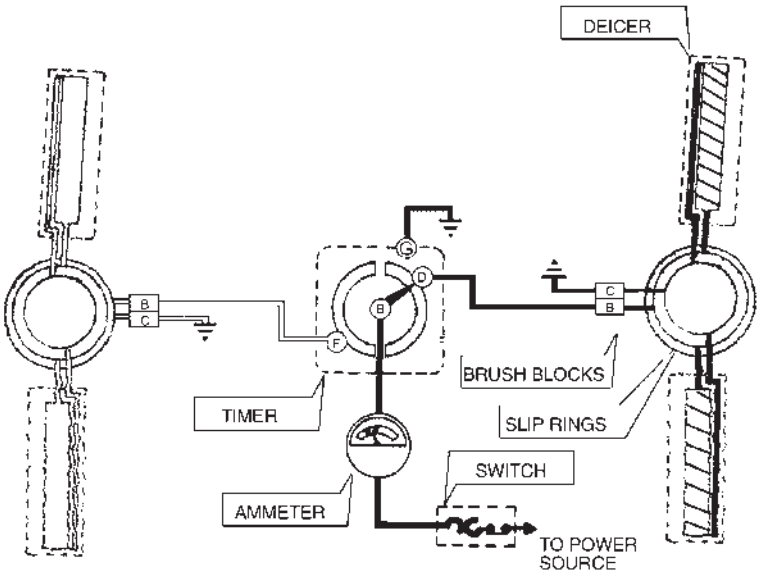
PHASE 3



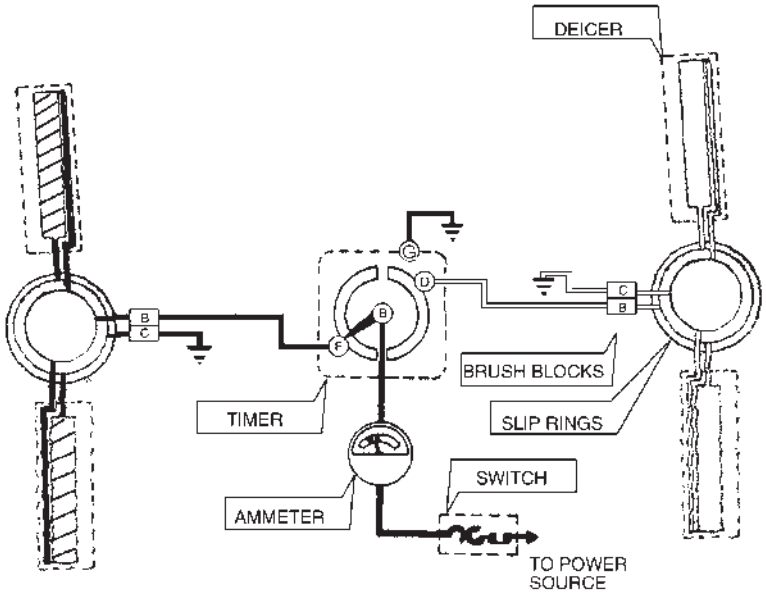
PHASE 4

Deicer Cycling Sequence
Figure 2 (Sheet 2 of 4)

ELECTRICAL DIAGRAM
SHOWING CYCLE SEQUENCE



PHASE 1

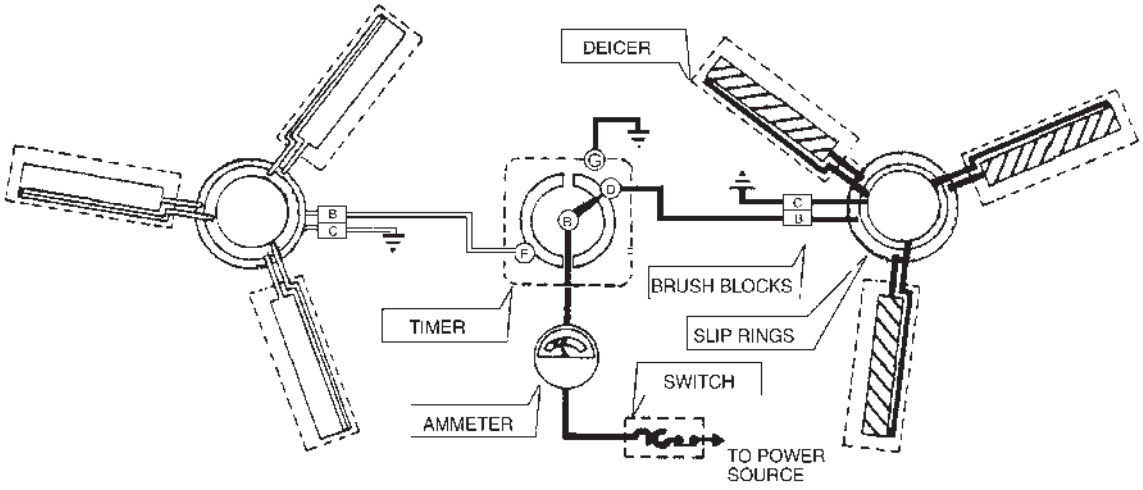


PHASE 2

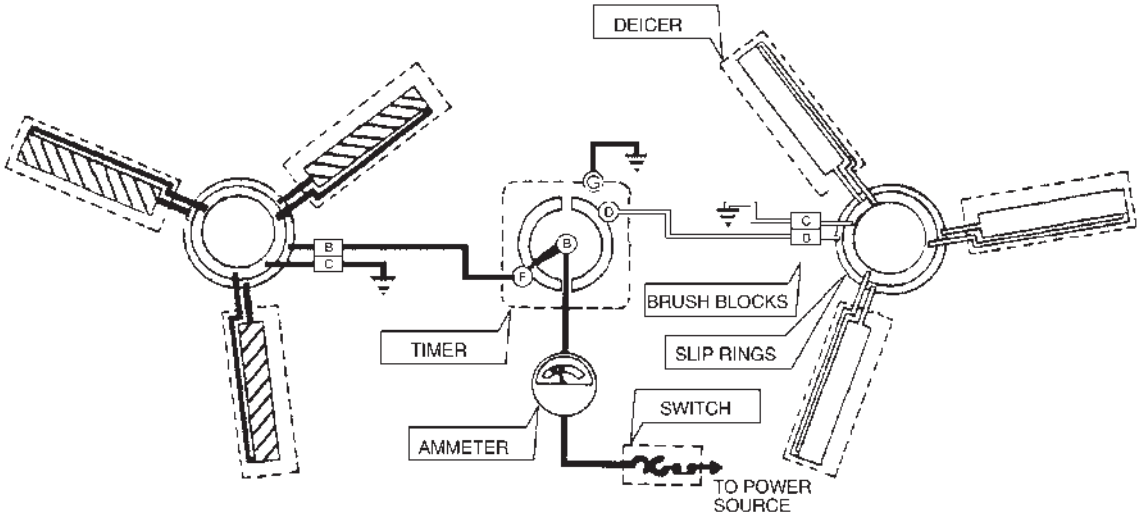
Effectivity
Seneca V - Two Blade

Deicer Cycling Sequence
Figure 2 (Sheet 3 of 4)

**ELECTRICAL DIAGRAM
SHOWING CYCLE SEQUENCE**



PHASE 1



PHASE 2

Deicer Cycling Sequence
Figure 2 (Sheet 4 of 4)

Effectivity
Three Blade

B. 100 Hour Inspection

(1) Remove cowling.

(2) Conduct 50 hour inspection.

(3) Check for radio noise or radio compass interference by operating the engine at near takeoff power with radio gear ON while turning deicer switch ON and OFF. If noise or interference occurs with deicer switch ON and disappears when switch is OFF, see troubleshooting chart.

(4) Ascertain that all clamps, clips. mountings and electrical connections are tight. Check for loose, broken or missing safety wire.

(5) Deicers: Closely check deicers for wrinkled, loose or torn areas, particularly around the outboard end and where the strap passes under the strap retainer. Look for abrasion or cuts, especially along the leading edge and the flat or thrust face. If heater wires are exposed in damaged areas or if rubber is found to be tacky, swollen or deteriorated (as from oil or solvent contact), replace the damaged deicer in accordance with the appropriate information in this chapter.

NOTE: Check the strap restrainers are correctly located and secure. Look for cracks or other damage. Operate propeller from “full pitch” to “feathering” and check that deicer lead straps do not come under tension or are pinched by propeller blade. (See “Figure 1” on page 30607.)

(6) Slip Rings: Check slip rings for gouges. roughened surface, cracks, burned or discolored areas, and for deposits of oil, grease or dirt.

(a) Clean greasy or contaminated slip rings with CRC 2-26 solvent. (This solvent is available from C.R.C. Chemical Division, Webb Inc. C-J10 Limekiln Pike, Dreshner, PA 19025.)

(b) If uneven wear is found or if wobble is noticed, set up a dial indicator as shown in “Figure 3” on page 306015 and check alignment of the slip rings to the propeller shaft as explained in this section.

(7) Brush Block - Brushes: Examine mounting brackets and housing for cracks, deformation or other physical damage.

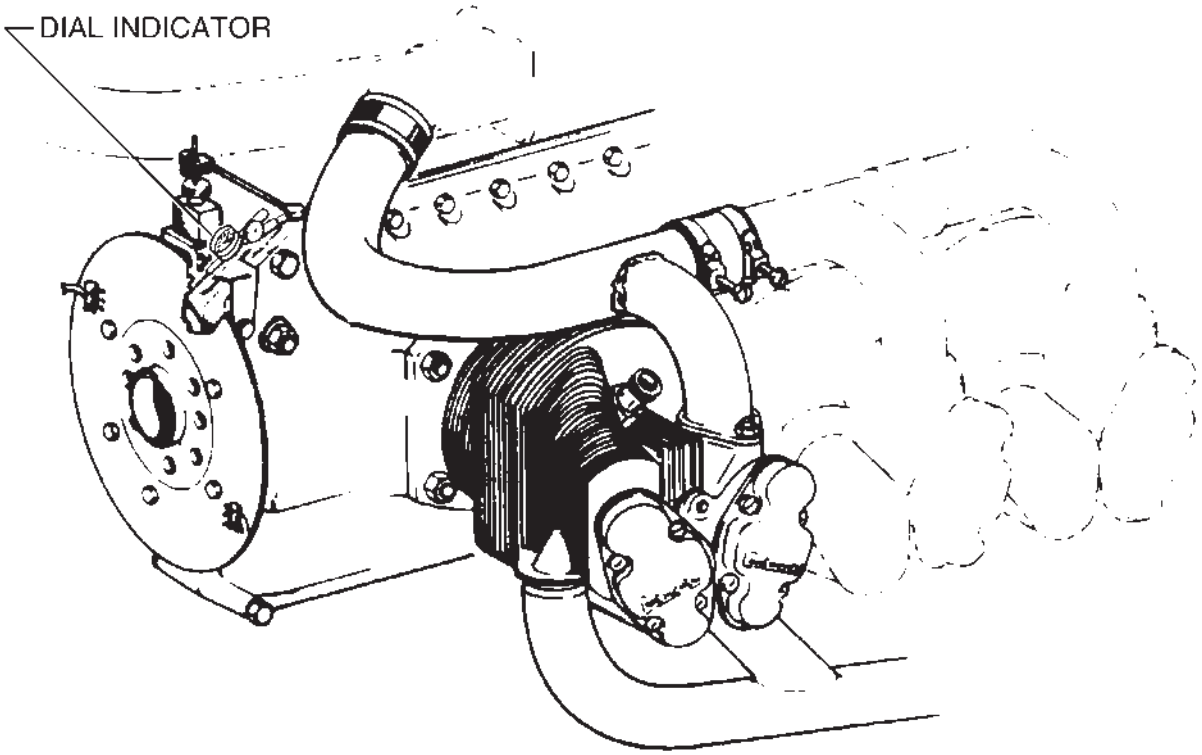
(a) Test that each brush rides fully on its slip ring over 360°. “Figure 4” on page 306015 shows the wear pattern if this condition is not corrected. If alignment is off, shim where brush block is mounted to bracket or adjust mounting bracket support arm.

NOTE: The shim is a series of laminates and may be peeled for proper alignment of brushes to slip ring.

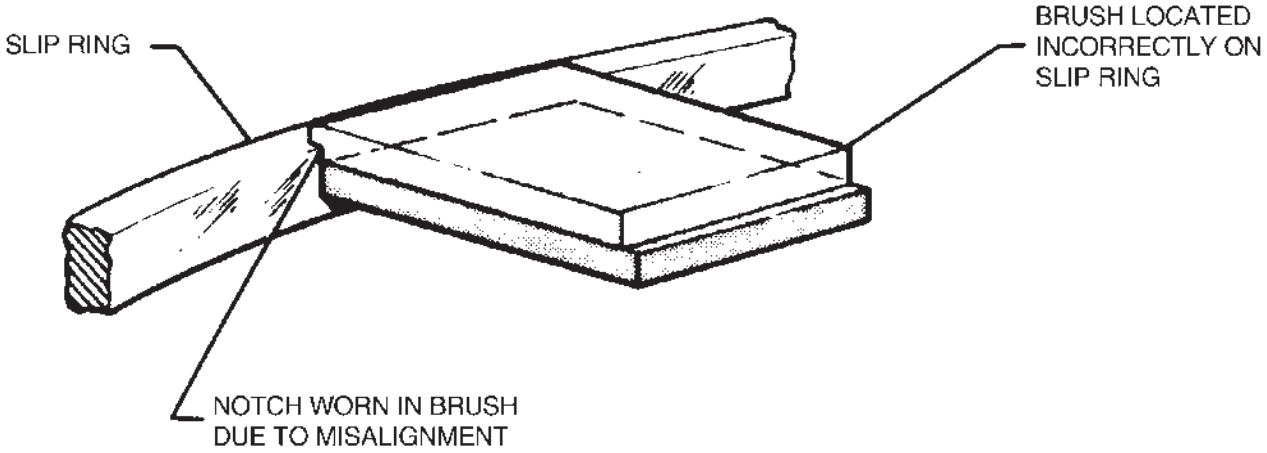
(b) Check for proper clearance of brush block to slip rings as shown in “Figure 5” on page 306016. If not correct, loosen mounting screws and move in elongated holes to correct block clearance before tightening securely.

(c) Visually check brush block for approximately 2° angle of attack. (See “Figure 5” on page 306016.) If not, loosen mounting screws and align block, but be sure to hold clearance limits shown when tightening.

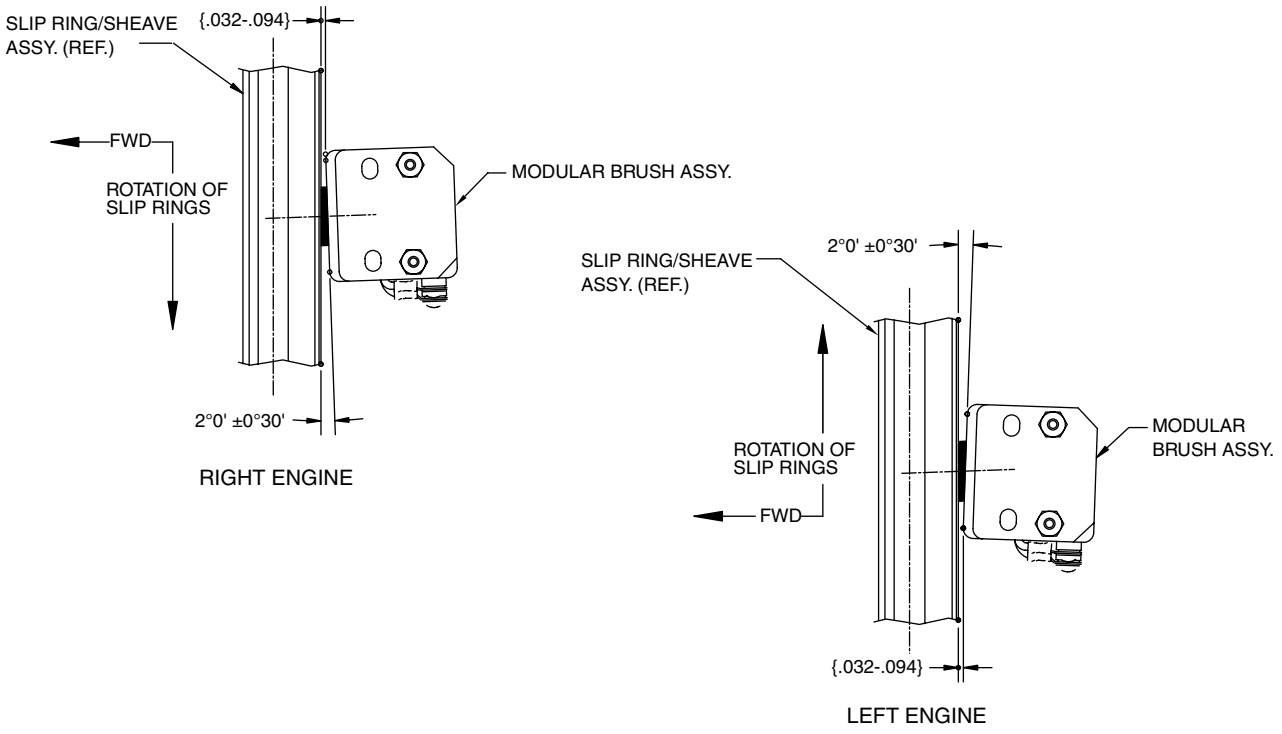
(8) System Wiring: With deicer system operating, have assistant observe ammeter while visually inspecting and physically flexing wiring from brush blocks through firewall, to timer, to ammeter, to switch and to aircraft power supply. The ammeter will flicker as the timer switches approximately every 34 seconds in the cycle. Jumps or flickers at other times indicates loose or broken wiring in the area under examination at that moment. In such case, check continuity through affected harness, while flexing and prodding each wire in the area that gave initial indication of trouble. Use the wiring diagrams in 91-30-60.



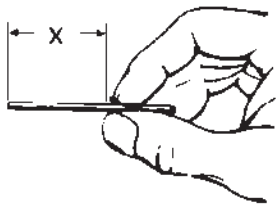
Using Dial Indicator
Figure 3



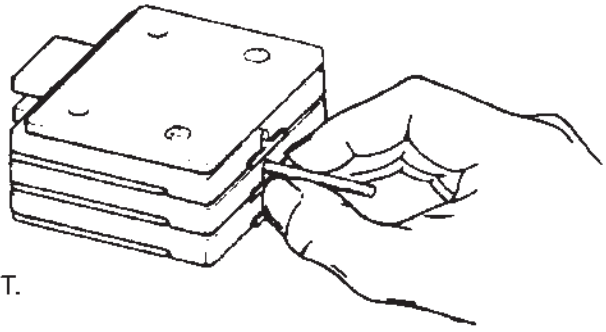
Centering of Brushes on Slip Rings
Figure 4



Modular Brush Assembly Alignment and Clearance
Figure 5



MEASURE BRUSH WEAR AS SHOWN, X - DIMENSION (INCHES) GIVEN INDICATES WHEN BRUSHES MUST BE REPLACED. DURING MEASUREMENT ONLY, 1/16 INCH OF BRUSH SHOULD BE ALLOWED TO PROTRUDE FROM BRUSH BLOCK. THIS IS NORMAL POSITION OF THE BRUSH WHEN INSTALLED ON THE AIRCRAFT.



X DIMENSION

BRUSH BLOCK ASSEMBLY	MUST REPLACE
3E-2042-1	23/64
3E-2062-2	23/64

Measuring Brush Assemblies
Figure 6

5. Brush Assemblies

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

A. Brush Module Replacement

Brush modules should be replaced when .375 inch of brush material remains; brush modules must be replaced when .250 inch remains. Measure the brushes as shown in “Figure 6” on page 306016. Replace brush modules as follows:

NOTE: Brushes are not offered individually as replacements. When a brush wears out, the module containing it should be replaced.

(1) Remove the modular brush assembly from the aircraft, by removing the attachment hardware, and disconnect the engine wire harness.

(2) Remove assembly screws and separate modules and spacers.

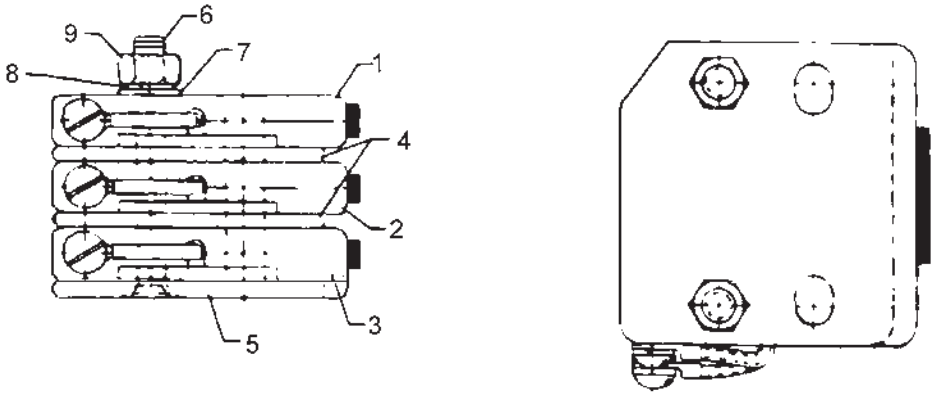
NOTE: The part number of each module is etched into the surface of the plastic housing; replace with the same part number module.

(3) Restack modules and spacers as shown in “Figure 7” or “Figure 8” on page 306018. If there is interference between adjacent ring terminals, reorient center module as shown in “Figure 9” on page 306018.

NOTE: Ascertain flat washer is positioned between star washer and housing.

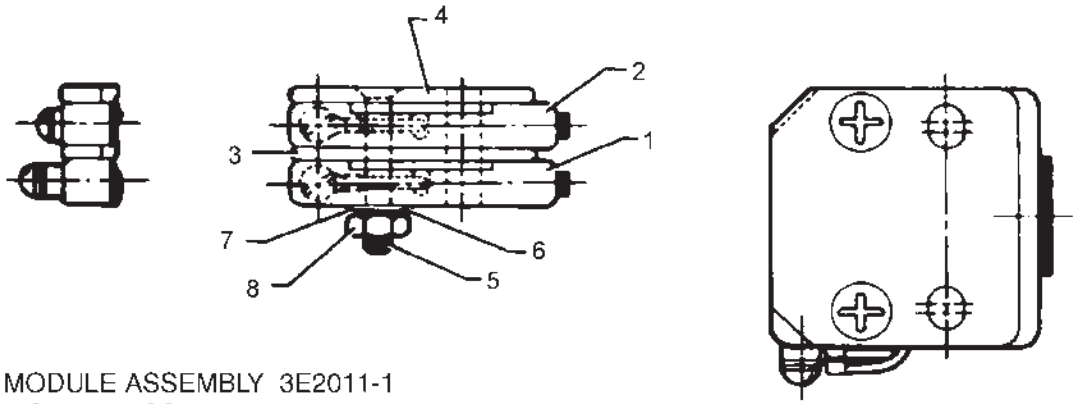
(4) Reconnect aircraft wire harness and ensure adjacent ring terminals are not touching.

(5) Install assembly on aircraft and check adjustment.



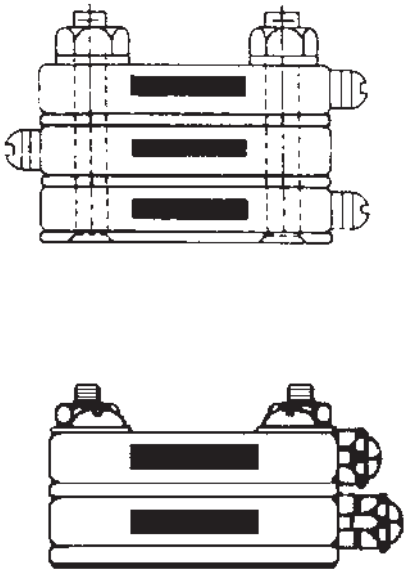
1. BRUSH MODULE ASSEMBLY 3E2011-1
2. BRUSH MODULE ASSEMBLY 3E2011-2
3. BRUSH MODULE ASSEMBLY 3E2011-3
4. SPACER
5. SPACER
6. SCREW
7. WASHER
8. LOCKWASHER
9. NUT

Modular Brush Assembly 3E2042-1
Figure 7



1. BRUSH MODULE ASSEMBLY 3E2011-1
2. BRUSH MODULE ASSEMBLY 3E2011-2
3. SPACER
4. SPACER
5. SCREW
6. WASHER
7. LOCKWASHER
8. NUT

Modular Brush Assembly 3E2062-2
Figure 8



Typical Module Stacking Arrangements
Figure 9

B. Alignment of New Brushes

Any time the brush block assembly is dismounted, the alignment at reinstallation must be checked as described in step (7) under “100 Hour Inspection.”

6. Slip Rings

A. Alignment

Excessive slip ring run-out will result in severe arcing between the slip ring and brushes and cause rapid brush wear. If the run-out is not corrected, rapid deterioration of the slip ring and brush contact surfaces will result and lead to eventual failure of the Deicing System. Check the slip ring run-out with a dial indicator securely attached to the engine with the pointer resting on the slip ring. (Refer to “Figure 3” on page 306015.) Rotate the propeller slowly noting the run-out indicated on the gauge. The total run-out must not exceed 0.005 inch ± 0.0025 inch and 0.002 inch in any 4 inch interval of slip ring travel.

NOTE: Some error may be induced in the readings by pushing in or pulling out on the propeller. Care must be taken to exert a uniform push or pull.

Small amounts of run-out may be corrected by varying the torque on the slip ring mounting bolts (AN4-7A) between 40 to 100 inch-pounds to obtain the required flatness.

B. Replacement

Slip ring assemblies that are open or shorted electrically, cracked or damaged structurally, or which have damaged surfaces beyond the scope of minor repair to clean up, should be replaced.

7. Deicer Boots

WARNING: CEMENTS AND SOLVENTS USED TO REMOVE AND INSTALL DEICERS ARE EXTREMELY FLAMMABLE AND TOXIC. EXTINGUISH OPEN FLAMES. AVOID SPARKS. USE IN WELL-VENTILATED AREA. AVOID SKIN CONTACT AND/OR PROLONGED BREATHING OF VAPORS. CONSULT MSDS FOR ADDITIONAL SAFETY INFO.

CAUTION: PROPELLER DEICER REPAIR IS LIMITED TO REFURBISHMENT OF EDGE SEALER. FOR HARTZELL BLADES, SEE LATEST REVISION OF HARTZELL'S PROPELLER ELECTRICAL DE-ICE BOOT REMOVAL AND INSTALLATION MANUAL, MANUAL NO. 182 (61-12-82).

GOODRICH PROPELLER DEICERS ARE NOT REPAIRABLE. DEICER SURFACE DAMAGE, SUCH AS PUNCTURES, CUTS, SCUFFS, EROSION, THAT EXPOSES THE WIRE OR ETCHED ELEMENT IS GROUNDS FOR REPLACEMENT, AS ADDITION OF REPAIR MATERIAL COULD AFFECT DEICER PERFORMANCE.

CAUTION: DISPOSE OF UNUSED MEK AND OTHER CHEMICALS AND SOLVENTS IN A MANNER CONSISTENT WITH LOCAL LAWS AND/OR ENVIRONMENTAL PROTECTION AGENCY REGULATIONS.

A. Resistance Check of Deicer Boots

Disconnect the deicer lead harness to measure heater resistances individually. See "Chart 2" on page 306021 or "Chart 3" on page 306021 for resistances. If this check is off limits, the deicer is damaged and must be replaced.

B. Replacement

If tests show the blade deicer to have an open circuit, to be the wrong resistance or to be visibly damaged beyond repair as outlined in this section, replace the deicer as directed in the following paragraphs.

C. Removal and Installation on Hartzell Propellers

NOTE: The following applies to factory propeller installations only.

See latest revision of Hartzell's Propeller Electrical De-ice Boot Removal and Installation Manual, Manual No. 182 (61-12-82).

CHART 2
DEICER ELECTRICAL RESISTANCE - SENECA IV

Resistance Check	Min.	Max.
Two Bladed Propeller		
Each Element	4.58	5.26
2 Elements in Parallel	2.29	2.63
Three Bladed Propeller		
Each Element	4.74	4.90
3 Elements in Parallel	1.58	1.63

CHART 3
DEICER ELECTRICAL RESISTANCE - SENECA V

Resistance Check	Min.	Max.
Two Bladed Propeller		
Each Element	4.74	4.90
2 Elements in Parallel	2.37	2.45
Three Bladed Propeller		
Each Element	4.74	4.90
3 Elements in Parallel	1.58	1.63

D. Removal (McCauley Propellers only)

NOTE: This and the following paragraphs E thru H apply exclusively to the removal and installation of deicers on McCauley propellers.

NOTE: Goodrich's Installation and Maintenance Manual for Prop De-Icing Systems ATA Report No. 30-60-02 and Removal and Installation Manual, Standard and FASTprop Electrothermal Propeller De-Icers ATA Report No. 30-60-07, provide approved, alternate procedures for removing and installing deice boots. Reference to them is highly recommended. They can be obtained online at https://customers.goodrich.com

NOTE: See "Chart 4" for required materials.

(1) Disconnect terminals of propeller deicer from studs on the spinner bulkhead.

CAUTION: DO NOT ALLOW SOLVENTS TO LEAK INTO PROPELLER HUBS AND CAUSE DAMAGE TO SEALS.

(2) Use MEK or Toulene to soften the adhesion line between the deicer and the propeller blade.

(3) Starting at one corner of the deicer, loosen enough of the deicer to grasp in the jaws of vise grip pliers or similar tool.

(4) Apply a steady pull on the deicer to pull it off the propeller surface. Continue using MEK or Toulene to soften the adhesion lines. Unless the deicer being removed is damaged and is to be scrapped, cushion the jaws of any pulling tool used to prevent damage to the deicer surface. Remove very slowly and carefully. If deicer has failed and is to be returned under request for warranty, extreme care should be exercised so that no additional damage is incurred to the deicer during and after removal.

(5) Remove residual cement from blade. Use Turco No. 3 or equivalent to help with dried cements.

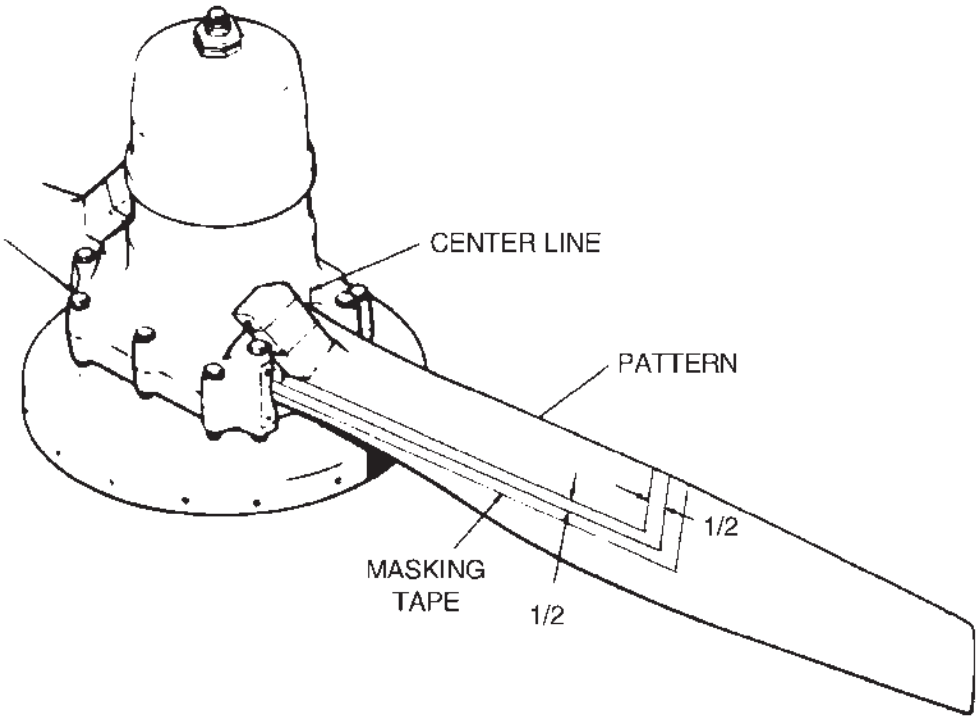
CHART 4
REQUIRED MATERIALS FOR REPAIR OF PROPELLER DEICER

The materials and tools listed below are commercially available and are not supplied by B.F. Goodrich in kit form:

Cement 1300L (Minnesota Mining & Mfg. Co.) (*Piper P/N 912-019)
Filler EC801 (6 Hour) *Piper P/N 279-047)
Sealer A-56-B *Piper P/N 912-018
Cleaning Solvent - MEK (MethylEthylKetone) or Acetone
Tackifying Solvent - Toulene or MEK (See Note)
Cleaning Cloth - any clean, lint-free cloth
1 inch paint brushes
2 inch rubber hand roller
1/4 inch hand stitcher
Masking tape

Piper Part Numbers reflect 1 quart containers.

NOTE: MEK may be used instead of Toulene to tackify cement, but it provides approximately 10 seconds working time for deicer applications, whereas Toulene provides approximately 40 seconds working time.



Installation of Deicer Boot
Figure 10

E. Blade Preparation (McCauley Propellers only)

CAUTION: CLEANLINESS OF METAL AND RUBBER PARTS CANNOT BE TOO HIGH STRESSED. ONLY PERFECTLY CLEAN SURFACES WILL ASSURE MAXIMUM ADHESION.

NOTE: All deicers on a single propeller must be located at same distance from the hub for rotational balance.

(1) Mark and cut from masking tape a pattern the size of the propeller deicer. (See “Figure 10” on page 306023.)

(2) Place a mark at the hub end of the blade in line with the blade leading edge. The location for this mark can be determined by sighting along the leading edge. Starting at the hub (See Note above), center the pattern on this mark and stick the pattern to the leading edge. Mark the position of the deicer harness.

CAUTION: DO NOT CLEAN PROPELLER BLADE TO BARE METAL. McCAULEY PROPELLERS HAVE A SPECIAL PROTECTIVE COATING NOT EASILY REMOVED BY SOLVENT. BOND DEICER DIRECTLY TO THIS PROTECTIVE COATING.

(3) Remove the pattern and clean the marked off area thoroughly, with MEK or acetone. For final cleaning, wipe the solvent off quickly with a clean, dry lint-free cloth to avoid leaving a film.

(4) Using a pencil or pen, mark a centerline at the hub of the propeller blade and on the tape at the outboard edge of the masked area.

F. Cement Application (McCauley Propellers only)

(1) Using a marker pencil, mark a centerline on the glossy side of the deicer.

(2) Moisten a clean cloth with MEK or acetone and clean the unglazed surface of the deicer, changing cloth frequently to avoid contamination of the clean area.

(3) Thoroughly mix the 1300L cement. Apply one even brush coat of cement to the unglazed back surface of the deicer. Cement one inch of the deicer lead strap. Allow to air dry for a minimum of one hour at 40°F (4.5°C) or above, when the relative humidity is less than 75%. If the humidity is 75% to 90%, allow two hours drying time. Do not apply cement if the relative humidity is higher than 90%. After allowing the proper amount of drying time, apply a second even brush coat of 1300L cement.

NOTE: If curling of the deicer edges is a problem, apply masking tape to the edges of the glazed side before applying cement to the unglazed side. Remove the tape before starting to install the deicer.

(4) Apply an even brush coat of 1300L cement on the cleaned surface of the propeller blade, immediately after the second coat of cement has been applied to the deicer. This timing is important for the cement on both surfaces to reach the tack stage at the same time.

G. Installation (McCauley Propellers only)

It is imperative that the following instructions be followed exactly to ensure maximum adhesion to the propeller blades:

(1) When the cement coats are tacky (slightly sticky to the touch - like masking tape), dry on both the propeller surface and deicer surface, position deicer on blade leading edge. Start at hub end, using centerlines as a guide. (See “Figure 10” on page 306023.)

(2) Make sure that the harness will fall in the previously marked position.

(3) Working outward toward the tip, tack the deicer centerline to the leading edge of the propeller blade.

(4) Use the tackifying solvent as necessary. If deicer is allowed to get off course, pull up with a quick motion and re-apply deicer.

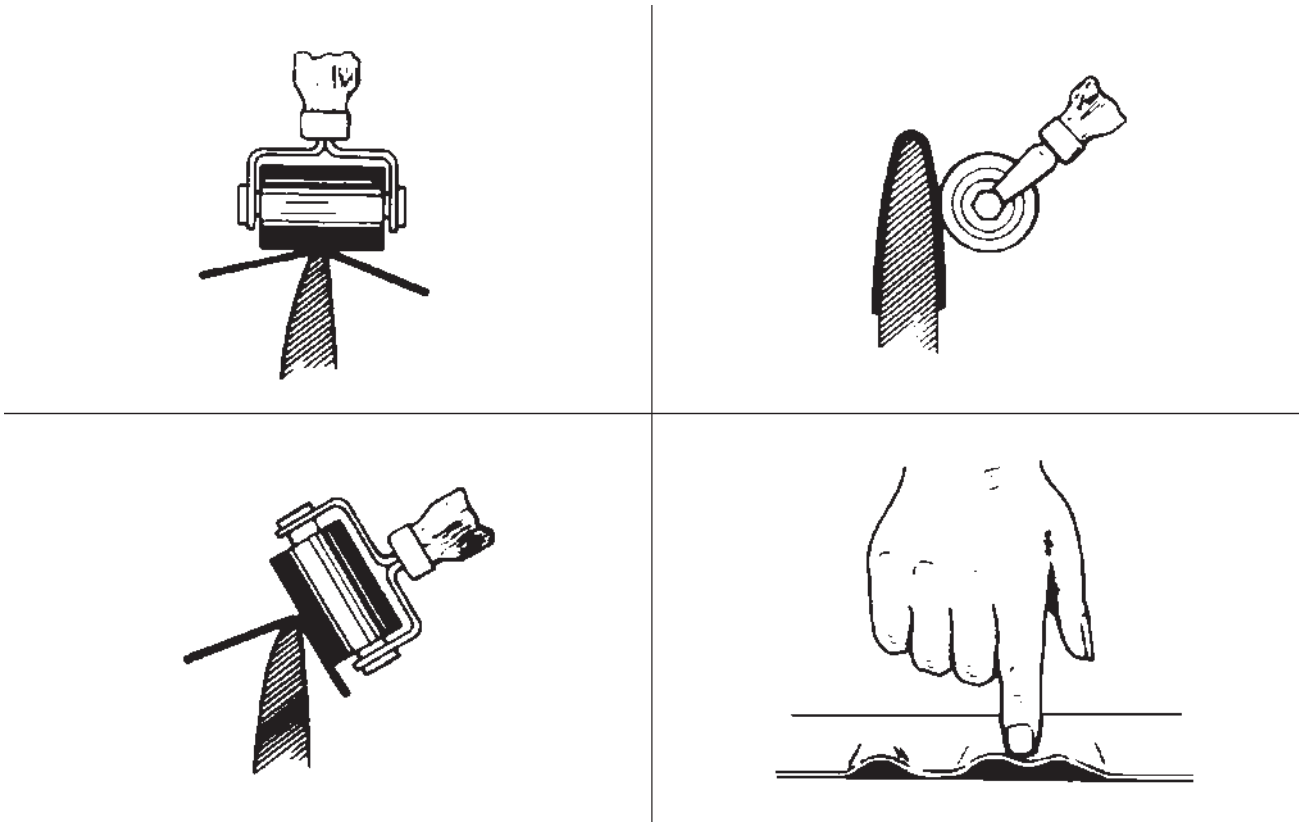
(5) If cement is removed from either surface, completely remove the deicer and re-apply cement as explained in the previous paragraph.

(6) When the deicer is correctly positioned, roll firmly along the centerline with a rubber roller. Refer to “Figure 11” on page 306025.

(7) Gradually tilt the rubber roller and carefully work the deicer over either side of the blade contour to avoid trapping air. Roll outwardly from centerline to edges. Be especially careful to work out excess material at outboard edge of deicer before other edges are completely rolled down. If excess material at edges tends to pucker, work out puckers smoothly and carefully with fingers.

CAUTION: TO AVOID DAMAGE TO DEICER RESISTANCE WIRES, DO NOT USE METAL STITCHER ON BODY OF DEICER. AREA WHERE METAL STITCHER IS PERMITTED NOT TO EXCEED 3/16” ALONG DEICER EDGE.

(8) Roll the tapered edges, especially inboard edge of the deicer with the metal stitcher.



Wrinkled Deicers
Figure 11

H. Preparation and Application of Sealer (McCauley Propellers only)

Deicers loosened due to destruction of adhesive bond by lubricants do not respond well to recementing. Therefore, removal, cleaning, and reinstallation of the deicers are recommended.

(1) Clean an area .500 of an inch wide around the circumference of the deicer down to the bare metal. Use MEK or Acetone and clean thoroughly.

(2) Clean outer .500 of an inch of all deicer edges and back under deicer about .250 of an inch on all sides past loosened areas with MEK or Acetone. For final cleaning, quickly wipe off solvent with a clean, dry lint-free cloth to avoid leaving a film.

(3) Recement loosened areas of deicers in accordance with the paragraph on cement application.

(4) Mix filler, sealer, or paint thoroughly and in the proper proportions by weight, see “Chart 5” on page 306026.

(5) Locate masking tape approximately .125 of an inch beyond the cemented area around the deicer to permit filler material to contact bare metal.

(6) Apply one even coat of filler to area around the inboard end and sides of the deicer. (See “Figure 12” on page 306027.) Immediately remove the masking tape and allow the filler to dry for six hours.

(7) Apply masking tape about .125 of an inch beyond filler or .250 of an inch beyond cemented area when no filler is used, to permit sealer to contact bare metal. Apply one even brush coat of sealer to the area around the deicer. (See “Figure 10” on page 306023 and “Figure 12” on page 306027.) Remove masking tape immediately and allow sealer to dry. Allow 12 hours cement curing time before starting engine, allow 24 hours cement curing time before operating the deicers.

CHART 5
MIXING OF MATERIALS

Material	Manufacturing & No.	Mixing Proportions
Filler	EC801 (6 Hr.) (Qt.) Piper P/N 279 047	Base 100 parts by weight. Accelerator 10 parts by weight.
Sealer Conductive Cement	A-56-B (Qt.)	None

I. Wrinkled Deicers

See “Figure 11” on page 306025.

If edge of deicer is found wrinkled or loose, try recementing. Use MEK or Toulene to loosen the bond for an additional 1/4 inch beyond the loose or wrinkled area. Apply one coat of 1300L cement to the deicer and propeller bonding surfaces and allow to air dry for one hour. Then apply a second coat of 1300L cement to both the deicer and bonding surface. Allow to dry. Retackify with MEK or Acetone and press with fingers to work out wrinkles or to secure loose edges. If material has stretched and will not cement flat, replace the deicer.

8. Electrical Check

Procedures

(1) Check the electrical resistance of each of the two elements within the deicer. (See Electrical Schematic, 91-30-60, and Charts 2 and 3.)

(2) Check for intermittent open circuits by tensioning the deicer strap slightly while measuring the resistance. Also, press lightly on the deicer surface in the area adjacent to the harness. Resistance must not vary.

(3) Identification of the circuits within the element may be confirmed by referring to the resistance values and schematic diagram. Proper identification is necessary in order to make the system cycle properly and to obtain the correct amperage values during system operation.

NOTE: These resistances apply only to deicers that are not connected to terminal studs.

9. Deicer Wiring Harness

A. Two Bladed Propellers

Refer to “Figure 13” on page 306028.

The deicer wiring harness must be installed to the propeller counterweight as follows:

(1) Install deicer harness through 9/32 inch hole in counterweight.

(2) Refer to view A. Install MS3367-1-0 strap between leads along length of plug. Do not tighten.

(3) Refer to view A. Install two MS3367-2-0 straps under MS3367-1-0 strap and around counterweight. Do not tighten.

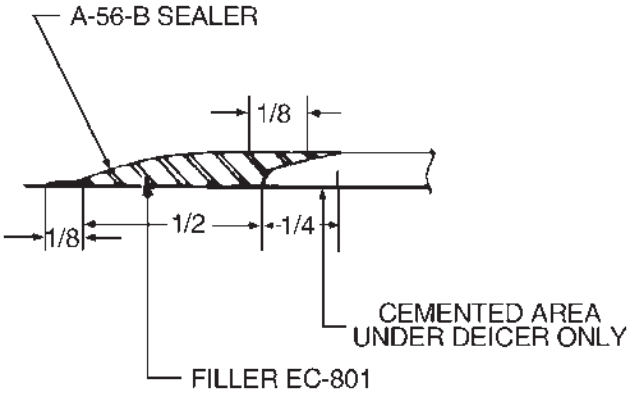
(4) Install tubing over deicer wire harness.

(5) Route the wire harness and protective tubing under both MS3367-2-0 straps. Tighten both tie straps.

(6) Tighten MS3367-1-0 strap around plug.

(7) Install deicer harness wire terminals to screws on spinner bulkhead. (Refer to view B.).

(8) Install lead clip over harness.



Sealer Application (Boot)
Figure 12

B. Three Bladed Propellers

Refer to “Figure 14” on page 306029.

(1) If necessary, install connector mounting bracket to propeller hub using appropriate number of screws and washers.

(2) Join connector plug on end of harness to connector plug extending from propeller deicer boot. Secure connection by installing a MS3367-2-0 strap around connector. (Refer to view B). Ensure that strap is routed between wires leads extending from each end of connector.

(3) Secure connector to connector mounting bracket using two MS3367-1-0 straps. (Refer to views A and B.) Route straps under MS3367-2-0 strap. Note position of strap buckles in “Figure 14” on page 306029.

(4) Using two MS3367-1-0 straps, secure harness and wires leading to propeller deicer boot to connector mounting bracket as shown in view B.

(5) Secure harness to spinner bulkhead using clamp, screw, washer, and nut. Clamp must be positioned 90° to radial line as shown in “Figure 14” on page 306029. Ensure clamp is installed in correct mounting hole with respect to propeller rotation.

(6) Connect harness to terminal strip.

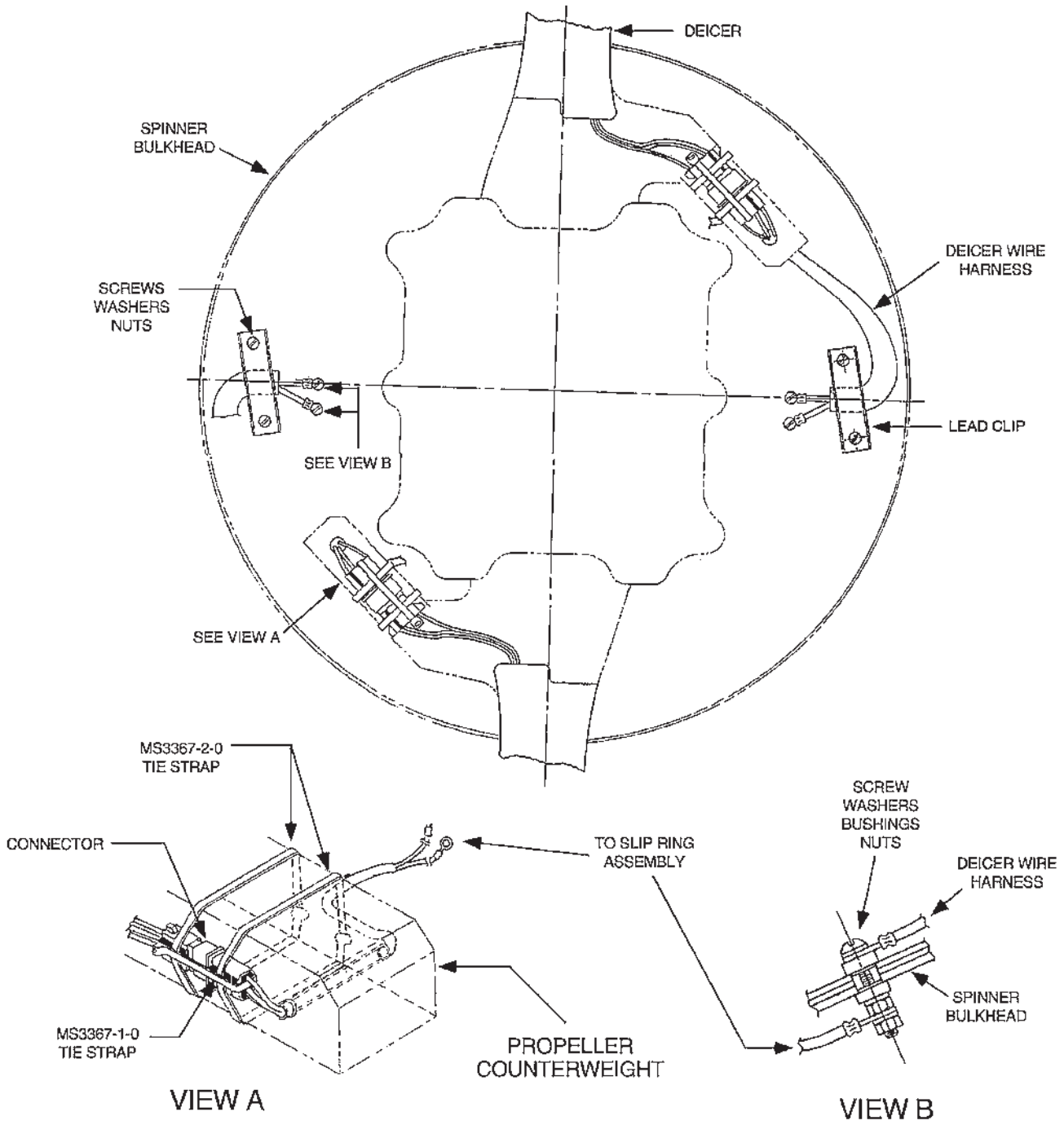
C. Balancing

To assure balance of the propeller assembly, the original balancing weights or their equivalents must be reinstalled. The weights must be left in the original position on the propeller hub. The restrainer and weights should not interfere with any part of the propeller assembly under any condition. If for any reason balance weights were removed, reinstall safety wire on screws. The deicer wire harness must be installed on the propeller as just described.

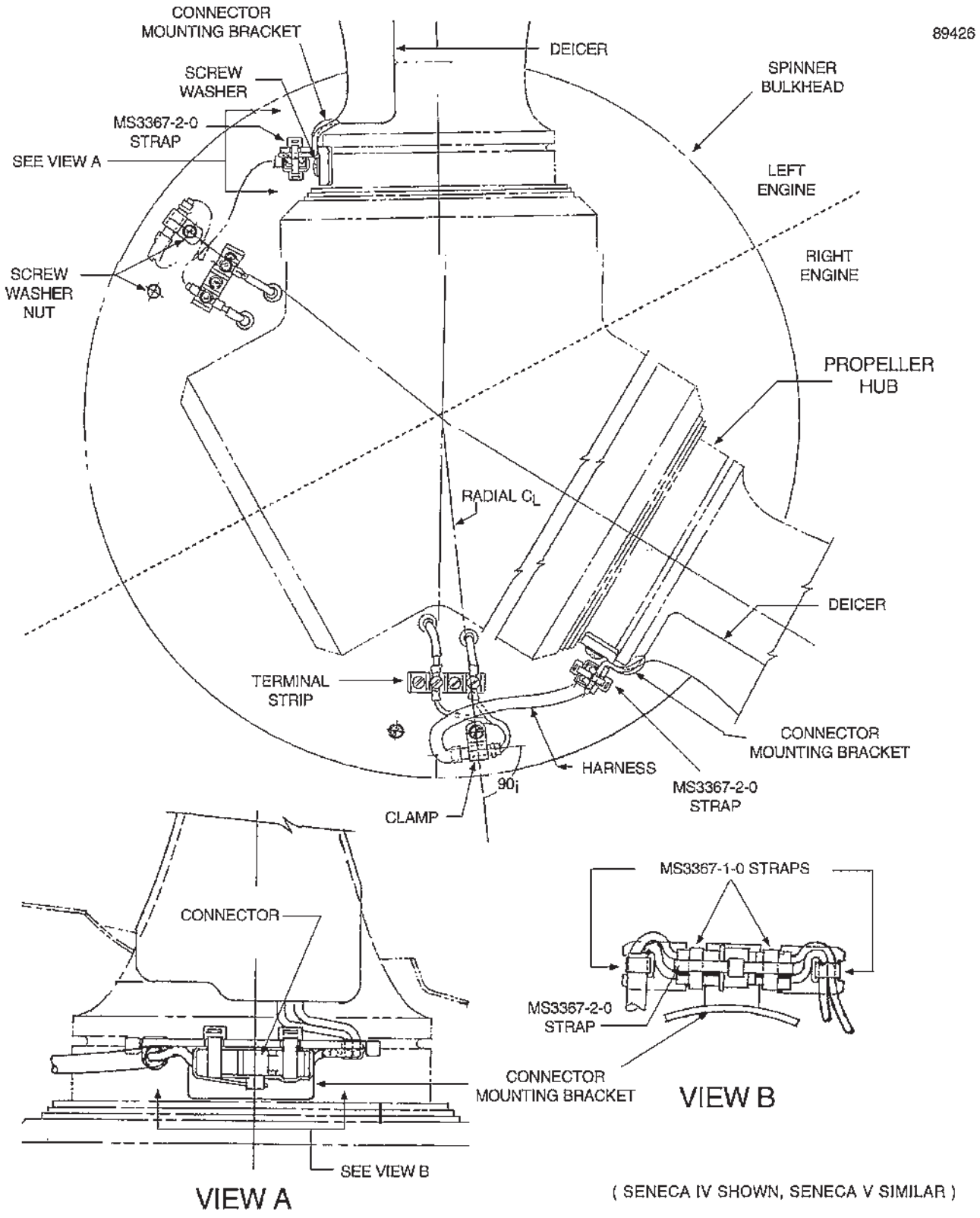
D. Final Electric Check

(1) Make certain that all terminals are tight. Do not over torque.

(2) Check the electrical resistance between the deicer terminals or between the slip rings. The reading should be per “Chart 2” on page 306021 or “Chart 3” on page 306021.



Two Bladed Propeller Deicer Harness Installation
Figure 13



Three Bladed Propeller Deicer Harness Installation
Figure 14

10. Other Components

Do not attempt internal repairs of the timer, ammeter or switch. If inoperative, these components must be replaced. For any other repair or maintenance problems not covered herein, inquire at De-Icing & Specialty Systems Division of the Goodrich Corporation or Hartzell Propeller, depending on which components are installed on your airplane.

11. Timer

The timer for controlling both the right and left systems is mounted in the nose of the aircraft. The unit is sealed and, if found inoperative, must be replaced. Timer field repairs are not authorized.

Testing

Field experience indicates that too often the timer is considered at fault when the true trouble lies elsewhere. Before removing a timer as defective, perform this test:

(1) Disconnect wire harness at timer and with deicer switch ON, check voltage from Pin B of harness plug to ground. If system voltage is not present, the fault is not in the timer. If system voltage is present at Pin B, check ground circuit using ohmmeter from Pin G to ground. If no circuit is shown, the fault is in ground lead, not in timer. If ground connection is open, the timer step switch will not change position.

(2) When power and ground circuits have been checked, connect a jumper wire from Pin B of harness to B contact of timer socket to power timer. Connect a jumper wire from Pin G of harness to G contact of timer socket to complete the power circuit. Now use voltmeter from ground to the timer socket and check that timer is cycling to deliver system voltage to C, D, E, and F contacts in that order. (The starting point is not important but sequence must be as given.) Each of these four contacts must deliver voltage for approximately 34 seconds, in turn, and there must be zero voltage on the three contacts not energized.

(3) If the timer meets these requirements, it is not the cause of the trouble. If it fails to perform as specified above replace timer.

CHAPTER

31

INDICATING / RECORDING SYSTEMS

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CHAPTER 31

LIST OF EFFECTIVE PAGES

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CHAPTER 31 - INDICATING / RECORDING SYSTEMS

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INDEPENDENT INSTRUMENTS

Carbon Monoxide (CO) Detector and Pulse Oximeter

Optional in S/N's 3449509 and up. See "Figure 1" on page 31202.

The Model 455 Pulse Oximeter and Carbon Monoxide Detector is designed to detect, measure, and provide a visual alert to the crew before the cockpit level of carbon monoxide (CO) reaches a critical level. Additionaly, the occupants of the aircraft are able to monitor their physiological condition using a pulse oximeter installed in the instrument panel. It measures SPO2 (oxygen saturation percentage in blood) and heart rate. CO PPM is displayed on the engine page of the MFD (see "Figure 2" on page 31203).

A. Troubleshooting

CO Detectors must be returned to CO Guardian for repair, calibration or overhaul.

B. Service Life

The CO Detector has a usable life of five (5) years. To ensure crew safety, and per the manufacturer's recommendation, replace the unit with a new or recalibrated unit every five (5) years.

C. Inspection/Maintenance

The carbon monoxide detector and associated equipment do not require periodic scheduled servicing or periodic scheduled preventive maintenance. Conduct a general visual inspection for system integrity, installation security, corrosion and chaffing.

NOTE: Be sure the vent on the faceplate is free of obstructions. Any failures of the system are evident to the pilot through a system failure message on the PFD.

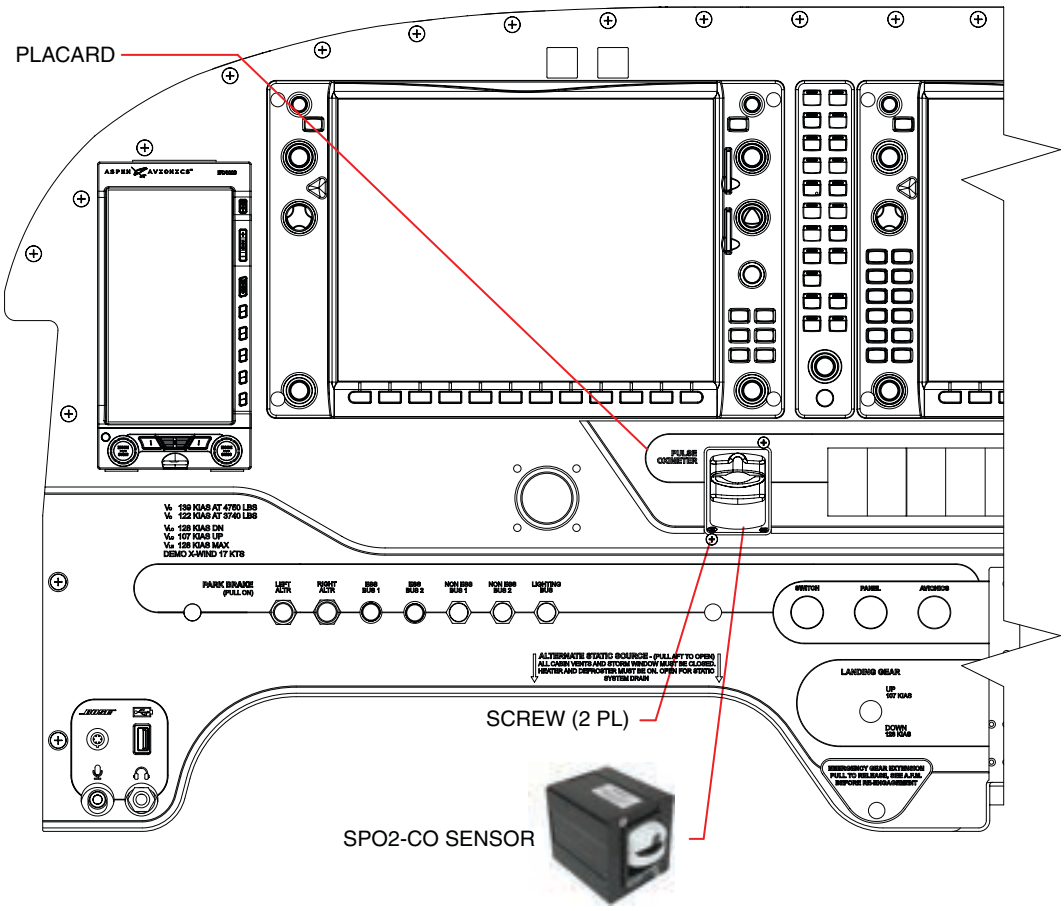
D. Removal

(1) Ensure aircraft power is off.
(2) Remove two (2) mounting screws and carefully pull the pulse oximeter away from the instrument panel to allow access to the electrical connector on the rear.
(3) Remove the electrical connector.

E. Installation

(1) Attach the electrical connector at the rear of the pulse oximeter.
(2) Feed the elctrical cable through the mounting hole in the instrument panel and place the pulse oximeter in the instrument panel.
(3) Secure the pulse oximeter using the two (2) mounting screws.

34F24A007 NEW



VIEW LOOKING FWD

Pulse Oximeter
Figure 1

F. Self-Test Sequence at Startup

(1) When the airplane master battery switch is selected ON, the 455 Detector goes through a self-test routine.

(2) The self-test checks for functionality of critical components such as the CO sensor, temperature sensor, pressure sensor, pulse oximeter, and integrity of the system and remote display will remain off if everything working properly. The RS232 MFD will show no CO on the CO Detector page.

G. Alarm Indicator

WARNING: ANY FAILURE OF THE SYSTEM WILL SHOW A SYSTEM FAILURE MESSAGE ON THE PFD. RETURN THE UNIT TO CO GUARDIAN FOR REPAIR OR REPLACEMENT.

(1) Relevant alert messages will display on the Primary Flight Display (PFD).

NOTE: The alert message will stay on until the CO level goes below 50 parts per million (PPM) by volume of carbon monoxide concentration.

(2) The CO ALERT can be reset on the engine page of the MFD (see “Figure 2”).



MFD Engine Page
Figure 2

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RECORDERS

Vision 1000 Flight Data Monitoring System

In S/N's 3449479 and up the Appareo Systems' Vision 1000 Flight Data Monitoring System may be installed. This system gathers data from global positioning, image capturing, flight attitude acquisition and ambient audio recording to provide high fidelity information regarding aircraft movement through space. Data collected is written in two locations in real time. One location is in a protected back up internal memory and the other a removable SD card.



Vision 1000 Unit
Figure 1

The installation consists of a Vision 1000 camera/recorder unit and a GPS antenna. An instrument panel mounted intercom system connected to passenger/copilot mic and phone jacks with an integral push to transmit button is also installed (see 23-50-00). That system is wired to give the pilot transmit priority over the passengers/copilot.

A. Vision 1000 Unit

See "Figure 1".

The Vision 1000 unit is mounted just aft of the co-pilot's head, to the right of the airplane centerline at F.S. 108.179. See "Figure 2" on page 31302.

(1) Removal

(a) Remove electrical power from the airplane.

(b) Disconnect the GPS antenna coax cable from the coax connector and the harness from the P1 Port.

(c) Remove nut, bolt, and two spacers and remove Vision 1000 unit.

(2) Installation

(a) Position Vision 1000 unit with clamping plate properly engaged and secure unit with bolt, nut, and two spacers.

(b) Connect the GPS antenna coax cable to the coax connector and the harness to the P1 Port.

(c) If the Vision 1000 unit installed is new or a replacement unit, configure it per "Configuration" on page 31304.

NOTE: Also document installation of a new or replacement unit with an appropriate logbook entry.

(d) Conduct Function Test, below.

B. GPS Antenna

The Vision 1000 GPS antenna is mounted to a bracket just above the cowl deck cover and to the right of the windshield center post. See "Figure 2" on page 31302.

(1) Removal

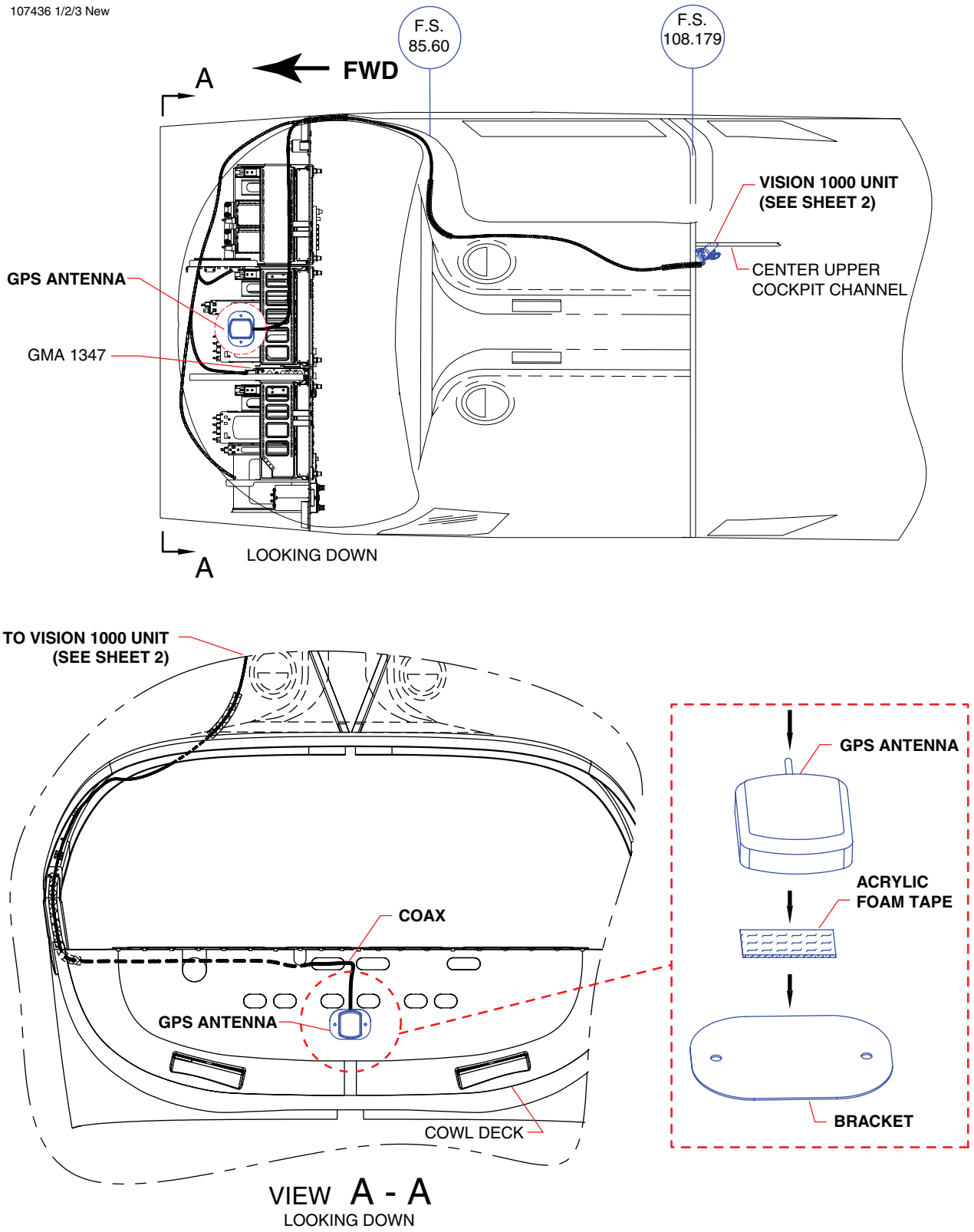
(a) Remove electrical power from the airplane.

(b) Behind the instrument panel and under the cowl deck, disconnect the SMB Connector in the GPS antenna coax cable.

(c) Loosen the jam screw on the antenna mounting bracket sufficient to disengage the mounting bracket hooks from the cowl deck.

(d) Remove the antenna and mounting bracket, taking care to feed the coax cable pigtail out through the gap between the cowl deck and the cowl deck cover.

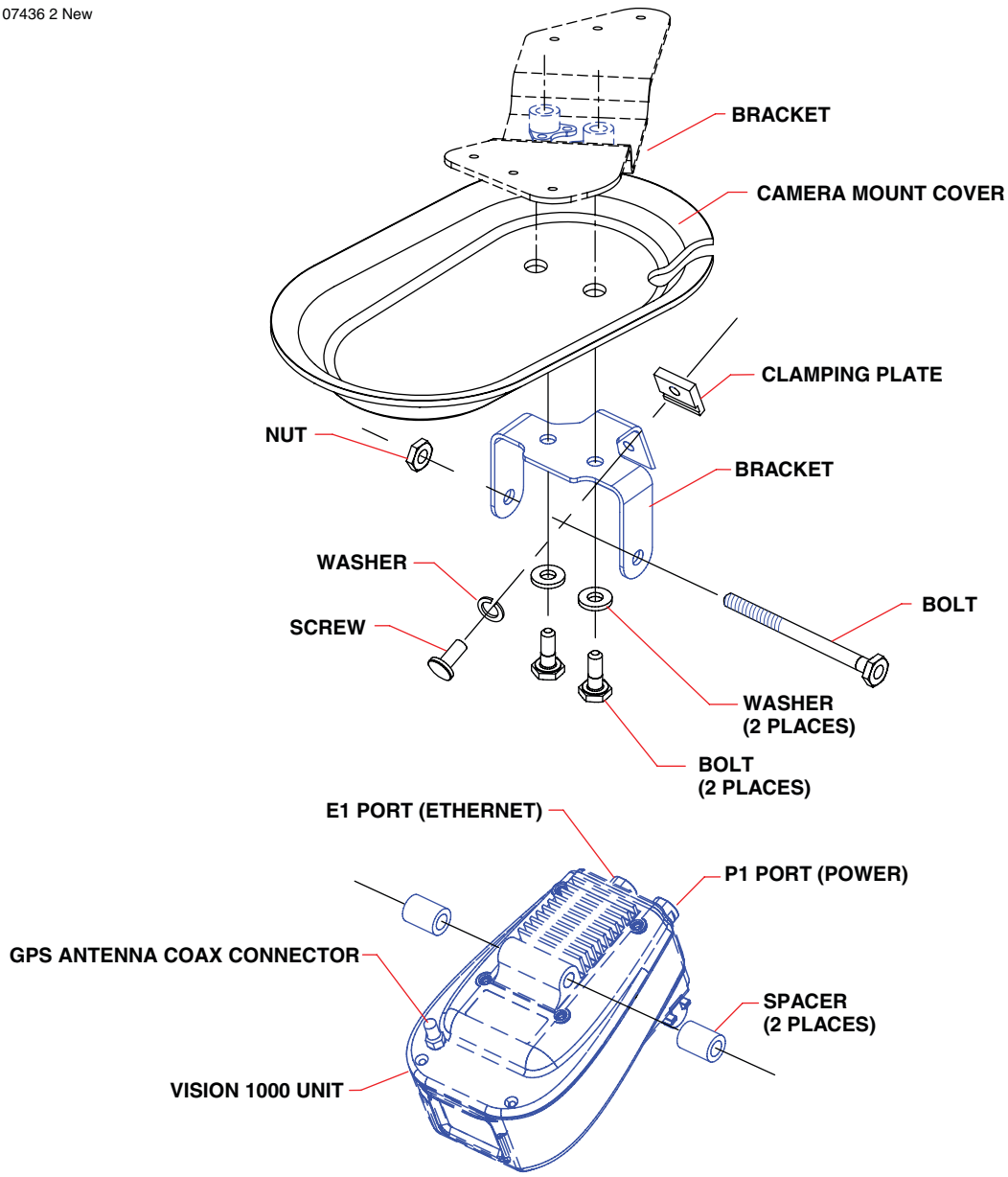
107436 1/2/3 New



Effectivity
3449479 and up

Vision 1000 System Installation (Optional)
Figure 2 (Sheet 1 of 2)

107436 2 New



Vision 1000 System Installation (Optional)
Figure 2 (Sheet 2 of 2)

Effectivity
3449479 and up

(2) Installation

(a) If replacing the antenna:

1) Separate the old antenna from the bracket with a knife.

2) Clean the double-stick foam tape residue from the bracket with an appropriate solvent such as Goo Gone.

3) Install the new antenna to the bracket with 3M VHB Foam Tape No. 5962 sufficient to cover the full footprint of the antenna.

(b) Feed the coax cable pigtail down through the vent hole in the cowl deck and the cowl deck cover.

(c) Engage the mounting bracket hooks over the inboard end of the cowl deck adjacent to the windshield hat assembly and secure with the jam screw.

(d) Behind the instrument panel and under the cowl deck, connect the GPS antenna coax cable at the SMB Connector.

(e) Perform Function Test, below.

C. Function Test

Conduct the functional test in an area where the airplane has an unimpeded view of the sky, so that a proper GPS fix can be established.

(1) Connect power to the aircraft.

(2) Ensure the Avionics Master Switch is ON.

(3) Open the SD access door.

Push SD door latch to the side and swing card door open.

(4) Insert SD card.

(a) Place SD card vertically into the SD slot with the contacts facing forward.

(b) Push SD card into the Vision 1000 unit until it is recessed into the enclosure and release. The card should return a flush position.

(5) Observe LED, blue LED should be on continuously until unit has booted.

(a) Solid Green LED indicates that the system has booted and is functioning.

(b) If a solid Yellow LED is seen, refer to “Chart 1” on page 31305.

(6) Close the card door and latch shut.

(7) Remove power from aircraft.

D. Configuration

When installing a new or replacement Vision 1000 Unit, it must be configured per the following procedure.

(1) Required Equipment

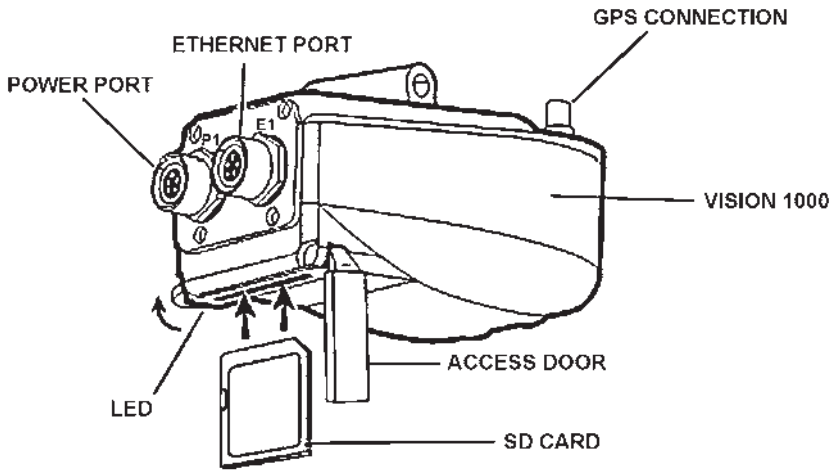
(a) Configuration Tool Software (see “Downloadable Software” on page 31307).

(b) Windows Laptop Computer

(c) Appaero System’s Ethernet Cord, P/N 153560-000004.

CHART 1
VISION 1000 LED STATUS

Item	LED Status	Configuration
1	Red	Contact Appareo Systems.
2	Blue	Booting.
3	Green	Operating.
4	Yellow	- SD card not inserted: insert SD card and verify Green LED. - SD card not formatted correctly: format SD to NTSF, verify Green LED. - GPS lock not received: Allow 15 minutes to clear, if problem persists contact Appareo Systems.
5	NO LED	- Make sure fuse is not blown: replace fuse. - Make sure avionics master switch is on. - Make sure avionics bus circuit breaker is closed. - Check power harness for continuity. - If all checks pass and LED is still not functioning, contact Appareo Systems.



Vision 1000 Unit Details
Figure 3

Effectivity
3449479 and up

(2) Procedure

The Vision 1000 must be powered ON during configuration.

(a) Connect a laptop to the Vision 1000 Unit:

1) Turn ON aircraft power and avionics master.
2) While the Vision 1000 is starting up, a blue LED located behind the SD card door will illuminate. After it has finished its start up, the LED will switch to green.
3) Once the LED has been green for approximately two (2) minutes, you may proceed.
4) Insert the appropriate end of the Ethernet cord into the computer's Ethernet port.

CAUTION: PINS ON THE ETHERNET CORD AND PORT MAY BEND IF THE ETHERNET CORD IS NOT HANDLED PROPERLY. ENSURE THE WHITE DOT OF THE ETHERNET CORD LINES UP WITH THE WHITE DOT ON THE E1 JACK ON THE BACK OF THE VISION 1000 UNIT.

5) Connect the other end of the Ethernet cord to the Vision 1000 E1 port, as shown in "Figure 3" on page 31305.

(b) Log in to the Vision 1000 configuration tool:

NOTE: The Configuration Tool can be obtained as specified under Downloadable Software, below.

1) Open Vision 1000 configuration tool on laptop.
2) Ensure the configuration tool is connected to a Vision 1000 and the Vision 1000 is powered ON (the status LED will be solid green if the Vision 1000 is powered on).
3) Wait until the Vision 1000 status orb is green and has the text "Device Connected" (top left corner).
4) Ensure that "Operator" is selected in the role field.
5) Type the password in the Password field. (Password = h2StAste .) (Password is case sensitive and cannot be altered.)

NOTE: Once logged in, you will have a live streaming image.

(c) Built-in-Test (BIT)

1) Select BIT tab on application screen.
2) Select "Start Built In Test" and ensure all tests pass.
   a) SD card must be inserted.
   b) Aircraft must be located where GPS signal can be received.
3) Overall status (top right of BIT screen) indicates:
   a) all the BITs have passed (displays a green orb with the text "pass"),
   b) a field-serviceable BIT has failed (displays a yellow orb with the text "warning")
   c) one of the non-serviceable BITs have failed (displays a red orb with the text "fail").

NOTE: If either b) or c), contact Appareo Systems.

(d) Configure the IMU Zero Point:

1) Level the airplane per 8-20-00.
2) Select configure tab; with aircraft stationary, enter zero for both roll and pitch and -6 for yaw in the appropriate fields.
3) Once you have entered the roll, pitch, and yaw click "configure".

NOTE: If the configure button is not available (appears grayed out) an IMU Zero Point value is missing.

(e) Enter the tail number:

1) Under the IMU Zero Point configure you will see an attitude indicator, below that you will see an aircraft tail number input.

2) Enter the airplane registration number in the Tail # field and click enter.

NOTE: This information will be used in the creation of file names recorded by the Vision 1000.

(f) You may now log out.

CAUTION: IN ORDER FOR THE CHANGES YOU MAKE WITH THE CONFIGURATION TOOL TO TAKE EFFECT, YOU MUST "POWER CYCLE" THE VISION 1000.

(g) Power cycle the Vision 1000:

1) Turn OFF avionics master switch, located on the aircraft's main circuit breaker panel, to power OFF the Vision 1000.

2) Turn ON avionics master switch to power ON Vision 1000.

3) Observe the Vision 1000's status LED light, located behind the SD card access door. (See "Figure 3" on page 31305.)

a) While the Vision 1000 is starting up, the LED status light will be solid blue for several minutes.

b) Once the Vision 1000 has properly booted up, the LED status light will be solid green. This signifies that the Vision 1000 is ready to record flight data.

E. Downloadable Software

Download the Vision 1000 Configuration Tool and the Vision 1000 Playback Utility software programs.

(1) Navigate to https://fdmportal.appareo.com/login/ in a browser.

(2) Sign in using an existing Appareo account, or click the page link to create a new account.

(3) Click Add Product.

(4) Select the appropriate software program from the product drop-down list (the software programs must be added to your account separately). Enter the appropriate product key from the fields below and click Submit.

(a) Software Product Key:

20420413 - 7EF14C60 - A6675 - 5E1020 - AEE1F

(b) Vision 1000 Configuration Tool Password:

h2StAste (Case Sensitive)

(c) Vision 1000 Playback Utility Software Product Key:

032517E8 - 42C84EAC - 98067 - 920010 - E4581

(5) Click Home (in the Manage Your Account column) and click the red download link for the product. Follow the on-screen instructions to finish the installation process. You may download the installer at any time using these links.

(a) If you do not currently have an Appareo Web account, you must create one before downloading these software program installers from the Appareo website. Creating an Appareo Web account is free and only takes a few minutes

(b) If you are unable to download these software programs from the Appareo website and a CD was not already included in your installation kit, contact Appareo to request an installer CD Vision 1000 Configuration Tool.

(6) Applying future Vision 1000 embedded software versions.

Appareo may release updated versions of the Vision 1000's embedded software in the future. Each embedded software release will be accompanied by a Service Bulletin that will include step-by-step instructions for how to apply the embedded software release to your Vision 1000; however, each company must use a unique key in order to download the release from the Appareo website. This embedded software key is included in the field below. Ensure that you retain this document so that you may apply any embedded software versions of your Vision 1000 in the future.

Vision 1000 Embedded Software Key: NA

CENTRAL WARNING SYSTEMS

A traditional annunciator panel is installed in Seneca IV models and in Seneca V S/N's 3449001–3449458 and 3449460–3449166.

Seneca V S/N's 3449459, 3449467 and up are equipped with Garmin G1000/G1000 NXi and annunciations are displayed as part of the Crew Alerting System (CAS) messages the MFD.

1. Annunciator Panel

See 91-31-50 for electrical schematics.

The annunciator panel provides a visual warning of possible malfunctions, including failure alert and precautionary warnings.

NOTE: Refer to Annunciator Panel figures to determine specific annumciator usage.

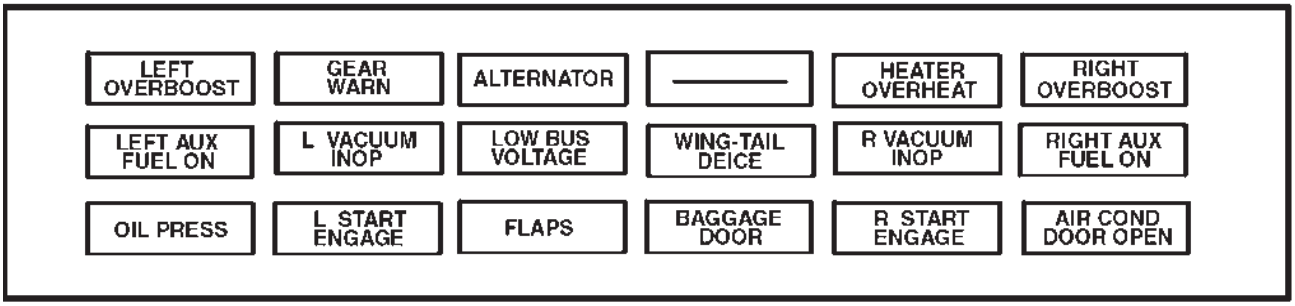
On Seneca IV models, an annunciator panel light cluster is mounted in the top of the instrument panel just above the pilot's altimeter. The PUSH TO TEST switch is located on the right side of the annunciator panel. A DAY - NIGHT toggle switch, located on the left side of the panel, operates the dimmer control box, enabling the pilot to dim the annunciator lights during night flight. The dimmer control box is located under the floor board on the right side of the fuselage near the flap torque tube brackets.

On Seneca V models, the annunciator system consists of two separate clusters mounted in the top center of the instrument panel. The PUSH TO TEST switch is located between the two clusters. The DAY - NIGHT toggle switch is also located between the two clusters, and operates the dimmer control box enabling the pilot to dim the annunciator lights during night flight. The dimmer control box is located under the floor board on the right side of the fuselage near the flap torque tube brackets.

Power is supplied to the annunciator panel through a five amp circuit breaker located on the main electrical bus circuit breaker panel.

A. Troubleshooting

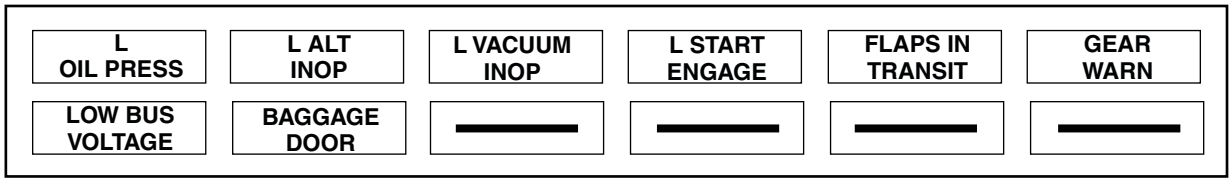
See "Chart 1" on page 31503.



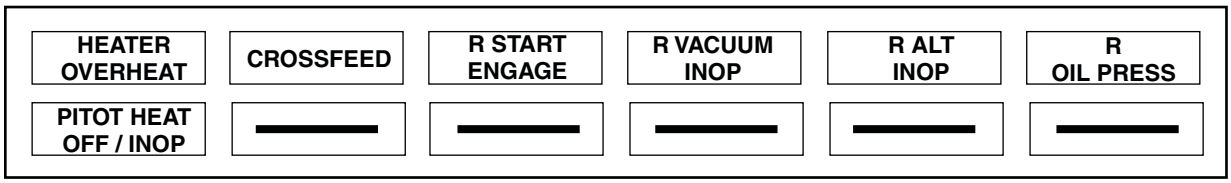
Annunciator Panel
Figure 1 (Sheet 1 of 2)

Effectivity
Seneca IV

89452 AC



LEFT



RIGHT

Effectivity
Seneca V

Annunciator Panel
Figure 1 (Sheet 2 of 2)

CHART 1
ANNUNCIATOR PANEL TROUBLESHOOTING

Trouble	Cause	Remedy
"Ground Applied" warning lights fail to operate.	ANNUN Circuit breaker disengaged (OUT).	Reset (push IN) breaker on circuit breaker panel.
All lights fail to operate.	No current.	Check all wire segments and connections at connectors on annunciator light assembly.
All the warning lights fail to extinguish after engine is running.	Test switch shorted.	Check terminals and wires for short. Replace switch if required.
Oil or gyro air warning light fails to extinguish.	Sensor activates at too high a setting.	Replace.
	Sensor terminals bridged.	Remove material between terminals.
	Defective sensor.	Replace.
Oil or gyro air warning	Lamp burned out. light fails to operate.	Replace.
	Sensor activates at too low a setting.	Replace.
	Defective Sensor	Replace.
Overboost warning light fails to extinguish. (Seneca IV only)	Manifold pressure gauge sensors set too low.	Check sensor activation. Sensors should activate at 39.5 to 40 inches of mercury.
Overboost warning light fails to activate. (Seneca IV only)	Lamp burned out.	Replace.
	Defective sensor switches.	Replace.
Alternator warning light fails to operate.	Lamp burned out.	Replace.
Alternator warning light fails to extinguish.	ALTNR FIELD Circuit breaker disengaged (OUT)	Reset (push IN) breaker on circuit breaker panel.
	Defective alternator output circuit.	Check and repair.
Test switch fails to activate warning lights.	Bad switch or connections.	Check wires and replace switch if necessary.

CHART 2
SENECA IV ANNUNCIATOR PANEL FUNCTION DESCRIPTIONS

Nomenclature	Color	Cause of Illumination
L or R OVERBOOST	AMBER	Illuminates when engines manifold pressure exceeds 39.5 to 40 inches of mercury.
GEAR WARN	RED	Illuminates when gear is in neither the full up nor the full down position.
ALTERNATOR	RED	Illuminates when either alternator fails or is selected “OFF.”
L or R AUX FUEL ON	AMBER	Illuminates when “HI” auxiliary fuel pump is on.
L or R VACUUM INOP	AMBER	Illuminates when either vacuum switch is activated. The vacuum switches are attached to regulators, set to close at 4 ± .25 In Hg. vacuum.
OIL PRESSURE	AMBER	Illuminates when engine’s oil pressure has decreased to 15 psi. or less.
FLAPS	AMBER	Illuminates when flap motor is in operation and flaps are in transit.
BAGGAGE DOOR	AMBER	Illuminates when nose baggage door is open, sensing latch pin position.
L or R START ENGAGE	RED	Illuminates when either left or right engine starter is activated.
AIR COND DOOR OPEN	AMBER	Illuminates when A/C control switch is “ON” and fan switch is in an operating position, indicating proper air conditioner condenser door activation.
LOW BUS VOLTAGE	RED	Illuminates when main bus voltage drops to battery voltage.
WING-TAIL DEICE	GREEN	Illuminates when wing or tail deice boots inflate to 8.0 psi.
HEATER OVERHEAT	RED	Illuminates when combustion heater overheats.
GEAR WARN	RED	Illuminates when gear is in neither the full up nor the full down position. Illuminates when gear selector switch is in up position while airplane is on ground with weight on wheels. Illuminates when power is reduced below approx. 14 inches of manifold pressure with landing gear up.

CHART 3
SENECA V ANNUNCIATOR PANEL FUNCTION DESCRIPTIONS

Nomenclature	Color	Cause of Illumination
GEAR WARN	RED	Illuminates when gear is in neither the full up nor the full down position. Illuminates when gear selector switch is in up position while airplane is on ground with weight on wheels. Illuminates when power is reduced below approx. 14 inches of manifold pressure with landing gear up.
L or R ALT INOP	RED	Illuminates when either alternator fails or is selected “OFF.”
L or R VACUUM INOP	AMBER	Illuminates when either vacuum switch is activated. The vacuum switches are attached to regulators, set to close at 4 ± .25 In Hg. vacuum.
LOW BUS VOLTAGE	RED	Illuminates when the electrical system drops from bus voltage to battery voltage. Fuse overload protection is provided to voltage monitor.
L or R OIL PRESSURE	AMBER	Illuminates when engine’s oil pressure has decreased to 15 psi. or less.
FLAPS IN TRANSIT	AMBER	Illuminates when flap motor is in operation and flaps are in transit.
BAGGAGE DOOR	AMBER	Illuminates when nose baggage door is open, sensing latch pin position.
L or R START ENGAGE	RED	Illuminates when either left or right engine starter is activated.
AIR COND DOOR OPEN	AMBER	Illuminates when A/C control switch is “ON” and fan switch is in an operating position, indicating proper air conditioner condenser door activation.
HEATER OVERHEAT	RED	Illuminates when combustion heater overheats.
CROSSFEED	AMBER	Illuminates when either fuel selector is in X-feed position.
PITOT HEAT OFF/INOP	AMBER	Illuminates when pitot heat fails or is selected off.

B. Light Tests

(1) Seneca IV

The annunciator lights may be tested as follows:

NOTE: The sequence of the test may be varied at the option of the mechanic.

(a) Press the annunciator test button to ensure that the annunciators illuminate.

(b) Start the right engine and observe that the right vacuum inop light extinguishes as the engine starts. Operate the engine at approximately 700-1000 rpm. Note that oil pressure, air pressure or vacuum, and alternator output are normal.

(c) Start the left engine and observe that the left vacuum inop light extinguishes as the engine starts. See that the oil pressure, air pressure or vacuum, and alternator output is normal.

(d) Idle one engine at a time while observing it's oil pressure gauge. The oil annunciator light should illuminate at 15 psi.

NOTE: The mixture control can be moved to cut-off, then to rich in order to get lower than normal idle speeds.

(e) Run both engines at approximately 900 rpm. Sequentially place the alternator switches in the off position. Check that either or both alternator switches in the off position cause the alt annunciator to illuminate.

(f) After ensuring that the propellers and the propeller blast will do no damage, run up the engines one at a time to check that each overboost annunciator comes on at 39.5 ± .5 inches of mercury manifold pressure.

(g) If the optional air conditioning system is installed, check that the air cond door open light illuminates when the air conditioner is on, the fan switch is in the high or low position, and the air conditioning door is open.

(h) Shut down the right engine and check that the right vacuum inop light illuminates just as the engine slows to approximately 300 rpm. Check that the other annunciator lights are on.

(i) Repeat step (h) for the left engine, ensuring that the left vacuum inop light illuminates.

(j) Locate the heater terminal strip on the upper forward side of the heater assembly. Attach a test jumper wire from terminal 1 (the most forward terminal) to terminal 4. Turn the aircraft's battery switch on, momentarily place the heater switch to the heat position and observe that the heater overheat annunciator illuminates. Return the battery and heater switches to the off position and remove the test jumper wire.

(k) Refer to 32-30-00, for the functional test for the gear unsafe light.

(2) Seneca V

(PIR-PPS60057-4, Rev. B.)

The annunciator lights may be tested as follows:

NOTE: The sequence of the test may be varied at the option of the mechanic.

(a) With the battery switch on, press the annunciator test button to ensure that all annunciators illuminate. Locate the Day/Night switch and activate the switch to both Day and Night positions with the annunciator with the annunciator test button pressed. Check that all annunciators illuminate bright for Day and dim for Night.

(b) Verify that the baggage door lamp illuminates when the baggage door is unlatched and extinguishes when the baggage door is latched.

(c) With fuel pumps OFF, move the left engine fuel control lever to the X feed position and check that the crossfeed annunciator illuminates. Move the left engine fuel control lever to the ON position. The crossfeed annunciator will extinguish. Move the right engine fuel control lever to the X feed position and check that the crossfeed annunciator illuminates. Move the right engine fuel control lever to the ON position. The crossfeed annunciator will extinguish.

(d) Make sure flaps are cleared and select a new position of the flap handle. The flaps annunciator light will illuminate. When the flaps stop moving, the annunciator will extinguish.

(e) Start the left engine. Check that L START ENGAGE annunciator is illuminated when start switch is depressed, and extinguishes when start switch is released. Observe the L OIL PRESSURE annunciator. The annunciator light should extinguish when oil pressure is above 15 psi and illuminate when oil pressure is 15 psi or less.

(f) Repeat step (e) checks with the right engine.

(g) With both engines running at approximately 900 rpm, turn the left alternator switch off. The L ALT INOP annunciator will illuminate. Turn the left alternator on, turn the right alternator off. The R ALT INOP annunciator will illuminate. Check that low bus voltage annunciator extinguishes when alternators are on.

(h) If the optional air conditioning system is installed, check that the A/C door open light illuminates when the air conditioner is on, the fan switch is in the HIGH or LOW position, and the air conditioning door is open. Check that the lamp extinguishes when the door is fully closed and the air conditioning switch is in the OFF position.

(i) Momentarily turn pitot heat on. Pitot heat OFF/INOP annunciator will extinguish. Check that pitot heat OFF/INOP annunciator is illuminated when pitot heat switch is off.

(j) Shut down the right engine and check that the R VAC INOP annunciator illuminates as the engine slows to approximately 300 rpm. Repeat for left engine. (Vacuum annunciator switches are set at 4 ± .25 inches of mercury.)

(k) Locate the heater terminal strip on the upper forward side of the heater assembly. Attach a test jumper wire from terminal 1 (the most forward terminal) to terminal 4. Turn the aircraft's battery switch ON, momentarily place the heater switch to the heat position and observe that the heater overheat annunciator illuminates. Return the battery and heater switches to the off position and remove the test jumper wire.

(l) Refer to 32-30-00 for the functional test for the gear unsafe light.

C. Rigging Fuel Crossfeed Annunciator (Seneca V only)

(1) Set the fuel control levers to "OFF." Move the fuel control levers aft to a point where any further movement aft would result in the levers falling into the crossfeed detent position.

(2) Adjust the fuel selector valve control cable to obtain .000 to .020 inch clearance between the actuator arm on the fuel selector valve and crossfeed "ON" stop on the fuel selector valve.

(3) Adjust the annunciator micro-switches to obtain illumination of the crossfeed annunciator light with the valve and levers in the above position.

(4) Re-inspect and adjust, as required, that the fuel control levers in the fuel "OFF" and fuel "ON" positions are still coordinated with the fuel selector valve.

D. Lamp Replacement

See 33-10-00.

E. Wiring

See Chapter 33 and the schematics presented in 91-31-50.

2. Annunciation in Garmin G1000 / G1000 NXi Equipped Airplanes

NOTE: The Garmin G1000 / G1000 NXi Pilot's Guides for the Piper PA-34-220T Seneca V contain detailed descriptions of the annunciator system (CAS and Non-CAS) and all warnings, cautions and advisories. See Vendor Publications under Supplementary Publications in the Introduction.

Annunciation is integrated into G1000 / G1000 NXi systems. The Crew Alerting System (CAS) consists of Master Warning and Master Caution Indicators operating in conjunction with CAS text messages. CAS text messages appear in the lower left area of the MFD during normal operations and in the right side area of the PFD's during reversionary mode operation. The Master Warning and Master Caution Indicators are illuminated push-button switches centered above the pilot's and copilot's PFDs in G1000 equipped airplanes. In G1000 NXi equipped airplanes the Master Warning and Master Caution indicators / switches have been integrated into the Gamin displays. The severity of CAS messages are categorized as Warning, Caution and Advisory as follows:

A. Warning (**red on black**) – Immediate crew awareness and action require.d; Master Warning triggered Warnings are accompanied by a continuous aural chime.

B. Caution (**amber on black**) – Immediate crew awareness and possible future corrective action required; Master Caution triggered Cautions are usually accompanied by a single aural chime of approximately 1.5 seconds in duration.

C. Advisory (**white**) – Advisory information; and are not accompanied by aural chimes. no immediate action required.

NOTE: The examples shown in "Chart 4" on page 31509 are representative. See the appropriate Garmin Pilot's Guide detailed descriptions for individual airplanes.

3. Aural Alerts

Aural alerts are provided to alert the crew and call for their attention:

- Master Advisory (single chime) / Master Caution (double chime) / Master Warning (triple chime).
- Stall Warning (steady tone).
- Gear Warning - In flight when the throttle is reduced to the point at which manifold pressure is approximately 14 inches of mercury or below and the landing gear are not in the DOWN position (beeping tone).
- Gear Warning - In flight when the flaps are extended more than 12 degrees and the landing gear are not in the DOWN position (beeping tone).
- Gear Warning - On the ground when the landing gear selector is in the UP position (beeping tone).
- When in a Vne overspeed condition ("Airspeed, Airspeed").
- Terrain cautions/warnings.
- Traffic Alerts.

CHART 4 (Sheet 1 of 2)
CREW ALERTING SYSTEM

— WARNINGS - RED - TRIPLE AURAL CHIME —	
Message	**Cause**
AC DOOR OPEN	Air conditioning condenser door is open.
BAGGAGE DOOR	Nose baggage door is not properly closed and latched.
CHECK GEAR	Landing gear are not down and locked.
GEAR SYS	A failure of a component of the landing gear system.
HTR OVRHEAT	Heater overheat as sensed by heater unit
L ALTR FAIL	Left ALTR switch selected ON and the alternator has failed.
R ALTR FAIL	Right ALTR switch selected ON and the alternator has failed.
L ENG CHT	Left engine CHT in warning range.
R ENG CHT	Right engine CHT in warning range.
L FUEL QTY	Left tank fuel quantity is less than or equal to 5.0 gallons while the engine is running.
R FUEL QTY	Right tank fuel quantity is less than or equal to 5.0 gallons while the engine is running.
L START ENGD	The L starter contactor is engaged for greater than 30 seconds during engine start or anytime while the engine is running.
R START ENGD	The R starter contactor is engaged for greater than 30 seconds during engine start or anytime while the engine is running.
PROP HT FAIL	A propeller heat fault has developed or the current is less than 14.0 amps.

CHART 4 (Sheet 2 of 2)
CREW ALERTING SYSTEM

— ADVISORIES - WHITE - SINGLE AURAL CHIME —

Message	Cause
AC DOOR OPEN	Air conditioning condenser door is open.
BAGGAGE DOOR	Nose baggage door is not properly closed and latched.
EMERG BATT ON	Emergency power in use.
FUEL IMBAL	Fuel quantity imbalance greater than 10 gallons while the engine is OFF.
FUEL X-FEED ON	Fuel cross-feed valve selected open.
MAINT MODE ON	System is in maintenance mode for ground tests.
SFC DEICE ON	Surface de-ice system is selected ON and power is being applied to the SURF DE-ICE timer.
L VAC FAIL	Left Vacuum pressure is less than approximately 2.0 in. of Hg.
R VAC FAIL	Right Vacuum pressure is less than approximately 2.0 in. of Hg.

— CAUTIONS - AMBER - DOUBLE AURAL CHIME —

Message	Cause
CHECK GEAR	Landing gear are not down and locked.
GEAR SYS	A failure of a component of the landing gear system.
HYDR PUMP ON	Hydraulic pump has been running for more than 16 seconds.
L FUEL QTY	Left tank fuel quantity is less than or equal to 5.0 gallons while the engine is OFF.
R FUEL QTY	Right tank fuel quantity is less than or equal to 5.0 gallons while the engine is OFF.
OXYGEN LOW	Oxygen pressure in caution range.
PITOT HT FAIL	Pitot heat has failed.
PITOT HT OFF	Pitot heat is turned OFF. No aural chime accompanies this action.
SFC DEICE FAIL	Surface de-ice system has failed to reach appropriate pressure.
VACUUM PSI	Vacuum system pressure is less than 4.5 in. HG.

CHAPTER

32

LANDING GEAR

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CHAPTER 32

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GENERAL

This chapter contains instructions for overhauling, inspecting and adjusting the various components of the PA-34-220T landing gear and brake system, with the exception of the hydraulic pump and lines - see Chapter 29 for those components. Also included are adjustments for the electrical limit, safety and warning switches.

1. Description

The PA-34-220T airplane is equipped with a retractable tricycle air-oil strut type landing gear, hydraulically raised or extended by an electrically powered reversible pump. A selector switch in the instrument panel to the left of the control quadrant is used to select gear UP or DOWN position.

When the gear is down and locked, gear positions are indicated by three green lights, located to the left of the selector switch. A red light, incorporated in the annunciator panel at the top of the instrument panel, illuminates when the gear is unsafe. Activation of all three down lock switches will shut the hydraulic pump off. The green lights, along with the red gear unsafe annunciator light will be dimmed by placing the DAY – NIGHT toggle switch, located to left of the annunciator lights, in the NIGHT position.

As manifold pressure drops below approximately 14 inches of mercury, and if the landing gear has not been extended, a throttle switch located in the quadrant will actuate a warning horn indicating to the pilot the landing gear is still up. The warning horn will continue to operate until the landing gear is down and locked, at which time three green lights on the instrument panel will energize.

The landing gear is extended and retracted by means of the gear selector knob. In the event of hydraulic or electrical failure, the gear can be extended by pulling the free-fall valve, thus permitting the gear to fall free. Once the gear are down a spring maintains pressure on the truss assembly in the locked position until released by hydraulic pressure.

While the airplane is sitting on the ground, should the gear selector knob be placed in the UP position with the BATT switch in the ON position, a safety switch (squat switch) located on the left main gear should prevent the hydraulic pump from actuating. When the plane leaves the ground, the safety switch will actuate as the oleo extends, permitting the hydraulic pump to raise the landing gear. In the event the airplane is placed on jacks and raised to the extent that the oleo extends in excess of 8 inches, the safety switch will actuate the hydraulic pump, thus raising the landing gear if the landing gear selector knob is placed in the UP position and the BATT switch is selected ON.

The nose gear is steerable by the use of the rudder pedals. As the gear retracts, the steering linkage becomes separated from the gear so that rudder pedal action with gear retracted is not impeded by the nose gear operation. A gear centering spring mechanism is incorporated in the nose gear steering mechanism.

The two main wheels are equipped with self-adjusting single disc hydraulic brake assemblies which are actuated by individual toe brake cylinders mounted on the rudder pedals. The cylinders are supplied hydraulic fluid from a reservoir located on the forward side of the cabin main bulkhead. The parking brake is engaged by depressing the toe brake pedals and pulling out the parking brake knob located on the lower left instrument panel. The parking brake is released by depressing the toe brake pedals and pushing in on the parking brake knob.

2. Troubleshooting

CAUTION: WHEN NECESSARY TO RAISE OR LOWER EITHER THE NOSE GEAR OR THE MAIN GEAR MANUALLY, THE FREE-FALL VALVE KNOB SHOULD BE PULLED FULL OUT TO PREVENT A BUILDUP OF UNNECESSARY PRESSURE ON THE ACTUATING CYLINDERS AND CONNECTING HYDRAULIC LINES. FAILURE TO COMPLY WITH THESE INSTRUCTIONS COULD RESULT IN THE BUILDUP OF SUFFICIENT PRESSURE TO UNLOCK THE DOWN LOCK MECHANISM ALLOWING THE GEAR TO COLLAPSE WHEN THE WING JACKS ARE REMOVED.

Mechanical and electrical troubles peculiar to the landing gear system are listed in “Chart 1” on page 32003. When troubleshooting, first eliminate hydraulic malfunctions as listed in 29-00-00, Chart 1. Then proceed to switch malfunctions, and last to the mechanical operation of the gear itself, both of which are included in this section.

CHART 1 (Sheet 1 of 4)
TROUBLESHOOTING LANDING GEAR

Trouble	Cause	Remedy
Green gear down lights fail to go out with gear in transit or retracted.	Gear down limit switch failed.	Replace switch.
One or more gear down lights out when gear is down and locked.	Burned out bulb.	Replace bulb.
	Improperly adjusted gear down light switch.	Adjust switch(es) as necessary.
	Open wire in gear light circuit.	Check annunciator light wiring.
Gear down lights do not dim when DAY/NIGHT annunciator switch is in NIGHT position.	Faulty DAY/NIGHT switch	Replace switch.
Gear warning light and horn fail to operate when throttle is near closed and landing gear is retracted.	Landing gear control circuit breaker open.	Reset circuit breaker and determine cause for open circuit breaker.
	Micro switch at throttle out of adjustment.	Adjust micro switch.
	Micro switch failed.	Replace switch.
	Warning horn and light circuit wire broken.	Check wiring.
Red gear unsafe light remains on with gear retracted and throttles advanced.	One or more of the landing gears not fully retracted. One or more gear up switches out of adjustment.	Check gears for full retraction. Adjust up switch(es) as necessary.
Landing gear doors fail to close completely.	Landing Gear not retracting completely.	Check adjustment of landing gear.
	Door retraction rods out of adjustment.	Check adjustment door retraction rods.

CHART 1 (Sheet 2 of 4)
TROUBLESHOOTING LANDING GEAR

Trouble	Cause	Remedy
Nose landing gear fails to fully lock down during emergency gear extension (i.e. - free fall).	Binding or distortion	Check that the trunnion, drag link, and all parts of the down lock linkage move freely on their pivots and no binding or distortion is present at the attachment points. Lubricate on condition.
	Damaged down lock link.	Inspect down lock link. Ensure that the end fittings are not bent or cracked and that the spring retaining pin has not elongated, distorted, or crushed the upper edge of the .191 dia hole in the link sleeve. Check the attachment holes for distortion or enlargement. Replace on condition.
	Retraction link retention spring worn out.	Inspect the retraction link retention spring for end wear and general condition. Using a suitable tension scale, check that the spring tension is: 20.5 lbs (± 2.0 lbs) when the spring is extended from its free length of 2.61 to 3.44 inches. 36.0 lbs (± 3.6 lbs) when the spring is extended from its free length of 2.61 to 4.31 inches. Replace on condition.
	Hydraulic actuator damaged.	Inspect the hydraulic actuator. Check that the rod is straight, free of distortion and that the bearings in the end fittings are not seized. Check that the piston rod can be moved smoothly in and out of the cylinder with hand pressure, and that no binding occurs. Replace on condition.

NOTE: Reinstall good components, replace bad components, and rig per 32-20-00, Nose Landing Gear, Installation and Rigging.

CHART 1 (Sheet 3 of 4)
TROUBLESHOOTING LANDING GEAR

Trouble	Cause	Remedy
Nose landing gear shimmies during fast taxi, takeoff, or landing.	Internal wear in centering springs.	Replace shimmy dampener.
	Center springs or bracket loose at mounting.	Replace necessary parts.
	Tire out of balance.	Check balance and replace tire if necessary.
	Worn or loose wheel bearings.	Replace and/or adjust wheel bearings.
	Worn torque link bolts and/or bushings.	Replace bolts and/or bushings.
Excessive or uneven wear on nose tire.	Incorrect operating pressure.	Inflate tire to correct pressure.
	Wear resulting from shimmy.	Inflate tire to correct pressure.
Nose gear fails to steer properly.	Oleo cylinder binding in strut housing.	Lubricate strut housing (See Lubrication Chart.)
		Chart and/or strut housing bushings damaged.
	One brake dragging.	Determine cause and correct.
	Steering arm roller sheared at top of strut.	Replace defective roller.
	Steering bellcrank loose on attachment plate.	Readjust and tighten.
	Steering bellcrank bearing and/or bolt worn.	Replace bearing and/or bolt.
	Centering springs galling or binding.	Replace.
Nose gear fails to straighten when landing gear retracts.	Steering arm roller sheared at top of strut.	Replace defective roller.
	Incorrect rigging of nose gear steering.	Check nose gear steering adjustment.
	Centering guide roller sheared.	Replace roller.
	Damaged guide.	Replace guide.

CHART 1 (Sheet 4 of 4)
TROUBLESHOOTING LANDING GEAR

Trouble	Cause	Remedy
Main landing gear shimmies during fast taxi, takeoff, or landing.	Tire out of balance.	Check balance and replace tire if necessary.
	Worn or loose wheel bearings.	Replace and/or adjust wheel bearings.
	Worn torque link bolts and/or bushings.	Replace bolts and/or bushings.
Excessive or uneven wear on main tires.	Incorrect operating pressure.	Inflate tire to correct pressure.
	Wheel out of alignment (toe in or out).	Check wheel alignment.
	Lower side brace link out of adjustment, allowing gear to slant in or out.	Check gear adjustment.
Strut bottoms on normal landing or taxiing on rough ground.	Insufficient air and/or fluid in strut.	Service strut with air and/or fluid.
	Defective internal parts in strut.	Replace defective parts.

MAIN GEAR AND DOORS

1. Main Gear Oleo

A. Disassembly

See “Figure 1” on page 32102.

The main gear oleo assembly may be removed and disassembled from the oleo housing with the gear removed for or installed in the airplane. The following instructions assume the gear is still installed.

(1) See 7-10-00 and jack up the airplane.

(2) Place a drip pan or other container under the affected gear to catch spillage.

(3) Depress air valve core pin until strut pressure has diminished. After removing the plug, insert a thin hose and siphon as much hydraulic fluid from the strut as possible.

(4) Disconnect the hydraulic brake line from the brake assembly and the clamps from the fork and torque link.

(5) Support the strut assembly and remove the cotter pin, nut, washer(s), and bolt connecting the upper and lower torque links and separate the links. Note the number and thickness of spacer washers between the two links and make sure to replace them upon reinstallation.

(6) Compress the piston tube slightly, to take off any load on the bearing assembly and make sure it is held in its position.

(7) In the bottom of the housing, release the snap ring from its slot in the housing.

(8) Carefully pull the piston tube and bearing assembly from the bottom of the housing.

(9) The bearing and other related assemblies can be removed from the strut position tube by removing the retaining snap ring and sliding the assemblies off the tube.

(10) The orifice tube is secured to the strut housing by a locknut where the tube extends through the top of the housing. Remove the locknut and washer.

(11) Draw the orifice tube with the teflon retainer ring out of the strut housing.

(12) As necessary, the orifice plate can be removed from the bottom of the orifice tube by releasing the snap ring holding the plate in position.

NOTE: It is not recommended that the piston tube (strut) and fork be separated due to shrink fit tolerance built in when manufactured.

B. Cleaning, Inspection, and Repair

(1) Clean all parts with a suitable dry type cleaning solvent.

(2) Inspect landing gear oleo assembly components for the following:

(a) Bearings and bushings for excess wear, corrosion, scratches and overall damage.

(b) Retaining pins for wear and damage.

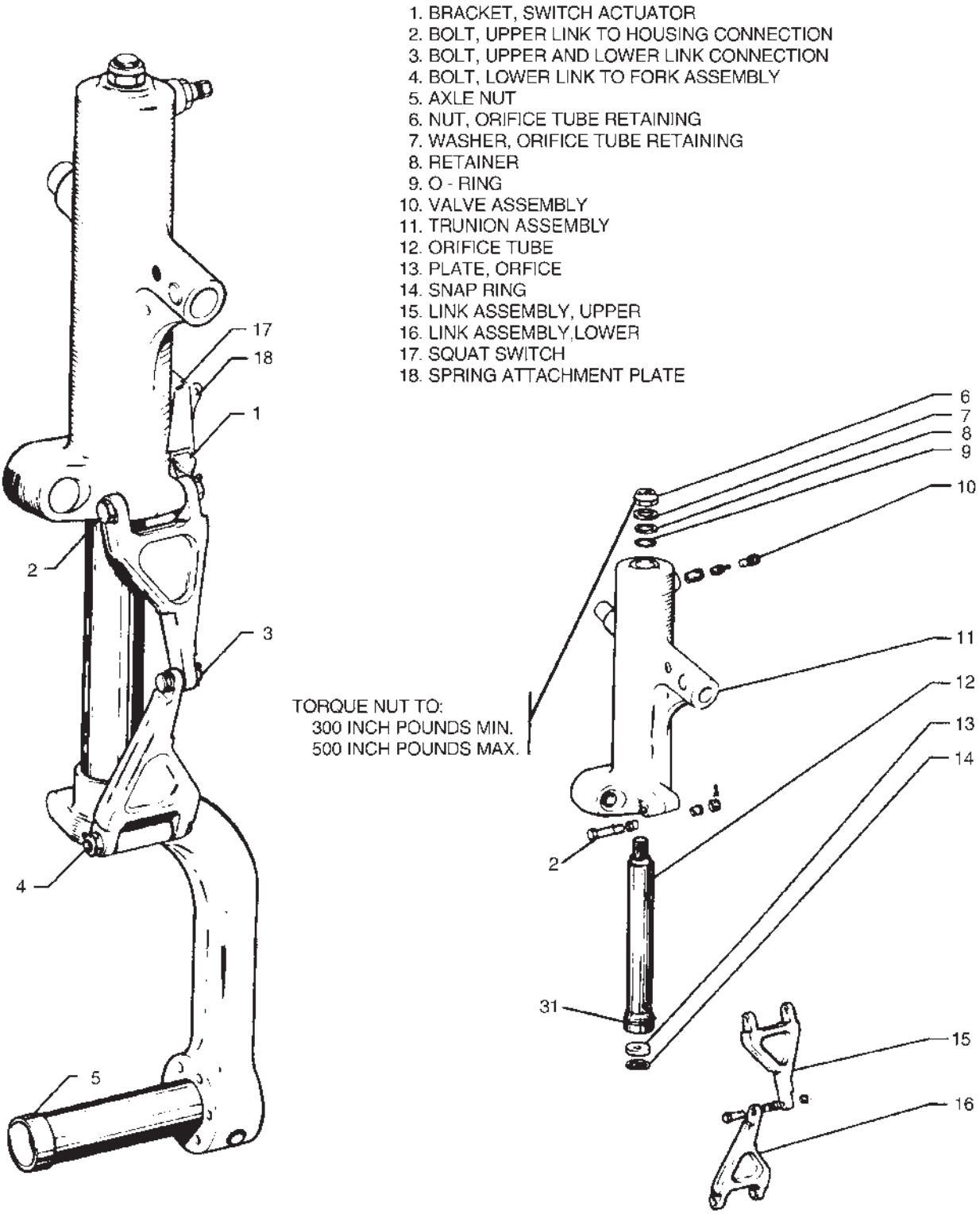
(c) Lock rings for cracks, burrs, etc.

(d) Cylinder and orifice tube for corrosion, scratches, nicks and excess wear.

(e) Orifice plate for hole restriction.

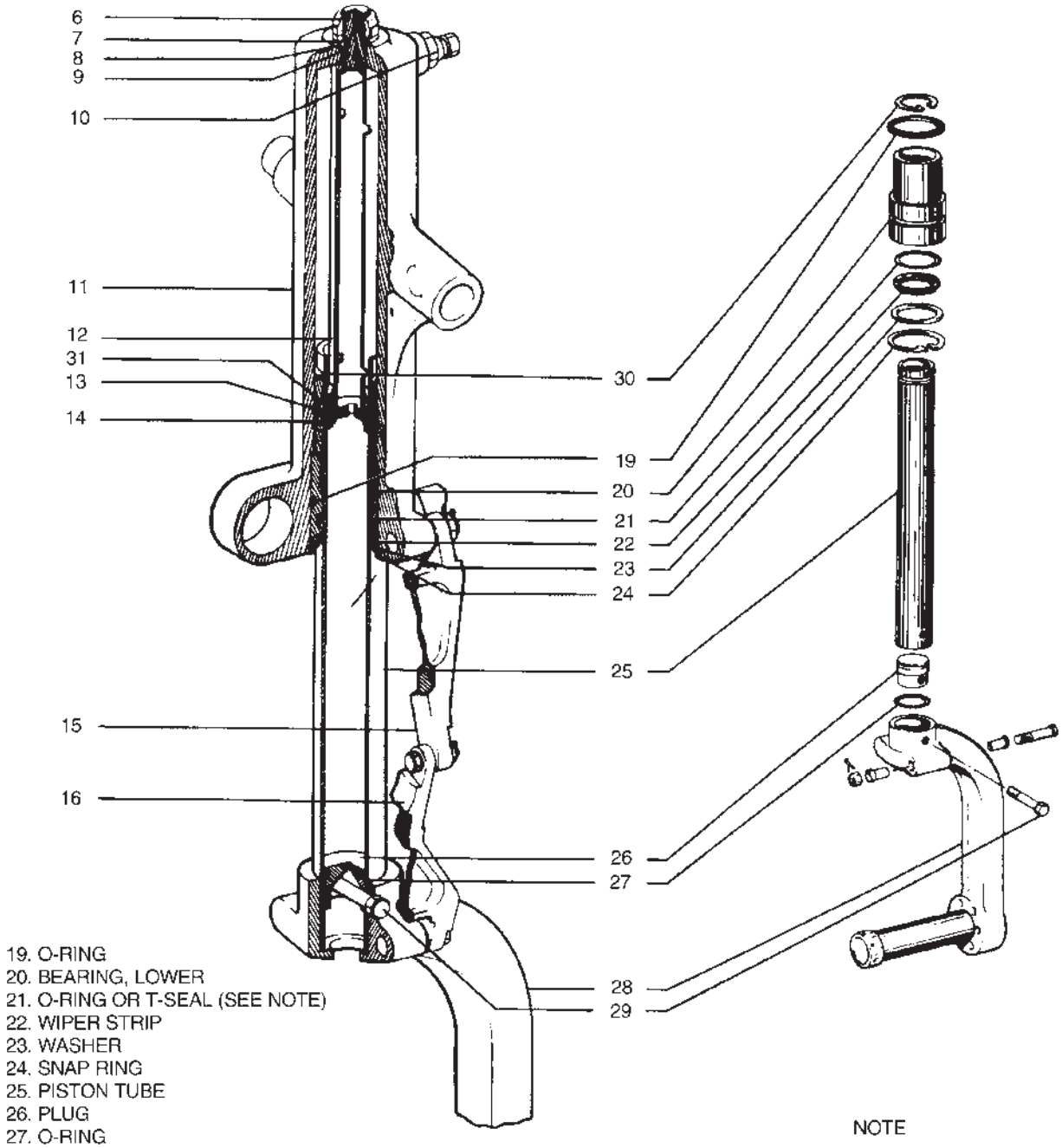
(f) Fork tube for corrosion, scratches, nicks, dents and misalignment.

(g) Air valve general condition.



Main Gear Oleo Strut Assembly
Figure 1 (Sheet 1 of 2)

2025



19. O-RING
20. BEARING, LOWER
21. O-RING OR T-SEAL (SEE NOTE)
22. WIPER STRIP
23. WASHER
24. SNAP RING
25. PISTON TUBE
26. PLUG
27. O-RING
28. FORK ASSEMBLY
29. BOLT ASSEMBLY
30. PISTON TUBE RETAINING RING
31. LUBERIZED BUTT JOINT RING, ORIFICE TUBE

Main Gear Oleo Strut Assembly
Figure 1 (Sheet 2 of 2)

(3) Repair of the oleo is limited to smoothing out minor scratches, nicks and dents and replacement of parts.

NOTE: Check all impregnated bearing surfaces, such as Garlock DU bearings (06U08 in the lower truss link assembly), for wear. Natural wear will show more and more bronze. Replace any bearing when at least 70% of the bearing surface is bronze and the bronze starts to smear. Do not apply grease. Use oil for aircraft kept in a highly corrosive environment. When oil is used it must be followed by scheduled lubrication.

C. Assembly

See “Figure 1” on page 32102.

NOTE: Use Parker Super O-Lube, P/N 197-507, to lubricate all O-Rings.

(1) Make sure all parts are cleaned and inspected as described in the last paragraph.

(2) If the orifice tube has been removed and maintained, proceed as follows:

(a) Make sure the old butt joint ring is replaced and its annular seat is clean and undamaged.

(b) Ensure there is no damage to the orifice plate seat in the orifice tube.

(c) Install orifice plate in the end of the orifice tube and secure with its snap ring.

(3) The orifice tube can be reinstalled in the housing as follows:

(a) Insert tube into strut housing and being careful so as not to damage the threads and mating surfaces push the stud end through the top of the housing.

(b) Install a new O-ring (see Parts Catalog) and reinstall the teflon retaining ring in the top of the housing. Make sure they are pushed one at a time as far into the recess as they will go.

(c) Install the washer over the stud and install a new locknut with a torque of 300 to 500 inch-pounds.

(d) Torque the orifice tube locknut 300 to 500 inch-pounds.

(4) The gear strut and its related assemblies are installed as follows:

(a) The bearing and its retaining hardware must first be placed in order over the piston tube. Place the bearing snap ring, retaining washer, and wiper strip in that order over the piston tube.

(b) Make sure the bearing annular internal and external O-ring seats are undamaged and clean. Replace the bearing if necessary.

(c) Install the new internal O-ring or T-seal and external O-rings. (See Parts Catalog.)

(d) Wet the piston tube surface with MIL-H-5606 hydraulic fluid.

(e) Wet the interior of the bearing with hydraulic fluid and slide the assembly over the piston tube with the wiper strip end towards the fork. Use care not to damage the internal O-ring.

(f) Install the retainer ring on the end of the piston tube.

(g) Align the piston tube opening with the end of the housing and the orifice tube. Carefully guide the piston tube strut halfway into the housing.

(h) Slide the bearing up the piston tube into the housing, and with it held in position, install the wiper strip, washer-retainer and snap ring.

(5) Ascertain that bushings are installed in the upper and lower torque link connecting holes and then connect the upper and lower torque links by installing the bolt, washer(s), and nut. Be sure to use the same number and type of spacer washers noted during disassembly (to maintain wheel alignment.) Tighten nut enough to permit no side play in the links, yet still allow bolt rotation. Secure nut with cotter pin.

(6) If the left gear has been maintained and the squat switch assembly has been disassembled, proceed as follows:
   (a) Install the switch actuator bracket on the upper link-to housing bolt.
   (b) Install the washers and the nut on the aforementioned bolt.
   (c) Install the squat switch bracket and spring attachment plate on the strut housing if removed.

(7) Reconnect the brake hose clamps and the fitting to the actuating cylinder. Bleed as described in section on “Brakes.”

(8) Lubricate gear assembly as described in 12-20-00.

(9) Service oleo strut with fluid and air as described in 12-10-00.

(10) Ensure the gear is down and locked and check alignment of gear as described in this chapter and check for proper gear retraction and proper activation of squat switch.

(11) Remove aircraft from jacks.

2. Main Landing Gear

A. Removal

See “Figure 2” on page 32106.

(1) Jack up the aircraft per instructions in 7-10-00.

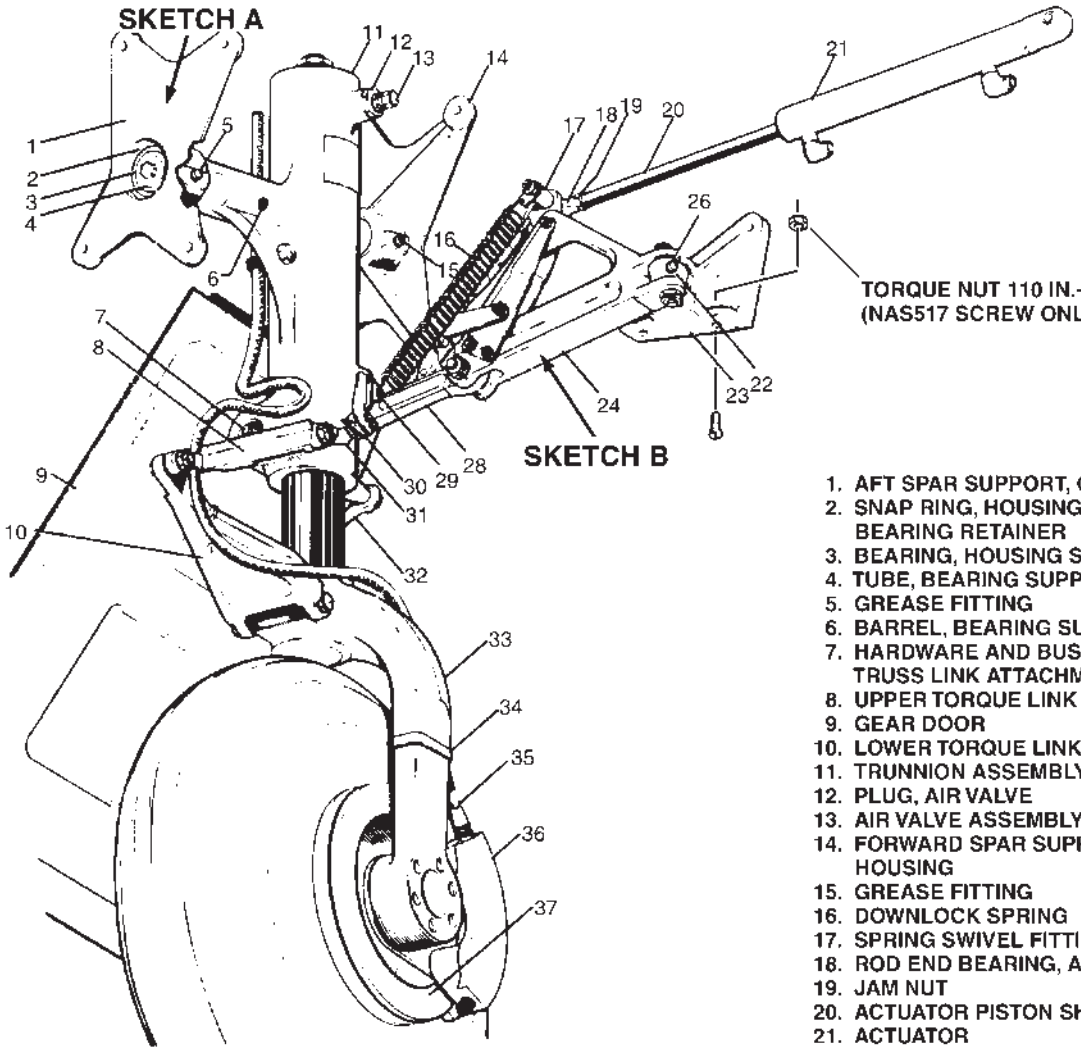
(2) If the side brace truss assembly is to be completely removed from the aircraft, proceed as follows:
   (a) With the gear extended disconnect the gear down lock spring.
   (b) Disconnect the actuator shaft rod end from the truss assembly’s upper link.
   (c) Take note of the bushings and washers on each side of the lower truss link’s end bearing where it connects to the strut housing.
   (d) Disconnect the lower truss link from the strut housing.
   (e) Disconnect the upper truss link fork from the truss support stud bearing.
   (f) Remove the assembly and maintain as required.

(3) The oleo strut assembly can be removed as follows:
   (a) Disconnect the brake line at the fitting in the gear well.
   (b) Disconnect the gear door actuating rod from the strut housing.
   (c) Remove the access plate located under the wing, aft of the landing gear.
   (d) Ensure the lower truss link is disconnected and clear of the gear housing.
   (e) Have the gear supported in a manner that as the supports are disconnected, the gear will not cant and cause unnecessary loads on the opposite support and structure.
   (f) Remove the bolts connecting the forward gear housing support to the forward spar.
   (g) The bearing support tube in the aft gear housing support can be removed as follows:
      1) Reaching through the access opening under the wing and the hole in the spar, remove the bolt securing the tube in its housing.
      2) Obtain a length of .040 safety wire and hook the end of it. Insert the hooked end of the safety wire through the bolt hole in the tube and, with the hook holding the tube, pull the tube from the support fitting.

(4) The aft support fitting can be removed by reaching through the access hole to hold the nuts and removing the bolts from the wheel well.

(5) Either of the oleo strut support bearings can be removed by removing one of the snap rings and pushing the bearing from its housing.

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1. AFT SPAR SUPPORT, GEAR HOUSING
2. SNAP RING, HOUSING SUPPORT BEARING RETAINER
3. BEARING, HOUSING SUPPORT
4. TUBE, BEARING SUPPORT
5. GREASE FITTING
6. BARREL, BEARING SUPPORT
7. HARDWARE AND BUSHINGS, LOWER TRUSS LINK ATTACHMENT
8. UPPER TORQUE LINK
9. GEAR DOOR
10. LOWER TORQUE LINK
11. TRUNNION ASSEMBLY
12. PLUG, AIR VALVE
13. AIR VALVE ASSEMBLY
14. FORWARD SPAR SUPPORT, GEAR HOUSING
15. GREASE FITTING
16. DOWNLOCK SPRING
17. SPRING SWIVEL FITTING
18. ROD END BEARING, ACTUATOR SHAFT
19. JAM NUT
20. ACTUATOR PISTON SHAFT
21. ACTUATOR
22. STUD BEARING, TRUSS SPAR SUPPORT
23. SPAR SUPPORT FOR TRUSS ASSEMBLY
24. UPPER LINK, TRUSS ASSEMBLY
25. MOUNTING PAD, DOWN LIMIT SWITCH
26. GREASE FITTING
27. ACTUATING PLATE, DOWN LIMIT MICRO SWITCH
28. LOWER LINK, TRUSS ASSEMBLY
29. BRACKET, DOWN LOCK SPRING
30. SQUAT SWITCH
31. ACTUATOR, SQUAT SWITCH
32. ROD, GEAR DOOR CONNECTING
33. STRUT ASSEMBLY
34. CLAMP SUPPORT FOR BRAKE LINE
35. BRAKE LINE
36. BRAKE ACTUATING CYLINDER

NOTE

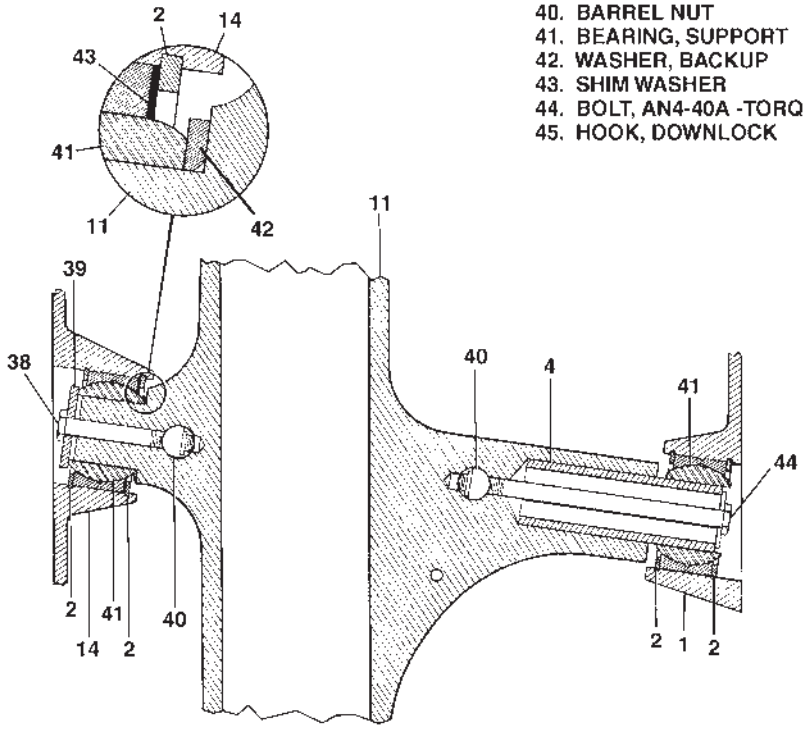
DURING ASSEMBLY IT MAY HAVE BEEN NECESSARY TO USE OVERSIZE HARDWARE TO MEET MANUFACTURING TOLERANCES. SPECIAL ATTENTION SHOULD THEREFORE BE GIVEN WHEN REMOVING HARDWARE TO INSURE THAT THE SAME HARDWARE (DIAMETER, ETC.) IS USED AT REASSEMBLY. STANDARD AN 4 OR AN 5 BOLTS ARE REPLACED BY ALTERNATE OVERSIZED BOLTS NAS 3004 OR NAS 3005 RESPECTIVLY, AND STANDARD FLUSH HEAD BOLTS MS 24694-505 ARE REPLACED BY NAS1604-9R BOLTS WHEN OVERSIZED HARDWARE IS REQUIRED.

Main Gear Installation
Figure 2 (Sheet 1 of 2)

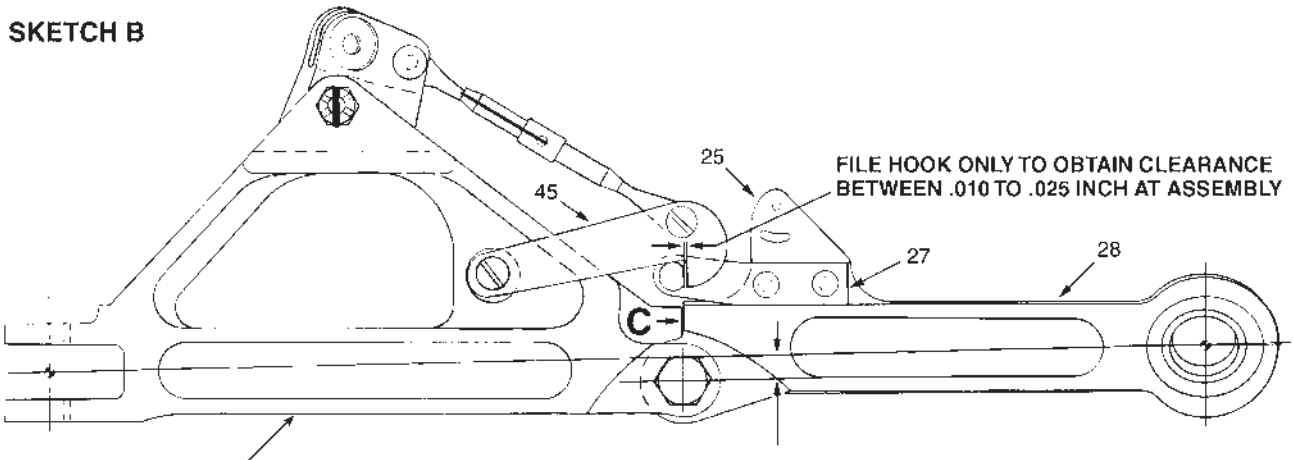
SKETCH A

37. BRAKE DISC
38. BOLT, AN4-17A -TORQUE 40T0 50 IN. - LBS.
39. WASHER, BACKUP
40. BARREL NUT
41. BEARING, SUPPORT
42. WASHER, BACKUP
43. SHIM WASHER
44. BOLT, AN4-40A -TORQUE 50 TO 70 IN.-LBS.
45. HOOK, DOWNLOCK

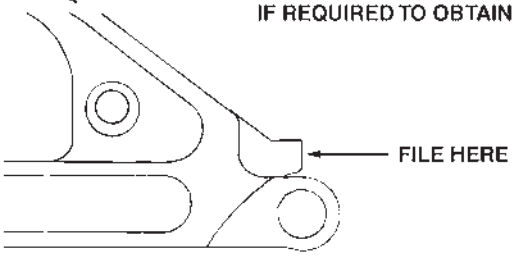
2026



SKETCH B



SKETCH C



Main Gear Installation
Figure 2 (Sheet 2 of 2)

B. Cleaning, Inspection, and Repair

(1) Clean all parts with a suitable dry type cleaning solvent.

(2) Inspect the gear components for the following unfavorable conditions:

(a) Bolts, bearing and bushings for excess wear, corrosion and damage.

(b) Gear housing, truss links, torque links and attachment plates for cracks, bends or misalignment.

(3) Inspect the gear down lock spring for the following:

(a) Excessive wear or corrosion, especially around the hooked ends of the spring. A spring should be rejected if wear or corrosion exceeds one-quarter the diameter of the spring. Clean away all corrosion and repaint.

(b) Check the spring for load tensions below minimum allowable tolerance. The minimum tension of the spring is 48 pounds pull at 7.9 inches. Measurement is taken from the inner side of each hook.

(4) Check the general condition of each limit switch and its actuator and wiring for fraying, poor connections or conditions that may lead to failures.

(5) Check side brace truss assembly over center travel by attaching the upper and lower links, setting them on a surface table and ascertaining that when the stop surfaces of the two links touch, linkage is .25/.28 of an inch over center. Should the distance exceed the required over center travel and bolt and bushings are tight, replace one or both links.

(6) Repair of the landing gear is limited to reconditioning of parts such as replacing components, bearings and bushings, smoothing out minor nicks and scratches and repainting areas where paint has chipped or peeled. Existing components should also exhibit limited free play when installed otherwise the bearings or bushings should be replaced.

C. Installation

See “Figure 2” on page 32106.

NOTE: Before and/or during assembly, lubricate bearings, bushings, and friction surfaces with the appropriate lubricants as described in 12-20-00.

(1) If either of the gear support bearings have been removed and need not be replaced, make sure they are clean. Grease and install using the appropriate snap rings to secure them.

(2) Check the bearings in both supports for side play. Add shim washers (P/N 62833-44) as necessary. See “Figure 2” on page 32106, Sketch A.

(3) Referring to “Figure 2” on page 32106, install the gear in the wing as follows:

(a) Place a new spacer washer on the short “forward” arm of the strut housing, ensuring the chamfered edge is toward the housing.

(b) Check that the barrel nut is properly positioned in the short arm of the housing and install the “forward” support fitting.

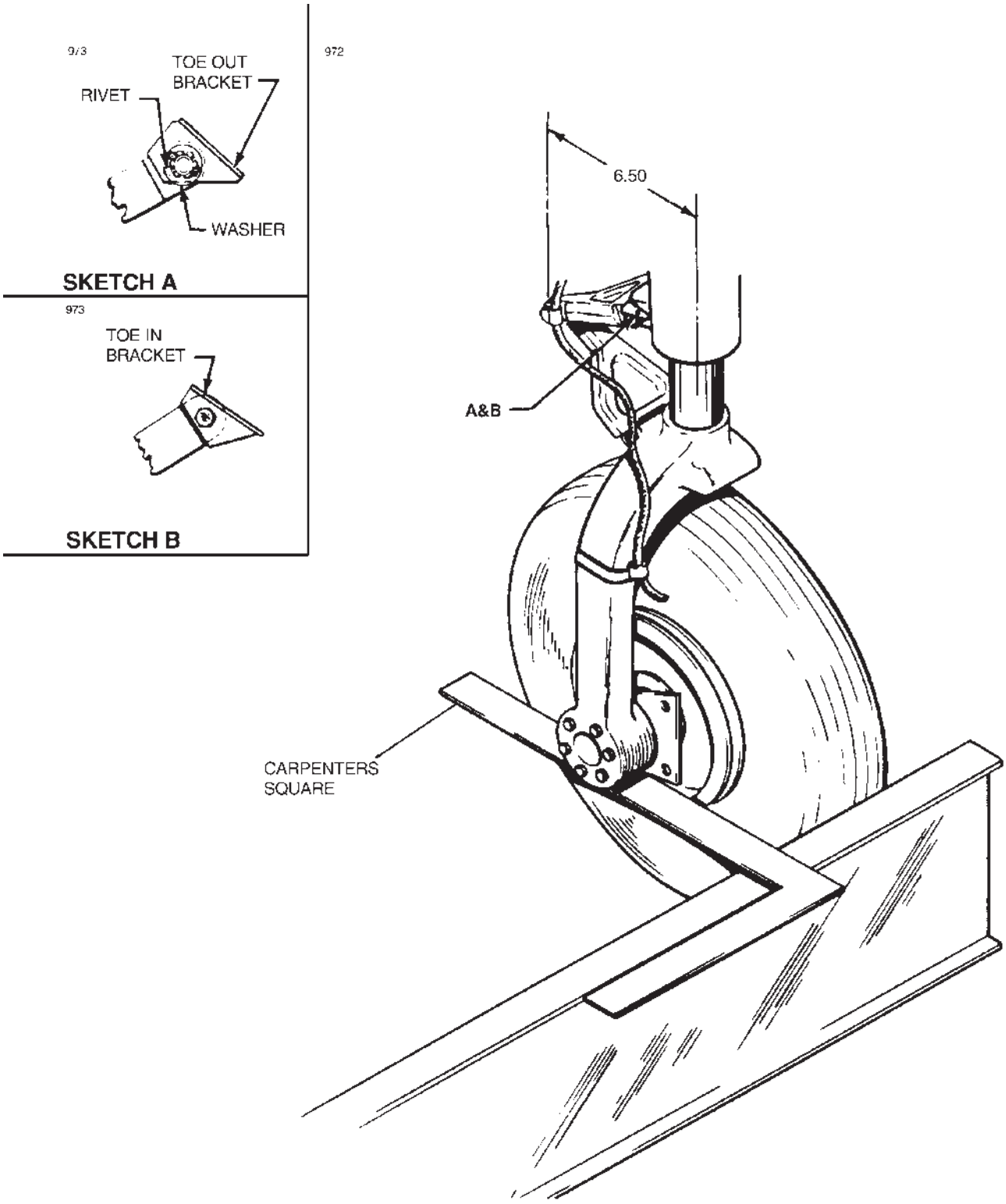
(c) Place the correct backup washer on the AN4-17A bolt and install the retaining bolt in the arm with a torque of 40 to 50 inch-pounds. Grease assembly from the fitting on the support.

(d) Position the aft support fitting at its attachment point in the wheel well and install the nuts and washers by reaching through the access opening under the wing.

(e) Holding the bearing support tube for the aft support, reach through the access opening under the wing. Insert the tube through its support fitting and the hole in the spar, making sure it does not extend so far through the support that the gear cannot be aligned.

(f) Position the gear in the well of the wing and support the gear in its aligned position.

(g) With the “forward” support fitting appropriately aligned on the main spar, install the bolts and washers.



Aligning Main Gear
Figure 3

(h) Through the access hole in the bottom of the wing, push the bearing support tube the rest of the way through the aft support fitting into the gear housing support arm.

(i) Check to ensure the barrel nut it properly installed in the long (aft) arm of the housing and slide the specific backup washer (P/N 67502-0) over the retaining bolt (AN4-40A).

(j) Reaching through the access hole under the wing, install the retaining bolt and backup washer with a torque of 50 to 70 inch-pounds.

(4) Reconnect the brake line in the gear well.

(5) The gear truss assembly may be installed as follows:

NOTE: If the reason for gear removal was due to wing replacement, it may be necessary to remove the upper truss link support from the old wing to install on the new wing. Should this be the case, the support should be partially installed to provide a guide to back drill one .250 and one .312 hole in the spar cap if not previously done at the factor. The outboard hole is .312 diameter and not countersunk. A bump is added to gear door to clear bolt head. Inboard hole is .250 countersunk. Countersink the hole in the exterior surface of the spar cap 100° x .500. The .250 screw head must be flush with the spar cap to allow the gear door to properly close.

(a) With the upper truss link support installed on the spar, ensure the attachment bolts are properly torqued. Check “Figure 2” on page 32106 for torque values.

(b) If necessary, assemble the upper and lower truss links and check the through center travel as described in the previous subject paragraph.

(c) Install the truss assembly to the gear and support fitting. Torque the nut at the housing 600 ± 25 inch-pounds.

(6) Connect the actuator rod end to the fitting in the upper truss link, making sure to properly install the spring swivel attachment.

(7) Install down lock spring and check adjustment of landing gear.

(8) Check wheel alignment per instructions in this chapter.

(9) Ensure the landing gear is lubricated per 12-20-00.

(10) Install access plates, reconnect gear door and lower airplane off jacks.

D. Alignment

See “Figure 3” on page 32109.

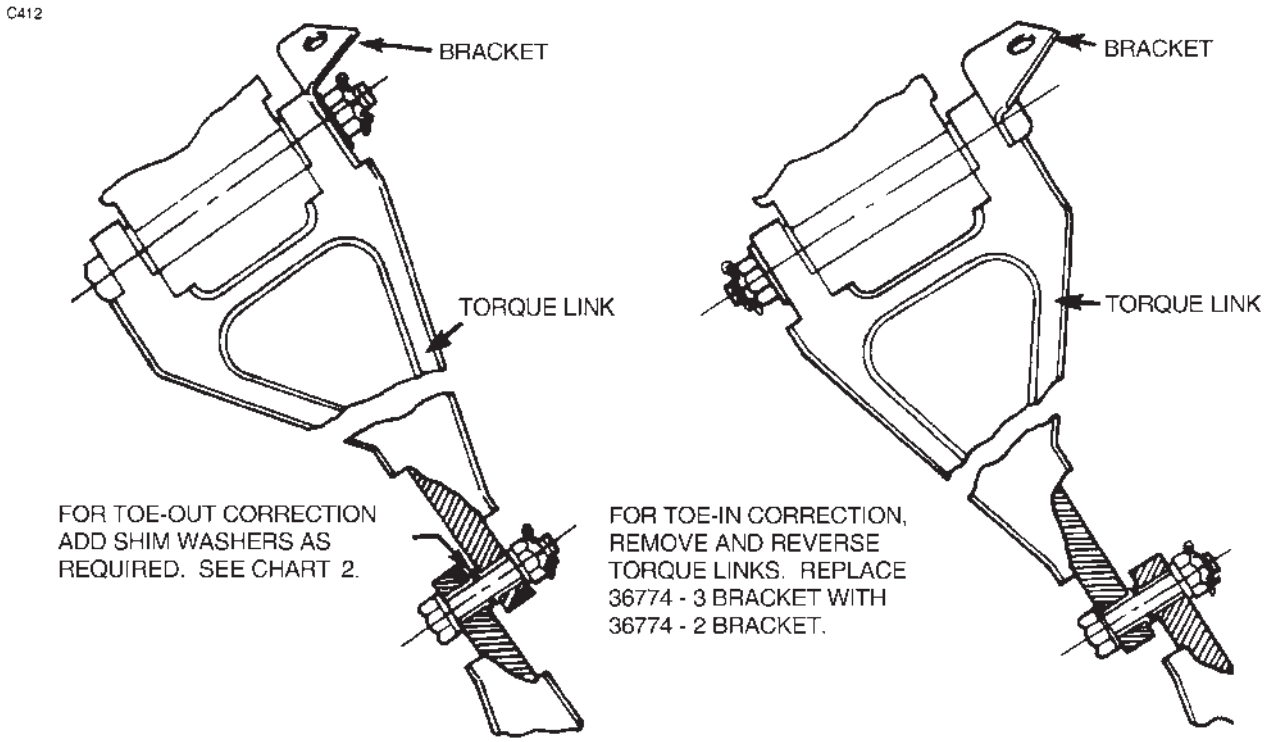
(1) Place a straightedge no less than twelve feet long across the front of both main landing gear wheels. But the straightedge against the tire at the hub level of the landing gear wheels. Jack the airplane up just high enough to obtain a six and one-half inch dimension between the centerline of the strut piston and the centerline of the center pivot bolt of the gear torque links. Devise a support to hold the straightedge in this position.

(2) Set a square against the straightedge and check to see if its outstanding leg bears on the front and rear side of the brake disc. (It may be necessary to remove the brake assembly to have clear access to the disc.) If it touches both forward and rear flange, the landing gear is correctly aligned. The toe-in for the main landing gear wheels should be 0 ± 1/2 degrees.

NOTE: Use a carpenter’s square for checking main landing gear wheel alignment.

(3) If the square contacts the rear side of the disc, leaving a gap between it and the front flange, the wheel is toed-out. If a gap appears at the rear flange, the wheel is toed-in.

(4) To rectify the toe-in and toe-out condition, remove the bolt connecting the upper and lower torque links and remove or add spacer washers to move the wheel in the desired position. See “Chart 1” on page 321011.



Toe-In/Toe-Out Adjustment
Figure 4

CHART 1
TOE-IN AND TOE-OUT CORRECTION CHART

TOE-IN TOE-OUT	SHIM WASHERS	WASHERS UNDER HEAD	WASHER UNDER NUT	AN 174 BOLT
0°		AN960-416	AN960-416 (3)	-14
0° 33”	AN960-416	AN960-416	AN960-416 (2)	-14
0° 48”	AN960-416L AN960-416	AN960-416	AN960-416	-14
1° 04”	AN960-416 (2)	AN960-416	AN960-416	-14
1° 19”	AN960-416L AN960-416 (2)	AN960-416L	AN960-416	-14
1° 35”	AN960-416 (3)	AN960-416	AN960-416 (2)	-15
2° 05” (Max. Allow.)	AN960-416 (4)	AN960-416	AN960-416	-15
AN960-416L Washers 0.31 Thick			AN960-416 Washers .062 Thick	

(5) Should a condition exist that all spacer washers have been removed and it is still necessary to move the wheel further in or out, then it will be necessary to turn the torque link assembly over. (See “Figure 4” on page 321011.) This will put the link connecting point on the opposite side allowing the use of spacers to go in the same direction.

(6) Recheck wheel alignment. If the alignment is correct, safety the castellated nut with cotter pin.

(7) If a new link on the top left main gear has to be installed or it had to be reversed during the alignment check, it will be necessary to check the gear safety switch (squat switch) bracket for engagement and locking in place. If link has to be reversed, then the bolt also has to be reversed and microswitch bracket P/N 36774-3 will be replaced by microswitch bracket 36774-2.

(8) Check adjustment of landing gear safety switch (squat switch).

3. Main Gear Doors

A. Removal

(1) With the landing gear extended, disconnect the door retraction rod from the door by removing nut, washers and bolt.

(2) Remove the door from the wing panel by bending the door hinge pin straight and from the other end pulling out the pin.

(3) The door retraction rod may be removed from the gear housing by cutting the safety wire and removing the attachment bolt and washer. Note the number of washers between rod end bearing and housing.

B. Cleaning, Inspection, and Repair

(1) Clean the door and retraction rod with a suitable cleaning solvent.

(2) Inspect the door for cracks or damage, loose or damaged hinges and brackets.

(3) Inspect the door retraction rod and end bearing for damage and corrosion.

(4) Repairs to a door may be replacement of hinge, repair of fiberglass and painting.

C. Installation

(1) Install the door by positioning the hinge halves of the door and wing and inserting the hinge pin. It is recommended a new pin be used. Bend the end of the pin to secure in place.

(2) Install the door retraction rod by positioning the rod at its attachment points at the door and strut housing. At the door attachment, thin washers are inserted at each side of the rod end bearing and it is secured with bolt, washer and nut. At the strut housing, place washers between rod end bearing and housing not to exceed .12 of an inch to obtain proper clearance and secure with bolt. Safety bolt with MS20995C41 wire.

(3) Check that the all around clearance between the door and the wing skin is not less than .032 of an inch.

D. Adjustment

(1) Place the airplane on jacks as described in 7-10-00.

(2) Determine that the main gear is properly adjusted for gear up as explained earlier in this section.

(3) Adjust retraction rod end at door so that door will pull up tightly when the gear is full up. Over-tightening may result in door buckling; however, if the door is too loose, it will gap in flight.

(4) Check all rod ends for adequate thread engagement, for safety and tightness of jam nuts.

(5) Remove airplane from jacks.

NOSE GEAR AND DOORS

1. Nose Gear Oleo Assembly

   A. Inspection

      (1) Clean all parts using a suitable dry type cleaning solvent.
      (2) Inspect components of the landing gear as follows:
         (a) Bearings and bushings for excessive wear, corrosion, scratches and overall condition.
         (b) Retaining pins for wear.
         (c) Lock rings for cracks, nicks, burrs and overall condition.
         (d) Cylinder and piston strut for excessive wear, corrosion, scratches and nicks.
         (e) Orifice hole for obstruction.
         (f) Fork for misalignment, cracks or other damage.
         (g) Air valve for general condition.

   B. Removal from Trunnion

      See “Figure 1” on page 32202.

      The nose gear oleo assembly can be removed and/or disassembled from the trunnion assembly with it mounted in or removed from the aircraft.

      (1) Jack up the airplane per 7-10-00.
      (2) Depress the air valve core pin to release the air pressure in the strut and remove the valve body from the plug.
      (3) The oleo housing can be evacuated by removing the valve, inserting a thin hose through the plug into the housing, and siphoning out the hydraulic fluid.
      (4) To remove the oleo assembly from the trunnion assembly, cut the safety wire and remove the four bolts at the top of the housing securing the tiller to the top of the oleo housing.
      (5) Remove the hardware connecting the centering spring rod end from the trunnion arm.
      (6) Pull the oleo assembly from the trunnion assembly.

   C. Installation in Trunnion

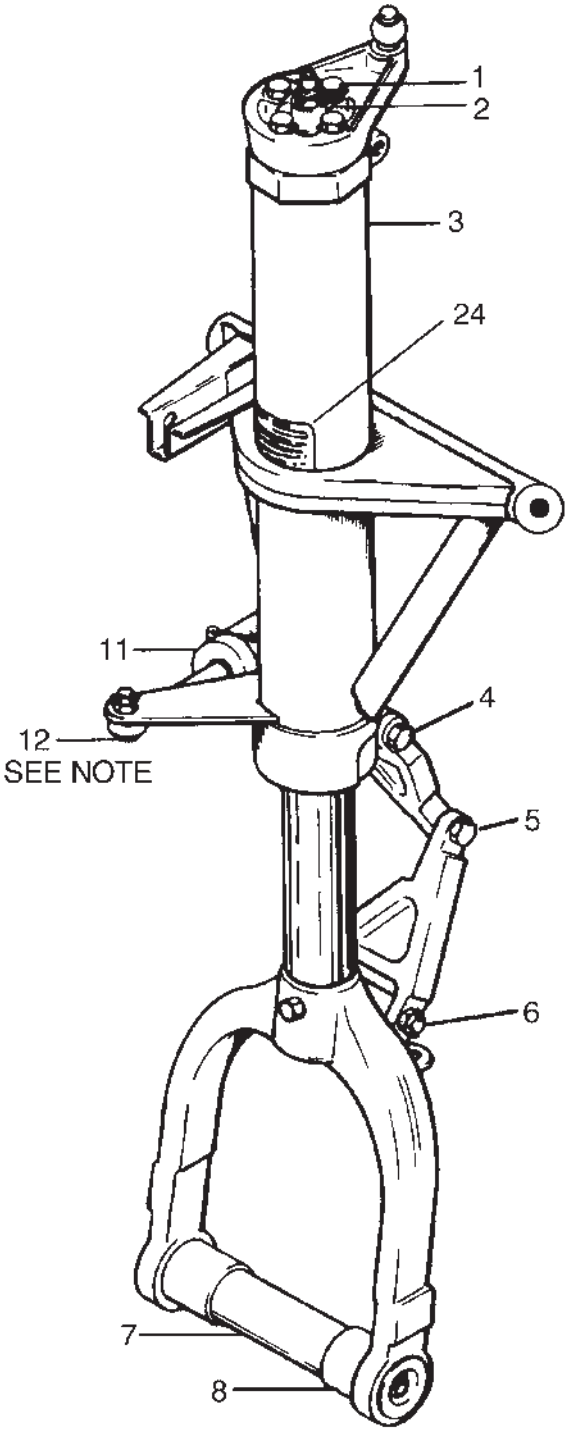
      See “Figure 1” on page 32202.

      (1) Assemble the components of the oleo strut as instructed in steps (1) through (8) of Assembly, below, and tape over or enclose the top of the oleo cylinder to prevent dirt from entering.
      (2) Remove the upper and lower shoulder bushings from the trunnion assembly.

         NOTE: Take proper care to avoid damaging the flanges and bearing surfaces of the bushings.

      (3) Coat the bearing surfaces of the lower shoulder bushing with the appropriate grease (see the appropriate Lubrication Chart in 12-20-00) and slide the bushing down the oleo cylinder until it bottoms above the upper torque link mount.
      (4) Being careful not to damage the shoulder bushing, insert the oleo assembly through the base of the trunnion, until the trunnion seats firmly on the lower shoulder bushing. Support oleo in trunnion.
      (5) Coat the upper shoulder bushing bearing surfaces with grease and carefully insert the upper bushing into its recess between the oleo cylinder and trunnion. Make sure the bushing seats firmly in trunnion.
      (6) Remove tape or covering from top of oleo cylinder and ensure the annular detent is clean and unmarred.

1. CAP BOLT & WASHERS
2. SAFETY WIRE
3. TRUNNION ASSEMBLY
4. BOLT, BEARING, WASHER & NUT
5. BOLT, WASHER, NUT & COTTER PIN
6. BOLT, BUSHING, WASHER, NUT & COTTER PIN
7. AXLE
8. TUBE - SPACER
9. PLUG - WHEEL END
10. AXLE BOLT
11. NOSE GEAR CENTERING SPRING
12. BOLT, WASHERS, NUT & COTTER PIN (BOLT HEAD DOWN)
13. AIR VALVE CAP
14. AIR VALVE CORE
15. AIR VALVE BODY
16. WASHER
17. BUSHING - SHOULDER (UPPER)
18. SCREW, WASHER & NUT
19. ROLLER ASSEMBLY & BUSHING
20. TILLER
21. WASHER
22. BUSHING - SHOULDER (LOWER)
23. TRUNNION ASSEMBLY
24. PLACARD
25. O-RING
26. OLEO CYLINDER
27. BEARING
28. PISTON STRUT
29. O-RING
30. BEARING
31. TORQUE LINK
32. O-RING OR T SEAL (REFER TO PARTS CATALOG)
33. RING - WIPER
34. WASHER
35. SNAP RING
36. FORK

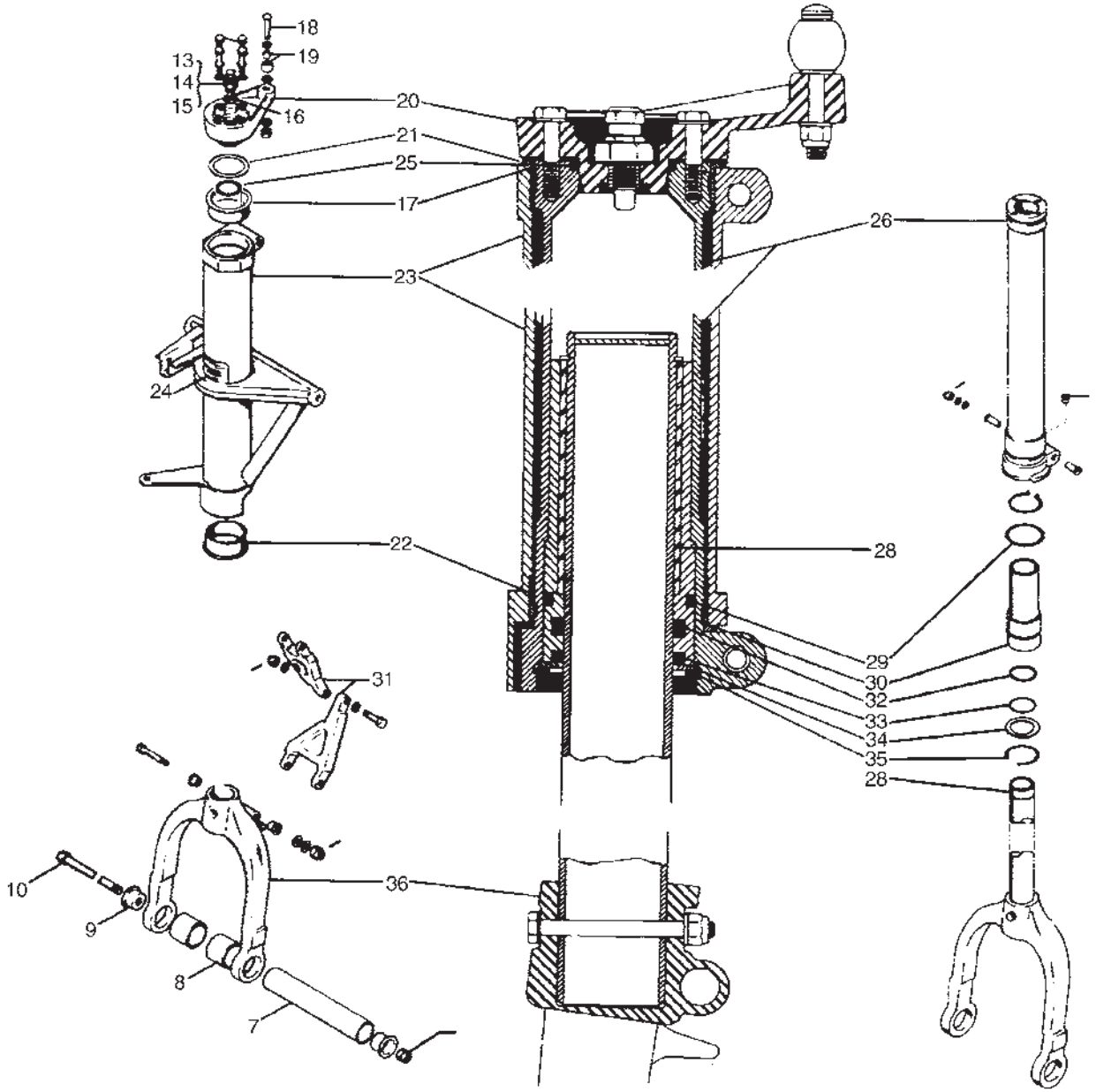


NOTE:

MAKE SURE THE TRUNNION ARM AND CENTERING SPRING ROD END ARE CONNECTED AS SHOWN WITH THE BOLT HEAD DOWN.

Nose Gear Oleo Strut Assembly
Figure 1 (Sheet 1 of 2)

3042



Nose Gear Oleo Strut Assembly
Figure 1 (Sheet 2 of 2)

(7) With the appropriate O-ring dipped in hydraulic fluid, fit it into the aforementioned annular slot in the oleo cylinder.

(8) Place the required spacer washers between the tiller and oleo/trunnion mounting surface. Grease them as specified in 12-00-00.

(9) Align and insert the bottom of the tiller in the oleo cylinder. Install and safety the four retaining bolts with 0.041 safety wire.

NOTE: The hardware used in connecting the centering spring rod end to the trunnion arm must be installed as shown in “Figure 1” on page 32202.

(10) See “Figure 1” on page 32202, and as shown, install nose gear centering spring assembly. Make sure the trunnion arm and centering spring rod end are connected as shown with the bolt head down.

(11) Lubricate the gear assembly per the Lubrication Charts in 12-20-00.

(12) Fill the oleo housing with 21 ± 1 oz. of MIL-H-5606 hydraulic fluid and move the strut up and down to ensure free movement. The weight of the wheel and fork should allow strut to extend.

(13) Install the air valve plug and body, as applicable, and torque the valve (body) in the plug 350 to 400 inch-pounds.

(14) Check nose gear for alignment and operation.

(15) With the aircraft at empty weight, inflate the strut until it exhibits 1.20 ± 0.25 inches visible strut extension.

D. Disassembly

See “Figure 1” on page 32202.

(1) If not already accomplished, release the air pressure and remove the hydraulic fluid as described in steps (2) and (3) of Removal from Trunnion, above.

(2) Remove the cotter pin, nut, washer(s), and bolt connecting the upper and lower torque links. Separate the two links.

NOTE: Unless absolutely necessary, the fork should not be disassembled from the piston tube.

(3) Remove the snap ring at the “bottom” of the oleo cylinder and carefully pull the strut to pull the bearing from the oleo cylinder.

(4) Remove the snap ring at the “top” of the strut piston and slide off the bearing assembly.

(5) Replace all O-rings and the wiper ring.

E. Assembly

See “Figure 1” on page 32202.

(1) Ensure all components are clean and inspected, especially the inside of oleo cylinder.

(2) Dip a clean sponge in clean hydraulic fluid and wipe down the strut piston tube, making sure the tube remains clean.

(3) Place the bearing retainer snap ring, retainer washer and wiper ring respectively over the piston tube.

(4) Make sure the bearing assembly is inspected for damage, especially the bearing surface for gouges and wipe down with hydraulic fluid.

(5) Install new O-rings or T-seals on bearing and slide the bearing, wiper end first, over the piston tube.

(6) Install retaining ring on end of piston tube. (See P/N 761-887, PA-34-220T Parts Catalog)

(7) Slide the bearing, wiper ring, bearing retainer ring, and snap ring up against the piston tube retaining ring. Carefully slide the assemblies into the oleo tube till the bearing bottoms and insert the bearing snap ring in the end of the oleo housing.

(8) Ascertain that bushings are installed in the upper and lower torque link connecting holes and then connect the upper and lower torque links by installing the bolt, washer(s), and nut. Tighten nut enought to permit no side play in the links, yet still allow bolt rotation. Secure nut with cotter pin.

(9) If the oleo strut assembly has not been removed from the trunnion and the tiller remains attached, fill the oleo housing with 21 ± 1 oz. of MIL-H-5606 hydraulic fluid. Move the strut to ensure that it operates freely.

(10) Reinstall air valve and torque to 350 to 400 inch-pounds. With the aircraft off the jacks and at empty weight, inflate the oleo until the strut exhibits 1.20 ± .25 inches visible strut extension.

F. Repair

Repair of the oleo is limited to smoothing out minor scratches, nicks and dents, and replacement of parts.

2. Nose Landing Gear

A. Modified Components

The following parts have been modified to increase their service life:

NOTE: Aircraft S/N's 3449262, 3449313 and up have had the modified parts installed at the factory. While they are already installed, they still need to be inspected.

NOTE: Immediate replacement of the currently installed part is not required, as long as the part meets inspection and time-in-service requirements. When replacement is required, use the parts identified below.

(1) Drag link-to-nose gear strut bolt, P/N 693-215 (NAS6207-50D) (was P/N 400-274 (AN7-35)).

The Nose Gear Installation has been revised to replace the AN7-35 bolt with the higher strength NAS6207-50D bolt.

(2) Steering Channel Assembly P/N 95394-000.

The Steering Channel has been revised to increase the material thickness by 25% and the height increased to prevent "ball-out-of-track" failures. The part number for the redesigned Steering Channel is 95394-005. The 95394-005 Steering Channel is approved for all PA-34 Series aircraft.

(3) Actuator Mount Bracket P/N 95724 and Bushing P/N 95061-89.

The Actuator Mount Bracket has gone through two major design changes, from aluminum (never installed in these airplanes) to steel, and a recent change adding reinforcements to increase the bearing surfaces for the P/N 95061-089 bushing.

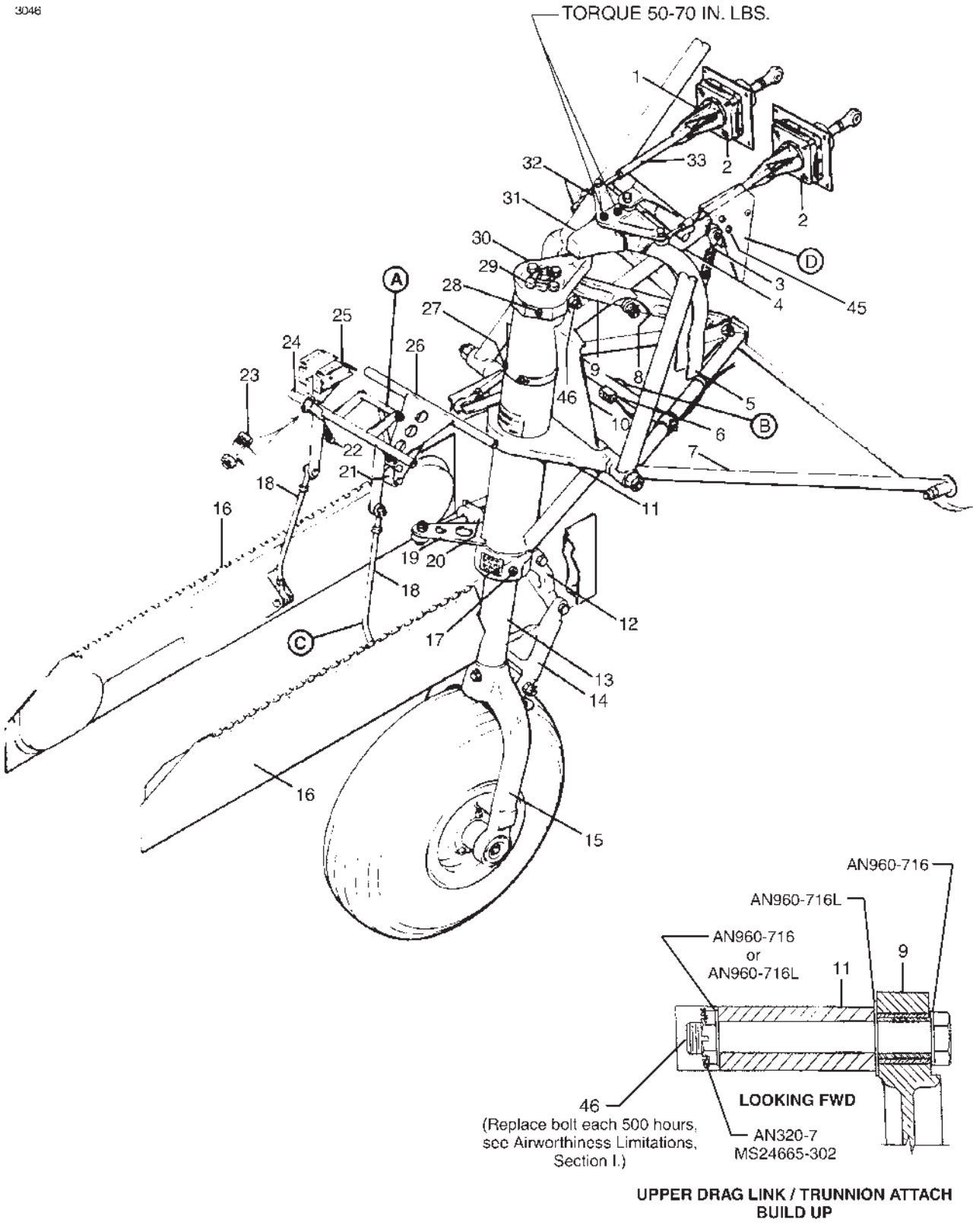
All PA-34-220T aircraft are subject to the 100 Hour Actuator Mount Bracket Inspection, below, regardless of which bracket is installed.

In the Seneca IV; and Seneca V S/N's 3449001–3449312, less 3449262: procure and install Piper Kit No. 767-359, if the steel Actuator Mount Bracket is discrepant and must be replaced. This kit includes the latest design actuator mount bracket (i. e., P/N 95724-006) and new, required, hardware for attaching the Nose Gear Retraction Link Assembly P/N 95712-004.

In Seneca V S/N's 3449262, and 3449313 & up: if the factory installed P/N 95724-006 bracket fails to pass inspection, replace it and any other components which are discrepant.

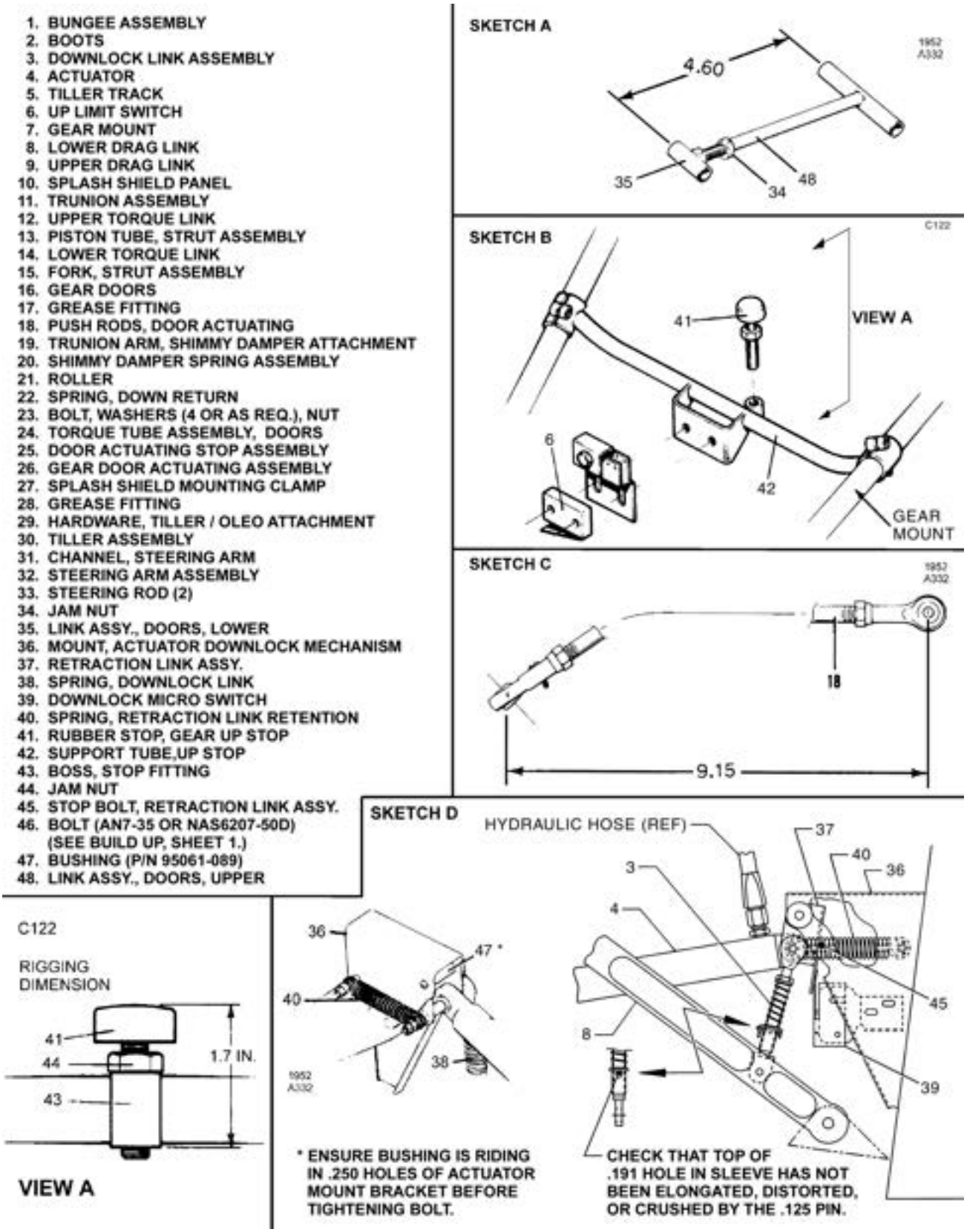
(4) Turn Limit Indicator

A Turn Limit Indicator was added to S/N's 3449061 and up to help reduce the incidence of turn stop boss failure (on the upper strut tube assembly Piper P/N 95720). Piper Kit No. 767-368 provides the Turn Limit Indicator for installation on all earlier PA-34 Series airplanes.



Nose Gear Installation
Figure 2 (Sheet 1 of 2)

1. BUNGEE ASSEMBLY
2. BOOTS
3. DOWNLOCK LINK ASSEMBLY
4. ACTUATOR
5. TILLER TRACK
6. UP LIMIT SWITCH
7. GEAR MOUNT
8. LOWER DRAG LINK
9. UPPER DRAG LINK
10. SPLASH SHIELD PANEL
11. TRUNION ASSEMBLY
12. UPPER TORQUE LINK
13. PISTON TUBE, STRUT ASSEMBLY
14. LOWER TORQUE LINK
15. FORK, STRUT ASSEMBLY
16. GEAR DOORS
17. GREASE FITTING
18. PUSH RODS, DOOR ACTUATING
19. TRUNION ARM, SHIMMY DAMPER ATTACHMENT
20. SHIMMY DAMPER SPRING ASSEMBLY
21. ROLLER
22. SPRING, DOWN RETURN
23. BOLT, WASHERS (4 OR AS REQ.), NUT
24. TORQUE TUBE ASSEMBLY, DOORS
25. DOOR ACTUATING STOP ASSEMBLY
26. GEAR DOOR ACTUATING ASSEMBLY
27. SPLASH SHIELD MOUNTING CLAMP
28. GREASE FITTING
29. HARDWARE, TILLER / OLEO ATTACHMENT
30. TILLER ASSEMBLY
31. CHANNEL, STEERING ARM
32. STEERING ARM ASSEMBLY
33. STEERING ROD (2)
34. JAM NUT
35. LINK ASSY., DOORS, LOWER
36. MOUNT, ACTUATOR DOWNLOCK MECHANISM
37. RETRACTION LINK ASSY.
38. SPRING, DOWNLOCK LINK
39. DOWNLOCK MICRO SWITCH
40. SPRING, RETRACTION LINK RETENTION
41. RUBBER STOP, GEAR UP STOP
42. SUPPORT TUBE,UP STOP
43. BOSS, STOP FITTING
44. JAM NUT
45. STOP BOLT, RETRACTION LINK ASSY.
46. BOLT (AN7-35 OR NAS6207-50D) (SEE BUILD UP, SHEET 1.)
47. BUSHING (P/N 95061-089)
48. LINK ASSY., DOORS, UPPER



Nose Gear Installation
Figure 2 (Sheet 2 of 2)

B. Inspection

(1) General

(a) Clean all parts with a suitable dry type cleaning solvent.

(b) Inspect gear components for the following unfavorable conditions:

1) Bolts, bearings and bushings for excessive wear, corrosion and damage.

2) Gear strut and cylinder, drag links and down lock link assembly for cracks, dents, bends or misalignment.

3) Down lock link assembly for damaged threads and bearing.

4) Roller assembly for freedom of movement and excessive wobble.

(c) Check down lock spring and down lock link spring for excess wear and corrosion, particularly around the hook portion. Springs should be discarded if wear or corrosion exceeds one-quarter the diameter of the spring. Remove corrosion and paint spring.

(d) Check down lock spring for adequate tension. This may be accomplished by observing several locking activations and checking for smooth operation with positive locking each time. If hook or down lock movement is slow or has a hesitation or jerky movement, the spring should be replaced. Check down lock link assembly for proper operation and cleanliness. Pin and hole should be carefully inspected for signs of wear or elongation.

(e) Check general condition of limit switches and actuators, wiring for fraying and poor connections or conditions which could lead to failures.

(2) Service Wear Limits

Use the wear limits in “Figure 3” on page 32209 to determine the condition of the listed parts when performing the following inspections. Visually inspect all bolts/pins for wear, damage, or corrosion. Replace as required.

(3) 100 Hour Actuator Mount Bracket Inspection

Each 100 hours time-in-service, visually inspect actuator mount bracket (Item 36, “Figure 2” on page 32206, Sketch D) for wear, cracks, loose mounting rivets, and elongation of .250 diameter holes (“Figure 3” on page 32209, Item 8,) where retraction link and P/N 95061-089 bushing (“Figure 2” on page 32206, Sketch D, Item 47) attach. See “Figure 3” on page 32209 for wear limits.

(4) 100 Hour Tunnel Bracket Installation Inspection

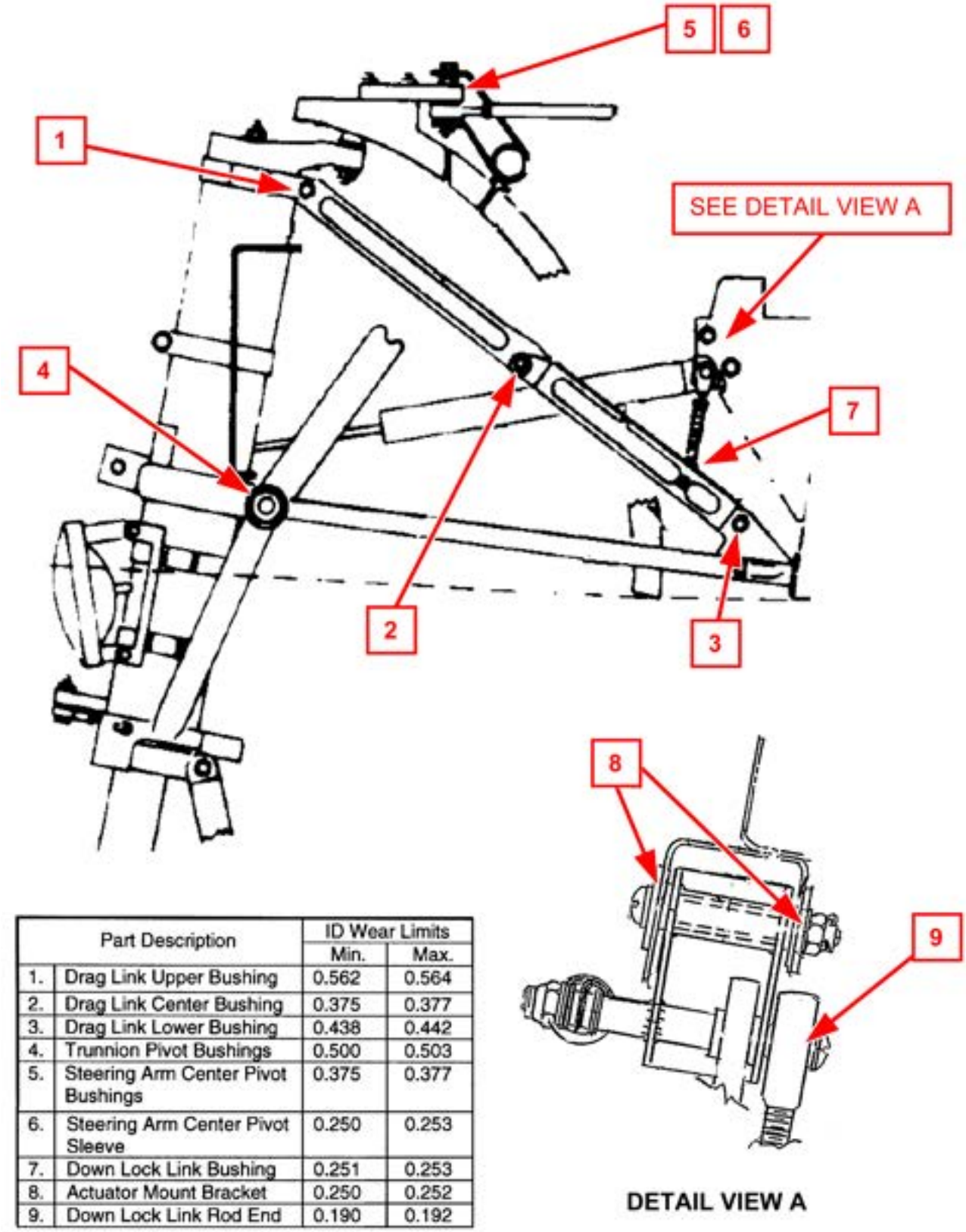
See “Figure 4” on page 322010.

Inspect the rivets shown in “Figure 3” on page 32209 for looseness. These rivets attach the Tunnel Bracket P/N 95554-000 and the Nose Gear Mount Fitting P/N 95555-000 to the tunnel aft of the bulkhead at F. S. 49.5. (See Items 90 and 350, respectively, in “Figure 4” on page 322010.)

(a) Place aircraft on jacks. Pull carpet away from the right side of the tunnel and secure out of the way. If present, remove access panel in bottom of fuselage, just aft of F.S. 49.5 at B.L. 00.0. Alternately, a mirror and flashlight, or a borescope, can be used to gain visual access through structural openings in the tunnel bracket (some removal of floor carpet may still be required).

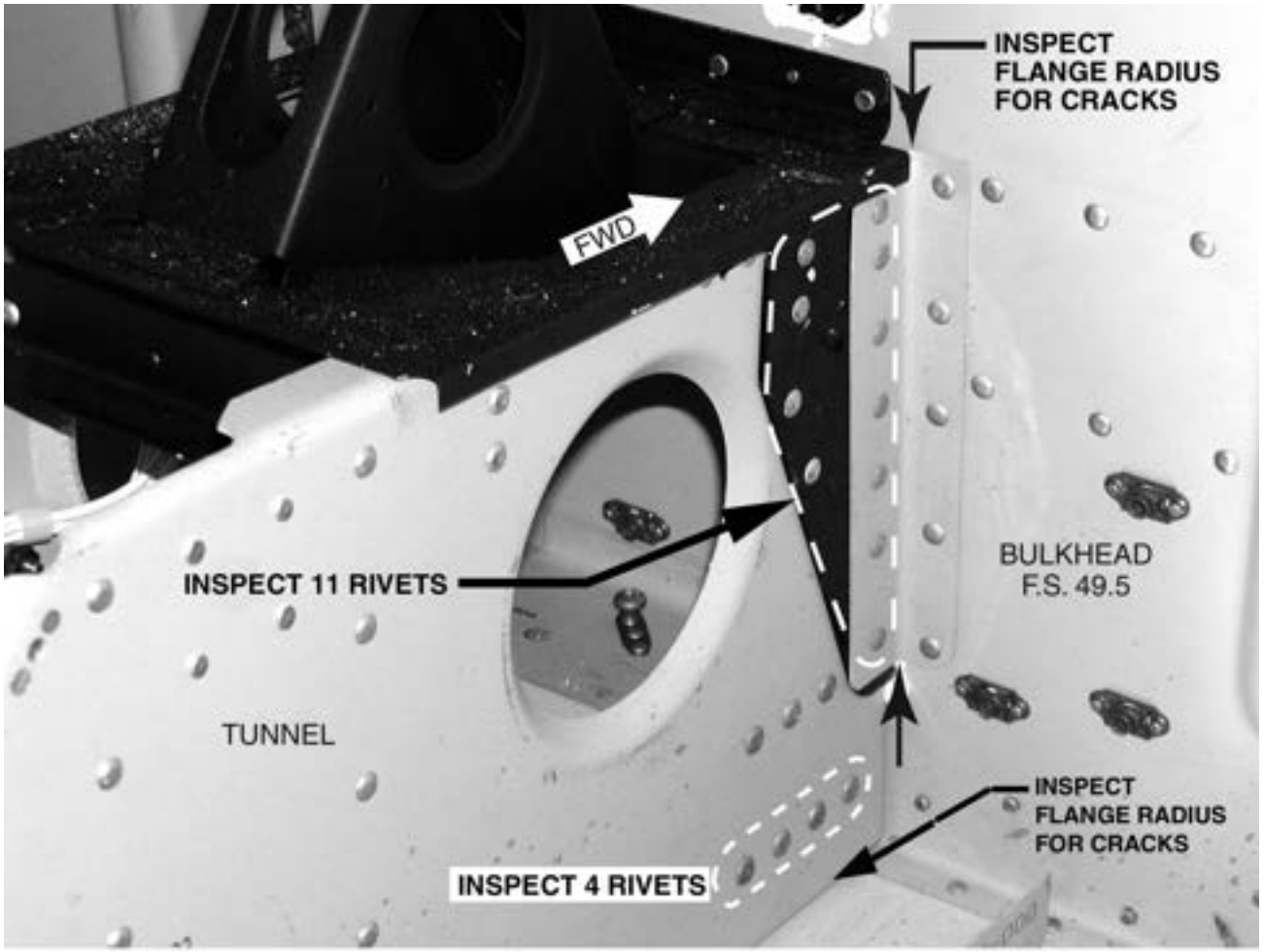
(b) Extend and retract landing gear through several cycles observing the rivets, tunnel bracket, and mount fitting, from both inside and outside the airplane. Inspect for any relative motion between the riveted components. No relative motion is permissible.

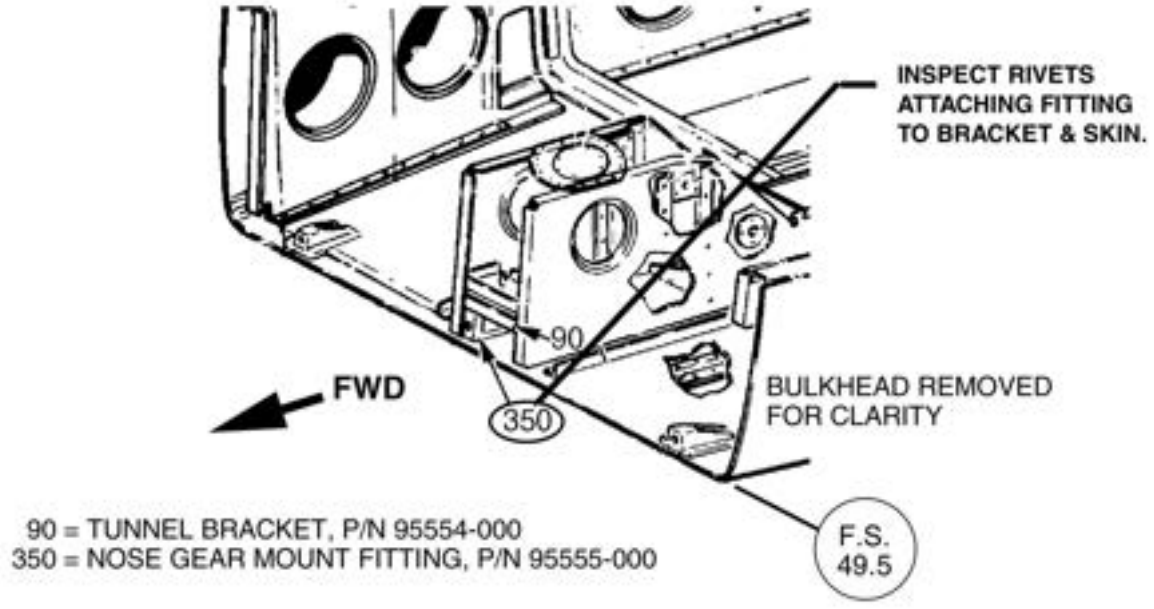
(c) Inspect the bracket attachment flange for cracks in the flange radius as shown in “Figure 3” on page 32209.



	Part Description	ID Wear Limits	
		Min.	Max.
1.	Drag Link Upper Bushing	0.562	0.564
2.	Drag Link Center Bushing	0.375	0.377
3.	Drag Link Lower Bushing	0.438	0.442
4.	Trunnion Pivot Bushings	0.500	0.503
5.	Steering Arm Center Pivot Bushings	0.375	0.377
6.	Steering Arm Center Pivot Sleeve	0.250	0.253
7.	Down Lock Link Bushing	0.251	0.253
8.	Actuator Mount Bracket	0.250	0.252
9.	Down Lock Link Rod End	0.190	0.192

Service Wear Limits
Figure 3





Tunnel Bracket Installation Inspection
Figure 4

(5) 100 Hour Upper Drag Link Bolt Inspection

(a) Each 100 hours, remove the nose gear drag link upper attach bolt AN7-35 or NAS6207-50D (P/N's 400-274 or 693-215, respectively), "Figure 2" on page 32206, Item 46 and with a 10X magnifier visually inspect for straightness, cracking or thread wear.

(b) If any of the above conditions exist, replace bolt (with new NAS6207-50D) and associated hardware (see Airworthiness Limitations, 4-00-00). Additionally, if replacing bolt, inspect associated bushings and bearings for deformation, cracking or wear, replace as required.

(6) 100 Hour Nose Gear Drag Link Inspection

See "Figure 5" on page 322012.

The drag links function to lock and retain the gear in its down position by traveling to a specific through center locked position. The "overcenter travel" is extremely important for safe operation and should be checked regularly. Use the following procedure to check the drag links for proper through center travel and condition.

(a) Jack up the airplane per 7-10-00.

(b) Check all hinge bosses on the links for cracks and security. The links should move freely but not exhibit looseness or stop at the bushings.

(c) Clean the gear with an appropriate cleaner and make sure the over center stops are unobstructed and hinge points are clean.

(d) See 12-20-00 and lubricate the gear.

(e) With reference to "Figure 5" on page 322012, continue as follows for over center measurement:

NOTE: The standard over center dimension is measured from the centerlines of the drag link mounting holes. When measuring off the bolts the different bolt sizes must be taken into account in order to obtain proper centerlines.

1) Unbolt the down lock link assembly from the lower drag link and tape it up against the retraction fitting support bracket.

CAUTION: UPPER DRAG LINK AN7-35 OR NAS6207-50D ATTACHMENT BOLT MUST BE REPLACED EACH 500 HOURS TIME-IN-SERVICE.

2) Without removing the bolts, remove the cotter pins from the three drag link bolts and loosen the nuts enough to withdraw them about 1/4 of an inch.

3) Lay a "true" straightedge across the exposed grips of the upper and lower drag link bolts.

4) Apply pressure to the top of the drag links to keep the over center stops in solid contact and measure the perpendicular distance between the top surface of the NAS464P620 bolt grip and the bottom of the straightedge.

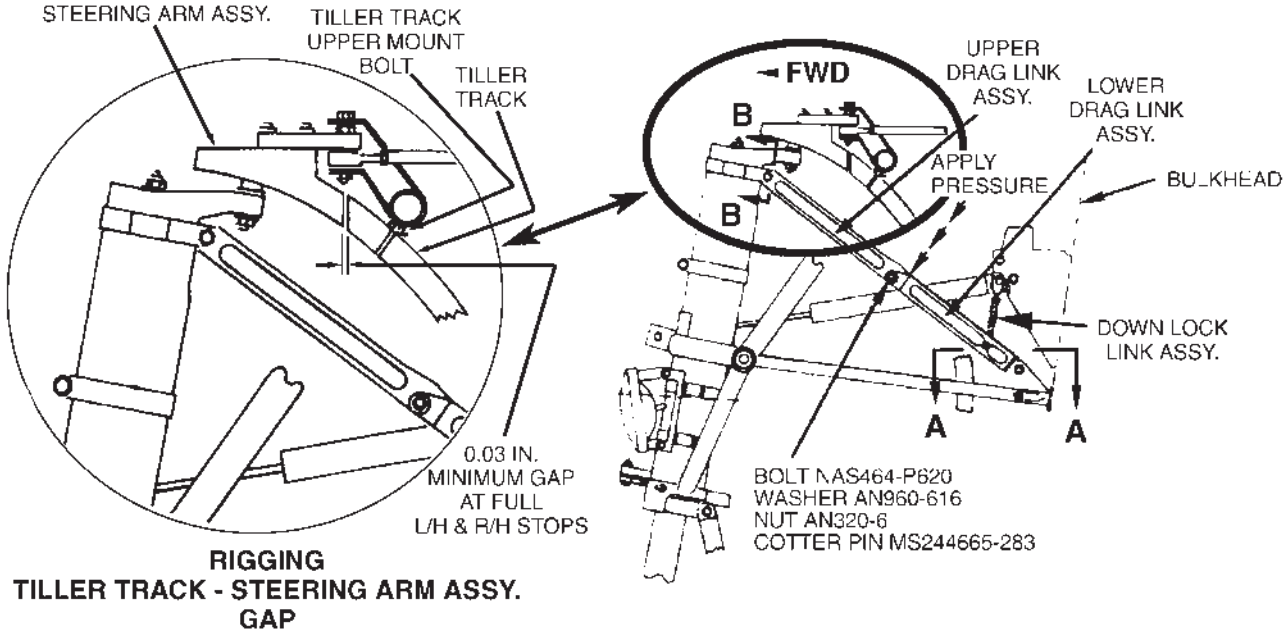
5) The measured distance should be 0.250 of an inch or greater. If the measured distance is less than 0.250 of an inch, replace drag links.

(7) 1000 Hour Nose Gear Trunnion Inspection

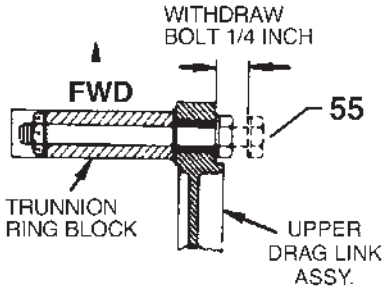
See "Figure 6" on page 322013.

Beginning at 2000 hours and each 1000 hours thereafter, visually inspect the nose gear trunnion (P/N 95723-005/006) for cracks in the areas shown in "Figure 6" on page 322013. Use supplemental lighting and a 10X magnifier.

DRAG LINK OVERCENTER MEASUREMENT

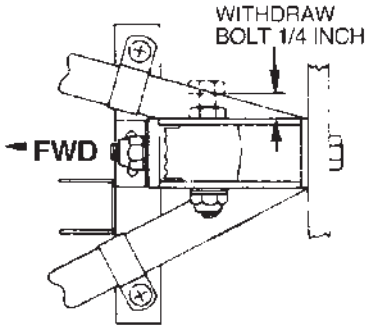


RIGGING
TILLER TRACK - STEERING ARM ASSY.
GAP



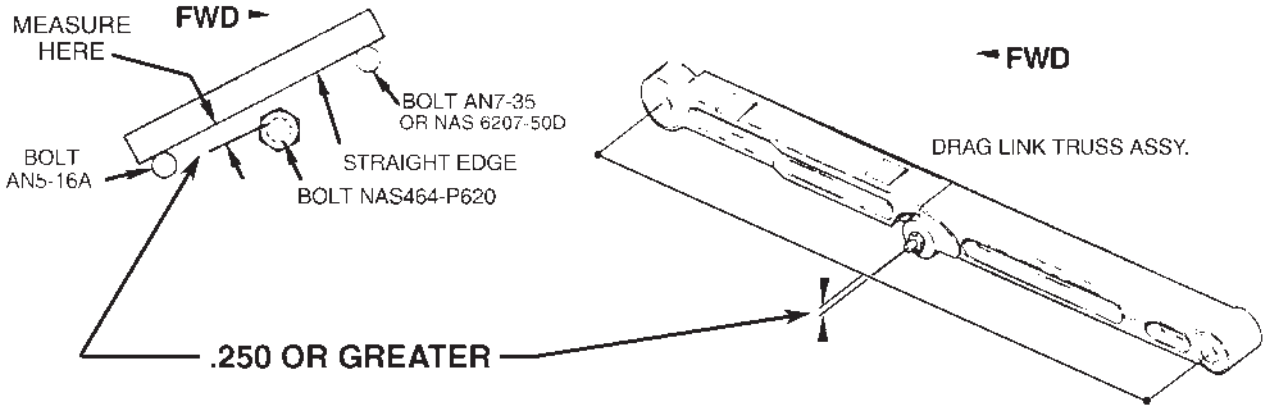
VIEW B-B

PREPARING FOR MEASUREMENT

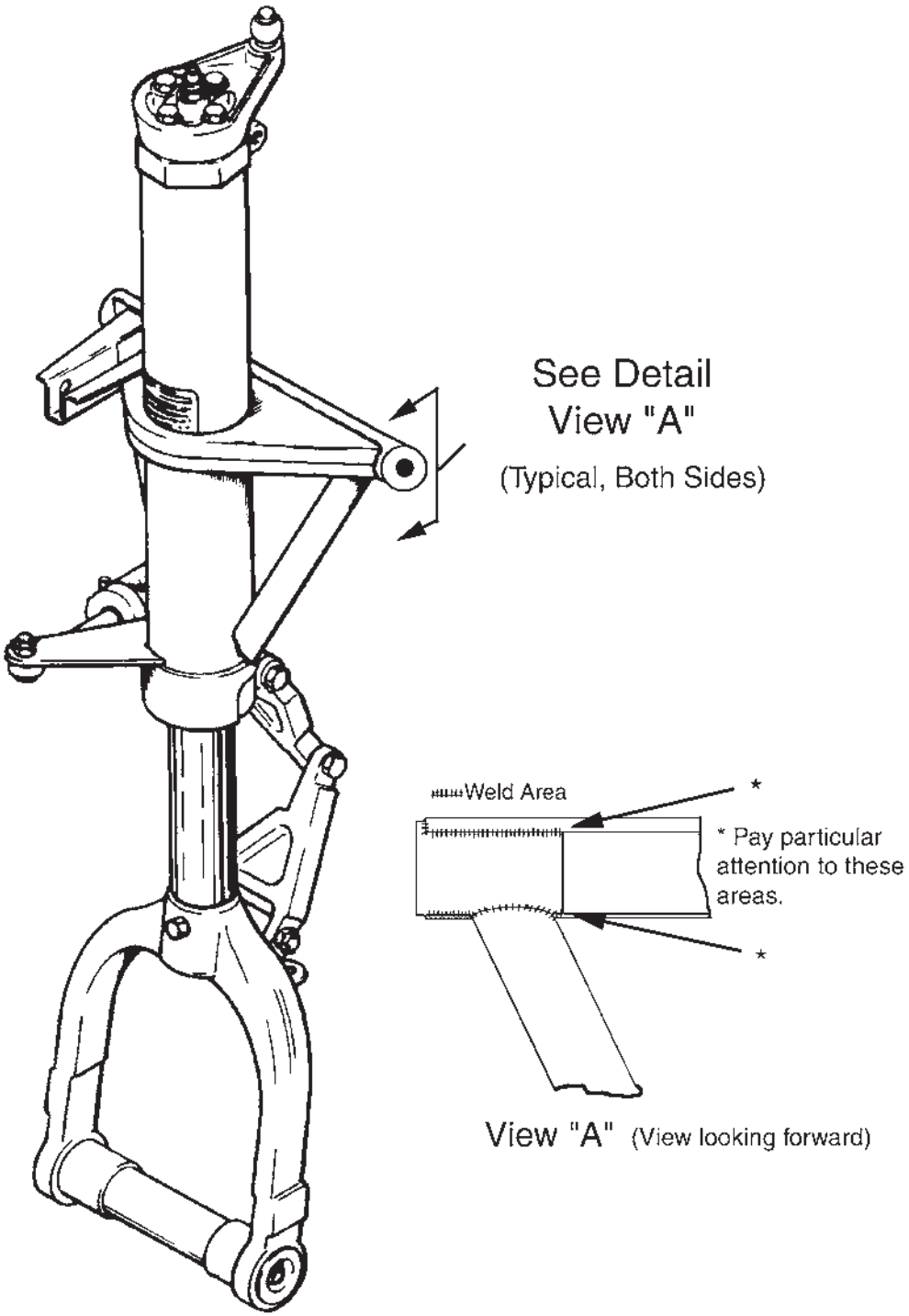


VIEW A-A

TAKING MEASUREMENT



Drag Link Installation and Adjustment
Figure 5



Nose Gear Trunnion Inspection
Figure 6

C. Removal

See “Figure 2” on page 32206.

WARNING: DO NOT ATTEMPT TO REMOVE THE SPRING FROM THE SHAFT ASSEMBLY OF THE NOSE GEAR CENTERING SPRING ASSEMBLY. THIS SPRING IS HELD UNDER COMPRESSION BY TWO BUSHINGS AND TWO PINS WITH FUSED HEADS.

NOTE: To gain access to the landing gear, remove access panels located in the forward baggage compartment.

(1) Place airplane on jacks.
(2) Disconnect leads to landing lights.
(3) Retract the landing gear far enough to permit unlocking the down lock mechanism.
(4) Disconnect down lock spring from aft end of gear actuating cylinder.
(5) Disconnect upper drag link from the strut housing.
(6) Disconnect the actuating cylinder from the strut housing.
(7) Remove attaching hardware at strut housing pivot point and remove landing gear from the airplane.
(8) To remove upper and lower drag links, disconnect down lock link from lower drag link and disconnect lower drag link from its attachment point.

D. Installation and Rigging

See “Figure 2” on page 32206 and “Figure 5” on page 322012.

NOTE: See Lubrication Charts, 12-20-00, and ensure affected parts of the landing gear such as bearings, bushings, etc, are lubricated prior to and following assembly.

(1) Position the gear assembly between the mounting points, making certain the tiller roller is properly inserted in the steering arm channel.
(2) Align the mounting points of the trunnion assembly with those on the mount and install the appropriate hardware. The bolt heads should be inboard, with the nuts outboard and just loose enough to allow the gear to swing freely.
(3) If necessary, assemble the drag links. Installed or removed, use the appropriate instructions in this section to check their rigging.
(4) The drag links are installed with their overcenter stops facing up, the upper link connection aligned with the right side of the gear’s upper ring block and the lower link connection positioned in its bracket on the bottom rear of the nose gear mount (See “Figure 2” on page 32206). With the links in position, install the connecting hardware and move the gear to assure free movement.
(5) At the actuator downlock mount (See “Figure 2” on page 32206, Sketch D), disconnect the retention spring from the retraction link fitting.
(6) Position the gear in its down locked position and check that the drag links have fully extended to their overcenter position with the stop surfaces in contact.
(7) With the actuating piston rod extended, adjust the piston rod end such that 0.25 of an inch of rod travel remains in the actuator before full extension. Connect the rod end to its mount on the gear assembly. The retraction link to which the actuator is attached should be near its stop.
(8) Reconnect the retention spring to the retraction link fitting.
(9) Install the down lock link (See “Figure 2” on page 32206) with the rod end connected to the retraction link and the other end to the bottom drag link.

(10) Adjust the down lock link as necessary until the guide pin is completely bottomed out at the top of its slot and the retraction link is moved against its stop.

NOTE: If the down lock link is adjusted properly, the retraction link will be moved completely to its stop by the down lock link therefore taking up some of the extra actuator piston rod travel and activating the down and locked limit switch.

(11) Retract and free fall the landing gear at least three times. Remove the down lock link, shorten it by 1/2 turn and reinstall.

(12) The down and locked limit switch should be adjusted to have it actuated when the retraction link is back against its stop.

(13) Using a tow bar to reach full travel against stops, rig the nose gear steering rod ends as necessary to allow full deflection.

(14) See “Figure 2” on page 32206, Sketch B and set the up stop to the dimension shown. Retract the gear and ensure the nose gear housing engages the stop under retraction pressure. Adjust as necessary.

NOTE: After any up stop adjustment, the gear must be cycled to ensure the strut engages the stop under pressure.

(15) Retract the gear and check that the up switch is just activated when the gear contacts the stop. Following this, adjust the switch upward another 0.02 to 0.04 of an inch.

(16) Support the gear in its up locked position and adjust the rod end of the actuator piston rod to allow a minimum of 0.06 of an inch actuator travel remaining with the gear up and locked.

(17) Cycle the gear a few times and check down lock, and up stop action, and switch activation. Include short pickup cycles which simulate gear sag pickup in flight. Leave gear up.

(18) Check up switch bracket override action to ensure proper activation.

(19) Extend the gear and check that the actuator piston travel remaining till full extension is not less than .15 of an inch.

(20) Retract and free fall the gear to ensure the retraction link retention spring moves to the assemblies aft causing the down lock link to fully compress and the drag links to lock in their over center position.

(21) Turn the nose gear full travel and make sure the clearance between the steering arm assembly and the tiller track is between 0.06 and 0.03 of an inch at both the left and right stops. (See the Rigging Tiller Track - Steering Arm Assy. Gap view, “Figure 5” on page 322012.)

CAUTION: THE TILLER, TILLER ROLLER, AND STEERING ARM CHANNEL CAN BE DAMAGED IF NOSE WHEEL TURN LIMITS ARE EXCEEDED WHEN TOWING THE AIRPLANE WITH POWER EQUIPMENT. INSPECT, ADJUST, REPAIR OR REPLACE AS REQUIRED.

(22) Verify free and correct movement of the tiller roller between the steering arm channel and the tiller track and up and down the tiller track.

(23) Check adjustment of the nose gear doors per Nose Gear Doors, Adjustment, below.

(24) Lube the system as specified in 12-20-00.

(25) Ascertain the gear is down and locked and check alignment of landing gear. See “Alignment” on page 322016.

(26) Remove airplane from jacks.

E. Alignment

(1) Park the airplane on a smooth level floor which will accommodate the striking of a chalk line.

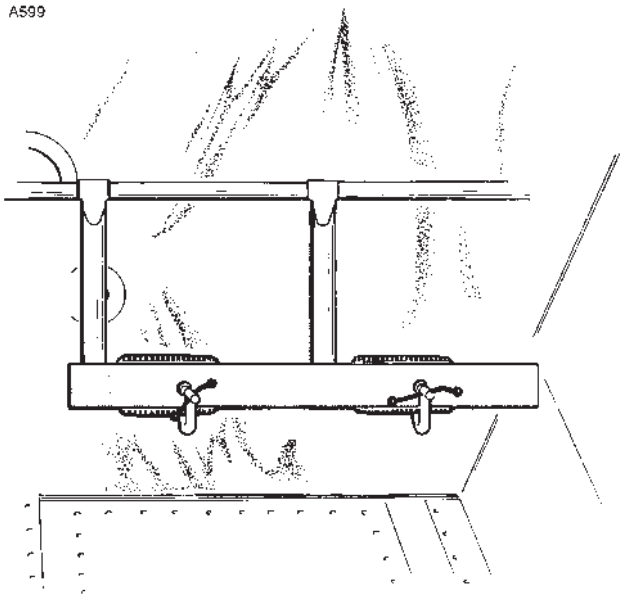
(2) Place airplane on jacks and level airplane laterally and longitudinally. (See 7-10-00 & 8-20-00.)

(3) Extend a plumb bob from the center of the tail skid and mark the contact point on the floor.

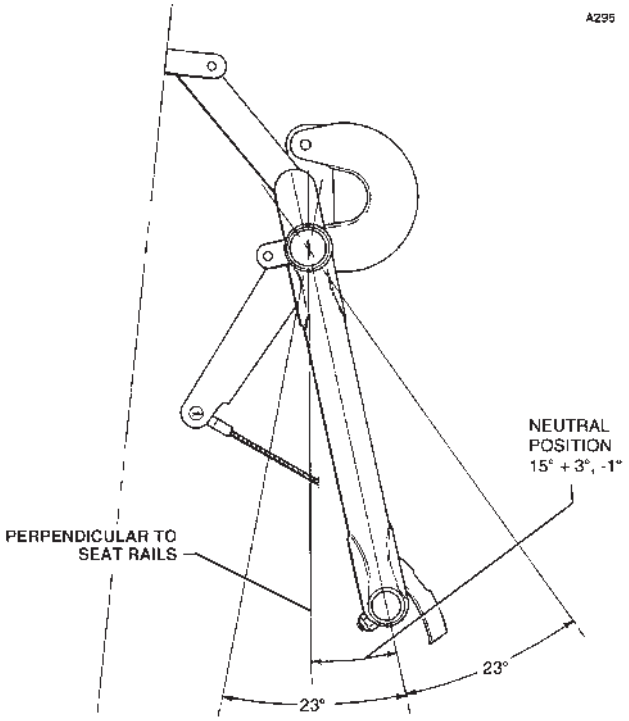
(4) Extend a chalk line from the mark on the floor below the tail skid to point approximately three feet forward of the nose wheel. Allow the chalk line to pass under the wheel at the centerline of the tire. Snap the chalk line.

(5) Clamp rudder pedals to align in a lateral position. (See “Figure 7” on page 322016.)

(6) Adjust the rod end bearings of each steering bungee to align the nose wheel with the chalk line and to bring the rudder pedals into neutral angle fore and aft (see “Figure 8”). There should be no load on the bungee springs. This condition exists when the overall measurement taken between the facing sides of the washers at each rod end of the bungee is 13.71 inches. To align the nose wheel straight forward, stand in front of the nose gear and align the center rib of the tire with the chalk line or lay a straightedge along the side of the tire and parallel with the chalk line. In neutral position, the rudder pedals are tilted aft as shown in “Figure 8”, with the airplane level. Place a bubble protractor against a steering tube to check this angle. One end of each steering bungee must be disconnected and the jam nuts loosened to make this adjustment; do not attempt to make the complete adjustment by means of one bearing, but divide the adjustment between the bearings at each end of each rod. Check that the rod ends have sufficient thread engagement by determining that a wire will not go through the check hole in the rod. Reinstall bungees, tighten and safety the jam nuts.



Clamping Rudder Pedals in Neutral Position
Figure 7



Rudder Pedals in Neutral Angle
Figure 8

(7) To check the nose gear steering for its correct degree of maximum travel, right and left use the wheel pivot point as the center point and draw a line at the travel degree angle on each side of the chalked centerline. Use a tow bar to turn the nose gear full travel left and right and overcome the bungee system. Should travel in one direction be excessive and not enough in the other, check the steering arm and steering bungees for damage. See 12-20-00 for lubrication instructions of the bungee assemblies.

F. Repair

Repairs to the landing gear are limited to reconditioning of parts such as smoothing out minor nicks and scratches, repainting areas where paint has chipped or peeled and replacement of parts.

3. Nose Gear Doors

See “Figure 2” on page 32206.

A. Inspection

(1) Clean all components with a suitable cleaning solvent.

(2) Inspect doors for damage, loose or damaged hinges and brackets.

(3) Inspect retraction rods for damage and rod end bearings for corrosion.

(4) Check door tension spring for wear and tension below minimum allowable tolerance. Reject spring if load tension is below 12 ± 2 pounds with spring extended to 4.1 inches.

(5) Check general condition of actuating tube assemblies and roller assembly.

B. Removal

(1) With nose gear extended, disconnect door retraction rods from doors by removing attaching hardware.

(2) To remove doors, straighten bent end of hinge pins and pull the pins out from the opposite end.

(3) Disconnect spring from arm of upper nose gear actuating tube assembly.

(4) Disconnect link assembly from the upper nose gear actuating tube and remove lower nose gear actuating tube assembly with roller attached.

(5) Remove upper actuating tube assembly.

C. Installation

(1) Install the upper nose gear actuating tube assembly in position between the two bearing blocks and secure with attaching hardware.

(2) Install lower nose gear actuating tube assembly in position between the two bearing blocks and secure with attaching hardware.

(3) Insert lower link assembly into upper link assembly and adjust as necessary to obtain a dimension of 4.60 inches between the centerline of each link. Tighten locknut. (See “Figure 2” on page 32206, Sketch A.)

(4) Install assembled link assembly between the two upright arms of the upper actuating tube assembly and secure with attaching hardware. Insert the lower link between the two upper holes in actuating tube assembly and secure with attaching hardware.

(5) Install roller directly below link on lower actuation tube assembly. Secure in position with attaching hardware making certain the roller is free to turn.

NOTE: If cracks or any signs of wear are evident, the roller must be replaced.

(6) Adjust both retraction rods to obtain a dimension of 9.15 inches as shown. Attach upper end of retraction rod to arm of upper actuation tube assembly. The lower end should be attached to the door bracket. Install tension spring.

(7) Install gear doors by positioning hinge halves and inserting hinge pin. A new hinge pin should be used. Bend end of hinge pin to secure door in place.

D. Adjustment

(1) Place airplane on jacks.

(2) With link assembly and retraction rods adjusted as explained in Installation, above. The gear should travel through the door opening with a clearance of 0.25 inch between the gear and door as their closest point.

(3) If clearance between gear and door is less than 0.25 inch, remove washers from stop bolt until the specified clearance is obtained. If clearance between the gear and door exceed the specified clearance, add washers to the stop bolt.

(4) If doors sag when fully retracted, tighten link assembly. If doors are too tight, loosen link assembly.

(5) Check all rod ends for adequate thread engagement and for safety and tightness of jam nuts.

E. Repair

(1) Repairs to doors are limited to replacing hinges and brackets.

(2) Repairs to the retraction mechanism are limited to painting and replacement of component parts.

EXTENSION AND RETRACTION

1. Nose Gear Actuating Cylinder

See “Figure 1” on page 32301.

A. Removal

(1) Place airplane on jacks. (See Jacking, 7-10-00.)

(2) Disconnect hydraulic lines from actuating cylinder and cover open line ends to prevent contamination.

(3) Disconnect operating rod end from the bracket on the trunnion assembly by removing attaching bolt and nut.

(4) Disconnect cylinder from the link assembly. The down lock spring and down lock link are also attached to this link assembly. After removing the cylinder, it is suggested the spring and link be temporarily reinstalled until the cylinder is ready for reinstallation.

(5) Remove the cylinder from the wheel well.

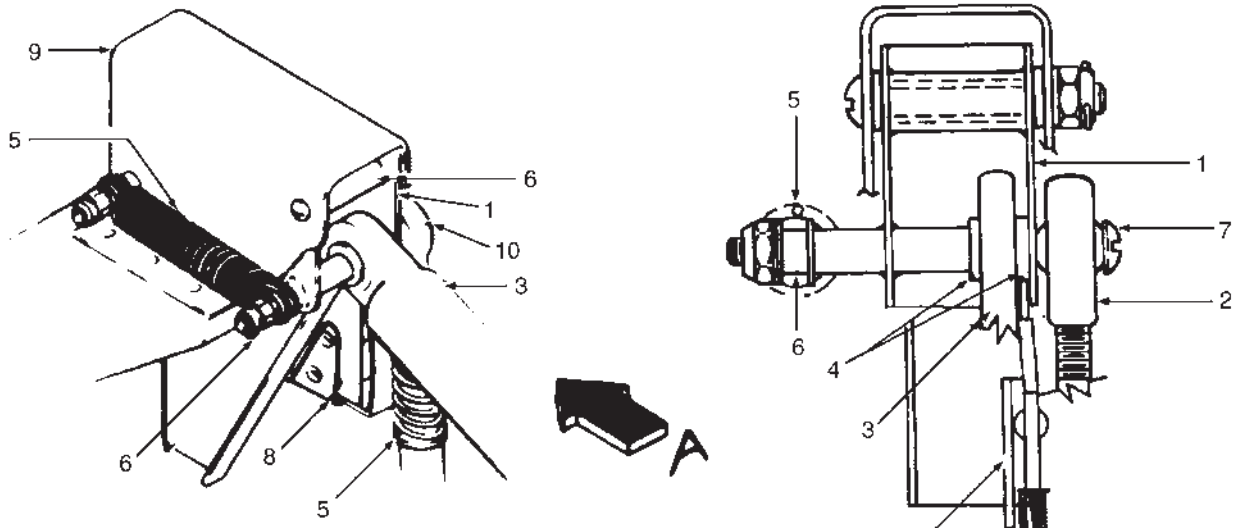
B. Installation

(1) See Step (4) in Removal, above. Remove bolt far enough to position clevis end of actuating cylinder in the link assembly. Insert bolt with callouts arranged as illustrated in “Figure 1” on page 32301.

(2) Insert the operating rod end into the bracket on the trunnion assembly and secure with bolt, nut and washers.

(3) Connect hydraulic lines to their respective fittings on the actuating cylinder.

(4) Check adjustment of cylinder rod end. (See 32-20-00, Nose Landing Gear, Installation and Rigging.)



1. Link Assembly
2. Link
3. Actuating Cylinder
4. Washers
5. Spring
6. Bushing
7. Bolt
8. Switch
9. Mount Assembly
10. Rod End Bearing

Nose Gear Actuating Cylinder Installation
Figure 1

(5) Operate pump to purge system of air and check fluid level in reservoir.

(6) Remove airplane from jacks.

2. Main Gear Actuating Cylinder

A. Removal

(1) Place airplane on jacks. (See Jacking, 7-10-00.)

(2) Disconnect hydraulic lines from actuating cylinder and cover open line ends to prevent contamination.

(3) Disconnect gear down lock spring from swivel fitting at upper end of spring.

(4) Remove down lock spring swivel fitting and disconnect cylinder operating rod end from upper side brace retraction fitting by removing attaching nut, washer and bolt.

(5) Disconnect cylinder from its attachment by removing nut and bolt.

(6) Remove cylinder from wheel well.

B. Installation

(1) Attach the cylinder to its attachment fitting in the wheel well using bolt and nut.

(2) Attach the operating rod end and down lock spring swivel fitting to the upper side brace retraction fitting using washer and nut. The swivel fitting must be free to rotate.

(3) Connect down lock spring to swivel fitting.

(4) Check adjustment of cylinder rod end.

(5) Operate pump and purge system of air. Check fluid level in reservoir.

(6) Remove the airplane from jacks.

3. Gear Actuating Cylinder Service

Disassembly, Assembly, and Cleaning, Inspection and Repair instructions for actuators P/N 96860-002 and 96860-003 (i.e., Cleveland SFA232-5) are found in Cleveland Wheel and Brakes publication: Component Maintenance Manual - CMSFA232-5 (011-00504) available from the vendor. See Vendor Publications in the Introduction.

4. Landing Gear Retraction System Functional Test

A. Seneca IV's and Seneca V S/N's 3449001 thru 3449300, less 3449161

(1) Setup

Before proceeding with this test, ensure that the:

(a) Tires and struts are properly inflated (See 6-00-00 and 12-10-00).

(b) The nose gear, main gear, limit switches and nose gear steering are properly adjusted (See the respective sections in this chapter).

CAUTION: TO PREVENT ACCIDENTAL ACTIVATION, PULL THE FOLLOWING CIRCUIT BREAKERS: SURFACE AND PROP DEICE, WSHLD HEAT, AND PITOT HEAT.

(c) The BATT switch and all other electrical switches are OFF and remain OFF throughout the test.

(d) Raise the aircraft on jacks per 7-10-00.

(e) Connect a 50-ampere minimum power source of the correct voltage to the aircraft's electrical system through the external power supply.

NOTE: During all tests, add hydraulic fluid MIL-PRF-5606 to reservoir as necessary to keep fluid level full.

CAUTION: AFTER FILLING RESERVOIR, TIGHTEN DIPSTICK, THEN BACK OFF 1 1/2 TURNS. THIS ALLOWS THE RESERVOIR TO BE VENTED.

(2) Procedure

NOTE: Perform the following tests and checks in the sequence shown. If the system fails to respond as indicated, the malfunction must be corrected before proceeding with the next step.

(a) Place the gear selector in the DOWN position.

(b) Place the throttles in a closed position.

(c) Ensure that the GEAR PUMP circuit breaker is IN. Check:

1) Three green gear safe lights – ON.
2) Red GEAR WARN annunciator is OFF.
3) Gear warning horn does NOT sound.
4) Hydraulic pump does NOT operate.

(d) Place gear selector switch in the UP position.

1) Three green gear safe lights – OFF.
2) Red GEAR WARN annunciator is ON.
3) Gear warning horn SOUNDS.
4) All gears retract fully and nose gear doors close.
5) Hydraulic pump STOPS operating

(e) Move left throttle to its mid-travel position. Check warning horn continues to sound and red GEAR WARN remains ON.

(f) Close left throttle. Move the right throttle to its mid-travel position. Check warning horn continues to sound and red GEAR WARN remains ON.

(g) Move both throttles to their mid-level position. Check warning horn STOPS sounding and red GEAR WARN goes OUT.

(h) Leave gear up for five minutes. Check hydraulic pump motor does NOT operate at any time. (If pump motor does operate, there is a leak in the up line or a malfunctioning component in the system.)

NOTE: One momentary pump operation is allowable during the five minute gear up period, provided that red GEAR WARN is NOT ON and there is no repeated pump operation for a subsequent fifteen minute period.

(i) Pull the GEAR PUMP circuit breaker OUT.

(j) Pull the gear free fall knob. Check all gears return to the down and locked position, with the down latches engaged.

(k) Place the gear selector switch in the DOWN position. Push the free fall knob in. Push the GEAR PUMP circuit breaker IN. Check:

1) Three green gear safe lights – ON.
2) Red GEAR WARN annunciator is OFF.
3) Gear warning horn does NOT sound.

(l) With the airplane on jacks, and the main gear struts extended, the squat switch on the left main gear will be in the IN-FLIGHT position. Use one of the following means to position the actuating arm in the GROUND position:

1) Insert a suitable wedge under the squat switch leaf.
2) Remove torque link connecting bolt and rotate upper half so the switch tang is free.
3) Loosen squat switch adjustment screws and rotate switch until tang is free.

4) Partially compress the left main gear shock absorber so that squat switch operating tang is free.

(m) Move throttles to closed position.

(n) Place the gear selector switch in the UP position. Check that the:

1) Hydraulic pump does NOT run.
2) Three green safe lights remain on.
3) Red GEAR WARN annunciator is ON.
4) Gear warning horn SOUNDS.

(o) Move throttles to mid travel position. Check the the:

1) Red GEAR WARN annunciator remains ON.
2) Gear warning horn does NOT sound.

(p) Place the gear selector switch down. Check to ensure that the:

1) Red GEAR WARN annunciator remains OFF.
2) The gear warning horn does NOT sound.

(q) Return the squat switch to the IN–FLIGHT position.

(r) Manually “break” the left main gear down lock. Check that the:

3) Hydraulic pump motor RUNS.
4) Correct green safe light is OUT.
5) Red GEAR WARN annunciator remains ON.
6) The horn does NOT sound.

(s) Repeat step above for the right main gear.

(t) Move the gear selector switch to its UP position and turn the landing light switches ON. Check that:

1) The hydraulic pump motor OPERATES.
2) The three green safe lights are OUT.
3) The Red GEAR WARN annunciator remains ON until all gear are UP; then OFF.
4) All gear retract fully in less than 10 seconds.
5) The pump motor STOPS operating after the gear is UP.
6) The warning horn does NOT sound.
7) The gear mounted landing lights are OFF when the gears are in the fully retracted position.

(u) Place the gear selector switch in the DOWN position. Check that:

1) The hydraulic pump motor OPERATES.
2) All gears return to the DOWN and LOCKED position.
3) The hydraulic pump motor STOPS operating after all the gears are DOWN and LOCKED.
4) The three green safe lights are ON.
5) The Red GEAR WARN annunciator is OFF.
6) The warning horn does NOT sound.

NOTE: Any momentary “blinking” of the red and/or green gear lights after the down locks are engaged indicates an improperly adjusted micro switch.

(v) Place annunciator panel DAY/NIGHT switch to NIGHT position. Check that the three green safe lights remain on but become dim.

(w) Place annunciator panel DAY/NIGHT switch to DAY position.

(x) Disconnect auxiliary power source from the aircraft's electrical system.

(y) Remove aircraft from jacks.

B. Seneca V S/N's 3449161 and 3449301 & up (Electro-Mechanical Equipped) (PIR-PPS60033-11, Rev. F.)

(1) Setup

Before proceeding with this test, ensure that the:

(a) Tires and struts are properly inflated (See 12-00-00).

(b) The nose gear, main gear, limit switches and nose gear steering are properly adjusted (see 32-10-00 and 32-20-00, respectively).

(c) Raise the aircraft on jacks per 7-10-00.

(d) Connect a 28 ± 1 vdc regulated power source capable of supplying a minimum of 50 amperes to the aircraft electric system.

NOTE: During all tests, add hydraulic fluid MIL-PRF-5606 to reservoir as necessary to keep fluid level full.

CAUTION: AFTER FILLING RESERVOIR, TIGHTEN DIPSTICK, THEN BACK OFF 1 1/2 TURNS. THIS ALLOWS THE RESERVOIR TO BE VENTED.

(2) Procedure

NOTE: Perform the following tests and checks in the sequence shown. If the system fails to respond as indicated, the malfunction must be corrected before proceeding with the next step.

Avionics Master switch "ON" (remains on through all tests).

(a) All other switches "OFF" and gear selector in the DOWN position.

(b) Move taxi light switch from "OFF" to "ON" position and cabin heat switch to fan position.

(c) LH engine gear warning manifold pressure switch jump wires G4E20 and G4F20 together.

(d) RH engine gear warning manifold pressure switch jump wires G4K20 and G4J20 together.

(e) "Landing Gear Pump" circuit breaker "IN". Verify the following:

1) Three green safe lights on.
2) Red gear unsafe light off.
3) Hydraulic pump does not operate.
4) Taxi light is illuminated.

(f) Gear selector switch "UP". Verify the following:

1) Three green safe lights off.
2) Red gear unsafe light on.
3) Gear warning horn sounds.
4) All gear retract fully and nose gear door closes.
5) Pump motor stops operating.
6) Taxi light extinguished.
7) Heater fan off.

NOTE: During gear retraction, a single momentary hesitation (less than 2 sec.) in pump operation prior to the landing gear reaching the fully retracted position is acceptable as long as total retraction time not exceed 10 seconds.

(g) Remove jumper on LH manifold pressure switch.

Verify that the warning horn continues to sound and red unsafe light remains on.

(h) Replace jumper on LH manifold pressure switch and remove jumper on manifold pressure switch RH.

Verify that warning horn continues to sound and red unsafe light remains on.

(i) Remove jumpers on both LH & RH manifold pressure switch.

Verify that warning horn stops sounding and red unsafe light goes out.

(j) Leave the gear up for five minutes.

Verify that pump motor does not operate at any time. (If the pump motor operates at any time during the five minute period, there is a leak in the “up” line of a malfunctioning component in the system.)

NOTE: One momentary pump operation is allowable during this five minute period, provided that the gear unsafe light is not lit and there is no repeated pump operation for a subsequent fifteen minute period.

(k) Pull “Landing Gear Pump” Circuit Breaker “OUT”.

(l) Pull gear free-fall knob.

Verify that all gear return to the “down and locked” position, with “down” latches engaged, taxi light illuminated, heater fan on.

(m) Turn heater fan “OFF”.

(n) Taxi light to “OFF” position.

(o) Gear Selector Switch “DOWN”. (See NOTE below.)

(p) Free-fall knob “IN”. (See NOTE below.)

(q) “Landing Gear Pump” Circuit Breaker “IN”. (See NOTE below.) Verify the following:

1) Three green safe lights on.
2) Red unsafe light off.
3) Warning horn does not sound.

(r) Partially compress the left main gear shock absorber so that the squat switch operating tang is free. Re-install jumpers on LH & RH manifold pressure switches per Item (c) and (d). (See NOTE below.)

(s) Gear selector switch “UP”. (See NOTE below.) Verify the following:

1) Pump does not run.
2) Three green safe lights remain on.
3) Red unsafe light on.
4) Gear warning horn sounds.

NOTE: Acceptable alternatives for steps (o) thru (s) are as follows:

a) Remove the torque link connecting bolt and rotate the upper half so the switch tang is free.

b) Loosen the squat switch adjustment screws and rotate the switch until the tang is free.

(t) Remove jumpers on both LH & RH manifold pressure switches. Verify the following:

1) Red unsafe light stays on.
2) Gear warning horn does not sound.

(u) Gear selector switch “DOWN”. Verify the following:

1) Red unsafe light off.
2) Gear warning horn does not sound.

(v) Release left main gear shock absorber.

(w) “Break” left main gear down lock manually. Verify the following:

1) Pump motor operates.
2) Correct green safe light out.
3) Red gear unsafe light on.
4) Horn does not sound.

(x) Repeat the above step at the right gear down lock.

(y) Repeat step (w) at the nose gear down lock.

(z) Gear selector switch “UP” and taxi light switches on. Verify the following:

1) Pump motor operates.
2) Three green safe lights out.
3) Red unsafe light on until all gear are up; then off.
4) All gear retract fully in less than 10 seconds.
5) Pump motor stops operating after gear is up.
6) Warning horn does not sound.
7) Taxi light off in full retracted position.

NOTE: During gear retraction, a single momentary hesitation (less than 2 sec.) in pump operation prior to the landing gear reaching the fully retracted position is acceptable as long as total retraction time does not exceed 10 seconds.

(aa) Gear selector switch “DOWN”. Verify the following:

1) Pump motor operates.
2) All gear return to the “down and locked” position in 15 seconds or less.
3) Pump motor stops operating after all gear are in the “down and locked” position.
4) Three green safe lights on.
5) Red unsafe light off.
6) Warning horn does not sound.

NOTE: Any momentary “Blinking” of the red/green gear lights after the down locks are engaged indicates an improperly adjusted microswitch.

(ab) Annunciator panel day-night switch in “NIGHT” position.

Verify that three green safe lights remain on, but become dim.

(ac) Rig squat switch on LH main gear so that the switch actuates when the oleo is 7.75-8.00 in. extended (see “Squat Switches Adjustment” on page 32603).

(ad) Master switch “OFF”.

(ae) Disconnect auxiliary power source from the aircraft’s electrical system.

(af) Remove aircraft from jacks.

C. Seneca V (Garmin G1000/G1000 NXi Equipped)

(PIR-PPS60033-11, Rev. F.)

(1) Setup

Before proceeding with this test, ensure that the:

(a) Tires and struts are properly inflated (See 12-00-00).

(b) The nose gear, main gear, limit switches and nose gear steering are properly adjusted (see 32-10-00 and 32-20-00, respectively).

(c) Raise the aircraft on jacks per 7-10-00.

(d) Connect a 28 ± 1 vdc regulated power source capable of supplying a minimum of 50 amperes to the aircraft electric system.

NOTE: During all tests, add hydraulic fluid MIL-PRF-5606 to reservoir as necessary to keep fluid level full.

CAUTION: AFTER FILLING RESERVOIR, TIGHTEN DIPSTICK, THEN BACK OFF 1 1/2 TURNS. THIS ALLOWS THE RESERVOIR TO BE VENTED.

(2) Procedure

NOTE: Perform following tests and checks in the sequence shown. If the system fails to respond as indicated, the malfunction must be corrected before proceeding with the next step.

Avionics Master switch “ON” (remains on through all tests).

Flap fully retracted (stowed position).

Power up the Garmin G1000 PFDs and MFD and enter the Maintenance Mode.

NOTE: Maintenance Mode is entered by entering the softkey sequence “1, 2, 3, 1” on the MFD when the MFD is displaying the initial splashscreen after power-up.

(a) All other switches “OFF”, and gear selector in the “DOWN” position.

(b) Move taxi light switch from “OFF” to “ON” position and cabin heat switch to fan position.

(c) Move air intake lever to open position.

CHECK: Combustion air blower operates.

(d) Set aircraft on jacks. (Ensure squat switch is defeated.)

NOTE: For any steps above where the squat switch operating tang is required to be free, acceptable alternates are:

a) Remove the torque link connecting bolt and rotate the upper half so the switch tang is free.

b) Loosen the squat switch adjustment screws and rotate the switch until the tang is free.

CHECK: Combustion air blower stops operating.

NOTE: Do not leave cabin heat on over 10 seconds after combustion air blower stops operating.

(e) Turn cabin heat switch “off”. Move air intake lever to closed position.

(f) “GEAR PUMP” circuit breaker “IN”.

CHECK: Three solid green circles on EIS strip (“Figure 2” on page 323013 & “Figure 4” on page 323014) and EIS page (“Figure 3” on page 323013 & “Figure 5” on page 323014).

Hydraulic pump does not operate.

Taxi light is illuminated.

(g) Pull PFD1 circuit breaker.

(h) While holding the ENT key on #1 PFD, restore power by closing the PFD1 circuit breaker.

(i) When the words “INITIALIZING SYSTEM” appear in the upper left corner of the PFD, release the ENT key.

(j) Using the FMS knob on the #1 PFD go to the GIA I/O Configuration page of the GIA group.

(k) Verify the DATA box next to CHANNEL “IN* 10” is green and the box next to “IN* 11” is black with the landing gear selected down.

(l) Activate the cursor and use the small FMS knob scroll to select GIA2. Press ENT.

(m) Verify the DATA box next to CHANNEL “IN* 10” is green and the box next to “IN* 11” is black with the landing gear selected down.

(n) Gear selector switch “UP”.

CHECK: While gear is retracting:

Three rectangles with black/white crosshatch on EIS strip (“Figure 6” on page 323014) and EIS page (“Figure 7” on page 323014).

When gear is fully retracted:

Three white hollow circles on EIS strip (“Figure 8” on page 323015) and EIS page (“Figure 9” on page 323015).

All gear retract fully and nose gear door closes.

Pump motor stops operating.

Taxi light extinguished.

NOTE: During gear retraction, a single momentary hesitation (less than 2 sec.) in pump operation prior to the gear reaching the fully retracted position is acceptable as long as total retraction time does not exceed 10 seconds.

(o) Verify the DATA box next to CHANNEL “IN* 10” is black and the box next to “IN* 11” is green with the landing gear selected up.

(p) Activate the cursor and use the small FMS knob scroll to select GIA1. Press ENT.

(q) Verify the DATA box next to CHANNEL “IN* 10” is black and the box next to “IN* 11” is green with the landing gear selected up.

(r) Gear selector switch “DOWN”.

CHECK: While gear is extending:

Three rectangles with black/white crosshatch on EIS strip (“Figure 6” on page 323014) and EIS page (“Figure 7” on page 323014).

When gear is down and locked:

Three solid green circles on EIS strip (“Figure 4” on page 323014) and EIS page (“Figure 5” on page 323014).

Pump motor stops operating.

Taxi light is illuminated.

(s) Taxi light to “OFF” position.

(t) Verify flaps are fully retracted and flap position indicator in the MFD shows 0°.

CHECK: Inboard stall vane shall not activate stall warning aural when lifted.

Outboard stall vane shall activate stall warning aural when lifted.

(u) Lower flaps to 10º position and verify flap position indicator in the MFD shows 10°.

CHECK: Inboard stall vane shall not activate stall warning aural when lifted.

Outboard stall vane shall activate stall warning aural when lifted.

(v) Lower flaps to 25º position and verify flap position indicator in the MFD shows 25°.

CHECK: Inboard stall vane activates stall warning aural when lifted.

Outboard stall vane shall not activate stall warning aural when lifted.

(w) Lower flaps to 40º position and verify flap position indicator in the MFD shows 40°.

CHECK: Inboard stall vane activates stall warning aural when lifted.

Outboard stall vane shall not activate stall warning aural when lifted.

(x) Retract flaps.

(y) Gear selector switch “UP”. Repeat paragraphs (t) through (x).

(z) Leave the gear up for five minutes.

CHECK: Pump motor does not operate at any time. (If the pump motor operates at any time during the five minute period, there is a leak in the “UP” line or a malfunctioning component in the system).

NOTE: One momentary pump operation is allowable during this five minute period, provided that the gear unsafe light is not lit and there is no repeated pump operation for a subsequent fifteen minute period.

(aa) Pull “GEAR PUMP” circuit breaker “OUT”.

(ab) Pull gear free-fall knob.

CHECK: Three solid green circle on EIS strip (“Figure 4” on page 323014) and EIS page (“Figure 5” on page 323014). All gear return to the “DOWN AND LOCKED” position, with “DOWN” latches engaged, taxi light illuminated.

(ac) Gear selector switch “DOWN”.

(ad) Free-Fall knob “IN”.

(ae) “GEAR PUMP” circuit breaker “IN”.

CHECK: Three solid green circle on EIS strip (“Figure 4” on page 323014) and EIS page (“Figure 5” on page 323014).

(af) Partially compress the left main gear shock absorber so that the squat switch operating tang is free. (See paragraph (d).)

(ag) Gear selector switch “UP”.

CHECK: Pump does not run.

Three solid green circle on EIS strip (“Figure 4” on page 323014) and EIS page (“Figure 5” on page 323014) remain on.

“CHECK GEAR” CAS message on PFD.

Aural “CHECK GEAR” is sounding.

(ah) Attempt to mute aural “CHECK GEAR” by depressing the “MASTER WARNING” switch. Confirm aural does not mute.

(ai) Gear selector switch “DOWN”.

CHECK: CAS message “CHECK GEAR” extinguished.

Aural “CHECK GEAR” stops sounding.

(aj) Release left main gear shock absorber.

(ak) Partially compress the right main gear shock absorber so that the squat switch operating tang is free. (See paragraph (d).)

(al) Gear selector switch “UP”.

CHECK: Pump does not run.

Three solid green circle on EIS strip (“Figure 4” on page 323014) and EIS page (“Figure 5” on page 323014) remain on.

“CHECK GEAR” CAS message on PFD.

Aural “CHECK GEAR” is sounding.

(am) Attempt to mute aural “CHECK GEAR” by depressing the “MASTER WARNING” switch. Confirm aural does not mute.

(an) Gear selector switch “DOWN”.

CHECK: CAS message “CHECK GEAR” extinguished.

Aural “CHECK GEAR” stops sounding.

(ao) Release right main gear shock absorber.

(ap) Gear selector switch “UP”.

(aq) Gear selector switch “DOWN”, pull “GEAR PUMP” circuit breaker “OUT” mid stroke.

CHECK: After 16 seconds, three solid red circle on EIS strip (“Figure 10” on page 323015) and EIS page (“Figure 11” on page 323015).

Aural Master Warning Tone is sounding.

NOTE: Timing begins when the gear selector is switched “DOWN”.

(ar) Acknowledge failure by depressing the “MASTER WARNING” switch.

CHECK: Three solid red circle on EIS strip (“Figure 10” on page 323015) and EIS page (“Figure 11” on page 323015)remain on.

Aural Master Warning Tone stops sounding.

(as) “GEAR PUMP” circuit breaker “IN”.

CHECK: Pump motor operates.

When pump stops running, three solid green circle on EIS strip (“Figure 4” on page 323014) and EIS page (“Figure 5” on page 323014).

(at) “BREAK” left main gear down lock manually.

CHECK: Pump motor operates.

When pump stops running, three solid green circle on EIS strip (“Figure 4” on page 323014) and EIS page (“Figure 5” on page 323014).

(au) Repeat the above step at the right gear down lock and nose gear down lock.

(av) Gear selector switch “UP” and taxi light switches on.

CHECK: Pump motor operates.

While gear is retracting:

Three rectangles with black/white crosshatch on EIS strip (“Figure 6” on page 323014) and EIS page (“Figure 7” on page 323014).

When gear is fully retracted:

Three white hollow circles on EIS strip (“Figure 8” on page 323015) and EIS page (“Figure 9” on page 323015).

All gear retract fully in less than 10 seconds.

Pump motor stops operating after gear is up.

Taxi light off in full retracted position.

NOTE: During gear retraction, a single momentary hesitation (less than 2 sec.) in pump operation prior to the landing gear reaching the fully retracted position is acceptable as long as total retraction time does not exceed 10 seconds.

(aw) Gear selector switch “DOWN”.

CHECK: Pump motor operates.

While gear is extending:

Three rectangle with black/white crosshatch on EIS strip (“Figure 6” on page 323014) and EIS page (“Figure 7” on page 323014).

When gear is down and locked:

Three solid green circle on EIS strip (“Figure 4” on page 323014) and EIS page (“Figure 5” on page 323014).

All gear return to the “DOWN AND LOCKED” position in 15 seconds or less.

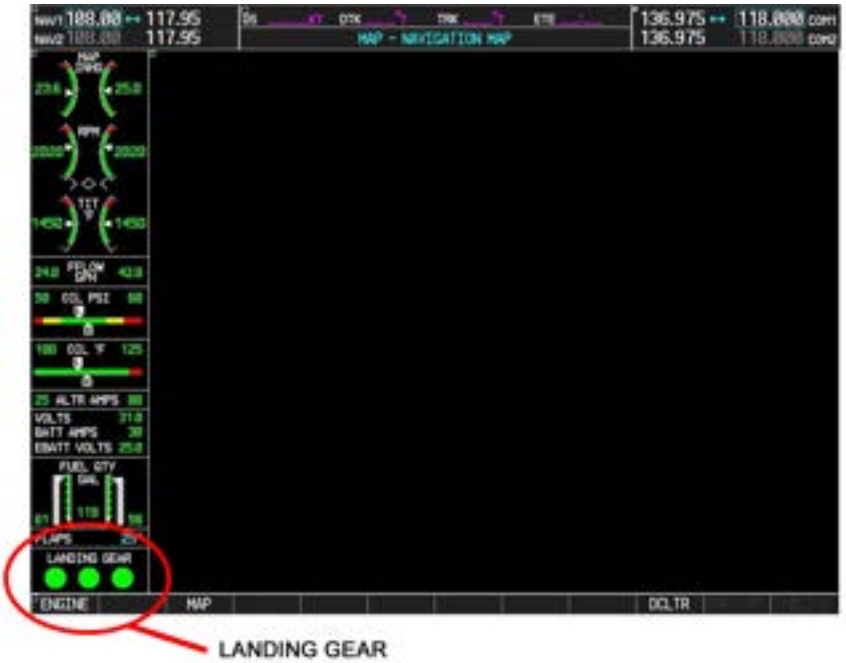
Pump motor stops operating after all gear are “DOWN AND LOCKED”.

(ax) Rig squat switch on LH & RH main gear so that the switch actuates when the oleo is 7.75–8.00 inches extended.

(ay) Avionics Master switch “OFF”.

(az) Disconnect the external (ground) power source.

(ba) Verify all gear are down and locked and remove airplane from jacks per 7-10-00.



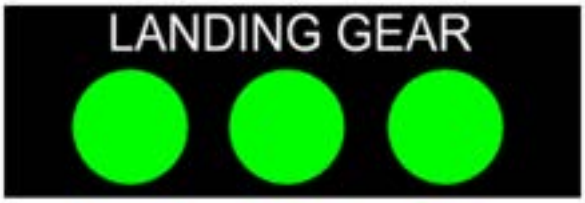
LANDING GEAR
DISPLAY
(SHOWN GEAR DOWN AND LOCKED)

EIS strip on PFD
Figure 2

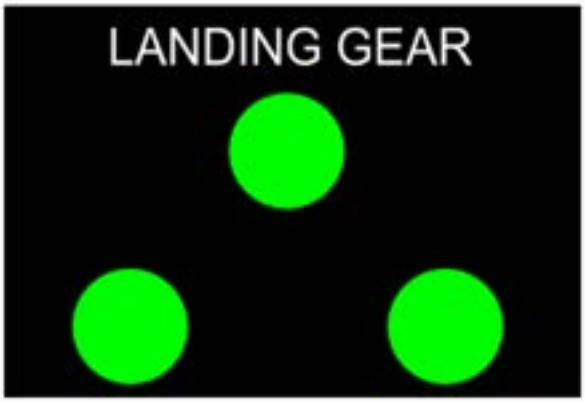


LANDING GEAR
DISPLAY
(SHOWN GEAR DOWN AND LOCKED)

EIS Page on MFD
Figure 3



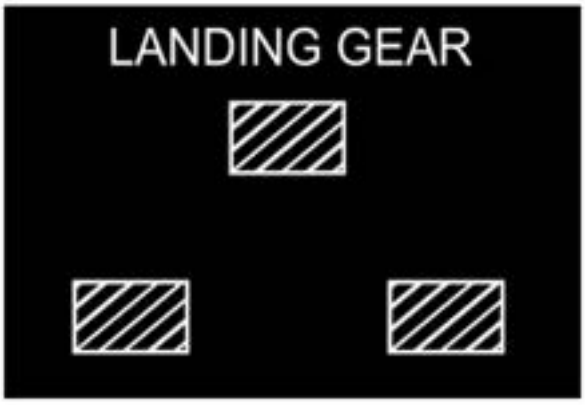
Indication Gear DOWN and LOCK on EIS strip
Figure 4



Indication Gear DOWN and LOCK on EIS Page
Figure 5



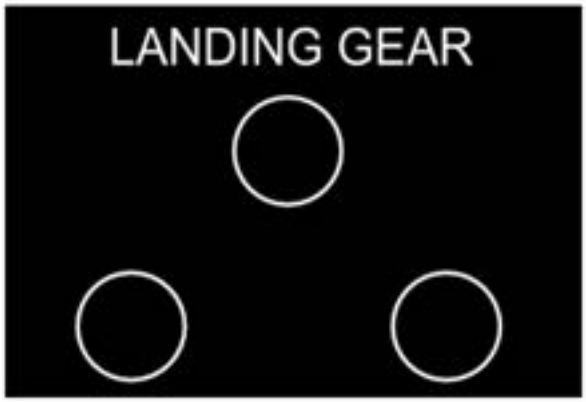
Indication Gear IN TRANSIT on EIS strip
Figure 6



Indication Gear IN TRANSIT on EIS Page
Figure 7



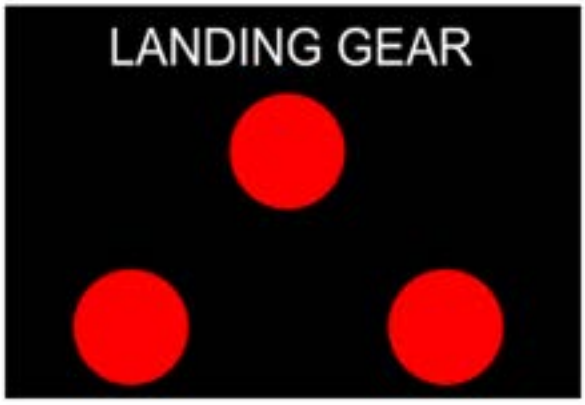
Indication Gear UP on EIS strip
Figure 8



Indication Gear UP on EIS Page
Figure 9



Indication Gear FAIL on EIS strip
Figure 10



Indication Gear FAIL on EIS Page
Figure 11

5. Hydraulic Landing Gear Filling and Purge Procedure

(PIR-PPS60175-1, Rev. B.)

WARNING: ONCE EACH OF THE FOLLOWING STEPS IN THIS SECTION HAS BEEN PERFORMED, THE SYSTEM MUST NOT BE SUBJECT TO AIR CONTAMINATION AGAIN. IF IT IS NECESSARY TO CHANGE THE ACTUATORS POSITION, THIS MUST BE DONE VIA HYDRAULIC POWER SUPPLY PUMP, MANUAL, OR ELECTRIC, USING THE PROPER FLUID AND TAKING CARE NOT TO INTRODUCE AIR BACK INTO THE SYSTEM. IN THE EVENT AIR IS INTRODUCED, THE APPLICABLE FILL AND PURGE PROCEDURE(S) MUST BE REPEATED.

NOTE: A 28 volt (direct current) regulated power source capable of connection to the external power receptacle of the aircraft and of supplying a minimum of 50 amperes and adequate supply of clean MIL-PRF-5606 Hydraulic Fluid, Hydraulic cart, and clean hydraulic fittings are required to accomplish this procedure.

A. Hydraulic Actuators

Either of the following actuator purging methods may be used below.

(1) Bench Method

(a) Connect hoses associated with actuator.

(b) With actuator in retracted position, connect hydraulic power supply pressure port to actuator extend hose. Leave actuator retract hose open to ambient.

(c) Fill actuator with fluid until it is fully extended.

(d) With actuator in extended position, connect hydraulic power supply pressure port to actuator retract hose, and connect hydraulic power supply return port to actuator extend hose.

(e) Using hydraulic power supply, fill actuator with fluid until it is fully retracted.

(f) Repeat process until actuator is full of fluid (no foam or bubbles). Extend actuator to its approximate installed length.

(g) Disconnect hydraulic power supply from actuator, and cap hoses. Ensure minimal air is introduced while capping. Cover rod with protective sheathing.

(2) Gear Swing Method

(a) Install actuator on gear assembly per aircraft rigging instructions.

(b) Connect hydraulic power supply to actuator through most practical available connections (power supply may be connected to actuator through hydraulic lines and hoses already installed on aircraft).

(c) Operate the hydraulic power supply, alternating between extend and retract modes, until fluid is clear (no foam or bubbles). Throughout operation, the fitting on the hydraulic actuator must remain oriented pointing up to prevent air pockets from forming in the actuator. Alternatively, it is permissible to connect actuator to a sump of hydraulic fluid and manually operate the actuator from extended to retracted to fill hydraulic actuator.

(d) Disconnect and cap each hose. Ensure minimal air is introduced while capping (to allow future attachment of system purging jumper hose).

B. Hydraulic Pump

(1) Fill hydraulic pump reservoir to upper white circumferential line when gear is up/retracted. If aircraft is oriented so that fluid in the power pack is not level/parallel with the white circumferential line, fluid level is to be checked below the fill cap on the reservoir.

(2) Attach clear plastic tubes to each pump port and dip in to a container of clean MIL-PRF-5606 Hydraulic fluid.

(3) Operate the pump, alternating between extended and retract mode, until clear fluid is circulated thru the pump (no foam or bubbles).

C. System

(1) Make sure all hydraulic tubes are connected and properly tightened.

(2) If actuator hoses are connected to system, disconnect and cap each hose. Ensure minimal air is introduced while capping.

(3) Install jumper hoses with ball valves in each wheel well where actuator hoses connect to system. Set all valve positions to close.

(4) Open emergency extend valve. Run pump for 10 seconds. DO NOT operate the emergency extend valve while hydraulic pump is running. Close emergency extend valve. Inspect hydraulic reservoir fluid level fill as necessary.

(5) Open nose gear ball valve. Run pump for 10 seconds. Close nose gear ball valve. Inspect hydraulic reservoir fluid level fill as necessary.

(6) Open right main gear ball valve. Run pump for 10 seconds. Close right main gear ball valve. Inspect hydraulic reservoir fluid level fill as necessary.

(7) Open left main gear ball valve. Run pump for 10 seconds. Close left main gear ball valve. Inspect hydraulic reservoir fluid level fill as necessary.

(8) Disconnect power from system.

(9) Disconnect jumper hoses and reconnect with actuator hoses to system. (Disconnect hoses one at a time or fluid will spill and system purge must be repeated). Retorque and inspect any fittings which were disconnected during the filling process.

(10) Once all actuator hoses are reconnect to system, swing landing gear 3 times and note if operation is smooth. If operation is not smooth, let system sit for one hour and retest. If system still does not operate smoothly repeat system purging procedure.

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WHEELS AND BRAKES

1. Wheels

(PIR-PPS50025, Rev. AM)

A. Main Wheel Assembly

(1) Removal and Disassembly

See “Figure 1” on page 32402.

(a) Place airplane on jacks. (See 7-10-00.)

(b) Remove the cap bolts joining the brake cylinder housing and the back plate/lining assemblies. Remove the back plate from between the brake disc and wheel.

(c) Remove the dust cover and cotter pin that safeties the axle nut. Remove the axle nut and bushing and slide the wheel from the axle.

WARNING : DO NOT ATTEMPT WHEEL DISASSEMBLY UNTIL THE TIRE HAS BEEN COMPLETELY DEFLATED. SERIOUS INJURY OR EQUIPMENT DAMAGE CAN OCCUR.

(d) Completely deflate the tire.

CAUTION: DO NOT PRY BETWEEN THE WHEEL FLANGE AND TIRE BEAD. THIS MAY RESULT IN DAMAGE TO THE WHEEL AND TIRE.

(e) To pull the wheel halves from the tire, first remove the inner half, then the outer half. To separate the wheel halves, apply pressure in even increments around the entire sidewall of the tire, as close as possible to the tire bead. Once the bead is broken, the tire should slide progressively easier.

(f) Mark the wheel halves to note the relationship to each other for reassembly. Remove the through bolts holding the wheel halves together. Remove the tire.

(g) Wheel bearing assemblies may be removed from each wheel half by removing snap rings, grease seal rings, felt grease seals, and bearing cone.

(h) Bearing cups should only be removed if they require replacement. See steps, below.

(2) Bearing Cup Replacement

(a) Removal:

1) Insert wheel half into boiling water for 15 minutes or place in an oven not exceeding 250°F (121°C) for 15 minutes.

2) Remove from source of heat and invert wheel half. If the cup does not drop out, tap the cup evenly from the axle bore with a fiber drift pin or suitable arbor press.

(b) Installation:

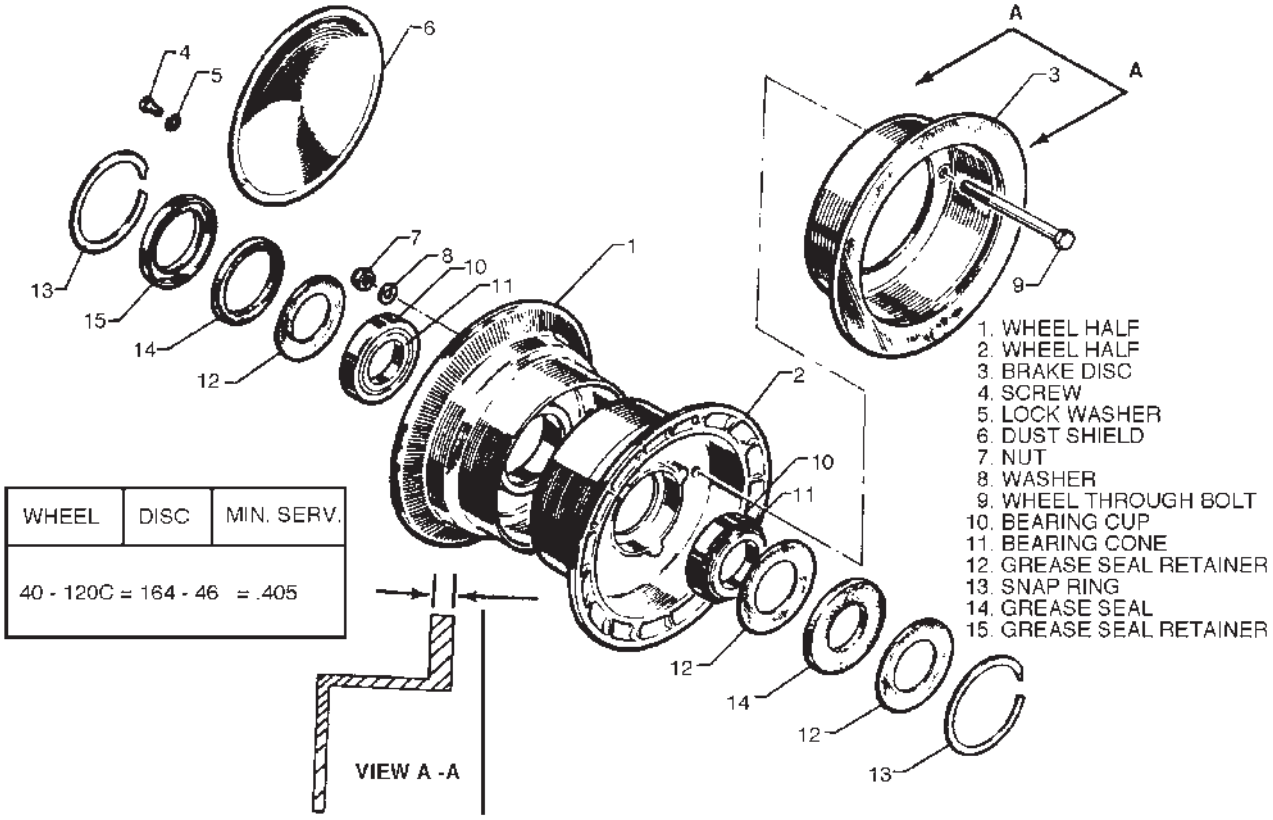
1) Insert the wheel half into boiling water for 15 minutes, or place it in an oven not exceeding 250°F (121°C) for 15 minutes. Chill a new bearing cup in dry ice for a minimum of 15 minutes.

2) When installing a new cup, apply one coat of fluid resistant epoxy primer to wheel half bearing bore.

NOTE: Never paint the working surfaces of bearing cups.

3) Remove the wheel half from the source of heat and the bearing cup from the dry ice.

4) Install the chilled bearing cup into the bearing bore of the heated wheel half. Tap gently into place using a fiber drift pin, making sure the cup is evenly seated against the shoulder of the wheel half. Avoid cocking bearing cup during installation. If the bearing cup will not seat properly in wheel half, repeat the procedure or replace the wheel half assembly.



WHEEL	DISC	MIN. SERV.
40 - 120C = 164 - 46 = .405		

Main Wheel Assembly
Figure 1

(3) Assembly and Installation

See “Figure 1” on page 32402 and “Figure 2” on page 32404.

NOTE: Piper recommends that all tires on an airplane be from the same manufacturer and that the two main tires be the same part number.

NOTE: Tubes must be from the same manufacturer as the tires. Do not mix tube and tire brands. Only new tubes should be used when installing a new tire. Tubes expand in service, taking a permanent set of about 25% larger than the original size. This makes a used tube too large to use in a new tire, which could cause a tube to wrinkle and lead to tube failure.

(a) Determine that the bearing cup is properly installed in each wheel half.

(b) Install slightly rounded inner tube into the tire.

(c) Align the valve on the tube with the red balance dot on the tire to ensure proper balance.

(d) Install the tire and tube on the outer wheel half (which includes the valve stem), and insert the valve stem through the valve hole.

(e) Place the opposite wheel half inside the tire. Align the wheel bolt holes and install the wheel bolts through the wheel halves.

NOTE: Inspect the bolts and washers prior to assembly. Plain or countersunk washers may be used. Some designs use bolts that have a radius between the head and shank, which requires the countersunk washers. Properly oriented washers with countersunk surfaces should sit flush against the bolt head.

(f) Install the washers and nuts and stagger tighten (draw up) the bolts to finger tight (bolt heads are to be on the brake disc side of wheel).

CAUTION: DO NOT USE POWER TOOLS TO TORQUE WHEEL NUTS. CONFIRM THAT THE WHEEL BOLTS ARE TORQUED TO WHEEL MANUFACTURER’S REQUIREMENTS BEFORE INFLATING.

(g) Remove all the pressure in the tire. Torque nuts in a star pattern according to the value specified on the wheel placard, initially to about 45 in-lb on each nut prior to setting the final torque.

(h) Inflate the tire to 70 psi initially to seat the tire bead, then deflate to the proper pressure (see 6-00-00).

(i) Deflate tire to equalize stretch.

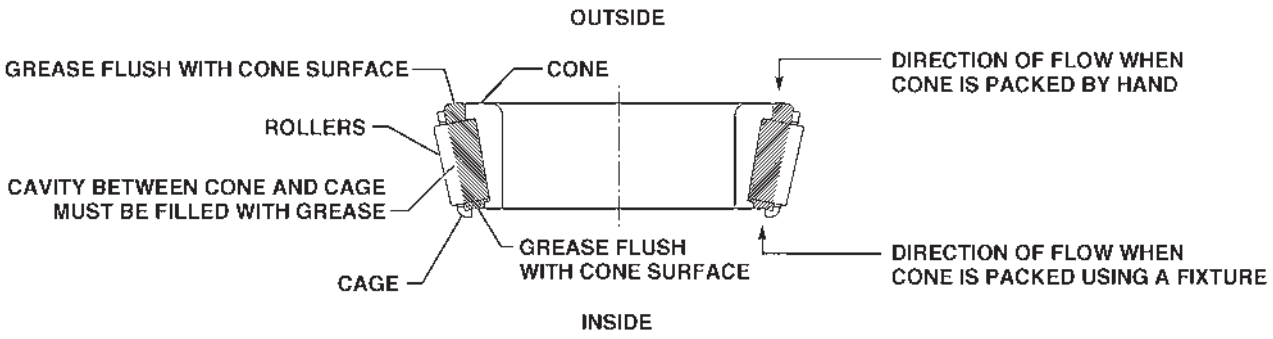
(j) Reinflate to the proper pressure (see 6-00-00).

(k) After a 12-hour stretch period, adjust the pressure, as necessary, to achieve the proper pressure.

NOTE: Be sure the valve stems are tightened after inflating tires. After assembly and inflation is complete, use a soap and water solution on the exterior of tire to perform a final leak check.

(l) After 24 hours, check tire for pressure loss.

NOTE: If the pressure decreases more than 5% in 24 hours, pressure loss could be caused by trapped air between the tire and tube, valve core leakage, or a damaged tube. The assembly should not be placed into service. Check tire and wheel assembly for damaged components and repair or replace as necessary.



Packing Wheel Bearings
Figure 2

(m) Bearing Installation

CAUTION: DO NOT MIX AVIATION WHEEL BEARING GREASES WITH EACH OTHER.

NOTE: Exxon Mobil Aviation Grease SHC 100 has been used by the wheel manufacturer since March 2007 and is the preferred grease. If not available, the preferred alternatives are NYCO GN 3058 and Aeroshell Grease 58 (both specified to SAE AMS3058). If old grease is unknown, thoroughly clean bearing components before packing with new grease.

CAUTION: HANDLE BEARING CONES WITH EXTREME CARE TO PREVENT CONTAMINATION OR DAMAGE.

1) Pack the bearing cones as follows:

NOTE: Pack the bearing cones just before installation to prevent contamination.

NOTE: Bearing cones can be packed by hand or by using a mechanical bearing greaser. The mechanical bearing greaser will do a more thorough job of packing the grease.

a) Thoroughly clean any existing grease or preservative oil from the bearings prior to repacking.

b) Push and force the grease up and out between the rollers, cone and cage.

c) The bearing is properly greased when no voids or daylight can be observed between the rollers and inner and outer races.

d) Disperse excess grease around each end and the tapered sides of each cone.

NOTE: The shaded area in “Figure 2” shows the recommended quantity of grease.

2) Liberally swab the bearing cup, bearing bore hub and grease seal/snap ring areas with bearing grease.

3) If felt seals are used, lightly coat all surfaces of the felt with the wheel bearing grease. If rubber lip seals are used, lightly coat the rubber surfaces with bearing grease.

4) Install the bearing cones, grease seals (felts and rings or rubber lip seals), and snap rings into the wheel hub.

5) Excess grease will squeeze out. Remove the excess grease with an inward rotating movement against the bearing cone ID. Disperse any small amounts of grease on the exterior surface of the grease seal and snap ring and remove any grease from the hub outside surface.

6) Protect bearing grease from contamination. Use caps or plugs as required.

(n) Installation of the wheel/tube assembly:

1) Install wheel assembly onto axle assembly.

2) Rotate the wheel and torque the axle nut to 150 to 200 in-lb to seat the bearing.

3) Back off the wheel nut to zero torque.

4) While rotating wheel, torque wheel nut to 30 to 40 in-lb torque.

5) If nut slot is not aligned with cotter pin hole, rotate wheel nut to align the nearest slot with the cotter pin hole.

6) Install cotter pin.

7) Recheck for the proper pressure (see 6-00-00).

(o) Position brake lining back plates between the wheel and brake disc and the brake cylinder on the torque plate. Insert spacer blocks between the back plates and cylinder and install four bolts to secure the assembly. If the brake line was disconnected, reconnect the line and bleed the brakes.

(4) Inspection

(a) Visually check all parts for cracks, distortion, defects and excess wear.

(b) Check wheel bolts for stripped or damaged threads.

(c) Check the internal diameter of felt grease seals. Replace the felt grease seal if surface is hard or gritty.

(d) Check the tire for cuts, internal bruises and deterioration.

(e) Check the bearing cones and cups for wear and pitting, and relubricate.

(f) Replace any wheel casting having visible cracks.

(g) Inspect the brake disc for cracks. excessive wear or scoring, rust, corrosion, and warpage.

1) Remove rust and blend out nicks, using fine 400 grit sandpaper.

2) Replace disc if cracked or when disc is worn below 0.450 in. (Refer to “Cleaning, Inspection, and Repair” on page 32409.)

B. Nose Wheel Assembly

(1) Removal and Disassembly

See “Figure 3” on page 32407.

(a) Jack the airplane enough to raise the nose wheel clear of the ground (see 7-10-00).

(b) Remove the nut from one end of the axle rod and slide out the rod and axle plugs.

CAUTION: EXERCISE CARE TO AVOID DAMAGING AXLE TUBE ENDS. THIS WILL MAKE REMOVAL AND INSTALLATION EXTREMELY DIFFICULT.

(c) Lightly tap the axle tube out from the center of the wheel assembly by use of an object of near equal diameter.

(d) Remove the spacer tubes and wheel assembly.

WARNING : DO NOT ATTEMPT WHEEL DISASSEMBLY UNTIL THE TIRE HAS BEEN COMPLETELY DEFLATED. SERIOUS INJURY OR EQUIPMENT DAMAGE CAN OCCUR.

(e) Deflate the tire completely.

CAUTION: DO NOT PRY BETWEEN THE WHEEL FLANGE AND TIRE BEAD. THIS MAY RESULT IN DAMAGE TO THE WHEEL AND TIRE.

(f) To separate tire beads from the wheel halves, apply pressure in even increments around the entire sidewall of the tire, as close as possible to the tire bead. Once the bead is broken, the tire should slide progressively easier.

(g) Mark the wheel halves to note the relationship to each other for reassembly. Remove the through bolts holding the wheel halves together. Pull the wheel halves from the tire by first removing the half opposite to the valve stem and then the other half.

(h) Wheel bearing assemblies may be removed from each wheel half by removing snap rings, grease seal rings, felt grease seals, and bearing cone.

(i) The bearing cups should only be removed for replacement. To remove, see “(2) Bearing Cup Replacement” on page 32401.

(2) Assembly and Installation

See “Figure 3” on page 32407.

NOTE: Piper recommends that all tires on an airplane be from the same manufacturer and that the two main tires be the same part number.

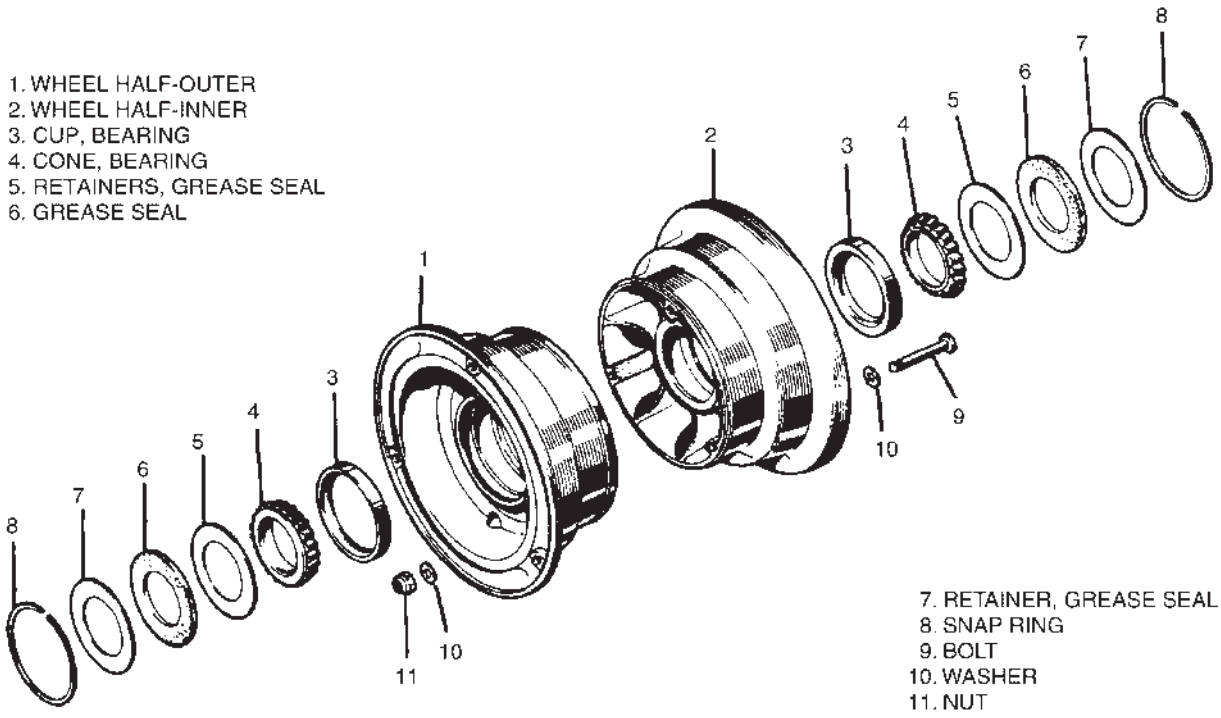
NOTE: Tubes must be from the same manufacturer as the tires. Do not mix tube and tire brands. Only new tubes should be used when installing a new tire. Tubes expand in service, taking a permanent set of about 25% larger than the original size. This makes a used tube too large to use in a new tire, which could cause a tube to wrinkle and lead to tube failure.

(a) Carefully install the bearing cups into each wheel half.

(b) Install slightly rounded inner tube into the tire.

(c) Align the valve on the tube with the red balance dot on the tire to ensure proper balance.

(d) Install the tire and tube on the outer wheel half (which includes the valve stem), and insert the valve stem through the valve hole.



Nose Wheel Assembly (Typical)
Figure 3

(e) Place the opposite wheel half inside the tire. Align the wheel bolt holes and install the wheel bolts through the wheel halves.

NOTE: Inspect the bolts and washers prior to assembly. Plain or countersunk washers may be used. Some designs use bolts that have a radius between the head and shank, which requires the countersunk washers. Properly oriented washers with countersunk surfaces should sit flush against the bolt head.

(f) Install the washers and nuts and stagger tighten (draw up) the bolts to finger tight (bolt heads are to be on the brake disc side of wheel).

CAUTION: DO NOT USE POWER TOOLS TO TORQUE WHEEL NUTS. CONFIRM THAT THE WHEEL BOLTS ARE TORQUED TO WHEEL MANUFACTURER'S REQUIREMENTS BEFORE INFLATING.

(g) Remove all the pressure in the tire. Torque nuts in a star pattern according to the value specified on the wheel placard, initially to about 45 in-lb on each nut prior to setting the final torque.

(h) Inflate the tire to 46 psi initially to seat the tire bead, then deflate to the proper pressure (see 6-00-00).

(i) Deflate tire to equalize stretch.

(j) Reinflate to the proper pressure (see 6-00-00).

(k) After a 12-hour stretch period, adjust the pressure, as necessary, to achieve the proper pressure.

NOTE: Be sure the valve stems are tightened after inflating tires. After assembly and inflation is complete, use a soap and water solution on the exterior of tire to perform a final leak check.

(l) After 24 hours, check tire for pressure loss.

NOTE: If the pressure decreases more than 5% in 24 hours, pressure loss could be caused by trapped air between the tire and tube, valve core leakage, or a damaged tube. The assembly should not be placed into service. Check tire and wheel assembly for damaged components and repair or replace as necessary.

(m) Grease and install wheel bearings per “(m) Bearing Installation” on page 32404.

(n) Place one spacer tube in each side of wheel and position wheel in fork. Align and slide axle tube through spacer tubes and wheel assembly. Install axle plugs and tie rod and secure with nuts. Tighten the nuts until no side play is felt, yet allowing the wheel to rotate freely.

(3) Inspection

(a) Visually check all parts for cracks, distortion, defects and excess wear.

(b) Check wheel bolts for stripped or damaged threads.

(c) Check the internal diameter of felt grease seals. Replace the felt grease seal if surface is hard or gritty.

(d) Check the tire for cuts, internal bruises and deterioration.

(e) Check the bearing cones and cups for wear and pitting, and relubricate.

(f) Replace any wheel casting having visible cracks.

(g) Inspect the brake disc for cracks. excessive wear or scoring, rust, corrosion, and warpage.

1) Remove rust and blend out nicks, using fine 400 grit sandpaper.

2) Replace disc if cracked or when disc is worn below 0.450 in. (Refer to “Cleaning, Inspection, and Repair” on page 32409.)

C. Repair of Main and Nose Wheel Assemblies

Repairs are limited to blending out small nicks scratches. gouges and areas of slight corrosion, plus the replacement of parts which are cracked or badly corroded.

NOTE: Remove rust and blend out small nicks, using fine 400 grit sandpaper.

Wheels may also be repainted if the parts have been repaired and thoroughly cleaned. Paint exposed areas with one coat zinc chromate equivilent primer and one coat of aluminum lacquer.

NOTE: Never paint working surfaces of the bearing cups.

2. Wheel Brake Assembly

Adjustment of brake lining clearance is unnecessary since they are self-adjusting. Inspection of the lining is necessary and may be inspected visually while installed on the airplane. The linings are of riveted type and should be replaced if the thickness of any one segment becomes worn below .099 of an inch or if signs of uneven wear are evident.

A. Removal and Disassembly

See “Figure 4” on page 324010 and “Figure 5” on page 324011.

(1) To remove brake assembly, first disconnect brake line from brake cylinder at the elbow.

(2) Remove bolts joining brake cylinder housing and back plate assembly. Remove back plate from between brake disc and wheel.

(3) Slide brake cylinder housing from torque plate.

(4) Remove pressure plate by sliding it off the anchor bolts of brake cylinder housing.

(5) The pistons may be removed by injecting low air pressure in the cylinder fluid inlet, forcing the pistons from the housing.

(6) Check anchor bolts for wear.

(7) The following procedure should be used when removing anchor bolts:

(a) Position cylinder assembly on a holding fixture. (See “Figure 5” on page 324011.)

(b) Use a suitable arbor press and remove the anchor bolt from the cylinder body.

B. Cleaning, Inspection, and Repair

(1) Clean the assembly with a suitable solvent and dry thoroughly.

(2) Check the wall of the cylinder housing and piston for scratches, burrs, corrosion, etc, that may damage O-rings.

(3) Check the general condition of the brake bleeder screw and lines.

NOTE: Heavy duty wheel assemblies and brake discs may be easily identified by six bolt hole pattern. Standard wheel assemblies and disc brakes have a three bolt hole pattern.

(4) Check the brake disc for wear, grooves, scratches, or pits. The minimum disc thickness of Disc 164-46 used on heavy duty Wheel Assembly 40-120 is 0.405. A single groove or isolated grooves up to 0.031 of an inch deep would not necessitate replacement, but a grooving of the entire surface would reduce lining life and would indicate that the disc should be replaced. Should it be necessary to remove the wheel disc, See Main Wheels, “Removal and Disassembly” on page 32401.”

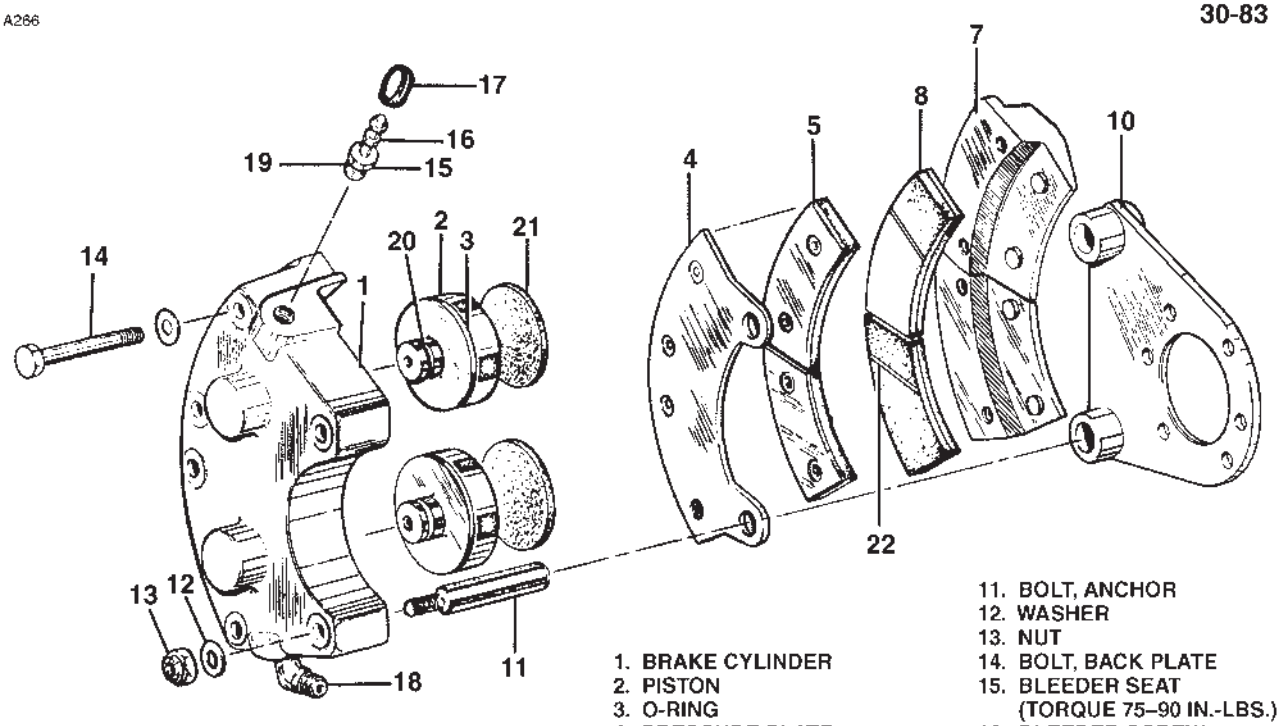
(5) At each periodic maintenance inspection, visually inspect both wearing surfaces of the brake disc for heat checks. Replace brake disc if crack length exceeds 0.800 or crack depth exceeds 0.210. If crack depth is not measurable, replace disc if crack length exceeds 0.400.

NOTE: Any crack, regardless of length and/or depth, extending into the welded seam between the flange and cup, is cause for immediate replacement.

(6) The riveted type lining may be removed from the backing plates by drilling out the old rivets using a 5/32 drill bit. Install a new set of linings using the proper rivets and a rivet set that will properly stake the lining and form a correct flair of the rivet. The snap-on type lining used on optional heavy duty assemblies may be removed by prying loose with a screwdriver or a thin flat wedge. Install the snap-on type by positioning onto the pins and applying pressure to snap into position.

A266

30-83



1. BRAKE CYLINDER
2. PISTON
3. O-RING
4. PRESSURE PLATE
5. LINING, PRESSURE PLATE
6. NOT USED
7. BACK PLATE
8. LINING, BACK PLATE
9. NOT USED
10. TORQUE PLATE ASSEMBLY
11. BOLT, ANCHOR
12. WASHER
13. NUT
14. BOLT, BACK PLATE
15. BLEEDER SEAT (TORQUE 75–90 IN.-LBS.)
16. BLEEDER SCREW
17. BLEEDER CAP
18. ELBOW
19. O-RING
20. SPRING FRICTION
21. INSULATOR
22. EXPANSION GROOVE

Wheel Brake Assembly
Figure 4

(7) Lining Conditioning Procedures

When new linings have been installed, it is important to condition them properly to obtain the service life designed into them. The metallic and organic linings are not conditioned in the same manner because they have different operating characteristics. Separate conditioning procedures are given for metallic and organic linings.

(a) Nonasbestos Organic Linings

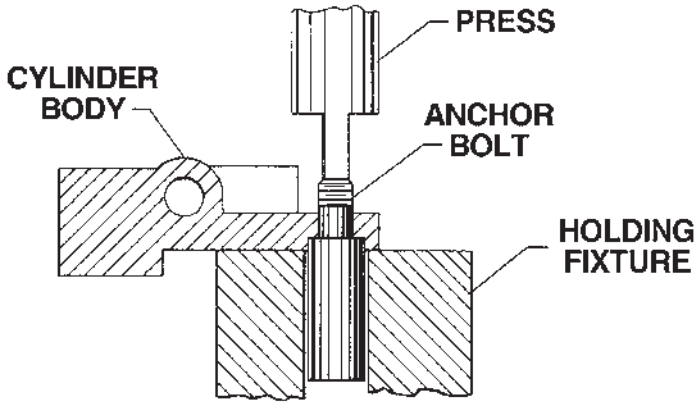
1) Taxi aircraft for 1500 feet with engine at 1700 rpm, applying brake pedal force as needed to develop a 5–10 mph taxi speed.

2) Allow the brakes to cool for 10 to 15 minutes.

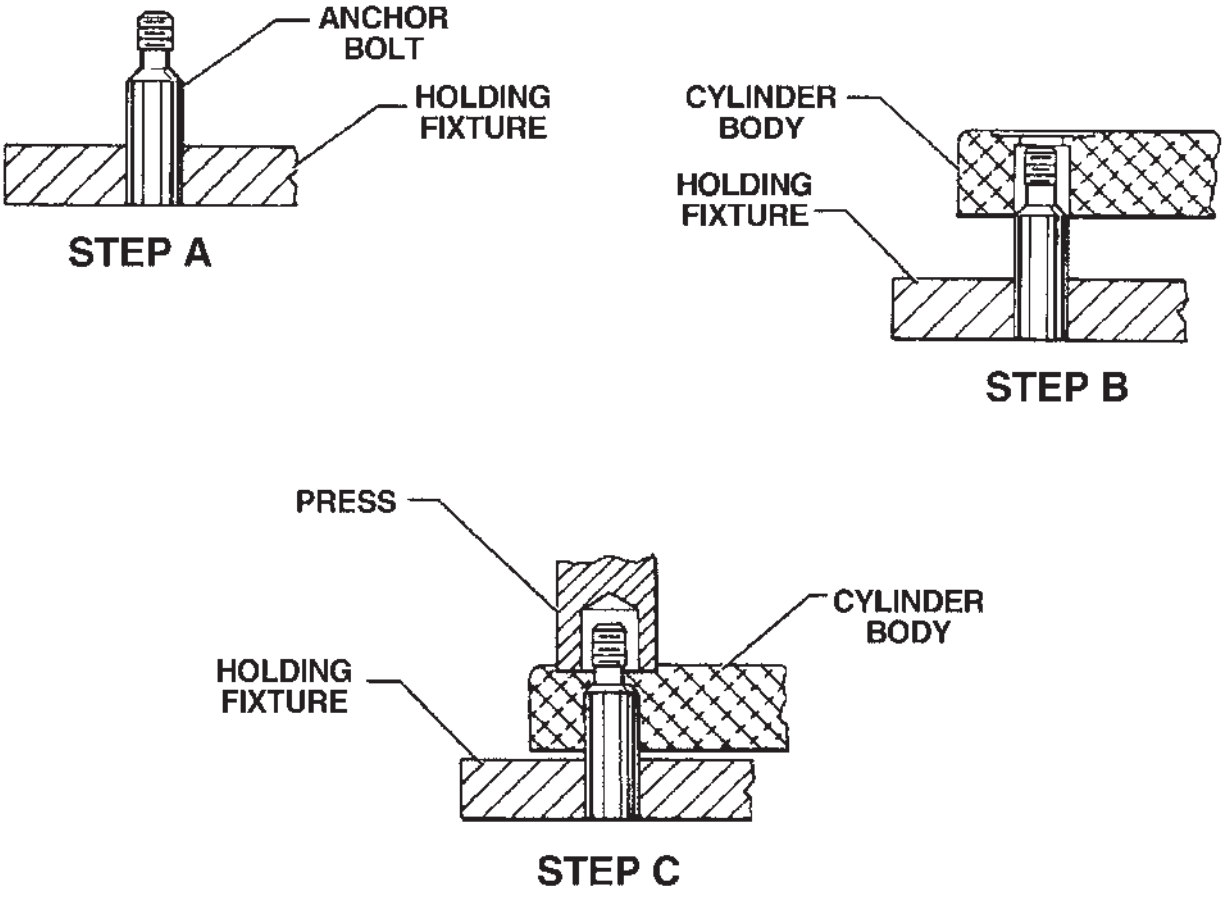
3) Apply brakes and check for restraint at high static throttle. If brakes hold, conditioning is complete.

4) If brakes cannot hold aircraft during static run up, allow brakes to completely cool, and repeat steps 1) through 3).

REMOVAL



INSTALLATION



Removal and Installation of Anchor Bolts
Figure 5

(b) Metallic Linings

1) Perform two (2) consecutive full stop braking applications from 30–35 knots. Do not allow the brake discs to cool substantially between the stops.

2) Allow the brakes to cool for 10-15 minutes.

3) Apply the brakes and check for restraint at high static throttle. If the brakes hold, conditioning is complete.

4) If brakes cannot hold aircraft during static runup, allow brakes to completely cool, and repeat steps 1) through 3).

This conditioning procedure will wear off high spots and generate sufficient heat to create a thin layer of glazed material at the lining friction surface. Normal brake usage should generate enough heat to maintain the glaze throughout the life of the lining.

Properly conditioned linings will provide many hours of maintenance free service. A visual inspection of the brake disc will indicate the lining condition. A smooth surface, one without grooves, indicates the linings are properly glazed. If the disc is rough (grooved), the linings must be reglazed. The conditioning procedure should be performed whenever the rough disc condition is observed. Light use, such as in taxiing, will cause the glaze to be worn rapidly.

C. Rebuilding and Installation

See “Figure 4” on page 324010 and “Figure 5” on page 324011.

(1) If anchor bolts have been removed, they should be reinstalled as follows:

(a) Support anchor bolt in a holding fixture. (See “Figure 5” on page 324011, Step A.)

(b) Align cylinder body over anchor bolt. (See “Figure 5” on page 324011, Step B.)

(c) Using a suitable arbor press, apply pressure on the spot face directly over the anchor bolt. (See “Figure 5” on page 324011, Step C.)

(d) Apply Sandstrom 28A dry film lubricant (MIL-PRF-46147) to anchor bolts and torque plate bushings. Ensure that no other surfaces become contaminated with lubricant. Allow lubricant to air cure in accordance with the manufacturer’s guidelines.

(2) Lubricate piston O-rings with hydraulic fluid (MIL-PRF-5606) and install on pistons. Slide piston into cylinder housing until flush with surface of housing.

(3) Slide pressure plate onto anchor bolts of housing.

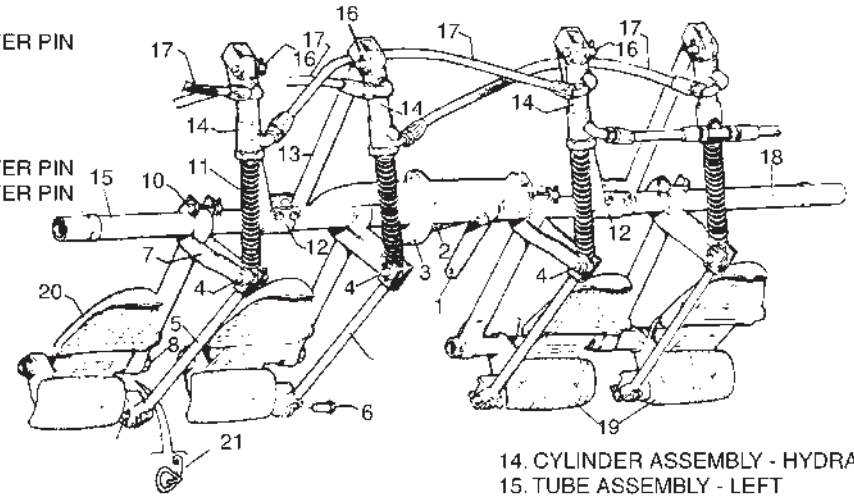
(4) Slide cylinder housing on torque plate.

(5) Position back plate between wheel and brake disc. Install bolts and dry torque to 90 inch-pounds to secure the assembly.

(6) Connect brake line to cylinder housing and bleed brake system as described in this section.

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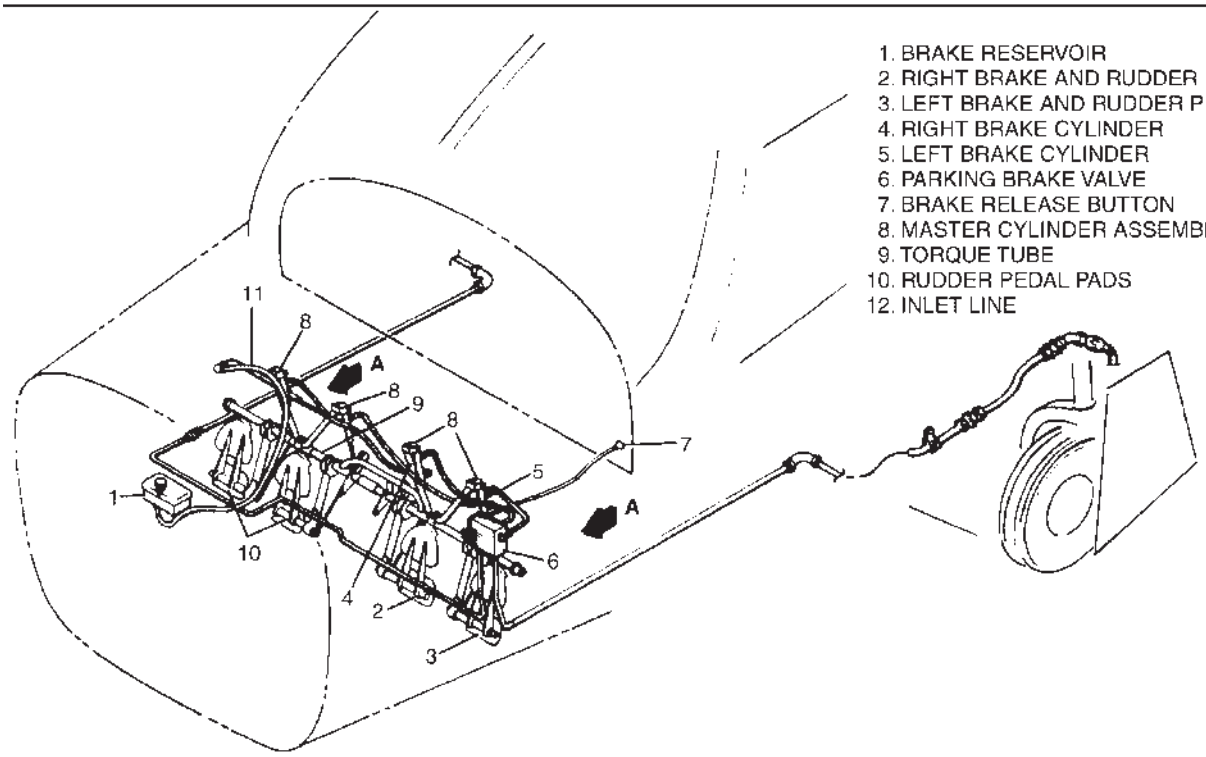
1. ARM, RUDDER CABLE ATTACHMENT
2. ARM, RUDDER CABLE ATTACHMENT
3. ARM, RUDDER STEERING
4. CLEVIS PIN, WASHER & COTTER PIN
5. CLEVIS ASSEMBLY
6. CLEVIS PIN
7. IDLER ARM
8. NUT
9. CLEVIS PIN, WASHER & COTTER PIN
10. CLEVIS PIN, WASHER & COTTER PIN
11. SPRING, RETURN
12. BRACKET
13. BRACE ASSEMBLY



14. CYLINDER ASSEMBLY - HYDRAULIC
15. TUBE ASSEMBLY - LEFT
16. CLEVIS PIN & COTTER PIN
17. HOSE ASSEMBLY
18. TUBE ASSEMBLY
19. PEDAL PADS
20. TOE BRAKE PEDAL
21. SPRING CLIP

VIEW A-A

1. BRAKE RESERVOIR
2. RIGHT BRAKE AND RUDDER PEDAL
3. LEFT BRAKE AND RUDDER PEDAL
4. RIGHT BRAKE CYLINDER
5. LEFT BRAKE CYLINDER
6. PARKING BRAKE VALVE
7. BRAKE RELEASE BUTTON
8. MASTER CYLINDER ASSEMBLY
9. TORQUE TUBE
10. RUDDER PEDAL PADS
12. INLET LINE



Toe Brake Installation
Figure 6

3. Toe Brake Cylinder

See “Figure 6” on page 324014.

A. Removal

(1) Disconnect upper and lower lines from the cylinder being removed. Cap lines to prevent fluid leakage or drain fluid from brake reservoir and master cylinder.

(2) Remove cotter pins and clevis pins securing brake cylinder in position; then remove brake cylinder.

B. Installation

(1) Position brake cylinder at its mounting points and secure in position with clevis pin. Safety clevis pins with cotter pins.

(2) Connect brake lines to cylinder fittings. Bleed brakes as explained in the following paragraphs.

C. Toe Brake Cylinder Service

See “Figure 7” on page 324016. (Cleveland cylinder number 10-30)

(1) Remove the cylinder, per Removal, above.

(2) Disassembly

(a) To disassemble the cylinder, first remove the piston rod assembly by removing the retaining ring from the annular slot in the cylinder housing. Draw the piston rod assembly from the cylinder.

(b) The piston rod assembly may be disassembled by first removing the retaining ring, sleeve, spring, and then the piston assembly, O-ring, and gland, and if desired, the return spring.

(c) Remove the O-rings from the piston and packing gland.

(3) Cleaning, Inspection, and Repair

(a) Clean cylinder components with a suitable solvent and dry thoroughly.

(b) Inspect interior walls of cylinder for scratches, burrs, corrosion, etc.

(c) Inspect general condition of fitting threads.

(d) Inspect piston for scratches, burrs, corrosion, etc.

(e) Repairs to the cylinder are limited to polishing out small scratches and burrs, and replacing seal and O-rings.

(4) Assembly

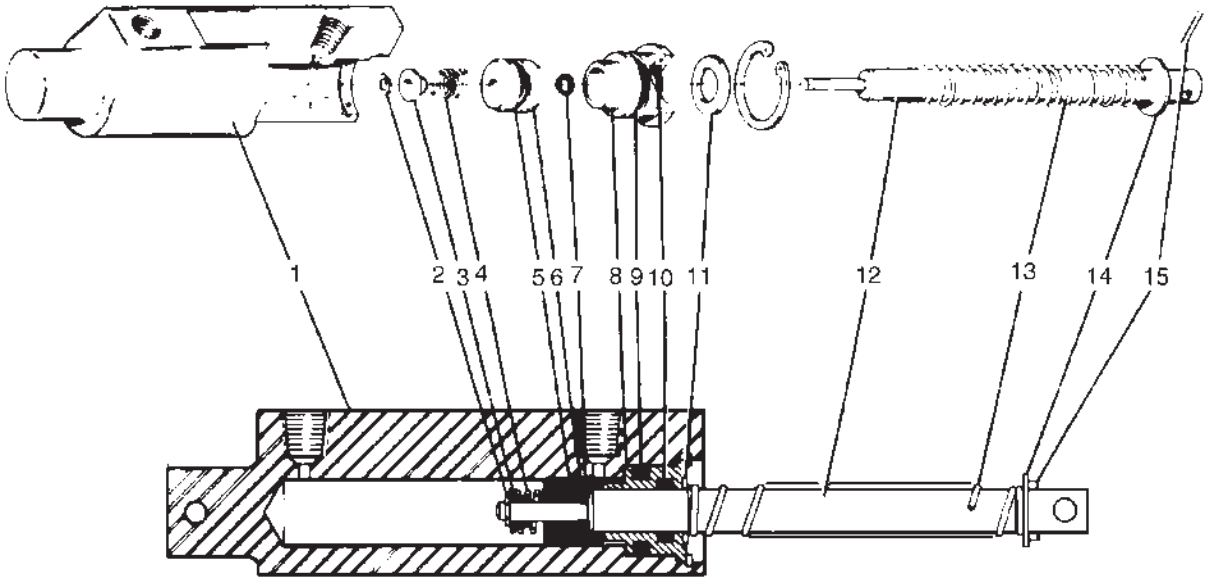
NOTE: Rub a small amount of hydraulic fluid (MIL-PRF-5606) on all O-rings and component parts for ease of handling during reassembly and to prevent damage.

(a) Install new O-rings on the inside and outside of the packing gland and on the outside of the piston.

(b) To assemble the piston rod assembly, install on the rod, in order, the roll pin, washer, spring, washer, packing gland with O-rings, seal, piston assembly with O-ring, spring, and roll pin.

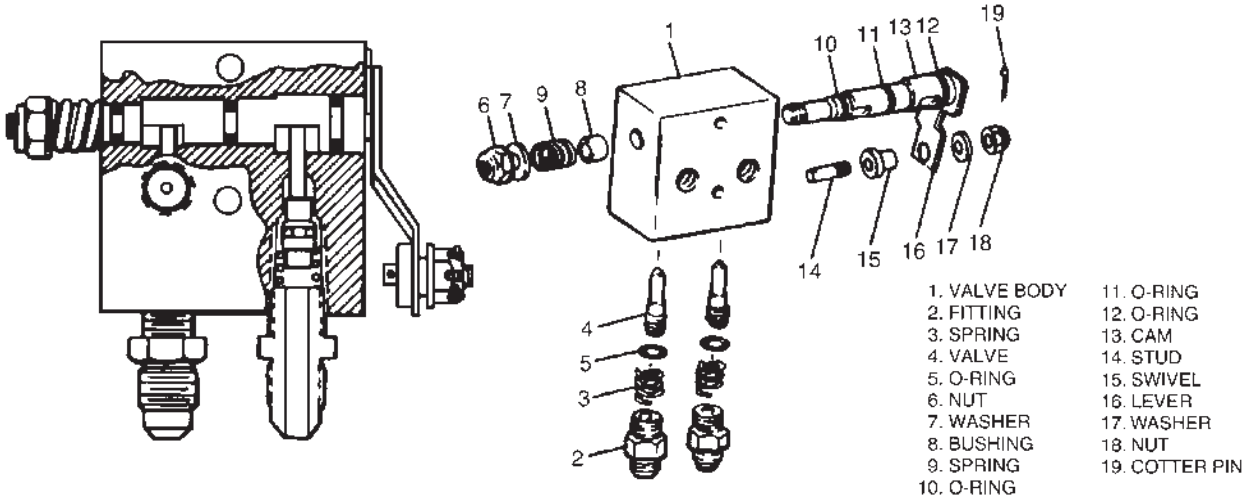
(c) Insert the piston rod assembly in the cylinder and secure with the retaining plug.

(5) Install the cylinder, per Installation, above.



1. HOUSING
2. RETAINING RING
3. SLEEVE
4. SPRING
5. O-RING
6. PISTON
7. O-RING
8. GLAND
9. O-RING
10. O-RING
11. WASHER WIPER
12. ROD
13. SPRING
14. WASHER
15. ROLL PIN

Cleveland 10-30 Toe Brake Cylinder
Figure 7



1. VALVE BODY
2. FITTING
3. SPRING
4. VALVE
5. O-RING
6. NUT
7. WASHER
8. BUSHING
9. SPRING
10. O-RING
11. O-RING
12. O-RING
13. CAM
14. STUD
15. SWIVEL
16. LEVER
17. WASHER
18. NUT
19. COTTER PIN

Parking Brake Valve Assembly
Figure 8

4. Parking Brake Valve

See “Figure 8” on page 324016.

A. Removal

(1) Disconnect the parking brake cable from the valve actuating arm.
(2) Disconnect the fluid lines from the valve.
(3) Remove the screws that attach the valve to its mounting bracket.
(4) Place a protective material over the line openings to prevent contamination of the system.

B. Installation

(1) Attach the valve to the bulkhead mounting bracket with screws.
(2) Connect the fluid lines to the valve.
(3) Connect the control cable to the valve lever and determine that when valve lever fits in the closed detent, parking brake handle is .03 to .06 inch of being full in against stop.

C. Parking Brake Valve Service

See “Figure 8” on page 324016.

Remove valve before servicing.

(1) Disassembly
(a) Remove the two fittings from the outside of the valve body. A valve spring is held in place by the fittings. Use caution not to loosen these when removing the fittings.
(b) From the valve body, remove the valve spring and valve.
(c) To remove the valve cam, remove the nut, washer, bushing and spring and pull the cam from the valve body.

(2) Cleaning, Inspection, and Repair
(a) Clean the valve parts with a suitable solvent and dry thoroughly.
(b) Inspect valve and seat surfaces of valve body for excessive wear and corrosion.
(c) Inspect the cam assembly for burrs, scratches, excess wear, loose operating lever, etc.
(d) Check general condition of valves and springs.
(e) Repair to the valve is largely limited to smoothing burred or scratched surfaces and replacing O-ring.

(3) Assembly
(a) Install O-rings on valve cam.
(b) Lubricate O-rings with fluid (MIL-PRF-5606), insert cam into valve body and secure with spring, bushings, washer and self-locking nut.
(c) Install O-ring on the valve, insert valve in hole of out port, install valve spring and secure with outlet fitting.

5. Brake Bleeding

A. Gravity Procedure

(1) On both main landing gear wheel brake assemblies, attach a clear plastic hose to the brake bleeders and extend into container partially filled with hydraulic fluid, MIL-PRF-5606. The ends of this hose should be submerged in the fluid. Open both bleeders approximately one and one-half to two turns.

(2) Fill the brake reservoir on the firewall with hydraulic fluid, MIL-PRF-5606.

(3) Disconnect the toe brake cylinders from the pedal connection by removing clevis pin, washer and cotter pin.

(4) Invert toe brake cylinder to aid in releasing trapped air in the top of the cylinder.

(5) Check toe brake pedals in the cockpit to ensure pedals are pulled full aft.

(6) Pull the hand brake handle, pumping the master cylinder very slowly approximately 25 times until fluid is observed passing through the clear plastic hoses at the wheel cylinder.

NOTE: Fluid level in the reservoir must be maintained to prevent air from entering the line.

(7) Tighten both wheel bleeders.

(8) Pull hand brake until a firm handle is maintained.

B. Pressure Procedure

See “Figure 9” on page 324019 and “Figure 10” on page 324019.

(1) Place a small clear plastic hose on the vent tube of the brake reservoir and place a second small clear plastic hose on the bleeder fitting on one main landing gear. Place the open ends of these tubes in a suitable container to collect the fluid overflow. Open the bleeder fitting one to two turns.

(2) On the other main gear, slide the hose of the pressure unit over the bleeder fitting then open the fitting one or two turns and pressure fill the brake system with MIL-PRF-5606 fluid.

(3) With fluid continually flowing through the brake system, SLOWLY and together actuate the hand brake and the toe brake pedal of the side being bled, several times, to purge the cylinders of air. On dual brake installations, both right and left pedals must be actuated.

NOTE: By watching the fluid pass through the plastic hose at the fluid reservoir and the bleeder fitting on the gear being bled, it can be determined whether any air is left in the system. If air bubbles are evident, filling of the system shall be continued until all the air is out of the system and a steady flow of fluid is obtained. Should the brake handle remain spongy, it may be necessary to disconnect the bottom of the toe brake cylinders (next to the pedal) and, rotating the cylinder horizontally or even above horizontal, and by use of the hand brake alone, purge the air from the system.

(4) Close the open bleeder fitting on the gear being bled. Close the open bleeder fitting to which the pressure hose is attached; then close the pressure unit and remove the hoses from the bleeder fittings. Check the brakes for proper pedal pressure. Replace the caps over the bleeder fittings.

NOTE: It may be necessary to remove any trapped air in the top of the wheel brake unit by applying pressure to the system with the brake hand lever and slowly opening the bleeder and releasing the hand lever.

(5) Repeat this procedure, if necessary, on the other gear.

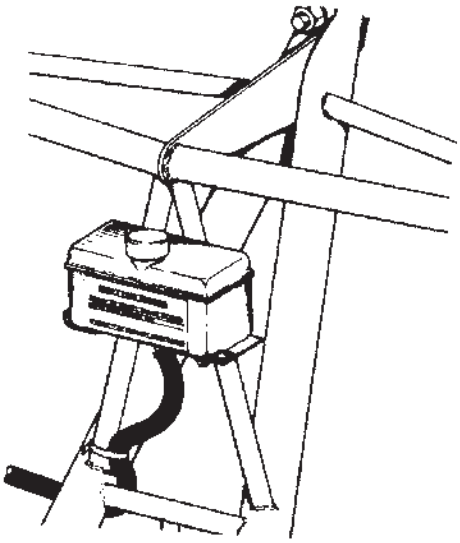
(6) Drain excess fluid from the reservoir to fluid level line with a syringe.

6. Brake System Leak Check

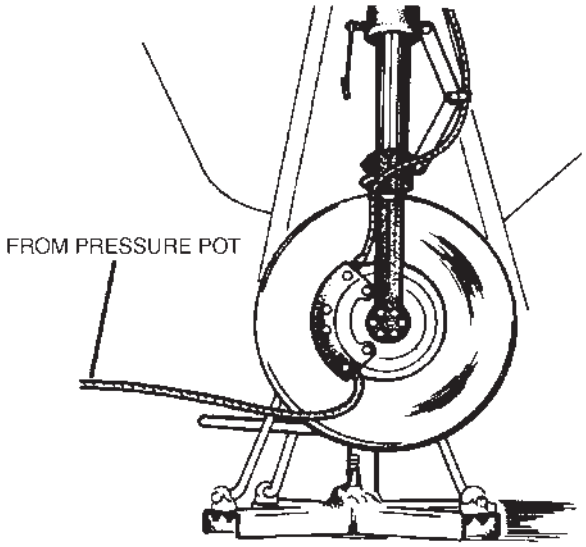
A. Pull firmly on the hand brake until resistance has built up and actuate the locking mechanism.

B. After the system has stood for approximately 10 minutes, the handle should be checked for the same resistance.

C. If the handle is able to be pulled easily or feels spongy, check the system for leaks. Especially check line joints. It is also possible the master cylinder or wheel brake assemblies may have internal leaks.



Brake Reservoir Bleeding
Figure 9



Bleeding Brake
Figure 10

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POSITION AND WARNING

1. Nose Landing Gear Limit Switches

NOTE: All limit switches adjustments should be made with airplane on jacks. Do not bend actuator springs mounted on the limit switches.

A. Up Limit Switch Adjustment

The gear up limit switch is mounted on a bracket attached to the lower inner left tubular member of the nose gear mount, adjacent to the gear roller track assembly or mounted on the stop assembly. (See “Figure 2” on page 32206.)

(1) To facilitate adjustment of the limit switch, disconnect gear doors.

(2) Turn the master switch on; move gear selector switch to the gear up position and raise the landing gear. Turn the master switch OFF.

(3) Block the nose gear in the up position and slowly pull the free fall knob away from the instrument panel. This will relieve hydraulic pressure and permit the main gear to drop.

(4) Loosen the attachment screws of the switch and rotate the switch toward the actuator tang until the switch is heard to actuate. Move the up limit switch upward .02 to .04 inches after actuation. Retighten the switch attachment screws. Remove the block from under the gear and allow it to extend slowly.

(5) Turn master switch on; raise gear and determine that gear limits function properly.

B. Down Limit Switch Adjustment

See “Figure 1” on page 32602.

The nose gear down limit switch is mounted on a bracket located on the forward side of the cabin bulkhead.

(1) With landing gear in the retracted position, pull the free fall valve knob permitting the gear to extend.

(2) Check to determine that the down lock spring returns the body end of the actuating cylinder aft.

(3) Ascertain that the down lock link assembly is fully retracted and that the drag link arms are over center.

(4) In this position, the nose gear green down lock light should energize when the master switch is turned on and the gear selector lever is in the down position.

(5) If the nose gear green down lock light does not energize, loosen the attachment screws of the switch and rotate the switch toward the actuator tang until it is heard to actuate. Tighten the adjustment screws.

2. Main Landing Gear Limit Switches

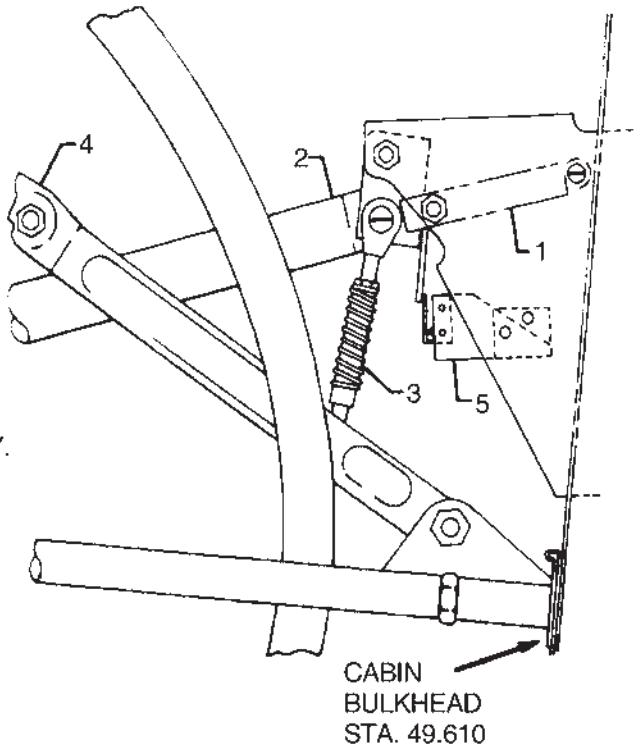
NOTE: All limit switches adjustments should be made with airplane on jacks. Do not bend actuator springs mounted on the limit switches.

A. Up Limit Switches Adjustment

Up limit switches are mounted in the outboard interior of each wheel well. Included in the right side installation is also an up limit or “Flight Switch” for the heater circuit.

The up limit switches are incorporated in the circuit such that the red Gear Unsafe light is extinguished when the switches are activated and the gear selector is in the up position.

The up limit or “Flight Switch” that is included with the right installation is tied into the heater circuit and is described in Chapter 21.



1. DOWNLOCK SPRING
2. ACTUATING CYLINDER
3. DOWNLOCK LINK ASSY.
4. DRAG LINK ARMS
5. MICRO SWITCH

Adjust Nose Gear Down Limit Switch
Figure 1

Adjustment of the up limit switches should be made by loosening the aft bolt and rotating the unit(s) in the correct direction to permit activation of the switches when the fork is 0.88 of an inch from full up.

B. Down Limit Switch Adjustment

See “Figure 2” on page 32604.

The gear down limit switch is mounted on the lower drag link of each main gear. The switch should be adjusted to allow it to actuate when the down lock hook has entered the locked position and is contacting the pin, thus turning the green light on in the cockpit. Adjustment, if necessary, should be made as follows:

(1) Bend/Adjust “Gear Down” micro switch support bracket to clear trunnion as required, ensuring engagement of micro switch tang when gear is extended.

(2) Rig down lock switch to remain actuated when the gear is in the down and locked position.

3. Squat Switches Adjustment

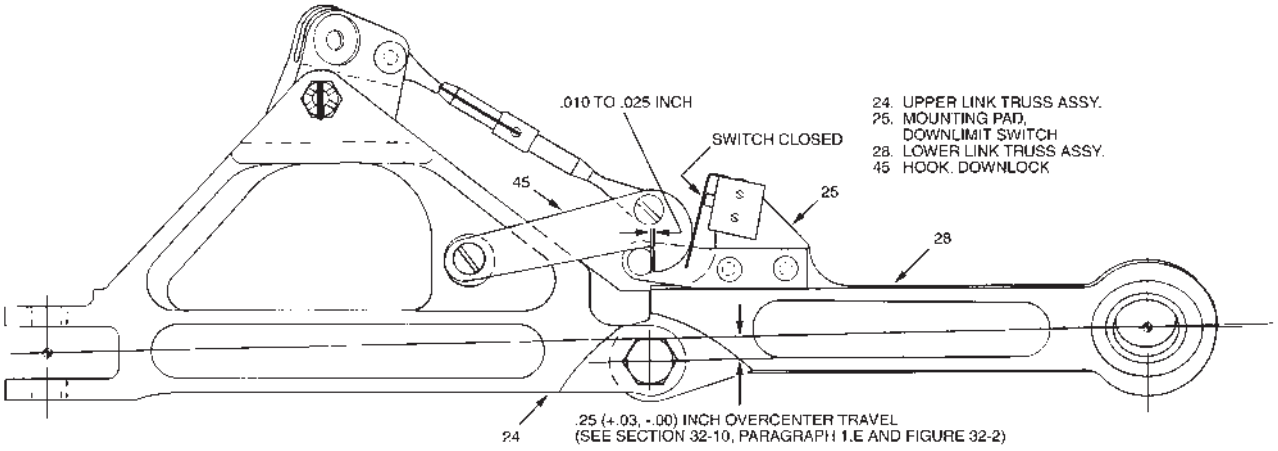
The squat switches are mounted to a bracket at the upper torque link attachment of the left gear and consists of two microswitches which control gear operation and stall warning. Each specific microswitch can be identified by the wire coding as referenced in the specific wiring schematic (See 91-32-60 for Electrical Schematic).

Adjust the squat switches as follows:

A. Compress the strut until the gap between the trunnion housing and fork mating surfaces is 7.75 to 8.00 inches. Retain the strut at this position.

B. Adjust the switches on their mounting bracket to actuate at this point.

C. Extend and compress gear to ensure proper operation.



Adjust Main Gear Down Limit Switch
Figure 2

4. Gear Warning Switches

A. Throttle Micro-Switch

See “Figure 3” on page 32606.

Seneca IV’s and Seneca V S/N’s 3449001 thru 3449300, less 3449161.

The throttle switch is utilized to activate the gear up warning horn if the gear is still up when the throttles are pulled below 14 ± 2 inches manifold pressure.

The microswitch is located in the control quadrant behind the throttle levers. Access to the switch can be made from below and behind the quadrant. The electrical schematic for the landing gear can be found in 91-32-60.

(1) Adjustment

Initially, the Gear Warning Throttle Switch is set for the airplane flying at approach speed and a relative altitude of 1000 feet AGL. When using the following procedures, the same limits should be used. Make sure that during this procedure the airplane is flown at or above 1000 feet AGL in case any problems should develop.

(a) With the aid of a qualified pilot, fly the airplane to an elevation of 1000 feet above the ground and come to approach speed with propellers set for high rpm. Retard the throttles to 14 ± 2 inches of manifold pressure and mark the quadrant cover adjacent to the throttle levers in such a manner so that the levers can be returned to the same position after the airplane is landed and the engines shut down.

(b) Place the airplane on jacks and retract the landing gear.

(c) Reposition the throttle levers at the location which gave the 14 ± 2 inches of manifold pressure per step (a).

(d) With the master switch turned on, loosen the two mounting screws securing the micro switch to the bracket. Move the switch in the direction necessary to make the warning horn operate and tighten the mounting screws.

(e) With the warning horn operating, lower the landing gear to determine whether the horn ceases to operate when the gear are down and locked. Turn OFF master switch and remove airplane from jacks.

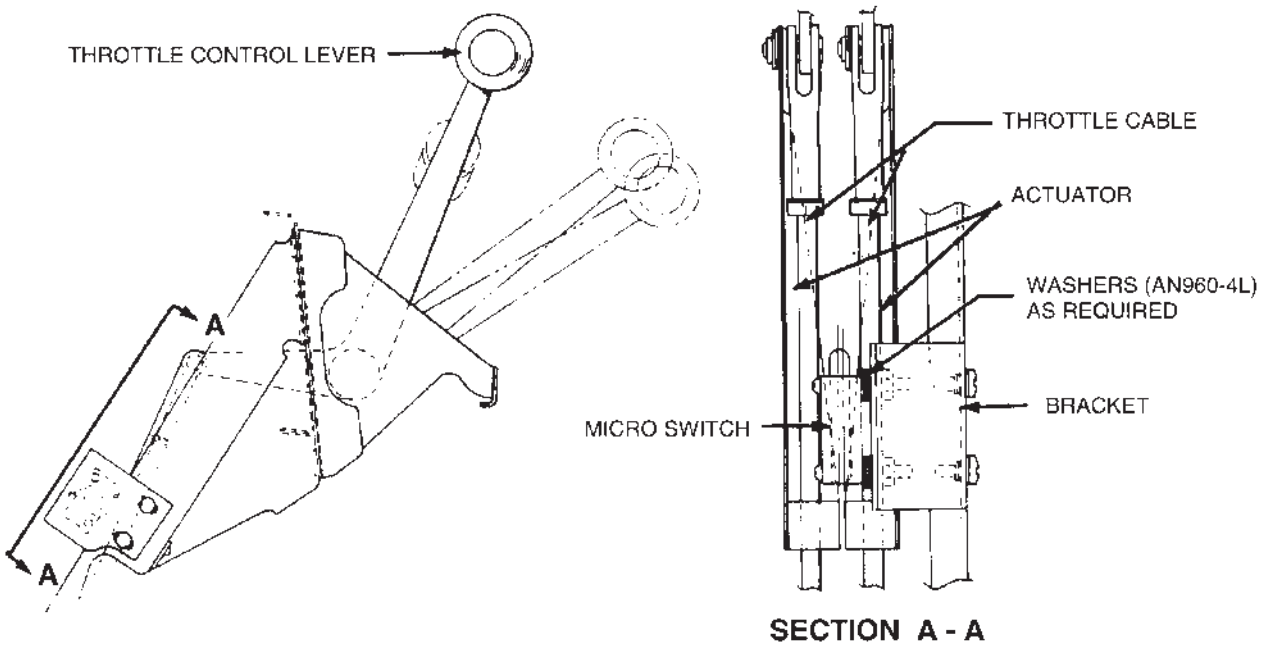
(f) Flight test the airplane to ensure proper operation of the gear warning horn with the gear up and power reduced below 14 ± 2 inches of manifold pressure.

(2) Replacement

(a) Determine and take note of how many washers are between the microswitch and the bracket.

(b) Disconnect electrical wires and mark microswitch position.

(c) Remove screws and install new switch in same position making sure to use same amount of washers.



Throttle Warning Switch
Figure 3

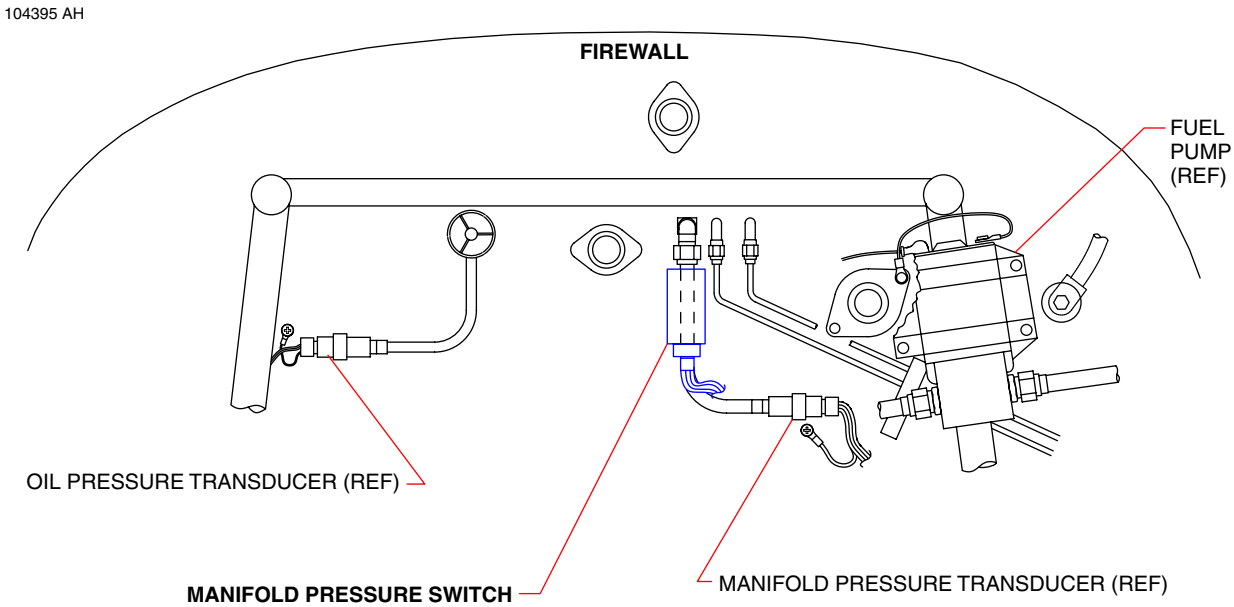
B. Manifold Pressure Switches

See “Figure 4”.

Seneca V S/N’s 3449161 and 3449301 & up.

The Manifold Pressure Switches activate the gear up warning horn if the gear is still up when the throttles are pulled below 14 ± 2 inches manifold pressure.

These switches are located on the back side of the engine firewall as shown. Remove the left or right nacelle hatch cover for access to the appropriate switch. The electrical schematic for the landing gear can be found in 91-32-60.



Gear Warning Manifold Pressure Switches
Figure 4

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CHAPTER

33

LIGHTS

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CHAPTER 33

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CHAPTER 33 - LIGHTS

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FLIGHT COMPARTMENT

This airplane is equipped with pilot and copilot overhead instrument panel/map lights, instrument post/wash lights, switch lights, and avionics lights. The instrument post/wash lights, switch lights, and avionics lights are controlled by independent dimmer controls located on the lower left instrument panel. The two overhead instrument panel/map lights are each controlled by its own dimmer control.

In Seneca IV airplanes and in Seneca V airplanes S/N's 3449001–3449458 and 3449460–3449166, an annunciator panel, located on the upper left side of the instrument panel (Seneca IV) or upper center of the instrument panel (Seneca V), provides a visual warning of possible malfunctions, including failure alert and precautionary warnings. See 31-50-00 for a complete description of its operation.

1. Troubleshooting

When checking any of these light systems the BATT switch must be ON in order to operate the lights. Ensure that the circuit breaker of the affected system is activated.

2. Dome Panel Flood/Map Lights

Two overhead instrument panel/map lights are mounted in the overhead forward fresh air vent panel adjacent to the speaker. A slide control incorporated in the forward part of each fixture may be utilized to expose a small square hole, which directs a beam of light onto the respective pilot's seat area. The light is controlled by a rheostat.

A. Bulb Replacement

(1) Ensure BATT switch is off. Disengage (pull OUT) FLOOD circuit breaker.
(2) Remove three screws retaining lens cover to light fixture.

NOTE: Observe there are two sets of three screws associated with each fixture; one set in an outer ring or circle and one set in an inner ring or circle. Be sure to remove screws from inner ring or circle.

(3) The light bulb is of a bayonet type and removed accordingly.
(4) Replace the bulb. Engage (Push IN) FLOOD circuit breaker. Ensure correct operation of light.
(5) If bulb does not light up, check the following.
  (a) Proper circuit breaker mode.
  (b) Continuity between housing socket and its electrical leads.
  (c) Proper dimmer operation.
(6) Clean lens as necessary.
(7) Install light fixture and secure with three screws previously removed.

B. Rheostat Replacement

(1) Ensure BATT switch is off. Disengage (Pull OUT) FLOOD circuit breaker.
(2) Loosen allen screw and remove switch knob.
(3) Loosen (do not remove) nut securing rheostat to dome panel.

CAUTION: PERMITTING DOME PANEL TO "FLIP" FRONT END DOWN DURING REMOVAL MAY RESULT IN TEARS OR RIPS IN HEADLINER.

(4) Remove and support dome panel. Drop panel just enough to gain access to rheostat. Do not permit panel to hang by electrical wires.
(5) Remove nut from rheostat. Remove rheostat from panel. Remove wires from rheostat.
(6) Install wires on new rheostat. Install rheostat in panel. Install security nut and finger tighten.

(7) Engage (push IN) FLOOD circuit breaker and check correct operation of light. Disengage (Pull OUT) FLOOD circuit breaker.

(8) Install dome panel. Tighten nut securing rheostat to panel

(9) Install rheostat knob and tighten allen nut.

(10) Engage (Push IN) FLOOD circuit breaker.

3. Instrument Post/Wash Lights

The primary means of illuminating flight and engine instruments is with post/wash lights.

A. Seneca IVs and Seneca V S/N's 3449001–3449458, 3449460–3449466

These airplanes are equipped with post lights.

(1) Bulb Replacement

(a) Ensure BATT MASTR switch is OFF and INST. PNL. circuit breaker is disengaged (pulled out).

(b) Remove light cover/shade by pulling straight out. If cover/shade is tight. a slight twisting or turning motion may be required.

(c) Bulb is located in cover/shade. Pull bulb out of cover. Insert new bulb into cover/shade.

(d) Push cover/shade into socket. Turn cover/shade as necessary so that light is directed onto associated instrument.

(e) Engage (push IN) INST. PNL. circuit breaker in.

(2) Socket

(a) Removal

1) Ensure BATT MASTR switch is OFF and INST. PNL. circuit breaker is disengaged (pulled out).

2) Gain access to back of instrument panel in vicinity of socket to be replaced.

3) Unscrew wire from machine threaded brass extension on rear of socket.

4) Remove small copper nut and lock washer from threaded brass extension on rear of socket.

5) Remove socket from mounting hole through front of instrument panel panel

(b) Installation

1) Ensure BATT MASTR switch is OFF and INST. PNL. circuit breaker is disengaged (pulled out).

2) Inserting end with threaded brass extension through hole in panel

3) Securing with lock washer and nut.

4) Snug nut just enough to flatten lock washer against panel to ensure proper ground.

5) Install wire by screwing onto end of threaded brass extension.

6) Engage (Push IN) INST. PNL. circuit breaker in.

B. Seneca V S/N's 3449459, 3449467–3449508

These airplanes are equipped with LED wash lights permanently bonded to the underside of the glareshield.

4. Electroluminescent Lighting

Electroluminescent placards are used in Seneca V S/N's 3449509 and up.

Each EL placard contains one or more electroluminescent lamps. Illumination by electroluminescence is accomplished by discharging an alternating current through a phosphor treated material. 120 Vac is supplied to each EL placard by an inverter mounted to a bracket on the right side of the fuselage between the instrument panel and the firewall. 24 Vdc power for the inverter comes through the PANEL DIMMER 7.5 amp circuit breaker.

A. Troubleshooting

Should any one electroluminescent lamp fail to illuminate, the complete placard must be replaced.

If all EL placards fail to illuminate, check that the PANEL DIMMER circuit breaker has been pulled out (i.e., tripped); reset, if necessary. If the PANEL DIMMER circuit breaker has not tripped, check the PANEL dimmer control unit.

If the PANEL dimmer control unit is good, ensure that no circuit breaker has been pulled out (i.e., tripped); reset as necessary.

Should all EL placards fail to illuminate, the probable fault is with the inverter. Replace the inverter.

If none of the above correct the problem, check all related wiring. Refer to 91-33-10 for appropriate electrical schematics.

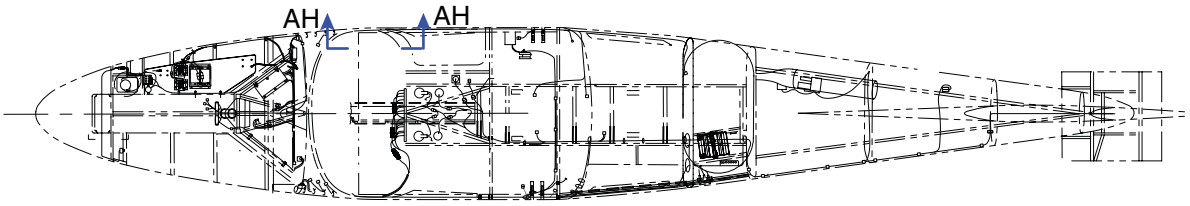
B. Inverter

An inverter mounted to a bracket on the right side of the fuselage between the instrument panel and the firewall supplies 120 Vac to the EL placards. See "Figure 1" on page 33103.

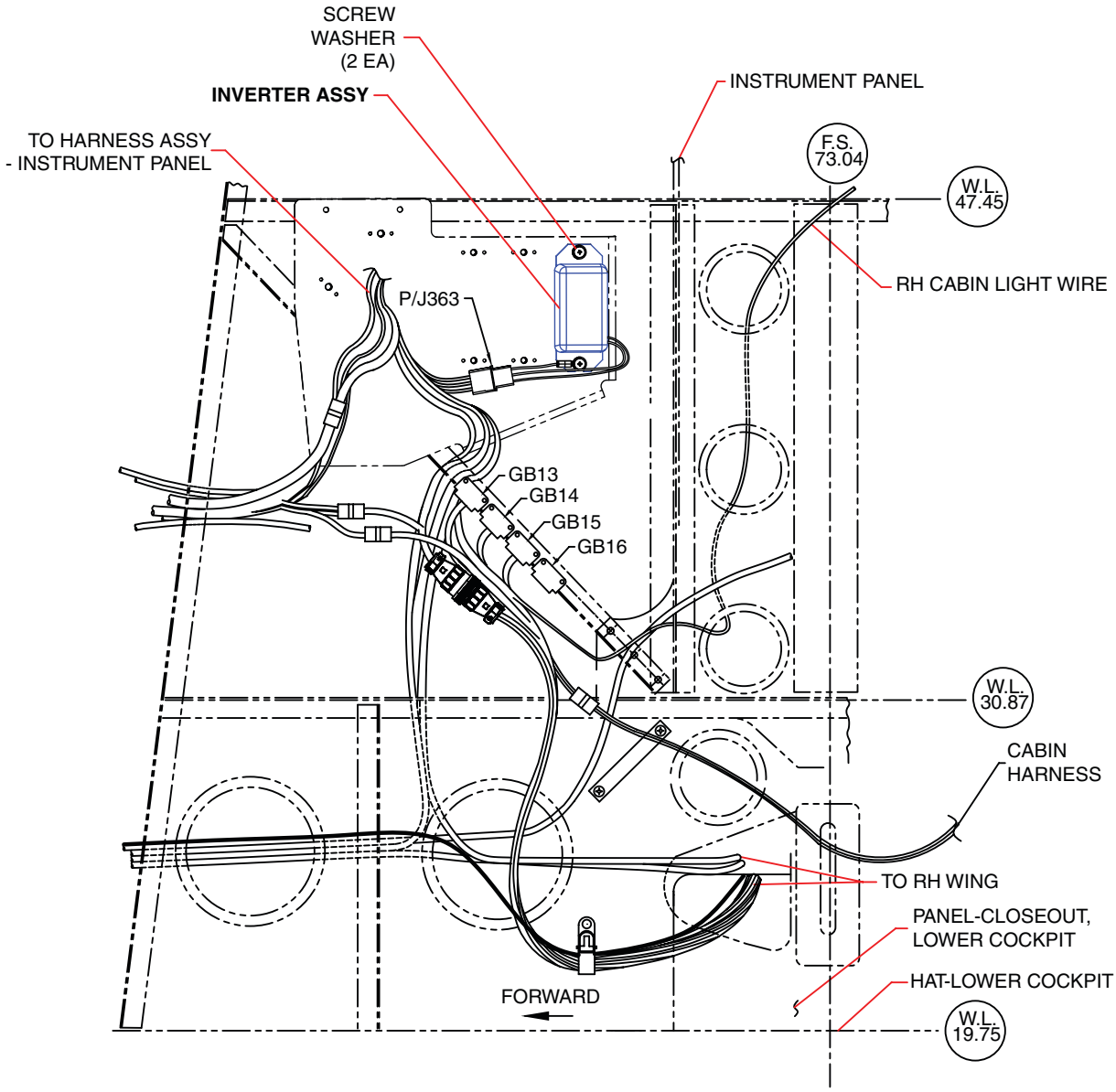
(1) Removal
   (a) Ensure that aircraft and/or ground power is OFF.
   (b) Locate the inverter, on the copilot side just forward of the instrument panel, see "Figure 1" on page 33104.
   (c) Disconnect electrical connector.
   (d) Remove two mounting screws. disconnect ground wire from bottom mounting screw.
   (e) Remove inverter.

(2) Installation
   (a) Place inverter on wall.
   (b) Install top mounting screw and tighten, attach ground wire to bottom mounting screw and tighten screw.
   (c) Connect electrical connector.
   (d) Power up system amd check operation.

34F24A000 NEW 4



LOOKING DOWN



VIEW AH-AH
LOOKING OUTBD AT RIGHT SIDE

Inverter
Figure 1

5. Switch Lights

All rocker type switches are internally lit with two miniature bulbs installed behind the switch cap.

CAUTION: BEFORE WORKING ON ANY SWITCH OR DIMMER CONTROL, ENSURE BATT SWITCH IS OFF AND BATTERY CIRCUIT BREAKER IS PULLED OUT.

A. Bulb Replacement

(1) Ensure BATT switch is OFF, and BATTERY and INST. PNL. circuit breakers are OUT.
(2) Position switch with bad bulb in ON position to expose small slit type notch in bottom of cap.
(3) Using finger nail or small screw driver, gently pull switch cap straight out.

NOTE: Top bulb is easier to replace with switch in ON position

(4) Pull bulb(s) to be replaced straight out.
(5) Push replacement bulb(s) into socket.
(6) Install switch cap by placing in position, with slot toward bottom of switch. Using a slight pressure, push cap onto switch.
(7) Position switch in OFF position. Push BATTERY and INST. PNL. circuit breakers IN.

B. Switch Replacement

(1) Ensure BATT switch is OFF.
(2) Disengage (Pull OUT) the following circuit breakers:
  (a) BATTERY.
  (b) INST. PNL.
  (c) Accessory operated by switch. (Example: If replacing NAV LIGHT switch also Pull NAV circuit breaker OUT.)
(3) Remove and save switch cap and bulbs. See To Replace Bulbs above.
(4) Gain access to back of instrument panel or overhead switch panel in vicinity of switch to be replaced.
(5) Remove switch from panel by pushing forward on back of switch, using a gentle wiggling motion, until free.
(6) Mark wire locations for later installation. Remove accessory wires from switch by removing screws. Remove lighting wires by unsoldering.
(7) Solder lighting wires to appropriate posts on new switch. Attach accessory wires to appropriate posts by installing screws (supplied with switch).
(8) Place switch in proper position in face of instrument panel. Using a slight pressure, press in on switch until spring clip engages panel.
(9) Install bulbs and switch cap.
(10) Engage (Push IN) the BATT, INST. PNL, and any required accessory circuit breakers.

6. Dimmer Controls

There are three dimmer control potentiometers; one each for switch lights, instrument post lights, and avionics. The potentiometers operate individual control boxes located forward of the instrument panel.

A. Removal

(1) Ensure BATT switch is OFF and BATTERY circuit breaker is disengaged (pulled OUT).

(2) If SWITCH or PANEL dimmer potentiometer is to be replaced, disengage (Pull OUT) INST PNL circuit breaker.

(3) If AVIONICS dimmer potentiometer is to be replaced, disengage (Pull OUT) AVIONICS circuit breaker.

(4) Remove dimmer potentiometer control knob by loosening set screw with appropriate size allen wrench.

(5) Remove nut securing dimmer potentiometer to instrument panel. Potentiometer should now hang down below bottom of panel.

(6) Mark wires attached to back of dimmer potentiometer for proper location. Unsolder wires from back of potentiometer.

(7) Remove potentiometer.

B. Installation

(1) Solder wires to proper contacts on back of new dimmer potentiometer.

(2) Position potentiometer in proper location and insert from back of instrument panel. Check that key lugs on switch align with key holes in back of panel.

(3) Install nut securing potentiometer to face of instrument panel.

(4) Install control knob. Check that knob pointer is in appropriate position. Tighten allen screw.

(5) Engage (Push IN) BATTERY, INST PNL and/or AVIONICS circuit breakers, as appropriate.

7. Annunciator Panel Lamp Replacement

It is not necessary to remove the annunciator panel assembly to replace lamp bulbs. Simply “PUSH-IN” on the function light until it “clicks”, and release pressure. The cover assembly will be partially ejected from the lamp base assembly. Pull the cover from the base and rotate to expose the lamp bulb. Replace defective bulb and reverse removal procedure. Depress “Test” function to verify lamp function.

NOTE: Test function only tests the bulbs, not the circuit.

PASSENGER COMPARTMENT

The three reading lights and both the forward and aft courtesy lights are standard equipment. They are each operated by individual switches.

The courtesy lights are wired directly to the battery so that they may be turned ON with the BATT switch OFF.

1. Courtesy and Reading Lights

See “Figure 1” on page 33202.

Courtesy lights are designed to light up each of the entrances. The lights are individually operated and located above the right forward entrance door and the left aft passenger/baggage doors. The left rear passenger light is designed to serve as a courtesy light.

The courtesy lights are powered off the battery bus. This enables illumination of both lights when the battery switch is off. The circuit is protected by a 5 amp fuse.

The aft courtesy light is operated by a switch on the left rear entrance door.

Reading lights are controlled by rocker type switches located in each seat’s armrest.

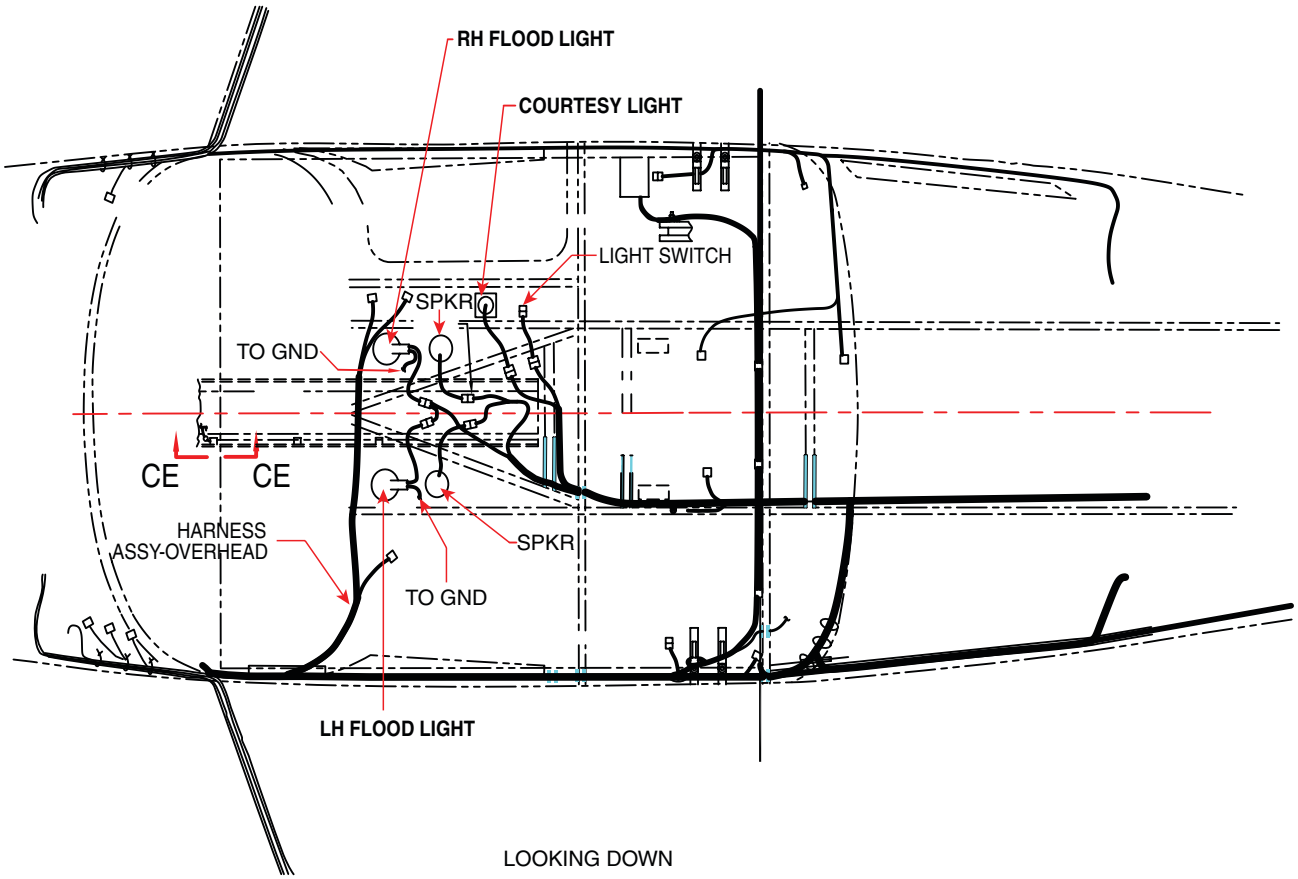
In addition to the aft courtesy light, three reading lights are installed in the passenger cabin that are powered through a five amp CABIN circuit breaker. Each light is independently controlled by a rocker type switch installed in the chair arm rest.

Bulb Replacement (Aft Courtesy or Reading Light)

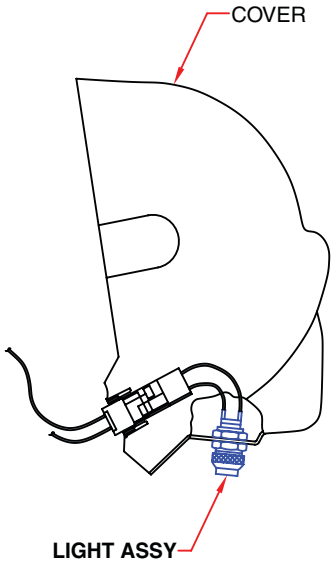
(1) Ensure BATT switch is off. Disengage (Pull OUT) CABIN circuit breaker.
(2) Remove entire light unit from the overhead by removing two attachment screws in bezel.
(3) Carefully withdraw unit from overhead.
(4) Disconnect the lamp holder from the cup. Twist bayonet bulb counterclockwise until it is released.
(5) Install new bulb. Engage (Push IN) CABIN circuit breaker. Position BATT and light switches ON. Ensure correct operation of light.
(6) If bulb does not light up, check the following.
    (a) Proper circuit breaker mode.
    (b) Use voltmeter to check that power is reaching bulb.
    (c) Check the continuity of the ground lead.
    (d) When light is operating satisfactorily, position the light and BATT switches OFF.
(7) Install light unit.

2. Troubleshooting

When checking any of these light systems, other than the courtesy lights, the BATT switch must be ON in order to operate the lights. Ensure that the circuit breaker of the affected system is activated.



LOOKING DOWN



VIEW CE-CE
LHS LOOKING INBD

Passenger Compartment
Figure 1

CARGO AND SERVICE COMPARTMENTS

1. Forward Baggage Compartment Lamp

The forward baggage compartment lighting system is standard equipment. The forward baggage compartment light illuminates automatically whenever the baggage compartment door is open.

This automatic system is operated by a switch installed on the forward door latch pin recess. When the forward pin is inserted into the latch recess hole, the switch is positioned OFF.

NOTE: The light will be on with the door closed, if the forward door latch pin is not engaged and the manual on/off switch is in the ON position, resulting in battery depletion.

On S/N's 3449508 and up, and S/N's 3449487- 34499507 with the improved forward baggage compartment light installed the light features a manually operated on/off switch, which is operated independently from the switch that is actuated by opening/closing the forward baggage compartment door. See "Figure 1" on page 33302. Installation of this light will allow operators to leave the compartment door open with the light off, to prevent draining of the battery. When the baggage compartment manual switch is turned On, the forward baggage compartment light illuminates automatically whenever the door is open.

The circuit is protected by a 5 amp fuse.

Refer to 91-33-30 for electrical schematics.

2. Removal and Installation of Baggage Light Bulb

NOTE: Observe the baggage light fixture has two sets of three screws; one set in an outer ring or circle and one set in an inner ring or circle. Be sure to remove screws from inner ring or circle.

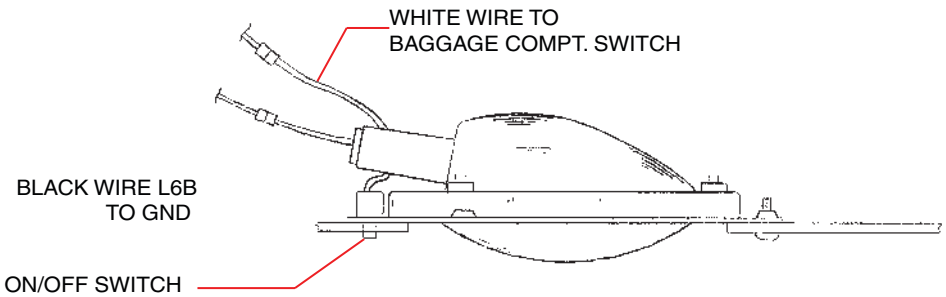
A. Remove the lens cover by removing the three attaching screws.

B. Remove bayonet type bulb.

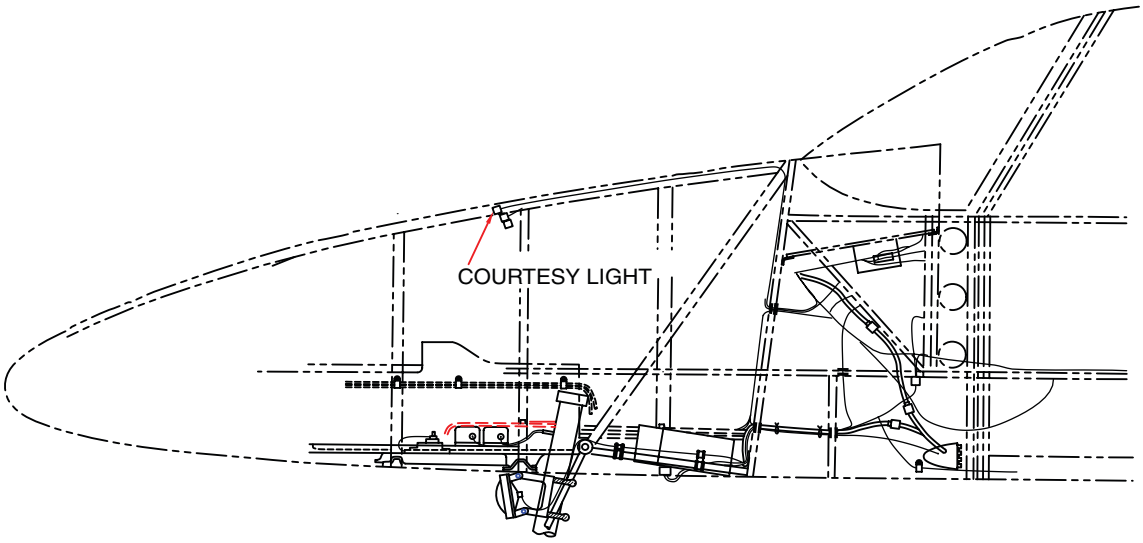
C. Install new bulb and ensure its operation.

D. Install lens cover by installing the three attaching screws.

107270 N 5
37874 Z



COURTESY LIGHT



Cargo and Service Compartment
Figure 1

EXTERIOR

Exterior lighting consists of navigation (position) lights, anti-collision (strobe) lights, wing ice light, recognition lights, and landing lights. With the exception of the wing ice light, switches for external lighting can be found on the main switch cluster in the instrument panel (Seneca IV), or in the overhead switch panel (Seneca V).

The wing ice light switch is located in the deice switch group located in the instrument panel above the throttle quadrant.

1. Troubleshooting

When checking any of these light systems the BATT switch must be ON in order to operate the lights. Ensure that the circuit breaker of the affected system is activated.

2. Landing / Taxi / Recognition Lights

See “Figure 5” on page 334011 and “Figure 4” on page 33409.

A. Description

(1) Seneca IV

In Seneca IV models, the landing and taxi lights consist of two 100 watt lamps which are located in a light assembly secured to the nose gear trunnion. Both lamps are used for landing and one lamp is used while taxiing. Each lamp is controlled by a separate switch mounted in the switch panel. (Refer to 39-10-00.) The lamps are wired to separate 10 amp circuit protectors mounted in the circuit protector panel. There is a safety switch mounted on the nose gear strut which will break the circuit to the lights when the nose gear is retracted in case the pilot forgets to turn the switches OFF.

Recognition lights are installed in the leading edge of both wing tips and controlled by a two-position (ON/OFF) RECOG LT switch.

(2) Seneca V

In Seneca V models, the landing and taxi lights consist of four 100 watt lamps, two of which are located in a light assembly secured to the nose gear trunnion, and one light assembly located in each wing tip. The landing lights in the wing tips also serve as recognition lights.

In S/N’s 3449001 thru 3449096, the lights are controlled by a three-position, rocker-type switch located on the overhead switch panel. When the switch is in the up (TAXI) position, the two lights on the nose gear illuminate. When the switch is in the center (OFF) position, all taxi and landing/recognition lights will be off. When the switch is in the down (LANDING) position, all nose gear lights and wing tip lights will illuminate.

In S/N’s 3449097 and up, two rocker-type switches in the overhead switch panel control these four lights. A two-position (on/off) TAXI lights switch allows the two nose gear mounted lights to be controlled independently for ground operation. A three-position Landing/Pulse lights switch controls all four lights in flight. When the switch is in the up (LANDING) position, all four lights (two on the nose gear and one in each wing tip) illuminate. When the switch is in the center (OFF) position, all four lights are off. When the switch is in the down (PULSE) position, the two nose gear mounted lights are off and wing tip lights will illuminate alternately (i.e., pulse) at 55 pulses-per-minute. (See “Figure 4” on page 33409.)

In all Seneca V’s, there is a safety switch mounted on the nose gear strut which will break the circuit to the nose gear lights when the nose gear is retracted, in case the pilot forgets to turn the switches off. The landing/taxi lights are wired to two separate 10 amp Landing/Taxi Lights circuit breakers.

B. Nose Gear Mounted Landing / Taxi Lights

Two landing / taxi lights are mounted in an assembly clamped to the nose gear trunnion. Refer to “Figure 1” on page 33403.

(1) Lamp Replacement

(a) Removal

1) Ensure that the master switch is turned to the OFF position prior to doing any work on the landing lights.

CAUTION: WHEN REMOVING THE FRONT PLATE ASSEMBLY, TAKE CARE NOT TO ALLOW THE LAMPS TO FALL OUT.

2) Removal of either lamp from the landing light assembly is accomplished by removing the six (6) screws securing the front plate assembly and removing the plate assembly.

3) Disconnect the electrical leads from the lamp being removed and remove the lamp.

(b) Installation

1) To install a lamp, attach the electrical leads to the lamp.

CAUTION: TIGHTEN THE SCREWS JUST ENOUGH TO ALLOW THE LAMPS TO FIT SNUG IN THE MOUNTING FIXTURE.

2) Place the lamp (or lamps) against the mounting pad(s) in the rear plate assembly and position the front plate assembly over the lamps and rear plate assembly.

CAUTION: TIGHTEN THE SCREWS JUST ENOUGH TO ALLOW THE LAMPS TO FIT SNUG IN THE MOUNTING FIXTURE.

3) Secure with six (6) screws.

NOTE: LED lamps are factory installed in S/N’s 3449509 and up; and authorized as service replacements in S/N’s 3447001–3447029 and 3449001–3449508. They are polarity sensitive. If they fail to light upon first installation, remove them and swap the electrical leads between the contacts on the lamp.

(2) Light Assembly

(a) Removal

To remove the complete assembly from the gear trunnion, disconnect the electrical leads from both lamps and release the clamps that secure the assembly to the trunnion.

(b) Installation

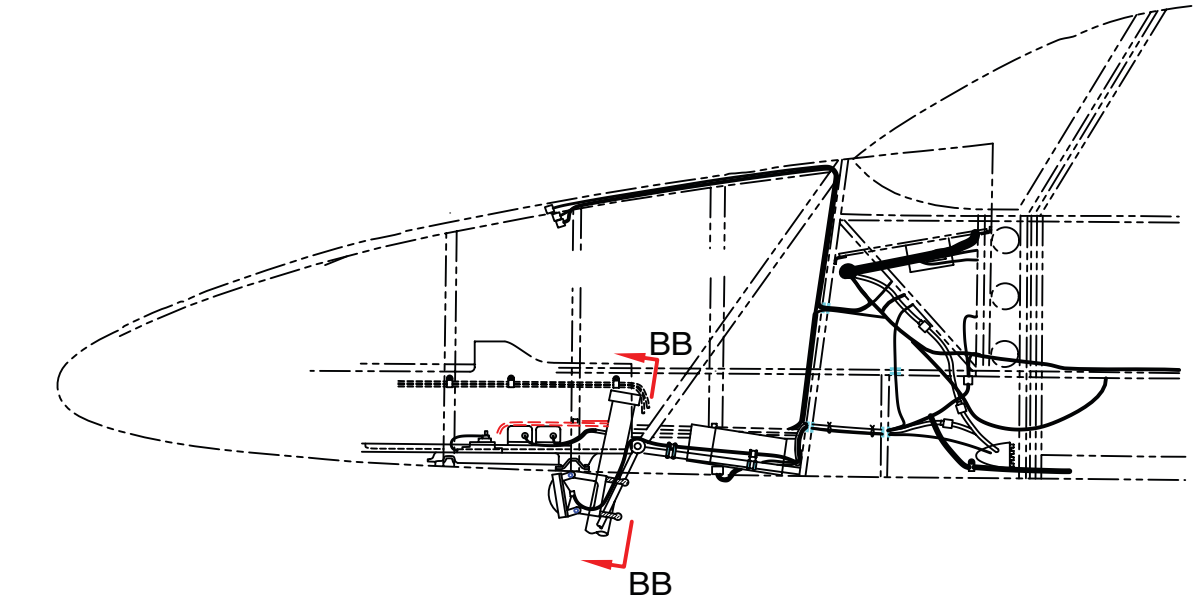
1) To install the landing light assembly to the trunnion, position the assembly against the trunnion and capture with clamps.

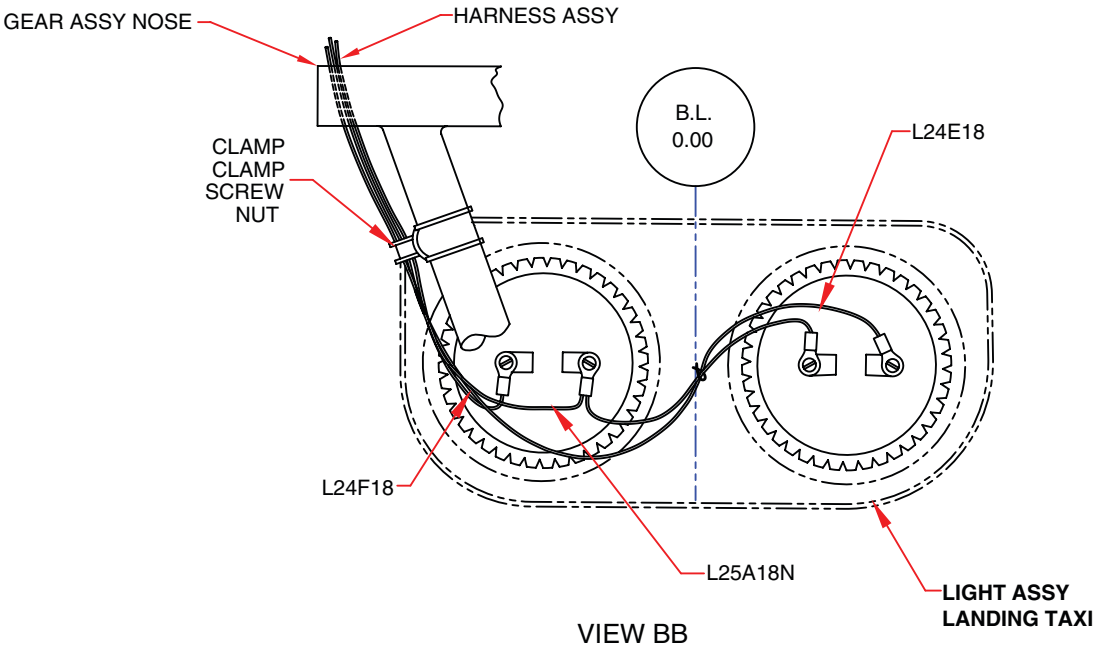
2) Align the light assembly longitudinally and align the top of the light assembly one (1) inch below the bottom of the trunnion pivot block plate.

(3) Adjustment

See “Figure 1 (Sheet 2 of 2)” on page 33404.

The light beam angle may be adjusted by the adjusting screw at each side of the light assembly and tilting the light assembly as desired.

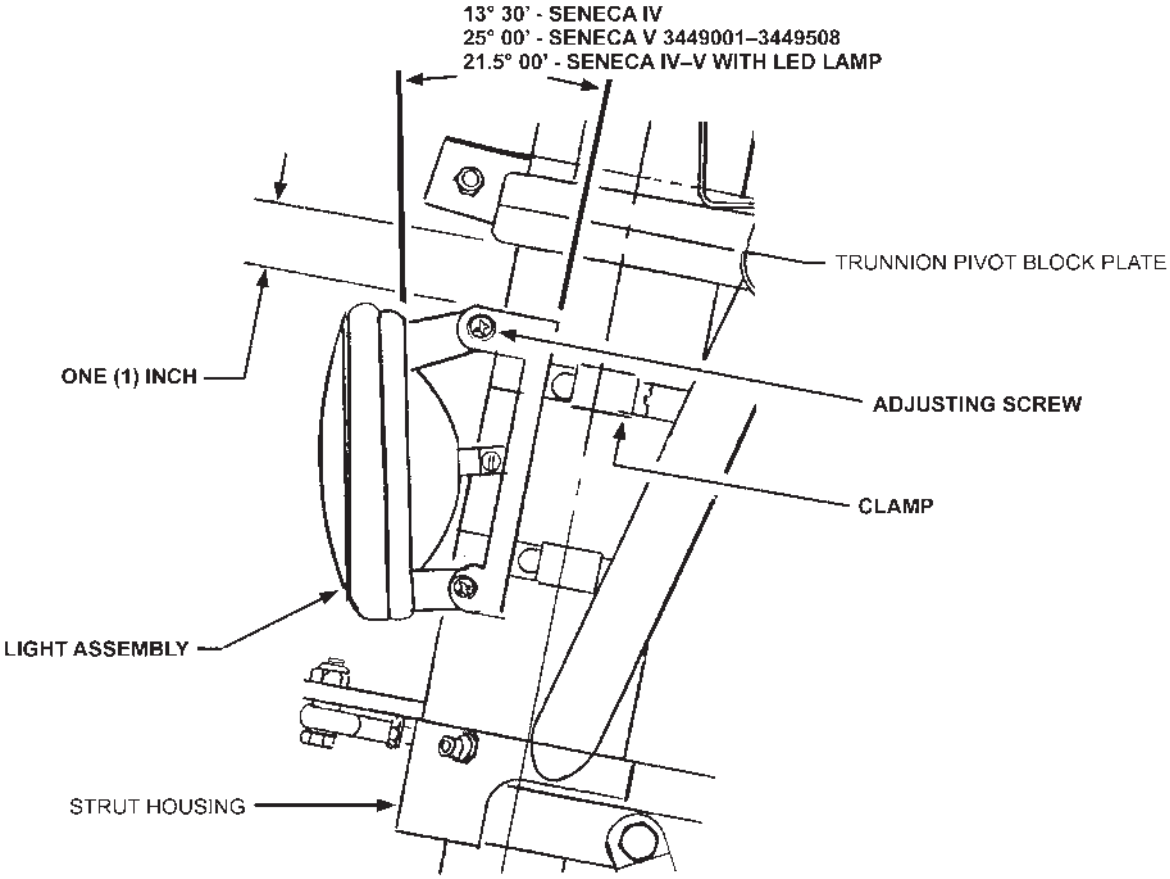




VIEW BB
LANDING LIGHTS
LOOKING FWD

Nose Gear Mounted Landing/Taxi Light Installation
Figure 1 (Sheet 1 of 2)

39460 V
96240 E



Nose Gear Mounted Landing/Taxi Light Installation
Figure 1 (Sheet 2 of 2)

C. Wing Tip Mounted Landing / Recognition Lights (Seneca V only)

A landing / recognition light is mounted in the leading edge of each wing tip. See “Description” on page 33401 for a summary of how they function. The following addresses removal and replacement of the lamp only. See “Figure 2” on page 33406.

(1) Removal

(a) Remove the appropriate Plexiglas lens by removing the nine (9) screws and washers.

NOTE: Do not disturb adjustment bolts when removing landing / recognition light lamp(s). Should adjustment bolt settings be disturbed, realign lights per “Adjustment (Alignment)”, below, after installing new lamp.

(b) Remove the three lamp retaining screws. Remove retaining ring.

(c) Pull lamp forward far enough to gain access to the electrical connection tabs on back of the lamp.

(d) Remove the two brass screws connecting the electrical leads to the lamp.

(e) Remove and discard defective lamp.

(2) Installation

(a) Connect electrical leads to replacement lamp using two brass screws.

NOTE: LED lamps are factory installed in S/N’s 3449509 and up; and authorized as service replacements in S/N’s 3449001–3449508. They are polarity sensitive. If they fail to light upon first installation, remove them and swap the electrical leads between the contacts on the lamp.

(b) Position lamp in place and install retainer ring.

(c) Install the three lamp retaining screws.

(d) Check lamp function by turning BATT MAST switch on and selecting LNDG LIGHT switch on.

(e) Place BATT MAST and landing light switch in OFF position.

(f) Install Plexiglas lens.

(3) Adjustment (Alignment)

See “Figure 2” on page 33406 and “Figure 3” on page 33408.

(a) Locate airplane 20 feet from a target wall as shown in “Figure 2” on page 33406.

(b) Level airplane both vertically and horizontally. (Refer to 8-20-00.)

(c) Remove the appropriate Plexiglas cover lens.

(d) Connect airplane to a 28 Vdc external power supply.

(e) Position the BATT MAST and LDG LIGHT switches ON.

(f) Adjust the three landing light adjustment bolts as necessary to move the landing light housing so that the center of the light beam illuminates the target wall at the following locations.

1) The left wing tip landing light beam is centered vertically at a point 32 ± 1 inches up from the floor; and, horizontally, at a point 17 feet ± 3 inches outboard from the airplane’s centerline.

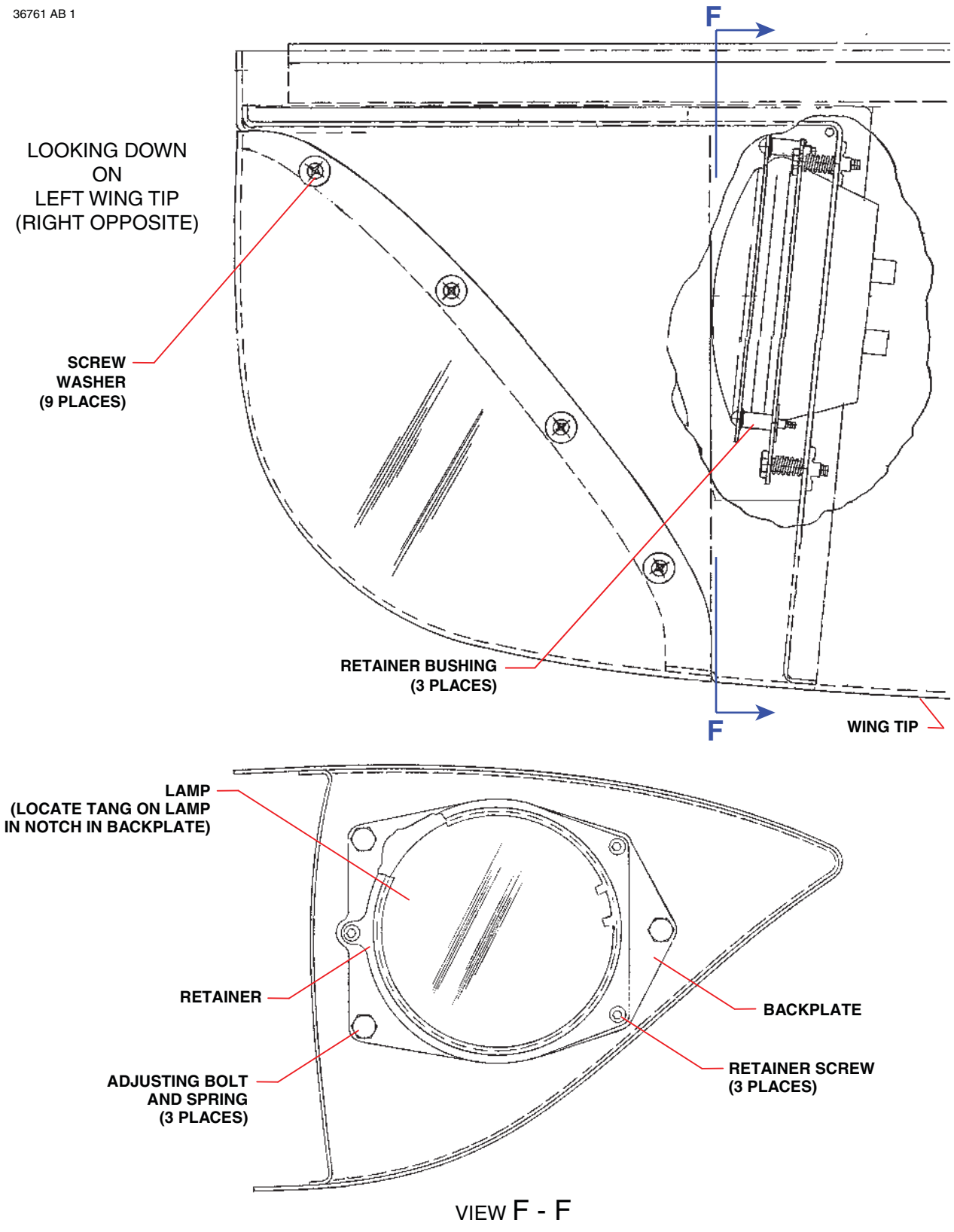
2) Center the right wing tip landing light beam vertically at a point 32 ± 1 inches inches up from the floor; and, horizontally, at a point 17 feet ± 3 inches outboard from the airplane’s centerline.

(g) Position the BATT MAST and LDG LIGHT switches OFF.

(h) Disconnect 28 Vdc external power supply.

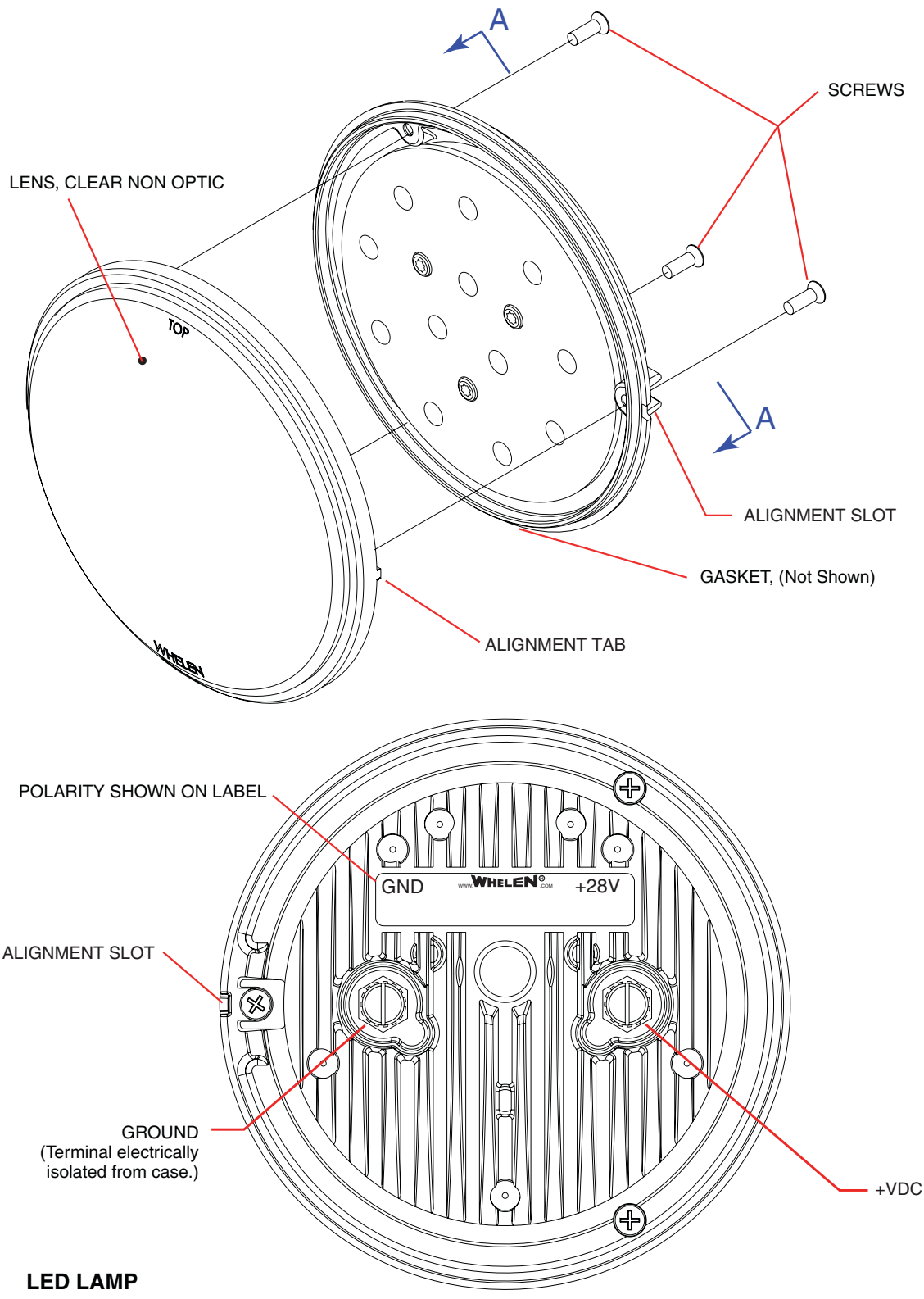
(i) Install Plexiglas cover lens.

(j) Remove any devices used to level the airplane. (Refer to Chapter 8.)



Effectivity
Seneca V

Wing Tip Landing / Recognition Light Installation
Figure 2 (Sheet 1 of 2)

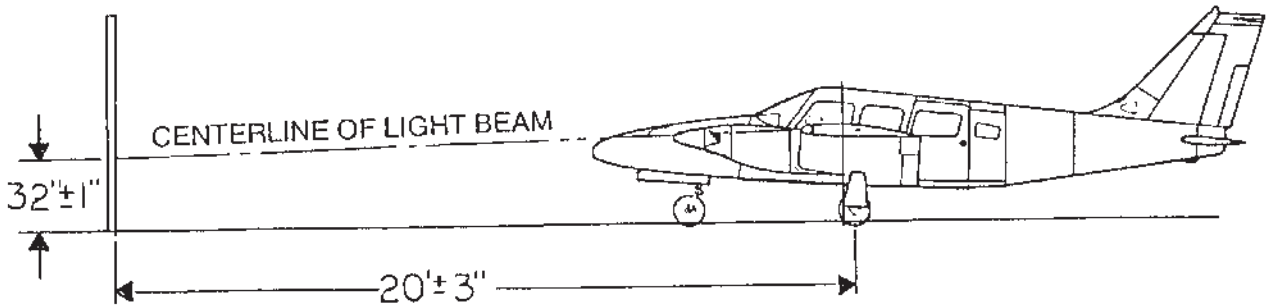


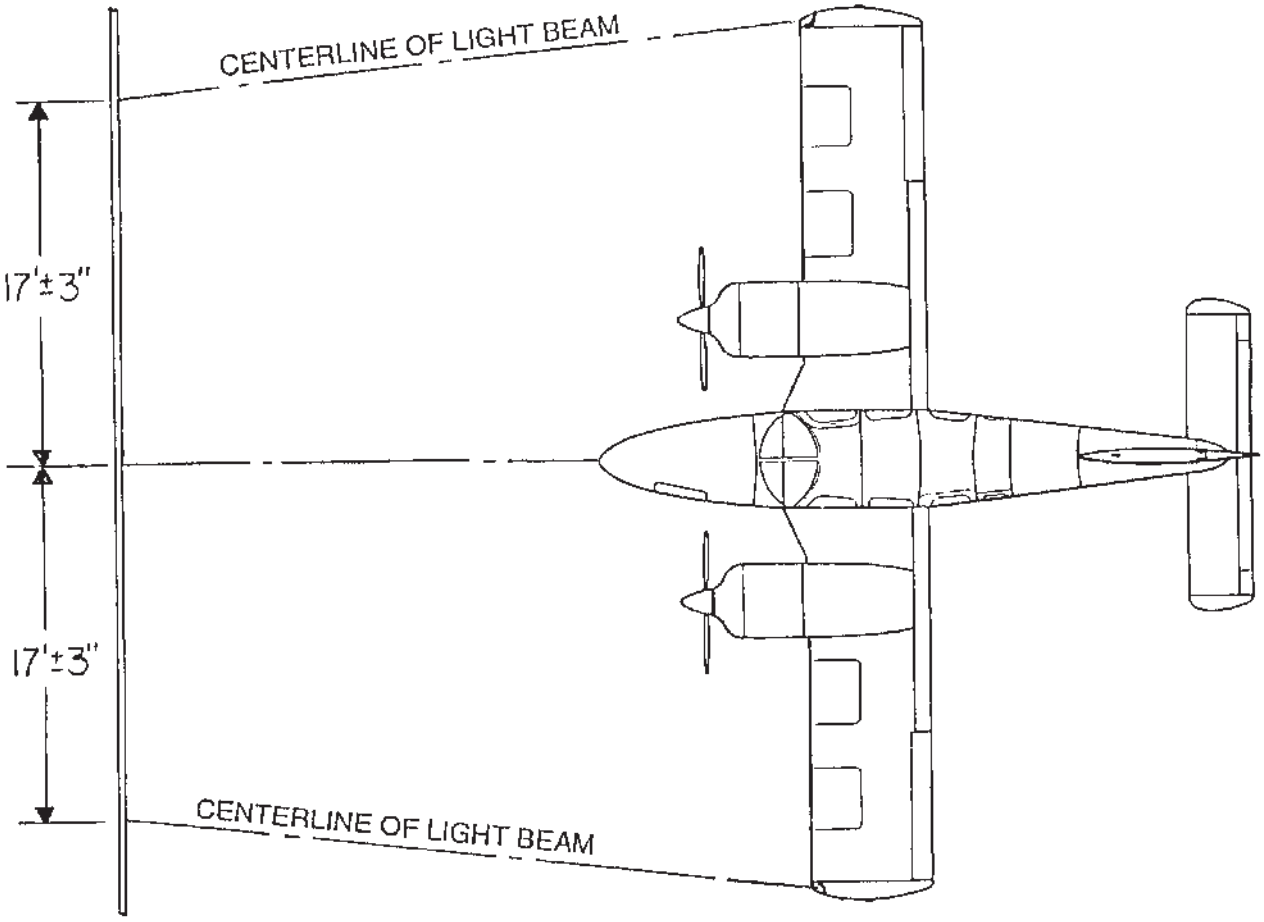
VIEW A - A

Wing Tip Landing / Recognition Light Installation
Figure 2 (Sheet 2 of 2)

Effectivity
Seneca V

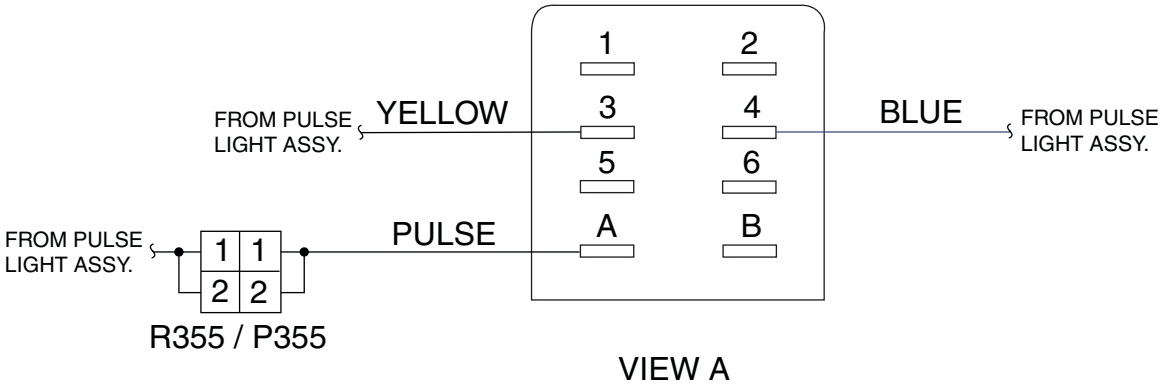
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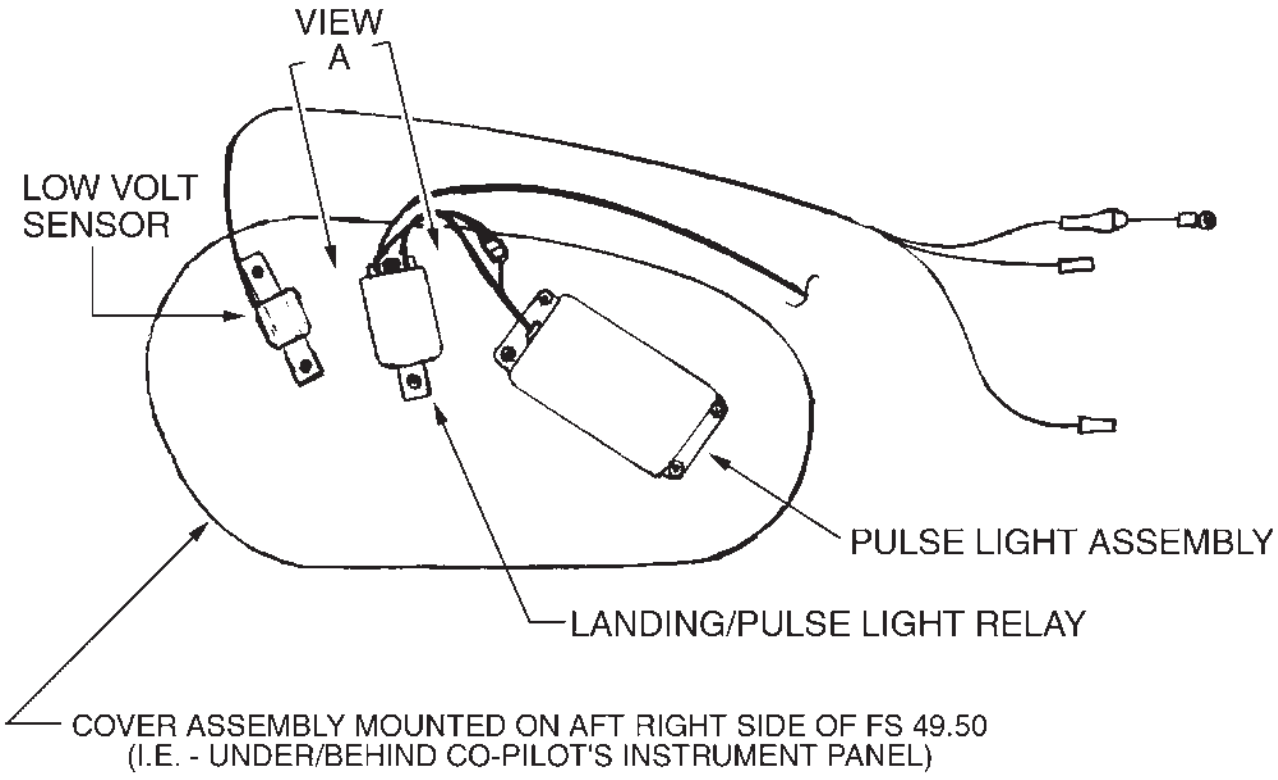




Effectivity
Seneca V

Wing Tip Landing / Recognition Light Adjustment
Figure 3





Landing / Recognition (Pulse) Lights Relay
Figure 4

Effectivity
3449097 & up

D. Wing Tip Mounted Recognition Lights (Seneca IV only)

Recognition lights are installed in the leading edge of each wing tip. See “Description” on page 33401 for a summary of how they function. The following addresses removal and replacement of the lamp/bulb only. See “Figure 5” on page 334011.

(1) Removal

CAUTION: TO AVOID CRACKING OR DAMAGING, DO NOT USE POWER TOOLS TO REMOVE OR INSTALL LENS.

(a) Ensure BATT and RECOG LIGHT switches are set to the OFF position.
(b) Disengage (i.e., Pull OUT) RECOG 7.5 amp circuit breaker.
(c) Remove the ten (10) screws and washers securing the lens to the wing tip.
(d) Remove lens.
(e) Remove lamp/bulb from bayonet base by pulling straight out.

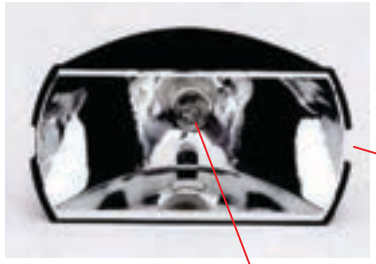
(2) Installation

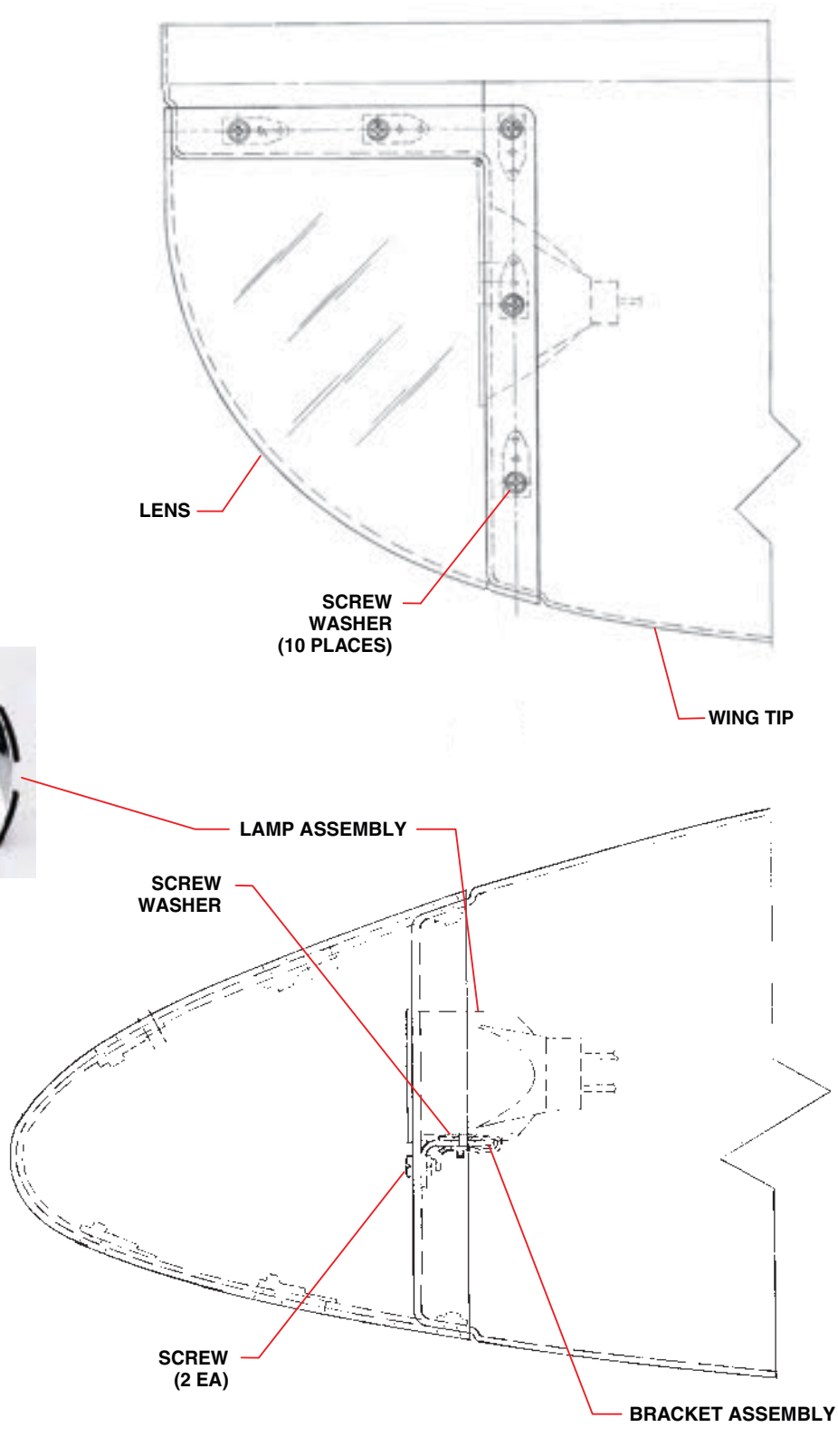
(a) Install replacement lamp/bulb by pushing straight into the socket.
(b) Engage (i.e., push IN) RECOG circuit breaker. Verify lamp illuminates when BATT and RECOG LIGHT switches are set to the ON position.

NOTE: If lamp does not illuminate, check wiring. Refer to 91-33-40 for schematic.

(c) Set switches to the OFF position and disengage (i.e., Pull OUT) circuit breakers.
(d) Position lens in recess in wing tip. Install ten (10) screws and washers and finger tighten.
(e) Ensuring lens is properly aligned in wing tip recess, tighten all attachment screws with a hand screwdriver.
(f) Engage (i.e., Push IN) RECOG circuit breaker.

36761 AB 2





Recognition Lights
Figure 5

Effectivity
Seneca IV

3. Navigation / Position Lights

Navigation and position lights are installed in a single light assembly which also contains an anti-collision (i.e., strobe) light in each wingtip. The strobe lights are addressed separately in “Anti-Collision (i.e., Strobe) Lights” on page 334020.

The navigation / position lights consist of: a red light forward and a white light aft in the left wingtip; and, a green light forward and a white light aft in the right wingtip. The lights are controlled by a rocker type switch (NAV LIGHT) switch mounted in the instrument panel switch panel (Seneca IV) or the overhead switch panel (Seneca V). The circuit is protected by a 10 amp NAV LIGHTS circuit breaker.

An optional white tail light mounted in the rudder tip fairing assembly was also available 1995–2018. When installed, the tail light is powered from and controlled by the same NAV LIGHT switch as the navigation/ position lights.

A. Wingtip

Three different installations exist. The following addresses removal and replacement of the navigation/ position lamps only.

(1) Installation A

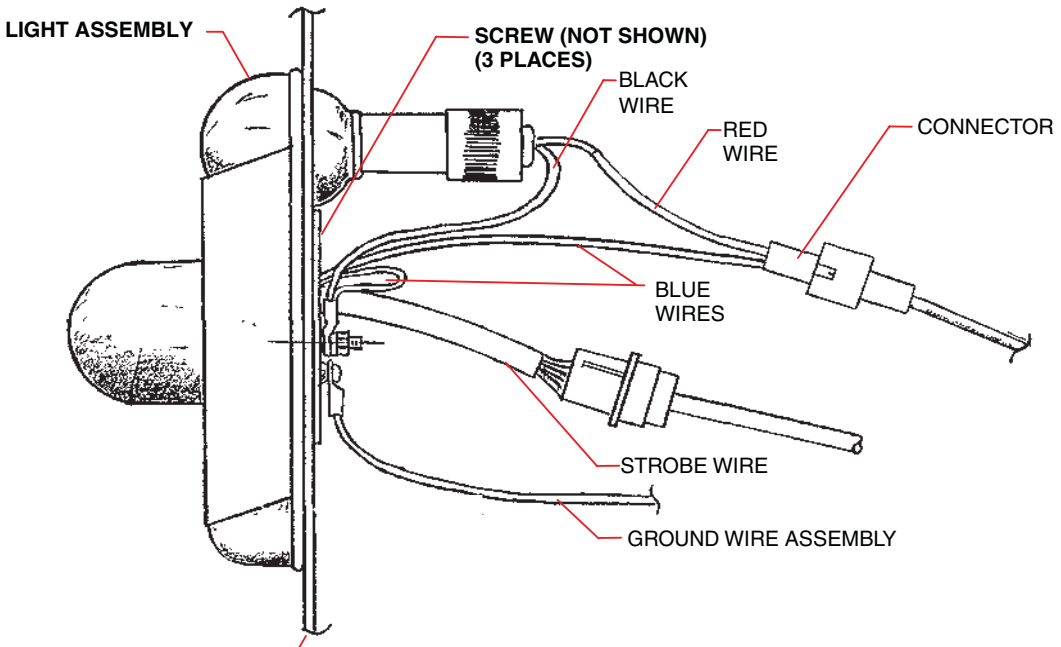
Incandescent navigation / position lamps combined with a Xenon flashtube are installed in Seneca IVs and in Seneca V S/N’s 3449001–3449458, 3449460–3449466. Standard and Garmin G600 equipped airplanes have slight differences in the wiring compared to Avidyne Entegra equipped airplanes as shown in “Figure 6” on page 334013. See also “Figure 7” on page 334014.

(a) Removal

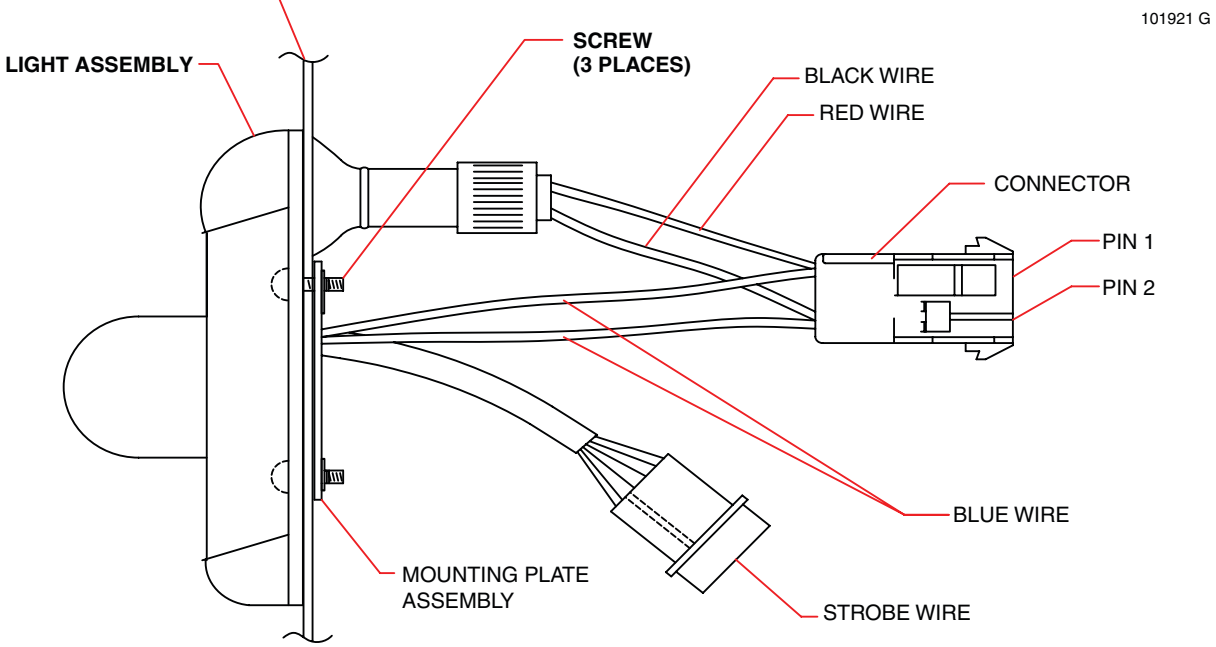
1) Ensure that the BATT MASTR, STROB LIGHT, and NAV LIGHT switches are in OFF position.
2) Disengage (pull out) NAV LIGHTS and ANTI-COLL circuit breakers.
3) Remove two (2) screw securing the lens retainer and remove the retainer.
4) Remove fore or aft lens as appropriate.
5) If replacing the forward lamp, push to bottom of socket and twist counter-clockwise to release and then pull out lamp.
6) If replacing the aft halogen lamp, pull straight aft from socket.

(b) Installation

1) If installing the forward lamp, align the lamp with the socket, then push to bottom of socket and twist clockwise to lock in place.
2) If installing the aft halogen lamp, align the lamp with the socket, then push straight forward into socket until lamp is seated.
3) Reinstall fore or aft lens as appropriate.
4) Place the lens retainer in position and secure with two (2) screws.
5) Engage (push IN) the NAV LIGHTS circuit breaker.
6) Switch the BATT MASTR and NAV LIGHT switches ON and ensure the navigation/ position LEDs illuminate.
7) Switch NAV LiGHT and BATT MASTR OFF and secure airplane.



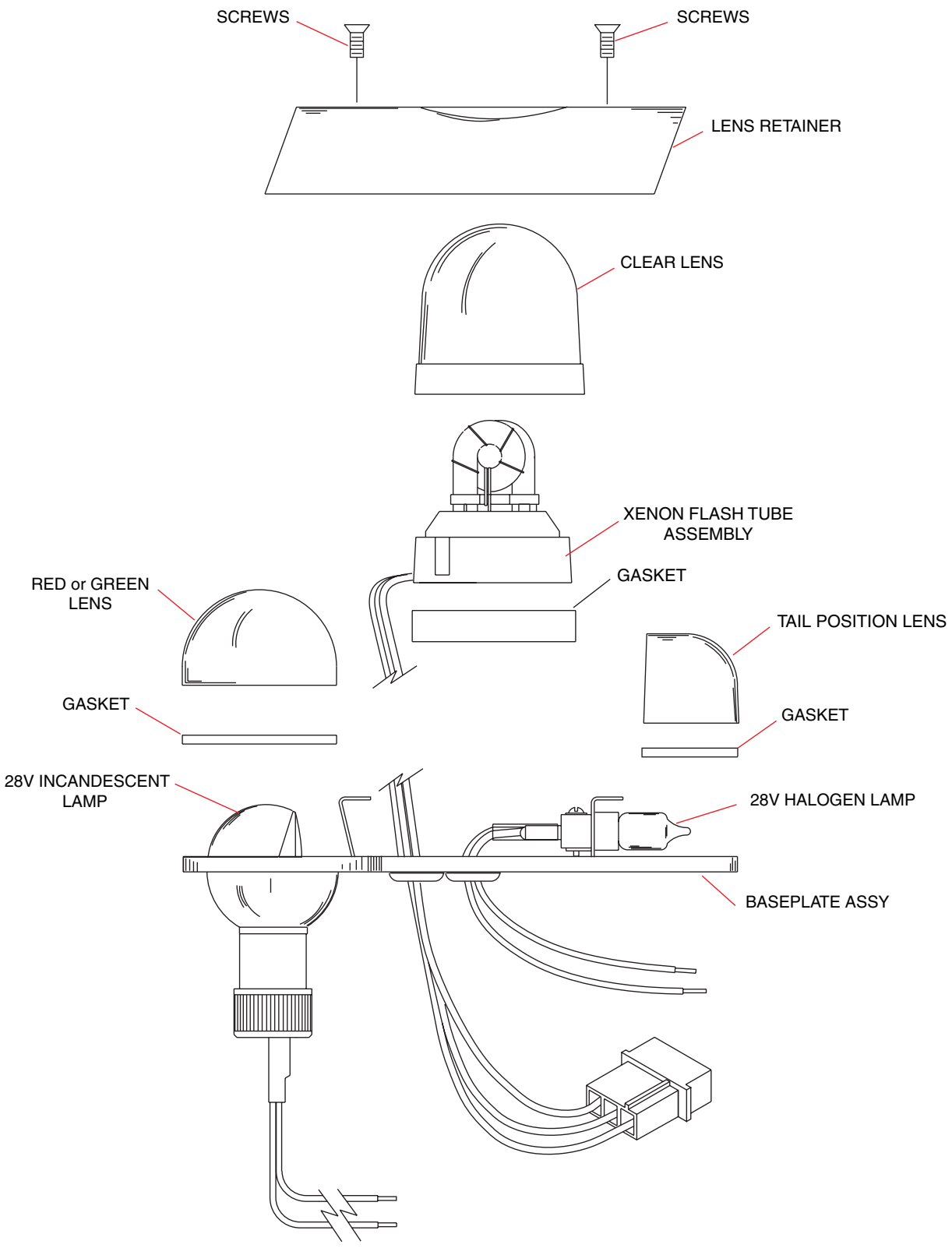
SENECA IV
SENECA V WITH STANDARD AND GARMIN G600



SENECA V WITH AVIDYNE ENTEGRA

Wing Tip Navigation/Position/Strobe Light Installation
Figure 6

Effectivity
Installation A



Effectivity
Installation A

Wing Tip Navigation/Position/Strobe Light Assembly
Figure 7

(2) Installation B

LED navigation / position lamps combined with a Xenon flashtube are installed in Seneca V S/N's 3449459, 3449467–3449508. See "Figure 8" on page 334016 and "Figure 9" on page 334017. Removal and replacement of either or both of the forward and aft navigation / position LED lamp assemblies requires removal and replacement of the entire LED lamp assembly base.

(a) Removal

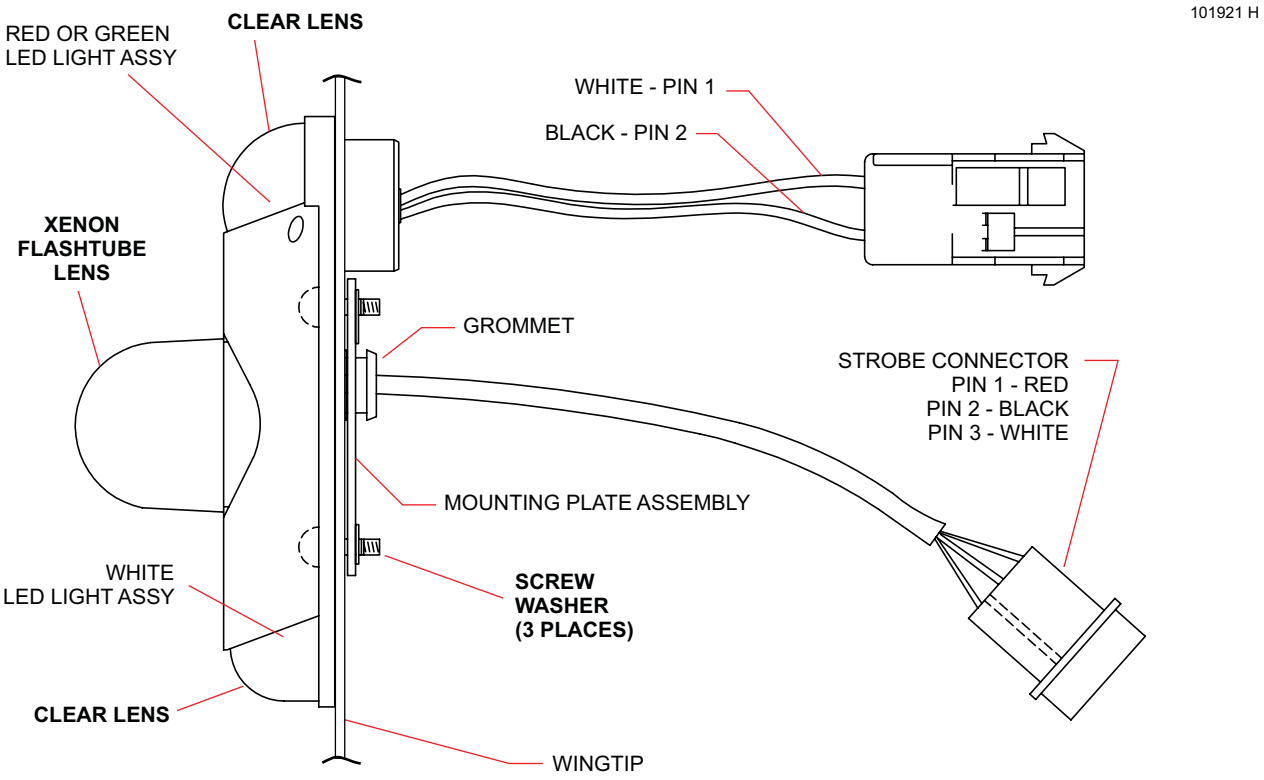
1) Ensure that the BATT MASTR, STROB LIGHT, and NAV LIGHT switches are in OFF position.
2) Disengage (pull out) NAV LIGHTS and ANTI-COLL circuit breakers.
3) Remove the two screws securing the lens retainer.
4) Remove the lens retainer and all three lenses.
5) Remove the three screws securing the light assembly to the mounting plate.
6) Break sealant and move the light assembly out of the way. Draw electrical leads out to expose connectors. Capture wing harness to ensure it does not retract into wing.
7) Disconnect the light assembly electrical leads from the wing harness and remove the light assembly from the wing tip.
8) Remove old sealant from mating surfaces of light assembly and wingtip.

(b) Installation

1) Ensure wingtip is clean and dry.
2) Connect the light assembly electrical leads (i.e., strobe and LED) to the wing harness. Replace any Ty-Wraps or sleeves previously removed.
3) Release the wing harness and carefully feed the LED and strobe lamp electrical leads through the hole in the base.
4) Put the light assembly in place and secure to mounting plate with three screws.

CAUTION: MATING SURFACES MUST BE SEALED TO PREVENT WATER INTRUSION.

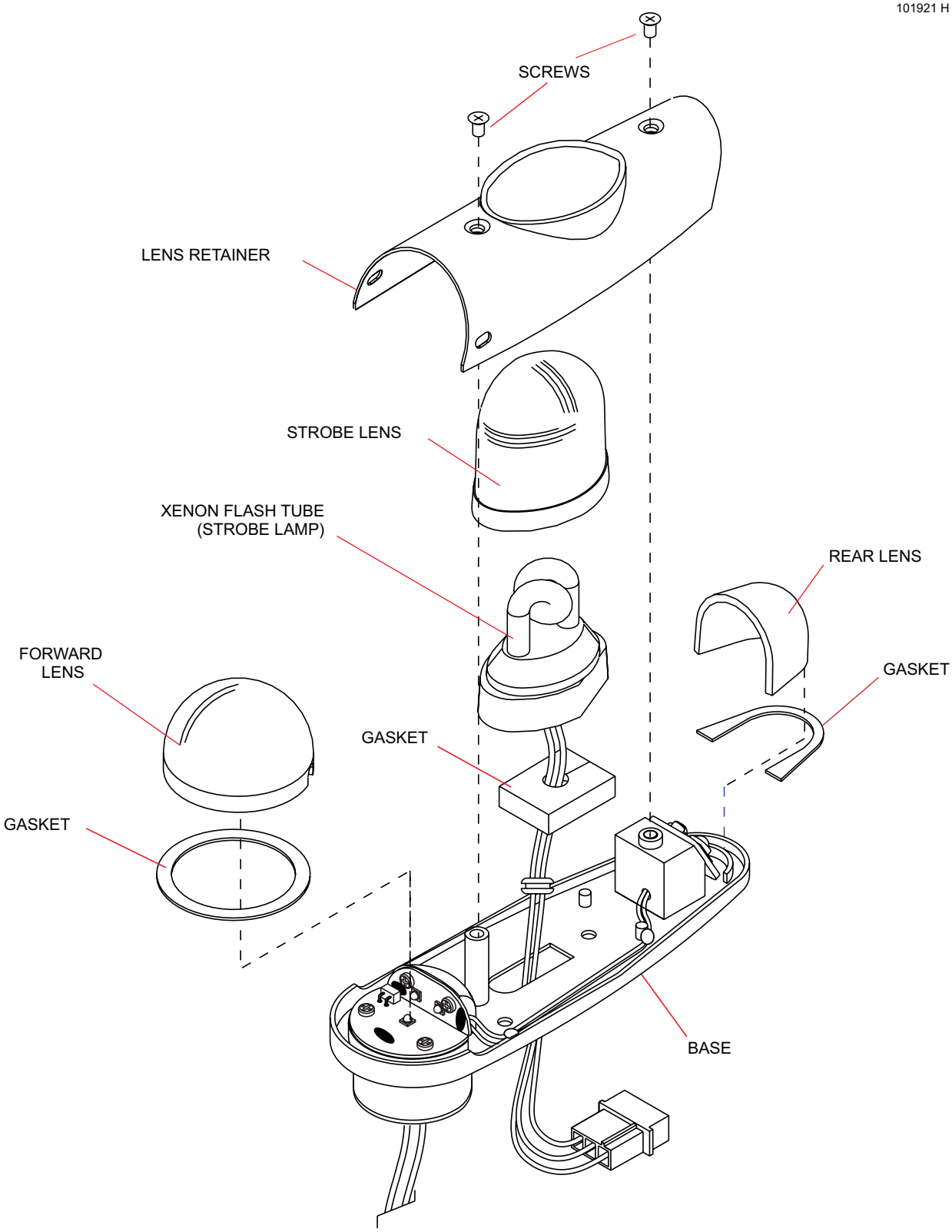
5) Fillet seal edges of mating surfaces of light assembly and wingtip. Use Sealant Type 4, Group 1, Chart 11 in 91-10-00.
6) Ensure xenon flashtube is correctly positioned and reinstall all three lenses.
7) Place the lens retainer in position and secure with two (2) screws.
8) Engage (push IN) the NAV LIGHTS circuit breaker.
9) Switch the BATT MASTR and NAV LIGHT switches ON and ensure the navigation/ position LEDs illuminate.
10) Switch NAV LiGHT and BATT MASTR OFF and secure airplane.



Effectivity
Installation B

Wing Tip Navigation/Position/Strobe Light Installation
Figure 8

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Wing Tip Navigation/Position/Strobe Light Assembly
Figure 9

Effectivity
Installation B

(3) Installation C

LED navigation / position lamps combined with a LED strobe are installed in Seneca V S/N's 3449509 and up. See "Figure 10" on page 334019 and "Figure 11" on page 334020. If any of the lights fail, the entire LED lamp assembly must be replaced.

(a) Removal

1) Ensure that the BATT MASTR, STROB LIGHT, and NAV LIGHT switches are in OFF position.
2) Disengage (pull out) NAV LIGHTS and ANTI-COLL circuit breakers.
3) Lens
   a) Remove two (2) socket head screws from aft end of light assembly and remove lens retainer.
   b) Slide lens aft approximately 1/2 inch and lift away from light assembly.
4) Light Assembly
   a) Remove lens as described above.
   b) Remove three (3) phillips head screws and separate light assembly from baseplate.
   c) Pull light assembly away from wing tip to the attached wiring harness out to exposethe connector. Capture the exposed wiring harness at the baseplate with tape to ensure it does not retract into the wing.
   d) Disconnect the connector and remove light assembly.
5) Baseplate

   Not required unless the existing baseplate is damaged or deteriorated.
   a) Remove Lens and Light Assembly as describe above.
   b) Remove three screws securing baseplate to wingtip and remove baseplate.
   c) Remove any remaining sealant from wingtip.

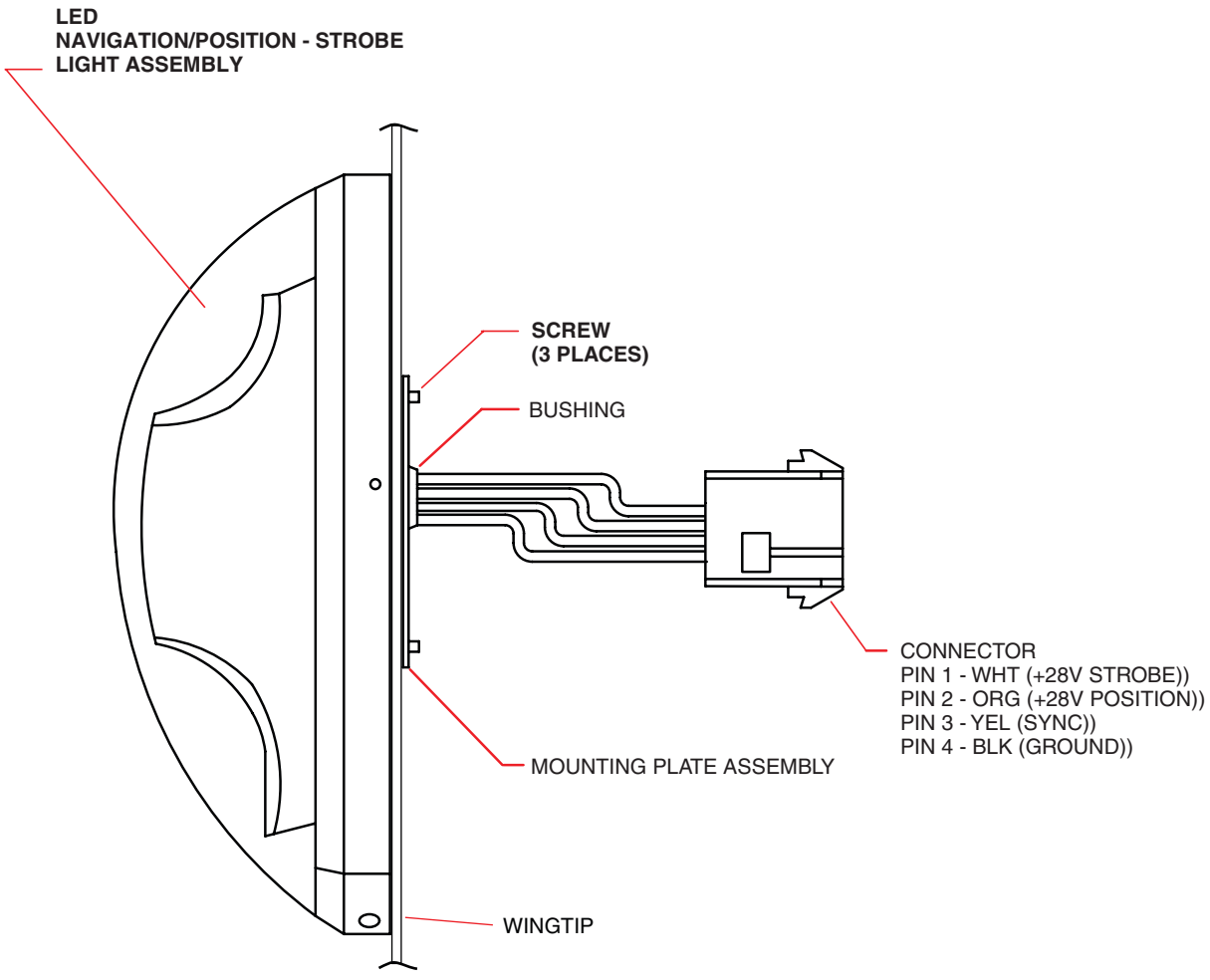
(b) Installation

1) Baseplate (If required.)
   a) Ensure wingtip is clean and dry.
   b) Feed exposed wing wiring harness through hole in baseplate and position baseplate on wingtip.
   c) Secure with three screws.
   d) Fillet seal edges of mating surfaces of light assembly and wingtip. Use Sealant Type 4, Group 1, Chart 11 in 91-10-00.
2) Light Assembly
   a) Connect navigation/postion/strobe light assembly electrical connector to the exposed harness wiring connector at the baseplate.
   b) Feed the wiring harness through the hole in the baseplate and place the new light assembly in position.
   c) Secure with three phillips head screws.
3) Lens
   a) Place lens approximately 1/2 inch aft of light assembly, slide forward into position.
   b) Place lens retainer in position over aft end of lens.
   c) Secure lens retainer and lens with two (2) socket head screws.
   d) Engage (push IN) the NAV LIGHTS circuit breaker.

e) Switch the BATT MASTR and NAV LIGHT switches ON and ensure the navigation/position LEDs illuminate.

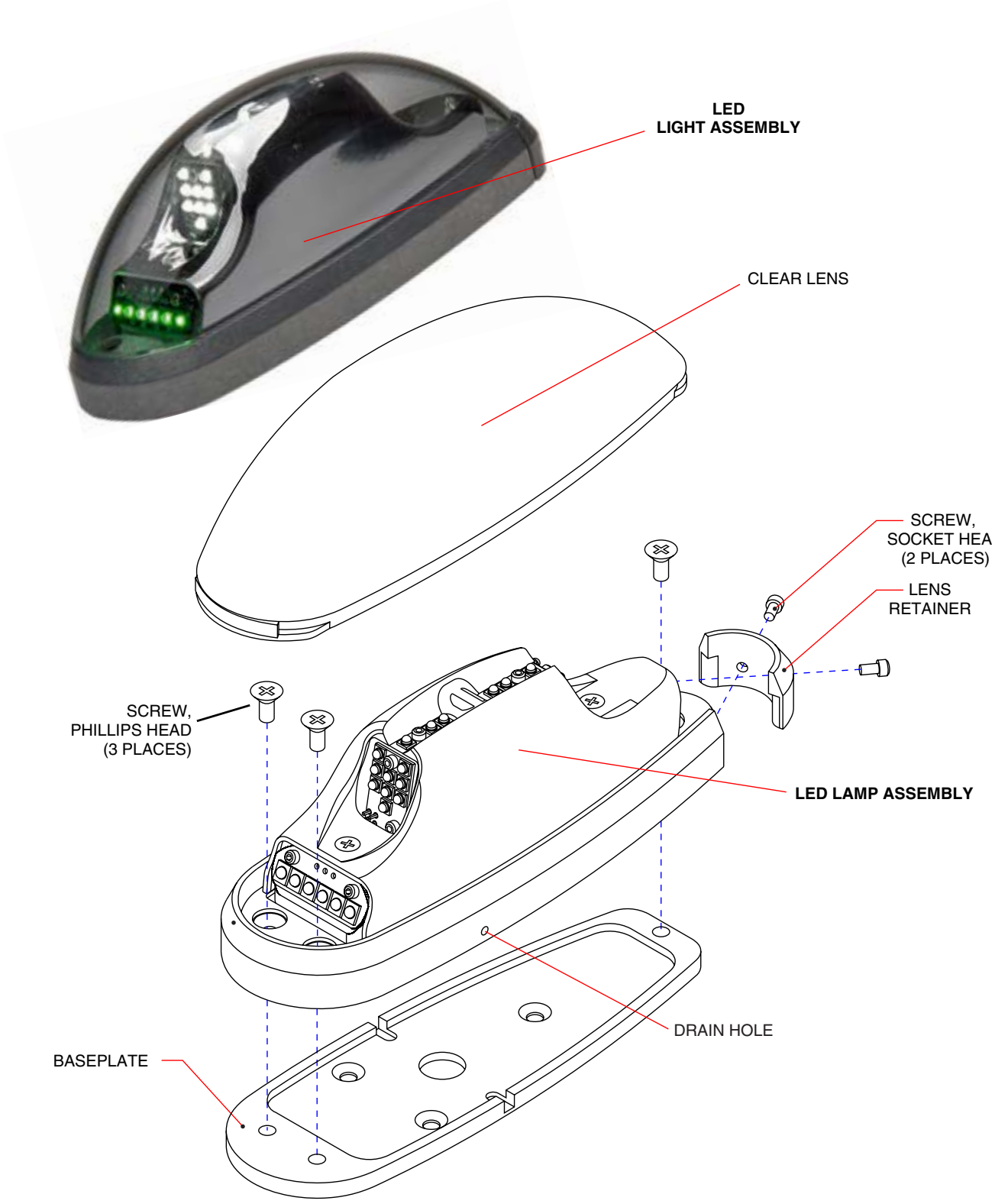
f) Switch NAV LiGHT and BATT MASTR OFF and secure airplane.

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Wing Tip Navigation/Position/Strobe Light Installation
Figure 10

Effectivity
Installation C



Effectivity
Installation C

Wing Tip Navigation/Position/Strobe Light Assembly
Figure 11

B. Tail Light (Optional)

A white tail light was available in Seneca IVs: and Seneca V S/N's 3449001–3449508. When installed it is mounted in the aft edge of the rudder tip fairing. See "Figure 12" on page 334022.

(1) Removal

Select BATT MASTR and NAV LIGHT switches OFF.

(a) Lamp

1) Remove the two screws securing the retainer mask.
2) Remove retainer mask, lens, and lens gasket from light assembly.
3) Pull the lamp straight out of the socket assembly.

(b) Light Assembly

1) At the rudder:
   a) Remove the twelve (12) screws securing the fairing assembly to the rudder.
   b) Lift the fairing assembly off the rudder and disconnect the tail light ground lead by removing the screw and washer securing it to the rudder rib.
   c) Disconnect the power lead at the splice.
2) At the bench:
   a) Remove the two screws securing the retainer mask.
   b) Remove retainer mask, lens, and lens gasket.
   c) Remove the light assembly housing from the fairing assembly.

(2) Installation

(a) Lamp

1) Align the new lamp with the socket assembly and press straight back until fully seated.
2) Install lens gasket, lens, and retainer assembly and secure with screws (2).

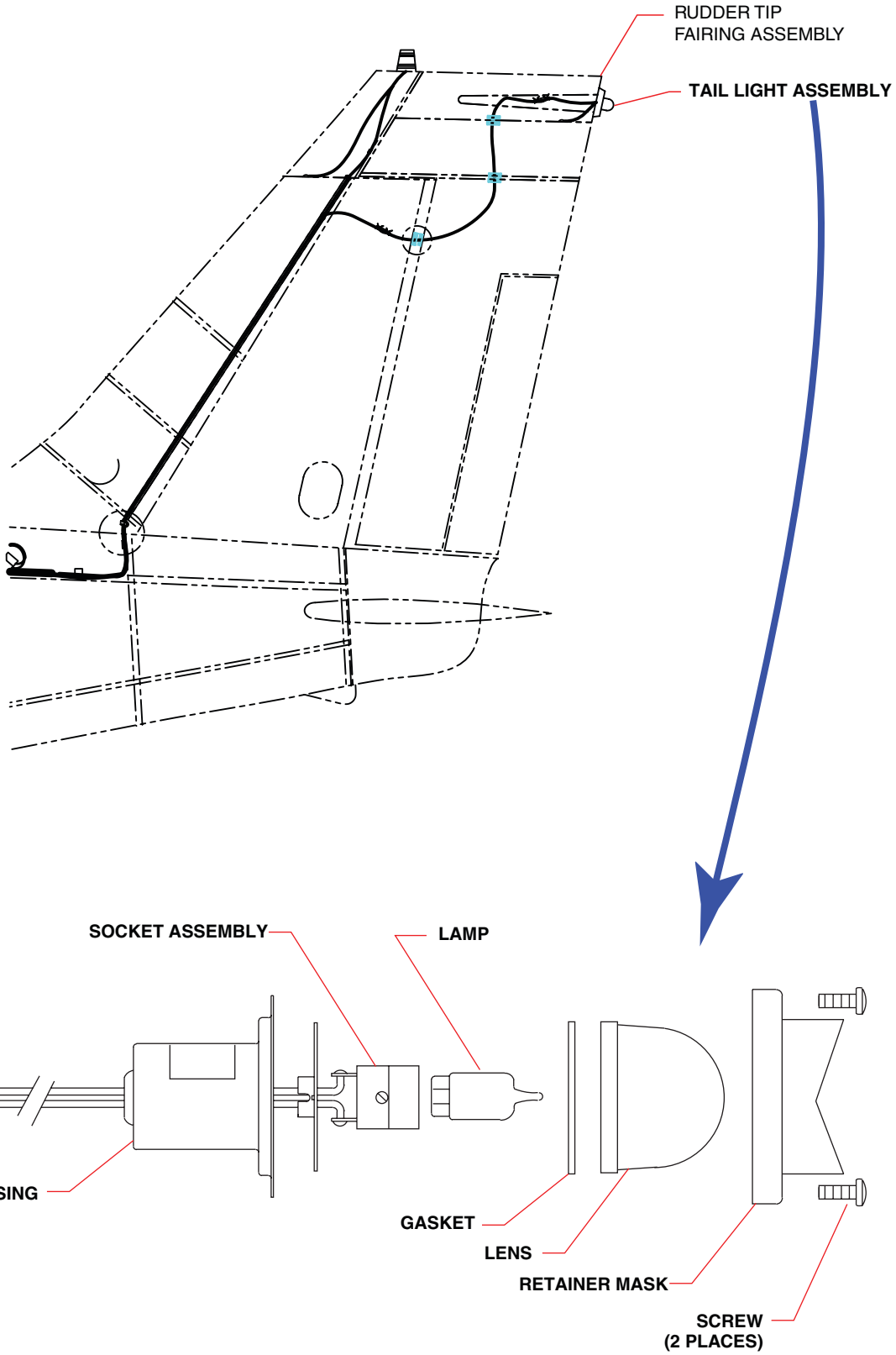
(b) Light Assembly

1) At the bench:
   a) Feed the electrical leads through the mounting hole in the rudder tip fairing assembly.
   b) Position light assembly housing in place in the fairing assembly.
   c) Position lens gasket, lens, and retainer assembly.
   d) Apply Loctite 222MS to screws (2).
   e) Secure light assembly to fairing assembly with screws (2).
2) At the rudder:
   a) Connect the light assembly ground lead to the rudder rib with screw and washer.
   b) Connect the power lead at the splice.
   c) Position fairing assembly in place atop the rudder and secure with screws (12).

(c) Select BATT MASTR and NAV LIGHT switches ON, verify tail light illuminates.

(d) Select BATT MASTR and NAV LIGHT switches OFF, secure airplane.

107270 R 5



Effectivity
3447001–3447029
3449001–3449508

Tail Light Assembly (Optional)
Figure 12

4. Anti-Collision Strobe Lights

A. Description

The exterior wingtip lighting assembly includes a pulsating anti-collision (i.e., strobe) white light. The strobe light is either a Xenon flashtube with a separate power supply; or, a LED strobe light. The strobe units are rated to flash approximately 50 times per minute. The anti-collision lights are controlled by an independent rocker switch (STROB LIGHT) in the overhead switch panel. The circuit is protected by a 5 amp ANTI-COLL LIGHTS circuit breaker.

A red fin tip ground reconigition/anti-collision (i.e., strobe) light is standard equipment. Again, the strobe light is either a Xenon flashtube with a separate power supply; or, a LED strobe light with no separate power supply. The unit is controlled by the three-position STROB LIGHT - OFF - FIN STROB rocker switch in the overhead switch panel. This allows the fin tip strobe to be activated with or independently of the wing tip strobes.

When installed the separate power supply unit is located in the aft section of the fuselage. See “Figure 17” on page 334033.

B. Troubleshooting (Xenon Flashtubes only.)

The strobe light assembly functions as a condenser discharge system. A condenser in the power supply is charged to approximately 450-volts DC, then discharged across the Xenon flash tube at intervals designed to hold off the 450-volts DC applied until the flash tube is triggered by an external pulse. This pulse is generated by a solid state timing circuit in the power supply.

When troubleshooting the strobe light system, it must first be determined if the trouble is in the flashtube or the power supply. Replacement of the flashtube will confirm if the tube is defective. A normal operating power system will emit an audible tone of 1 to 1.5 kHz. If there is no sound emitted, check the system according to the following instructions. When troubleshooting the system, utilize the appropriate schematic in 91-33-40.

CAUTION: WHEN DISCONNECTING POWER SUPPLY, ALLOW TEN MINUTES OF BLEED DOWN TIME BEFORE HANDLING UNIT.

(1) Ensure that the input voltage at the power supply is 28 volts.

CAUTION: WHEN DISCONNECTING AND CONNECTING POWER SUPPLY INPUT CONNECTIONS, DO NOT REVERSE CONNECTIONS. REVERSED POLARITY OF INPUT VOLTAGE FOR JUST AN INSTANT WILL PERMANENTLY DAMAGE POWER SUPPLY. REVERSED POLARITY DESTROYS A PROTECTIVE DIODE IN THE POWER SUPPLY, CAUSING SELF-DESTRUCTION FROM OVERHEATED POWER SUPPLY. THE DAMAGE IS SOMETIMES NOT IMMEDIATELY APPARENT, BUT WILL CAUSE SYSTEM FAILURE IN TIME.

(2) Check for malfunction in interconnecting cables.

(a) Ensure BATT and other appropriate switches are OFF.

(b) Disengage (PULL) appropriate circuit breakers.

NOTE: A short of the type described in Steps (2), (c) and (d) will not cause permanent damage to the power supply but the system will be inoperative if such a short exists. Avoid any connection between pins 1 and 3 of the interconnecting cable as this will discharge the condenser in the power supply and destroy the trigger circuit.

(c) Ensure that pins 1 and 3 of interconnecting cable are not reversed.

(d) Using an ohmmeter, check continuity between pins 1 and 3 of interconnecting cable. If a reading is obtained on the meter, the cable is shorted and should be replaced.

(3) Check interconnecting cables for shorts.

(a) Disconnect the output cables from the power supply outlets.

(b) Use an ohmmeter for the following continuity checks.

1) Check from pin 1 to pin 1, pin 2 to pin 2, and pin 3 to pin 3. If no continuity exists, the cable is broken and should be replaced.

2) Check for continuity between pins 1 and 2, 1 and 3, and 2 and 3 of the interconnect cable. If continuity exists between any of these connections, the cable is shorted and should be replaced.

3) Check for continuity from pins 1, 2, and 3 to airplane ground. If continuity exists, the cable is shorted and should be replaced.

(4) Check tube socket assembly for shorts.

(a) Disconnect tube socket assembly of the anti-collision light from the interconnecting cable.

(b) Use an ohmmeter for the following continuity checks.

1) Check continuity between Pin 1 of AMP connector to Pin 1 of tube socket. If no continuity exists, tube socket assembly is broken and must be replaced.

2) Check Pin 2 of AMP connector to Pins 6 and 7 of tube socket.

3) Check Pin 3 of AMP connector to Pin 4 of tube socket.

4) When making these tests, if no continuity exists, tube socket assembly is broken and must be replaced.

(5) Clear all possible shorts at the power supply, by disconnecting the output cables from the power supply outlets, and connect an operating strobe lighthead assembly or a strobe check unit directly to the power supply outlet, then apply the required voltage to the power supply input. If this application proves positive the power supply is in working condition, and the problem may be with the interconnecting cables.

(6) Most power supplies are protected against a short or open circuit on the output. In either case, the power supplies will effectively turn themselves off when subjected to a shorted output of a Xenon flash tube that refuses to flash.

(7) Strobe light power supplies are meant to be used, not to remain in an inactive state. Use them at all times, this will improve their proper functioning. Any strobe light power supply that has been out of service for a long period of time is subject to failure because the electrolytic condenser loses the polarity formation.

If this is the case, it is recommended to start operating the system on a voltage that is reduced by 25 percent for 10 to 15 minutes before putting the power supply into normal service. This will prevent overheating of the condenser while they reform. If the power supply, after a long period of non use, is operated at full voltage immediately, there is an excellent possibility that the condenser will become overheated

C. Wing Tip

Three different installations exist.

(1) Installation A

Incandescent navigation / position lamps combined with a Xenon flashtube are installed in Seneca IVs and in Seneca V S/N's 3449001–3449458, 3449460–3449466. Standard and Garmin G600 equipped airplanes have slight differences in the wiring compared to Avidyne Entegra equipped airplanes as shown in "Figure 6" on page 334013. See also "Figure 7" on page 334014.

(a) Removal

Removing the Xenon flashtube requires removing the entire light assembly.

1) Ensure that the BATT MASTR, STROB LIGHT, and NAV LIGHT switches are in OFF position.
2) Disengage (pull out) NAV LIGHTS and ANTI-COLL circuit breakers.
3) At the wing tip:
   a) Remove the two screws securing the lens retainer.
   b) Remove the lens retainer and the clear lens covering the strobe, as well as the lenses covering the forward and aft navigation/position lamps.
   c) Remove the three screws securing the light assembly to the mounting plate.
   d) Break sealant and move the light assembly out of the way. Draw electrical leads out to expose connectors. Capture wing harness to ensure it does not retract into wing.
   e) Disconnect the light assembly electrical leads from the wing harness and remove the light assembly from the wing tip.
   f) Remove old sealant from mating surfaces of light assembly and wingtip.
4) At the bench:

   Pull the strobe lamp (i.e., Xenon flashtube) and gasket away from the light assembly while carefully feeding the electrical leads through the hole in the base.

(b) Installation

1) At the bench:
   a) Ensure the gasket (and grommet, if any) are positioned properly.
   b) Carefully feed the strobe lamp (i.e., Xenon flashtube) electrical leads through the hole in the base.
2) At the wing tip:
   a) Connect the light assembly electrical leads to the wing harness. Replace any Ty-Wraps or sleeves previously removed.
   b) Release the wing harness and carefully feed the connector and strobe lamp electrical leads through the hole in the base.
   c) Put the light assembly in place and secure to mounting plate with three screws.

      CAUTION: MATING SURFACES MUST BE SEALED TO PREVENT WATER INTRUSION.

   d) Fillet seal edges of mating surfaces of light assembly and wingtip. Use Sealant Type 4, Group 1, Chart 11 in 91-10-00.
   e) Replace the clear lens over the strobe and the forward and aft navigation / position lens over their respective lamps. Capture lenses with the lens retainer.
   f) Secure the lens retainer with two screws.

(2) Installation B

LED navigation / position lamps combined with a Xenon flashtube are installed in Seneca V S/N's 3449459, 3449467–3449508. See "Figure 8" on page 334016 and "Figure 9" on page 334017.

Removal and Installation of the strobe lamp (i.e., Xenon flashtube) is the same as that shown under "Installation A" on page 334025.

(3) Installation C

LED navigation / position lamps combined with a LED strobe are installed in Seneca V S/N's 3449509 and up. See "Figure 10" on page 334019 and "Figure 11" on page 334020. If any of the lights fail, the entire LED lamp assembly must be replaced.

See "Removal" and "Installation" under "Installation C" on page 334018.

D. Fin Tip

A red fin tip strobe is installed atop the vertical stabilizer as standard equipment and functions primarily as a Ground Recognition Light. Three different installations exist.

(1) Installation A

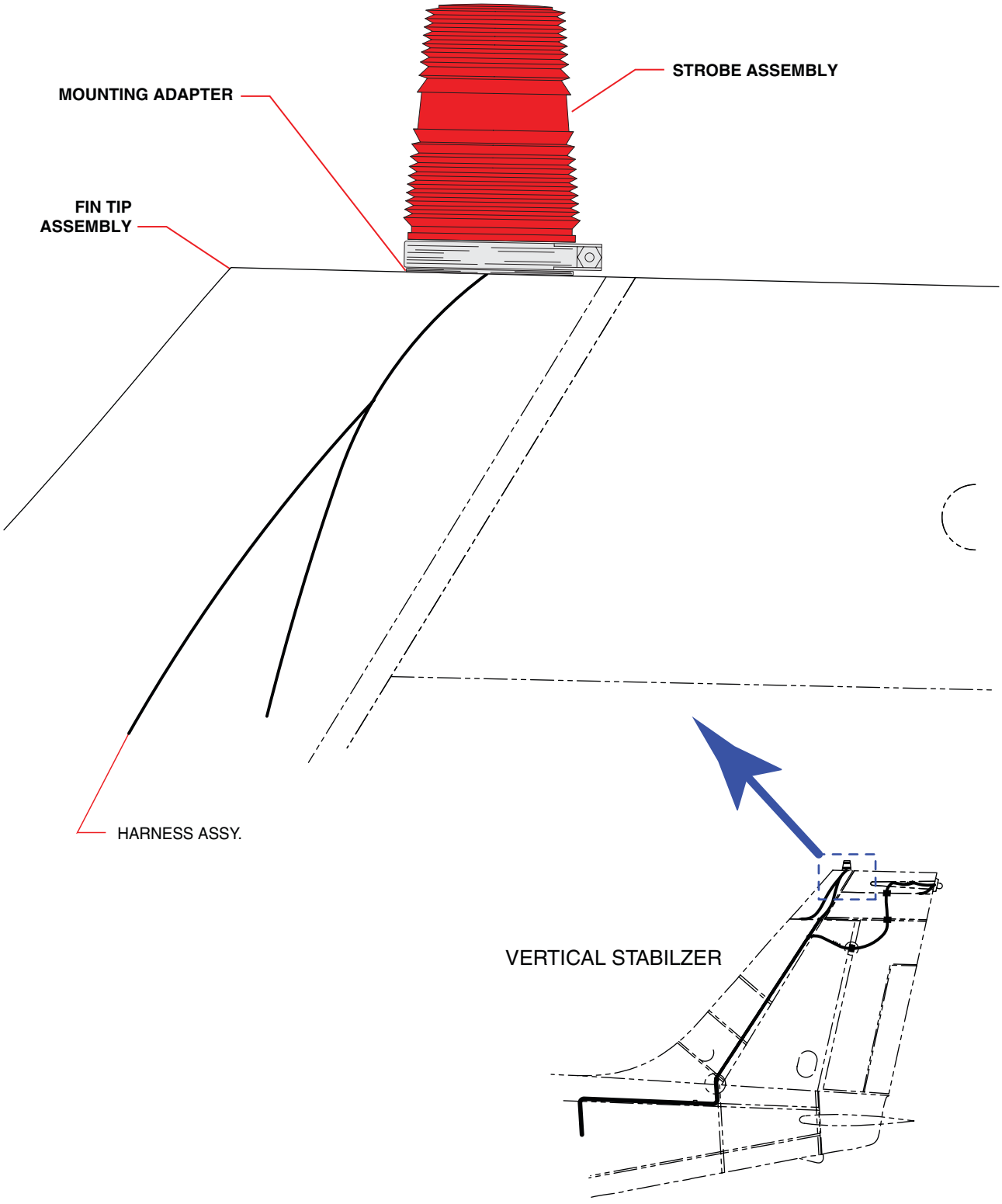
A xenon flashtube assembly is installed in Seneca IVs. See "Figure 13" on page 334027 and "Figure 14" on page 334028.

(a) Removal

1) Ensure that the BATT MASTR, STROB LIGHT, and NAV LIGHT switches are in OFF position.
2) Disengage (pull out) NAV LIGHTS and ANTI-COLL circuit breakers.
3) Remove the screw and nut securing clamp around the base of the strobe light assembly and the mounting plate.
4) Remove clamp and remove the strobe light assembly from the fin tip.
5) Pull the flashtube assembly and gasket out of the the lens.

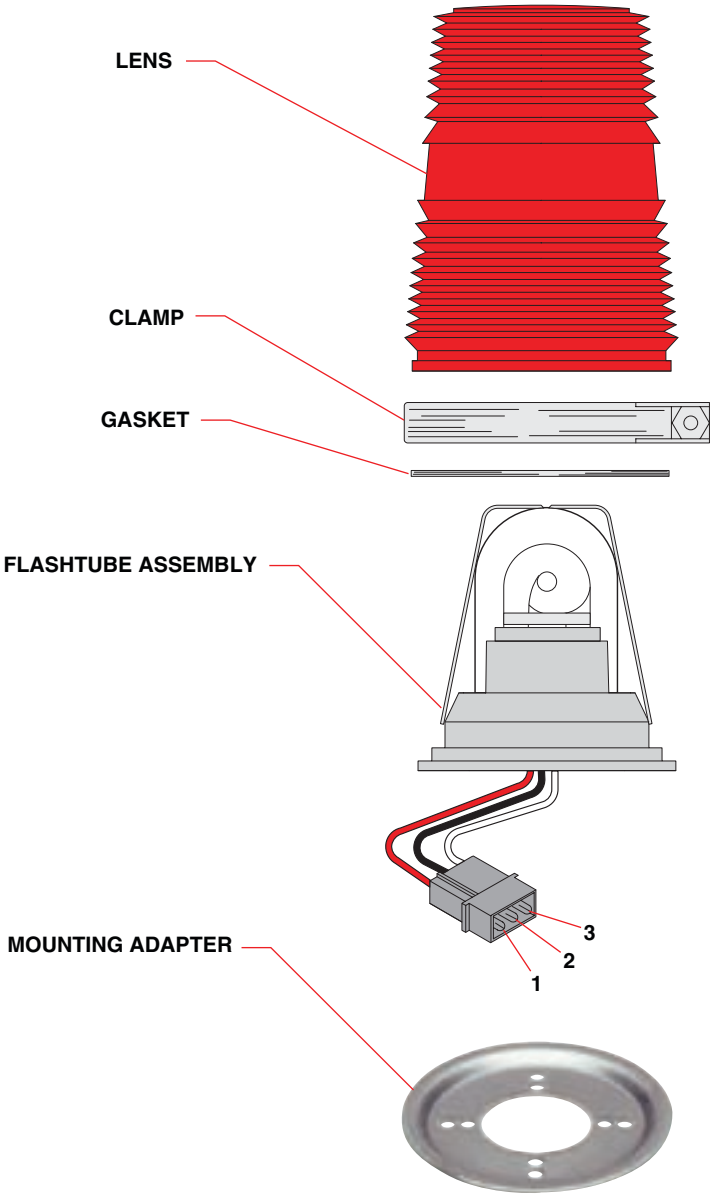
(b) Installation

1) Position the gasket properly.
2) Push flashtube assembly into the lens.
3) Put the light assembly in place atop the fin tip and secure to mounting plate with clamp, screw, and nut.
4) Engage (push in) the ANTI-COLL circuit breaker.
5) Switch ON the BATT MASTR and STROB LIGHT switches. Verify the wing tip and fin tip strobes flash.
6) Switch the STROB LIGHT switch OFF and then to FIN. Verify the only fin tip strobe flashes.
7) Select STROB LIGHT switch OFF. Set (push in) NAV LIGHTS circuit breaker.
8) Select BATT MASTR switch OFF.



Fin Tip Strobe / Ground Recognition Light Installation
Figure 13

Effectivity
Seneca IV



Effectivity
Seneca IV

Fin Tip Strobe / Ground Recognition Light Assembly
Figure 14

(2) Installation B

A xenon flashtube assembly is installed in Seneca V S/N's 3449001–3449508. See "Figure 15" on page 334030 and "Figure 16" on page 334031.

(a) Removal

1) Ensure that the BATT MASTR, STROB LIGHT, and NAV LIGHT switches are in OFF position.
2) Disengage (pull out) NAV LIGHTS and ANTI-COLL circuit breakers.
3) Remove four (4) screws securing the lens cover assembly and remove the lens cover assembly.
4) Lift the xenon flashtube assembly off its mounting base and withdraw the electrical leads until the connector is exposed.
5) Capture the ship's harness assembly to prevent it retracting into the vertical stablizer.
6) Disconnect the electrical leads and remove the flashtube assembly.

(b) Installation

1) Connect the electrical leads from the flashtube assembly to the ship's harness assembly.
2) Release the harness assembly and feed the electrical leads through the hole in the mounting base until the flashtube assembly is properly seated.
3) Reinstall the lens cover assembly and secure with four (4) screws.
4) Engage (push in) the ANTI-COLL circuit breaker.
5) Switch ON the BATT MASTR and STROB LIGHT switches. Verify the wing tip and fin tip strobes flash.
6) Switch the STROB LIGHT switch OFF and then to FIN. Verify the only fin tip strobe flashes.
7) Select STROB LIGHT switch OFF. Set (push in) NAV LIGHTS circuit breaker.
8) Select BATT MASTR switch OFF.

(3) Installation C

A LED strobe light assembly is installed in Seneca V S/N's 3449509 and up. See "Figure 15" on page 334030 and "Figure 16" on page 334031.

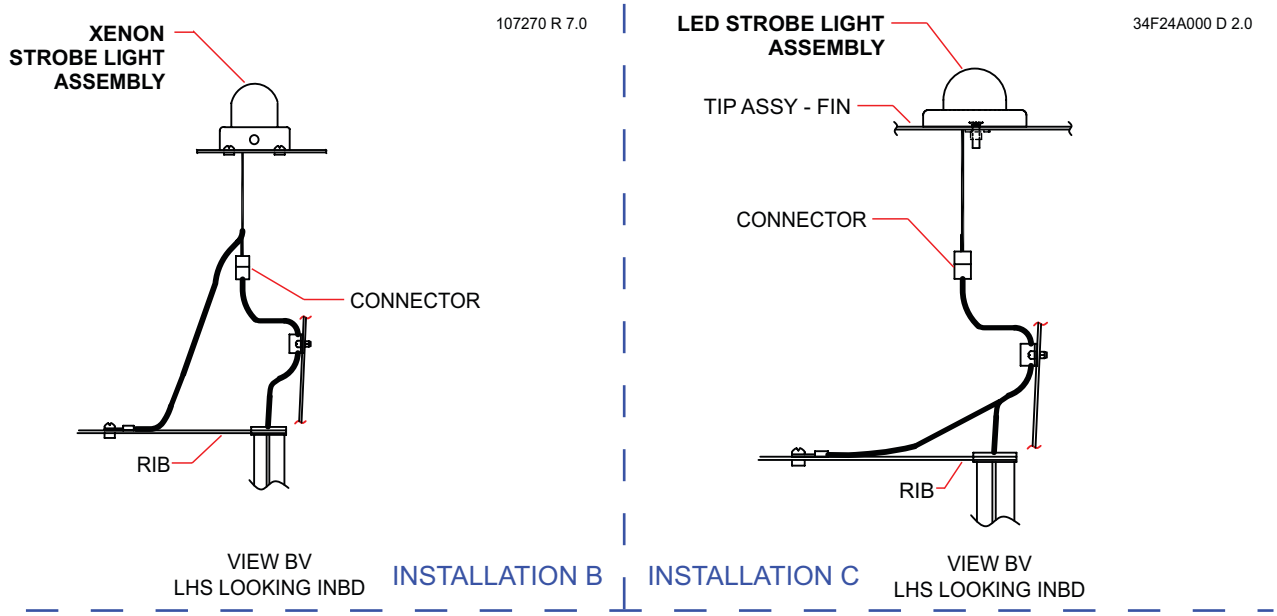
If the light fails the entire light assembly must be replaced.

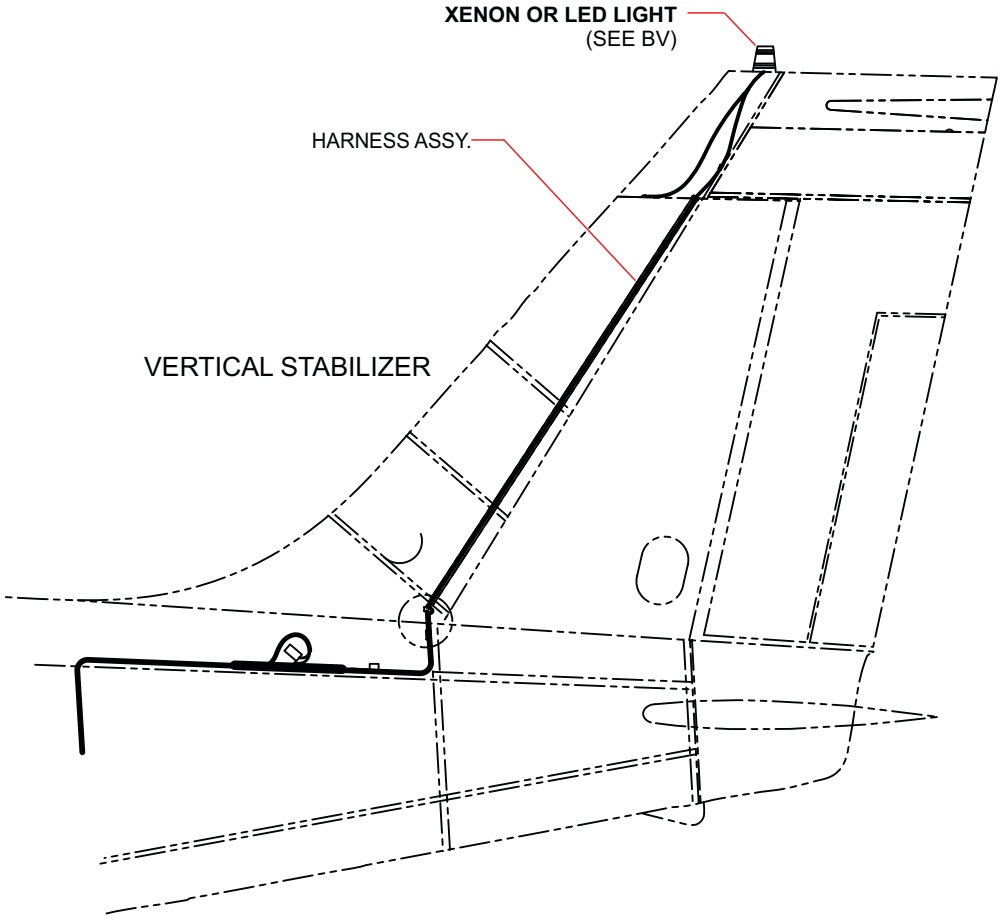
(a) Removal

1) Ensure that the BATT MASTR, STROB LIGHT, and NAV LIGHT switches are in the OFF position.
2) Disengage (pull out) NAV LIGHTS and ANTI-COLL circuit breakers.
3) Remove two (2) screws securing the lens retainer and remove the retainer, lens, and lens gasket.
4) Remove two (2) screws securing the LED assembly to the fin tip.

CAUTION: TAKE CARE NOT TO DAMAGE THE MOUNTING GASKET BONDED TO THE FIN TIP.

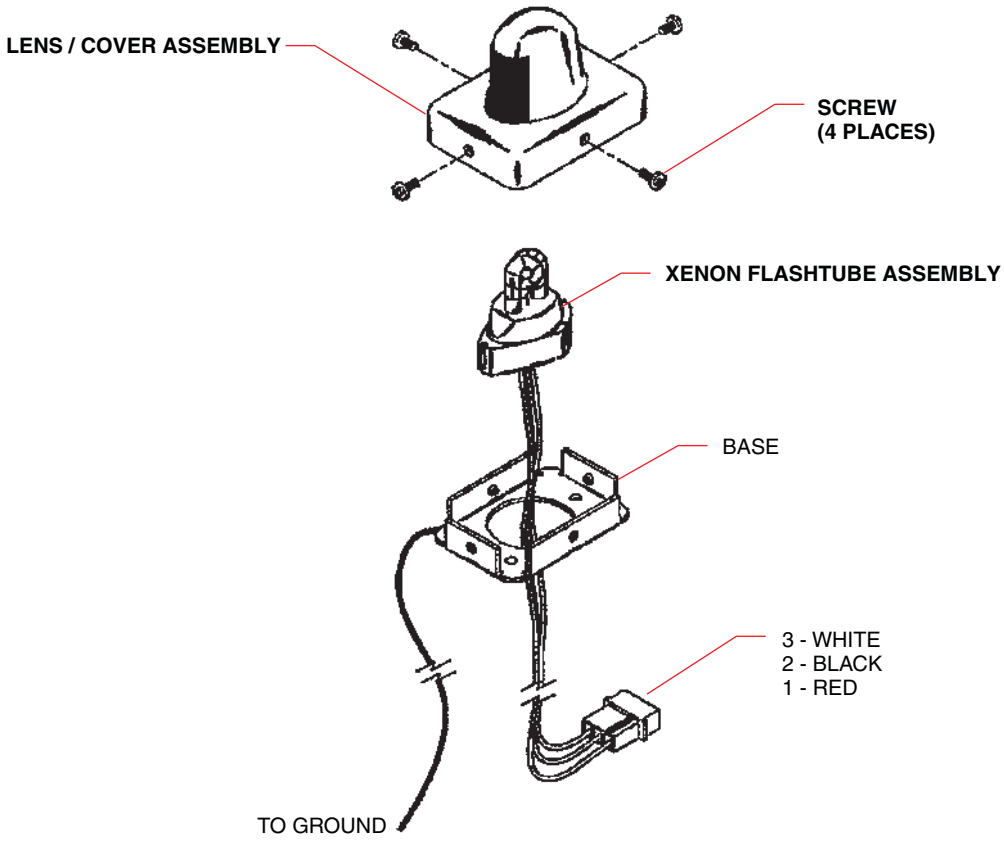
5) Lift the LED assembly off the fin tip and withdraw the electrical leads until the connector is exposed.
6) Capture the ship's harness assembly to prevent it retracting into the vertical stablizer.
7) Disconnect the electrical leads and remove the LED assembly.





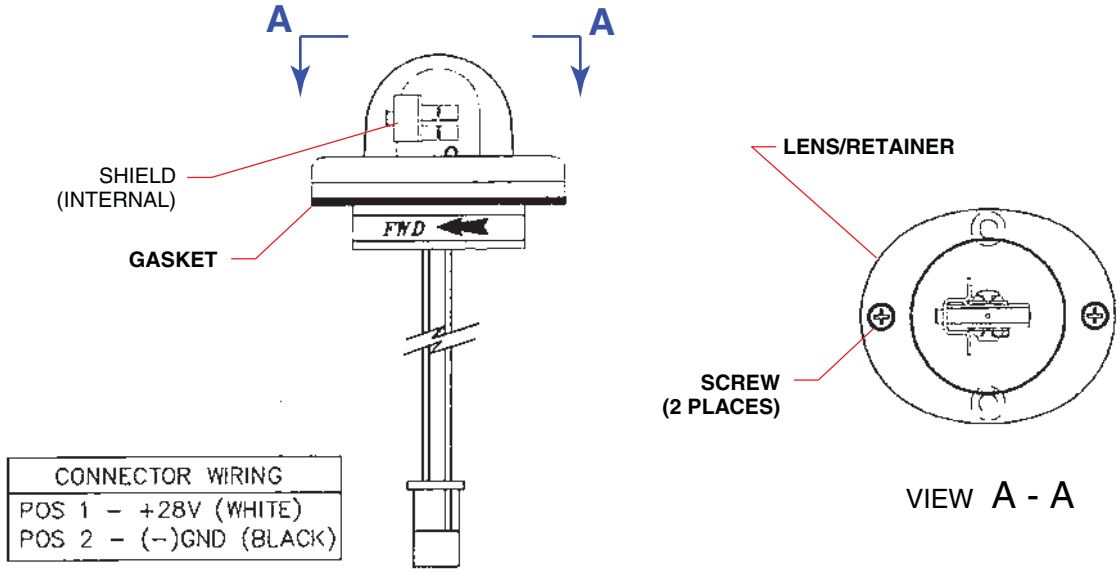
Effectivity
Seneca V

Fin Tip Strobe / Ground Recognition Light Installation
Figure 15



INSTALLATION B

INSTALLATION C



Fin Tip Strobe / Ground Recognition Light Assembly
Figure 16

Effectivity
Seneca V

(b) Installation

1) Examine the mounting gasket bonded to the fin tip for damage and condition.

a) If it is acceptable, proceed to Step 2).

b) If damaged, it must be replaced. Remove it and clean any residual adhesive from the fin tip. Bond a new mounting gasket in place with 3M Marine Adhesive/Sealant Fast Cure 4000 UV.

2) Connect the electrical leads from the LED assembly to the ship's harness assembly.

3) Release the harness assembly and feed the electrical leads through the hole in the mounting gasket until the LED assembly is properly positioned.

NOTE: Ensure the FWD arrow on the LED assembly is pointing forward and the internal shield on the LEDs is on the forward side of the LED assembly.

4) Secure with the LED assembly to the fin tip with two (2) screws.

5) Reinstall the lens gasket, lens, lens retainer, and secure with two (2) screws.

6) Engage (push in) the ANTI-COLL circuit breaker.

7) Switch ON the BATT MASTR and STROB LIGHT switches. Verify the wing tip and fin tip strobes flash.

8) Switch the STROB LIGHT switch OFF and then to FIN. Verify the only fin tip strobe flashes.

9) Select STROB LIGHT switch OFF. Set (push in) NAV LIGHTS circuit breaker.

10) Select BATT MASTR switch OFF.

E. Strobe Power Supply

In Seneca IVs; and, Seneca V S/N's 3449001–3449508, a strobe power supply is in the aft section of fuselage on the left side of the airplane mounted to a bracket fixed to the side and bottom of the fuselage. See "Figure 17" on page 334033.

(1) Removal

(a) Ensure that the BATT MASTR and STROB LIGHT switches are in OFF position.

(b) Disengage (pull out) ANTI-COLL LIGHTS circuit breaker.

(c) Remove access panel to aft section of fuselage in the baggage compartment to gain access to the power supply.

(d) Note and mark location; then disconnect the strobe harness plugs.

(e) Remove screws and washers (4 ea.) holding power supply to bracket.

NOTE: Ground wires are connected to the screws adjacent to the strobe harness plugs. Note and mark their location as well.

(f) Remove power supply.

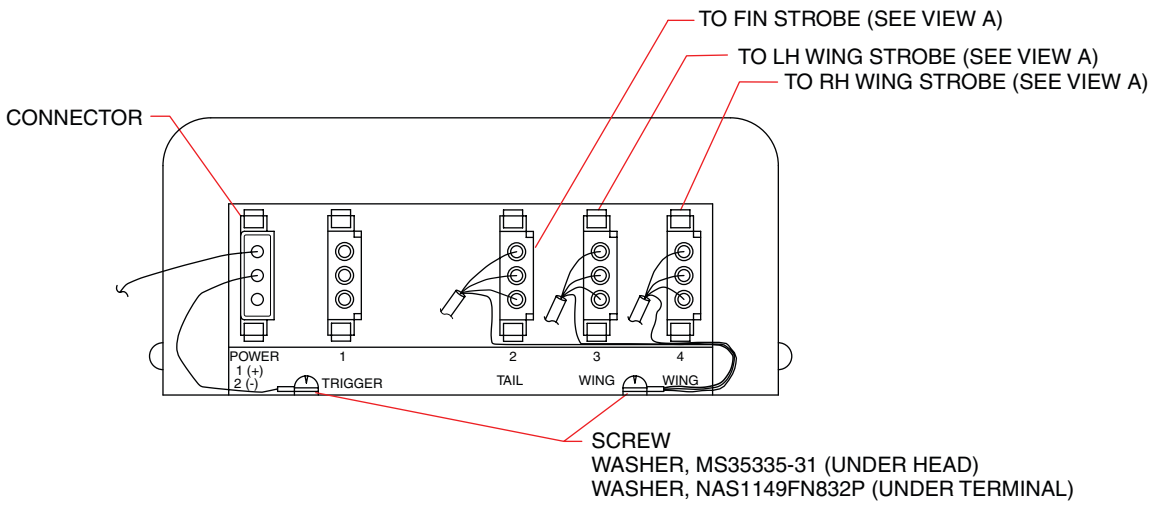
(2) Installation

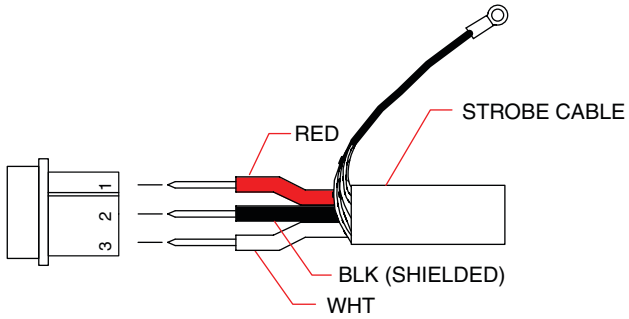
(a) Position power supply in place and secure with screws and washers (4 ea.) being sure to capture the appropriate ground wires.

(b) Connect strobe harness plugs in proper place.

(c) Set (push in) ANTI-COLL LIGHTS circuit breaker.

89424 V
101816 H





VIEW A

Xenon Strobe Power Supply Connections
Figure 17

5. Ice Detection Light (Optional)

This light is installed with the pneumatic deicing system and will aid the pilot to detect any ice formation on the left wing leading edge during night flying operations.

The light is mounted in the left outboard edge of the left nacelle just above leading edge of the wing. The light is controlled from a push ON–push OFF type switch mounted in the deice switch grouping in the center of the instrument panel. The light is positioned in the nacelle to illuminate the leading edge of the wing when the switch is activated in the cockpit. There are three separate installations.

A. Installation A

In Seneca IVs; and, Seneca V S/N's 3449001–3449066, the wing ice light is riveted to the inside of the left nacelle skin. See "Figure 18 (Sheet 1 of 4)" on page 334035.

(1) Removal

(a) Set the BATT MASTR and ICE LIGHT switches to the OFF position.

(b) Remove the top access panel from the left nacelle.

(c) Within the nacelle, remove the three (3) screws securing the base assembly to the retainer.

(d) Pull the socket and lamp free of the retainer.

(e) Push in and twist the lamp counterclockwise and remove the lamp.

(2) Installation

(a) Align the lamp with the socket of the base assembly, push in and twist clockwise until seated.

(b) Insert the lamp into the retainer and align the base assembly with the screw holes.

(c) Secure the base assembly to the retainer with three (3) screws.

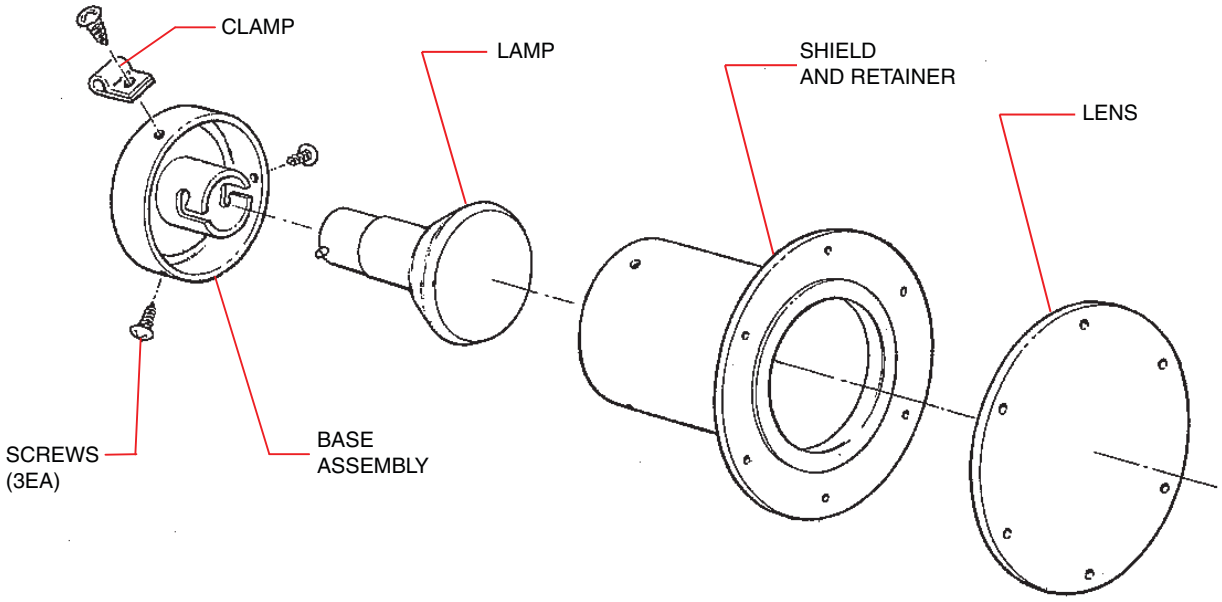
(d) Set the BATT MASTR and ICE LIGHT switches to the ON position.

(e) Verify the ice light illuminates.

(f) Set the BATT MASTR and ICE LIGHT switches to the OFF position.

(g) Replace the nacelle access panel with the attachment hardware.

472-900



Ice Light Installation
Figure 18 (Sheet 1 of 4)

Effectivity
3447001–3447029
3449001–3449066

B. Installation B

In Seneca V S/N's 3449067–3449458 and 3449460–3449466, the wing ice light is attached to the inside of the left nacelle skin with four (4) screws. See "Figure 18 (Sheet 2 of 4)", and "Figure 18 (Sheet 3 of 4)" on page 334038.

(1) Removal

(a) Set the BATT MASTR and ICE LIGHT switches to the OFF position.

(b) Remove the top access panel from the left nacelle.

(c) Within the nacelle, unscrew the end cap bushing from the reflector housing assembly.

(d) Pull the lamp assembly socket and lamp free of the reflector housing.

NOTE: Take care not to lose the O-ring or lamp bushing.

(e) Pull the lamp straight out to remove.

(2) Installation

(a) Align the lamp with the socket and push straight in until seated.

(b) Insert the lamp bushing and lamp assembly into the reflector housing assembly and secure with the end cap bushing.

(c) Set the BATT MASTR and ICE LIGHT switches to the ON position.

(d) Verify the ice light illuminates.

(e) Set the BATT MASTR and ICE LIGHT switches to the OFF position.

(f) Replace the nacelle access panel with the attachment hardware.

C. Installation C

In Seneca V S/N's 3449459, 3449467 and up, the LED wing ice light is attached to the inside of the left nacelle skin with four (4) screws. If the light fails, the entire assembly must be replaced. See "Figure 18 (Sheet 2 of 4)", and "Figure 18 (Sheet 4 of 4)" on page 334039.

(1) Removal

(a) Set the BATT MASTR and ICE LIGHT switches to the OFF position.

(b) Remove the top access panel from the left nacelle.

(c) Inside the nacelle, disconnect the J349 connector.

(d) Have an assistant hold the light assembly and shim inside the nacelle.

(e) From outside the nacelle, remove the four (4) screws securing the LED ice detection light assembly to the nacelle skin.

(f) The assistant can now remove the LED ice detection light assembly and shim.

(2) Installation

(a) Align the shim and LED ice detection light assembly with the holes in the nacelle skin.

(b) Have an assistant insert the four (4) screws from outside the nacelle and tighten them with a handheld screw driver.

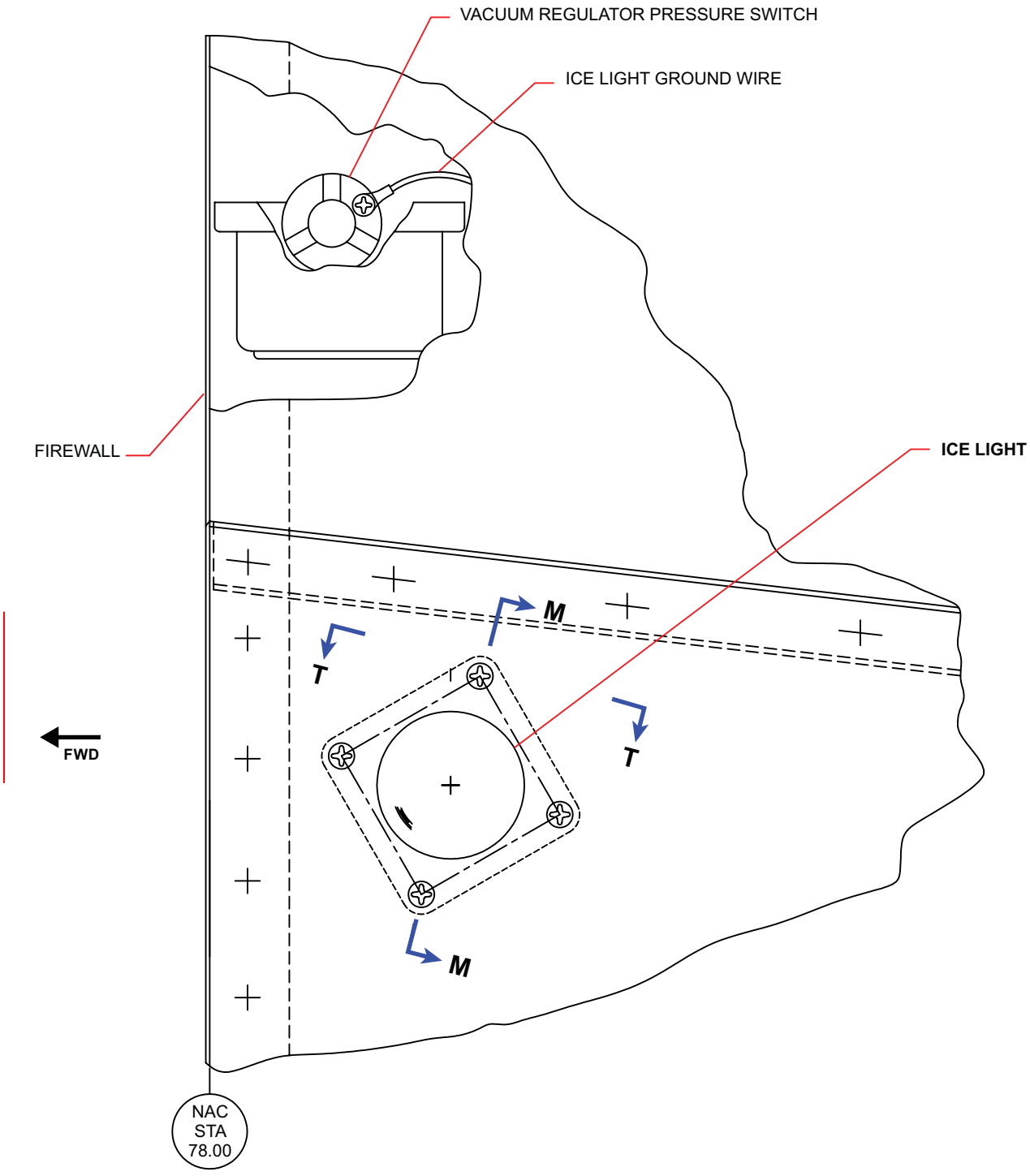
(c) Set the BATT MASTR and ICE LIGHT switches to the ON position.

(d) Verify the ice light illuminates.

(e) Set the BATT MASTR and ICE LIGHT switches to the OFF position.

(f) Replace the nacelle access panel with the attachment hardware.

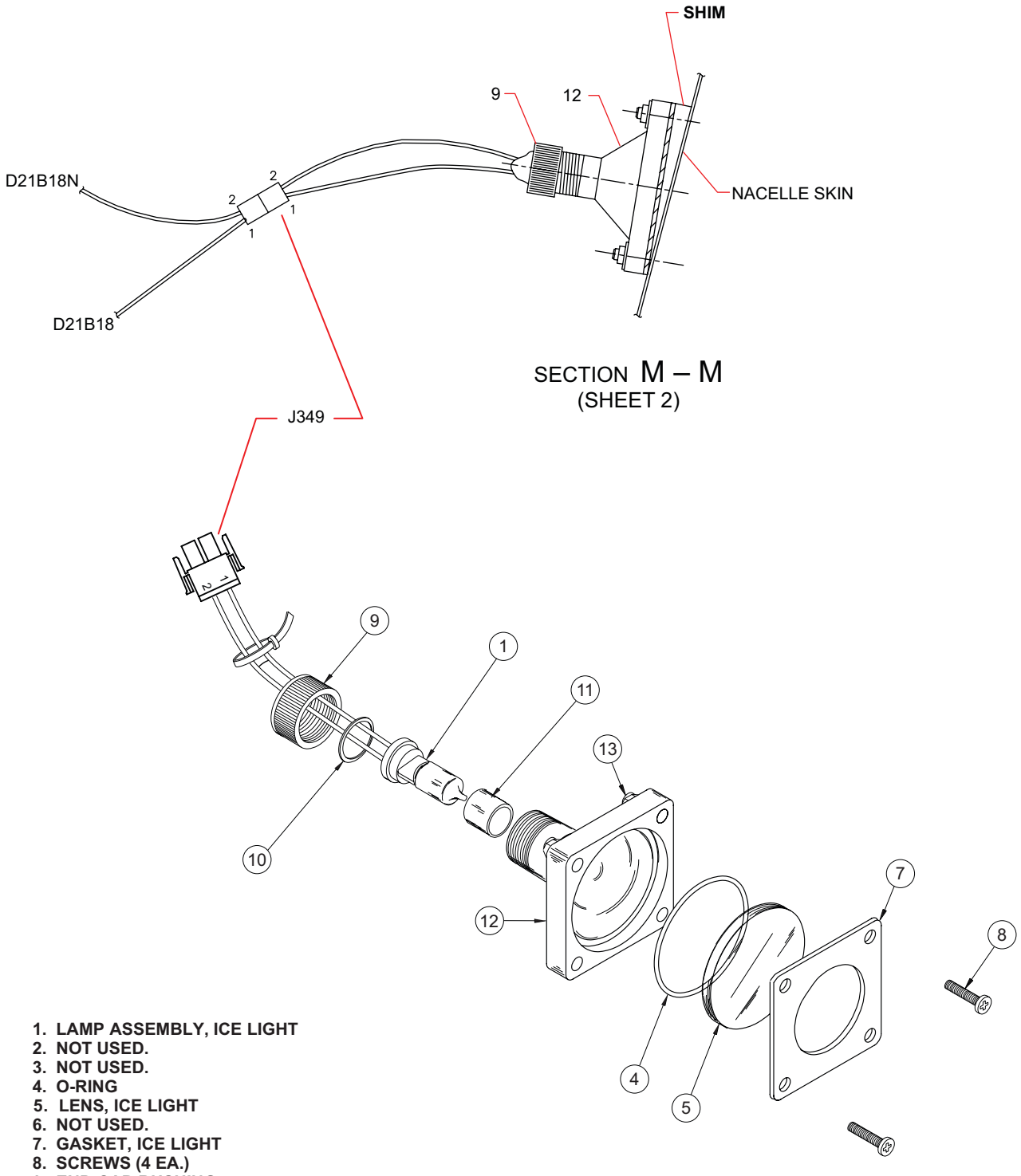
87291 Y



LOOKING AT LEFT SIDE OF LEFT NACELLE

Ice Light Installation
Figure 18 (Sheet 2 of 4)

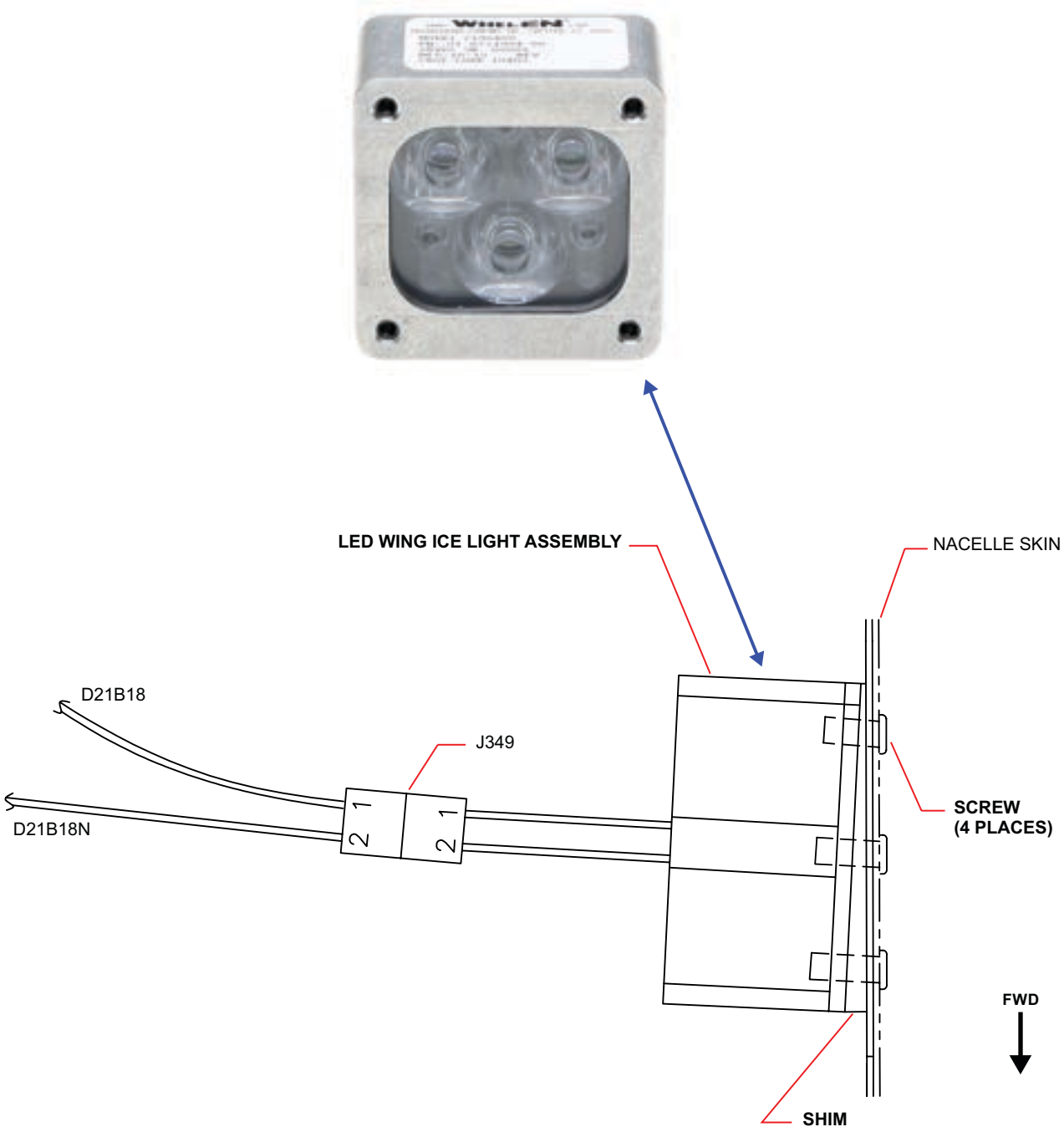
Effectivity
3449067 and up



1. LAMP ASSEMBLY, ICE LIGHT
2. NOT USED.
3. NOT USED.
4. O-RING
5. LENS, ICE LIGHT
6. NOT USED.
7. GASKET, ICE LIGHT
8. SCREWS (4 EA.)
9. END CAP BUSHING
10. GASKET, LAMP ASSEMBLY
11. LAMP BUSHING
12. REFLECTOR HOUSING ASSEMBLY
13. NUT (4 EA.)

Effectivity
3449067–3449458,
3449460–3449466

Ice Light Installation
Figure 18 (Sheet 3 of 4)



SECTION T – T
(SHEET 2)
(ROTATED 90 DEGREES COUNTERCLOCKWISE)

Ice Light Installation
Figure 18 (Sheet 4 of 4)

Effectivity
3449509 and up

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CHAPTER

34

NAVIGATION

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CHAPTER 34

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GENERAL

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

1. Standard

Seneca IVs; and Seneca V S/N's 3449001–3449310, 3449312–3449322, and 3449324–3449325.

Traditional 3.25 inch flight instruments are installed as standard equipment. These instruments are face mounted, see 39-10-00 for removal and installation.

2. Meggitt Single-side EFIS Option

S/N's 3449269 and up.

The Meggitt Avionics next Generation Integrated Cockpit (MAGIC) Electronic Flight Instrument System (EFIS) is available as an option on the pilot's side only. When installed, it's the primary flight and navigation instrumentation. The MAGIC EFIS consists of three primary components:

A. Primary Flight Display (PFD) [i.e. - Electronic Attitude Director Indicator (EADI)].

B. Navigation Display (NAV) [i.e. - Electronic Horizontal Situation Indicator (EHSI)].

C. Air Data & Attitude Heading Reference System (ADAHRS) [i.e. - Attitude Heading Reference System (AHRS)].

Line maintenance of this system is limited to basic troubleshooting and that is addressed in 34-21-00.

3. Avidyne Entegra IFDS

S/N's 3449311, 3449323, 3449326 thru 3449409, 3449411 thru 3449414, and 3449420.

The Avidyne Entegra Integrated Flight Display System (IFDS) was installed as the primary flight and navigation instrumentation. This system uses two large 10.4-inch diagonal, high-resolution, sunlight-readable full color displays (PFD and MFD), to provide primary flight and engine information

Line maintenance of this system is limited to basic troubleshooting and that is addressed in 34-22-00.

4. Garmin G600 Flight Display System

S/N's 3449410, 3449415–3449419, 3449421–3449458, and 3449460–3449466.

The Garmin G600 Flight Display System replaces the standard six-pack flight instruments, along with any external CDIs and ADF indicator. Standby instruments (ADI, airspeed indicator, altimeter and non-stabilized magnetic heading indicator (magnetic compass)) are provided.

Line maintenance of this system is limited to basic troubleshooting and that is addressed in 34-23-00.

5. Garmin G1000 Integrated Avionics System

S/N's 3449459, 3449467–3449508.

The G1000 system is an integrated avionics system in the Seneca V and has a three panel configuration with 2 GDU1040 10.4-inch LCD panels as Primary Flight Display (PFD's), and one GDU1040 10.4-inch LCD panel as the Multi-Function Display (MFD). Functions provided by the system presents flight instrumentation, position, navigation, communication, and identification information. The system consists of the following Line Replaceable Units (LRUs):

- GIA 63W Integrated Avionics Unit (2)
- GEA 71 Engine/Airframe Unit (2)
- GDC 74A Air Data Computer (ADC) (1) Standard, (2) Optional
- GTP 59 Outside Air Temperature (OAT) Probe (1) Standard, (2) Optional
- GRS 77 Attitude and Heading Reference System (AHRS) (1) Standard, (2) Optional
- GMU 44 Magnetometer (1) Standard, (2) Optional
- GMA 1347 Audio Panel
- GTX 33 Mode S Transponder
- GWX 68 Weather Radar (optional)
- GTS 825 TAS (optional)
- GSR 56 IRIDIUM (optional)
- GDL 69A Satellite Data Link Receiver (optional)
- GRT 10 XM Remote Control (optional)

An Aspen Avionics Standby Instrument System is also installed as secondary, independent source of attitude, altitude, airspeed, vertical speed and heading information.

Line maintenance of this system is limited to basic troubleshooting and that is addressed in 34-25-01.

6. Garmin G1000 NXi Integrated Avionics System

S/N's 3449509 and up.

The G1000 NXi system is an integrated avionics system in the Seneca V and has a three panel configuration with two GDU 1050 LCD panels as Primary Flight Displays (PFD1 and PFD2), and one Garmin GDU 1054 (with GFC 700 w/o Yaw damper) or GDU 1055 (with GFC 700 with Yaw Damper) as the Multi-Function Display (MFD). Functions provided by the system presents flight instrumentation, position, navigation, communication, and identification information. The system consists of the following Line Replaceable Units (LRUs):

- GIA 64 Integrated Avionics Unit (2)
- GEA 71B Engine/Airframe Unit (2)
- GDC 72 Air Data Computer (ADC) (1) Standard, (2) Optional
- GTP 59 Outside Air Temperature (OAT) Probe (1) Standard, (2) Optional
- GRS 79 Attitude and Heading Reference System (AHRS) (1) Standard, (2) Optional
- GMU 44 Magnetometer (1) Standard, (2) Optional
- GMA 1360 Audio Panel
- GTX 335R Mode A, C, S Transponder with ADS-B out.
- GTX 345R Mode A, C, S Transponder with ADS-B out. (Bluetooth ADS-B in Traffic and Weather to smart devices (Optional).
- GWX 75/80 Weather Radar (optional)
- GTS 825 TAS (optional)
- GSR 56 IRIDIUM (optional)
- GDL 69eA Satellite Data Link Receiver (optional)
- FS-510 Flightstream Bluetooth and WiFi Connectivity (optional).

An Aspen Avionics Standby Instrument System is also installed as secondary, independent source of attitude, altitude, airspeed, vertical speed and heading information.

Line maintenance of this system is limited to basic troubleshooting and that is addressed in 34-25-02.

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FLIGHT ENVIRONMENT DATA

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

The instrument air system consists of pitot air and static air sources. The system supplies both pitot and static air pressure for the airspeed indicator, and static pressure for the altimeter and vertical speed indicator. When installed, pitot and static air pressure is also provided for the Meggitt Single-side EFIS Air Data and Attitude Heading Reference System (ADAHRS), the Avidyne Entegra IFDS ADAHRS, or the Garmin Air Data Computer (ADC).

1. Pitot and Static System

A. Description and Operation

See “Figure 1” on page 34102.

The pitot air system consists of a pitot mast located on the underside of the left wing, with its related plumbing. Impact air pressure entering the pitot is transmitted from the pitot inlet through hose and tubing routed through the wing to the airspeed indicator on the instrument panel and, if installed, to the ADAHRS unit located on the aft equipment shelf (Meggitt), to the PFD (Avidyne Entegra), or to the Air Data Computer (Garmin). A partially or completely blocked pitot head will give erratic or zero reading on the instruments.

Static air system consists of two static ports located on the sides of the aft fuselage just forward of station 220.164. The static ports are directly connected to the airspeed indicator, altimeter and vertical speed indicator on the instrument panel and, if installed, to the ADAHRS unit located on the aft equipment shelf (Meggitt), to the PFD (Avidyne Entegra), or to the Air Data Computer (Garmin) by means of hose and tubing routed along the top of the fuselage center line to station 138.627, where it is then routed to, down and along the left side of the fuselage to the back side of the instrument panel. An alternate static air source is located below the instrument panel to the right of the power quadrant (when equipped with standard instrumentation, the optional Meggitt, the Avidyne Entegra IFDS, or the Garmin G600 Flight Display) or below the pilot’s PFD (when equipped with the Garmin G1000 / G1000 NXi IAS). The alternate static source is part of the standard system and has a shutoff valve which closes the port when it is not needed. A placard giving instructions for use is located on the instrument panel near the control. Pitot and static lines can be drained through separate drain valves located on the left lower side of the fuselage interior.

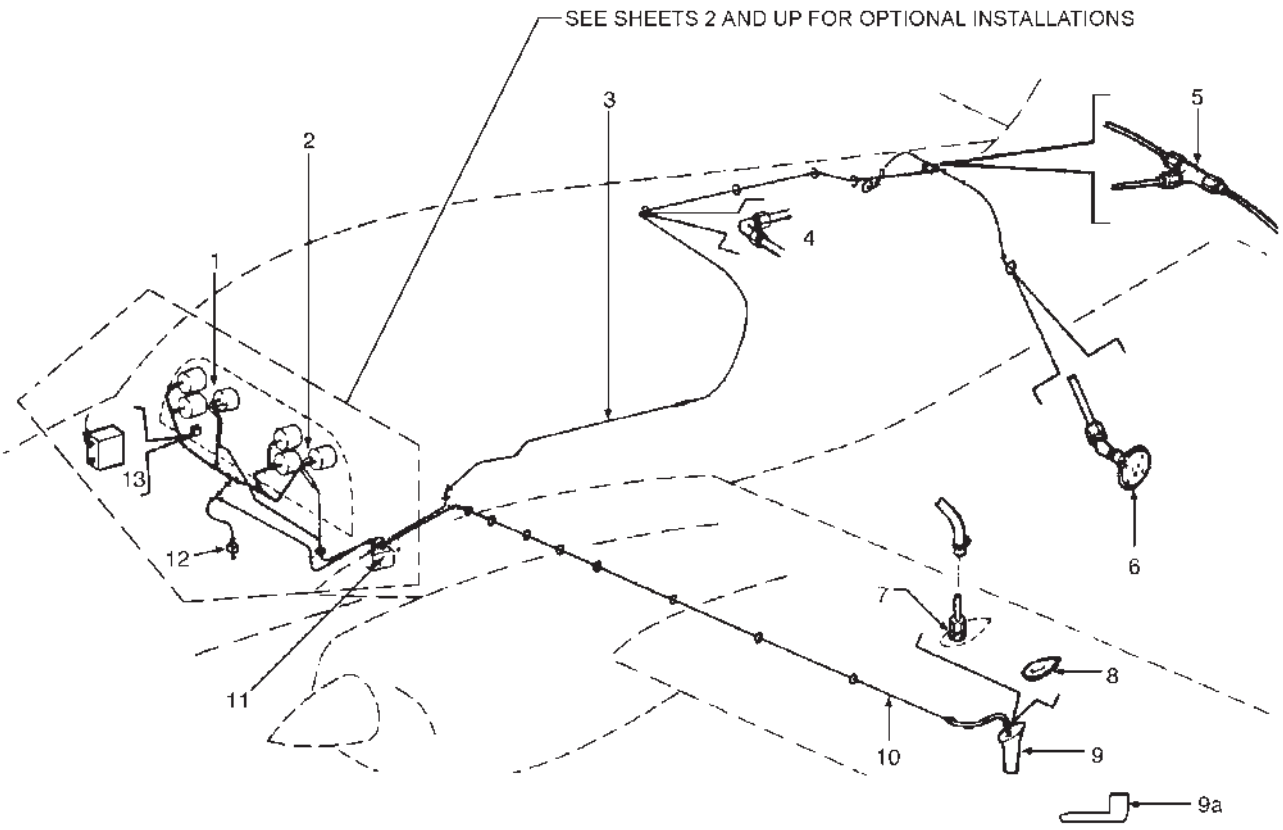
The holes in the sensors for pitot and static pressure must be fully open and free from blockage. Blocked sensor holes will give erratic or zero readings on the instruments.

The heated pitot head is standard equipment. The pitot heat switch is located on the right overhead switch panel. Static source pads have been demonstrated to be non-icing; however, in the event that icing does occur, selecting the alternate static source will alleviate the problem.

B. Troubleshooting

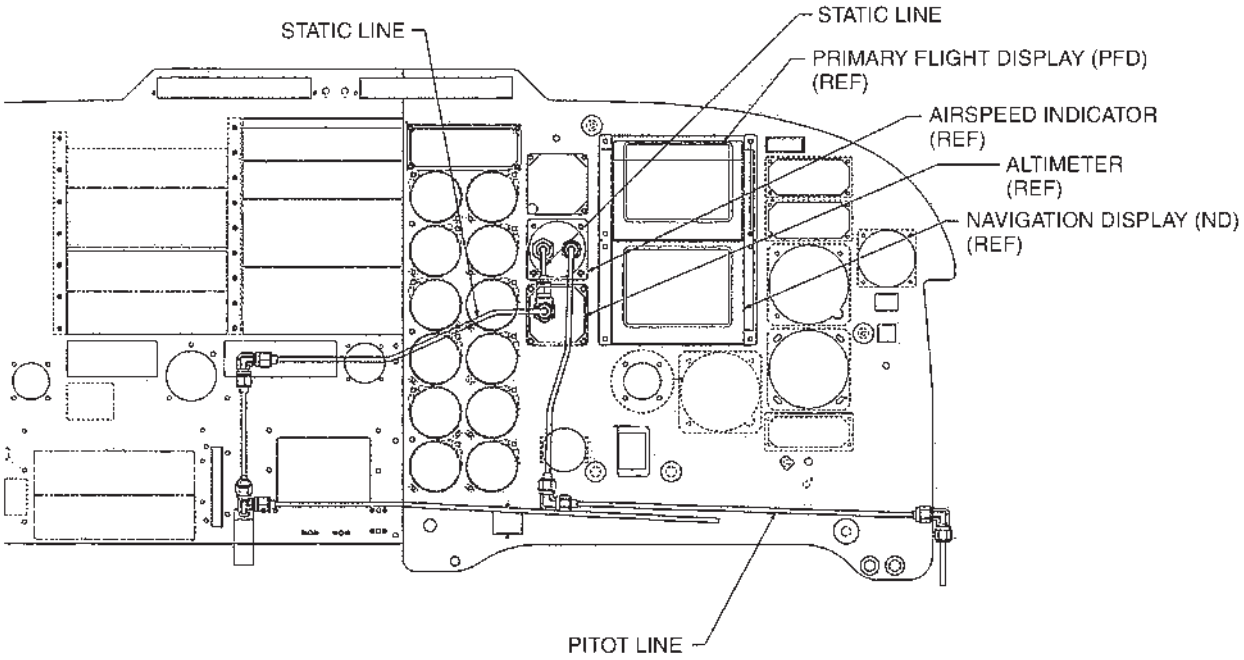
See “Chart 1” on page 34107.

NOTE: If any connection in the pitot / static system is opened for maintenance, the entire system must be rechecked per Pitot / Static System “Test” on page 34108. With the optional Meggitt Single-side EFIS installed, see also Appendix 1.

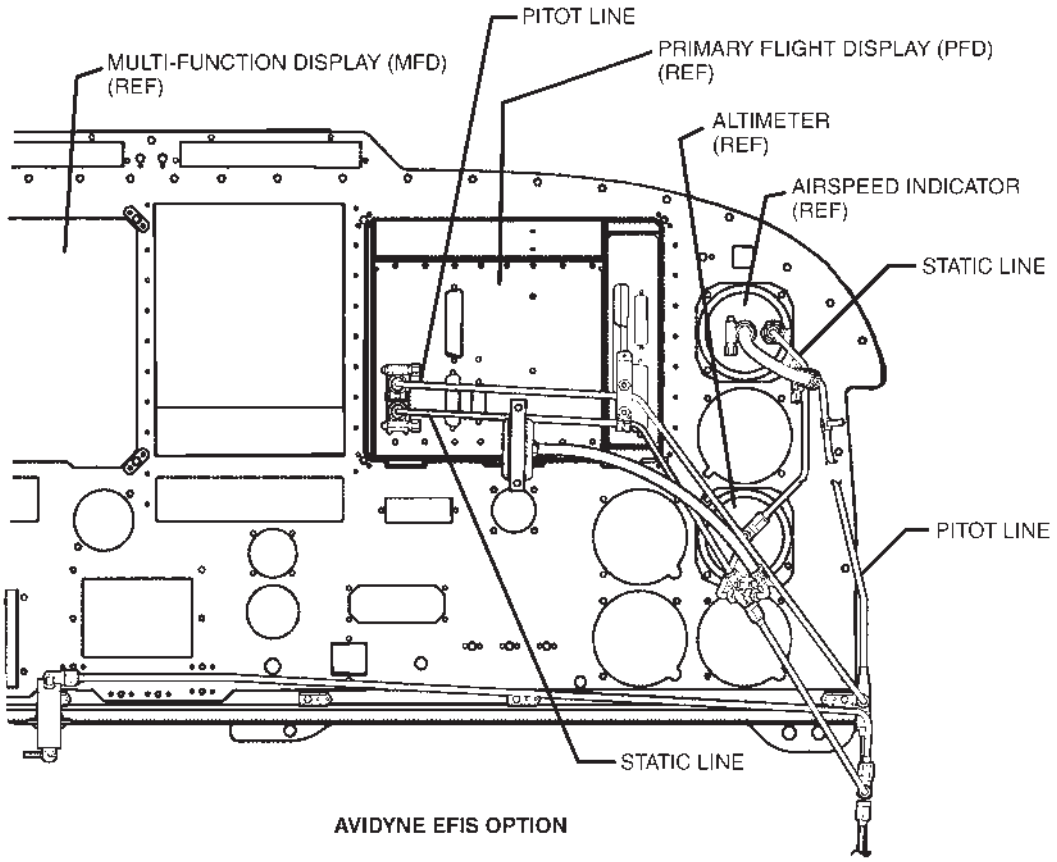


1. CO-PILOT'S INSTRUMENT
2. PILOT'S INSTRUMENT
3. STATIC LINE
4. ELBOW
5. TEE
6. STATIC BUTTON
7. CONNECTOR
8. GASKET
9. PITOT HEAD (SENECA IV)
9a. PITOT HEAD (SENECA V)
10. PITOT LINE
11. SUMP ASSEMBLY - DRAIN
12. VALVE - ALTERNATE STATIC SOURCE
13. PITOT HEAT CIRCUIT BREAKER

Pitot-Static System
Figure 1 (Sheet 1 of 5)

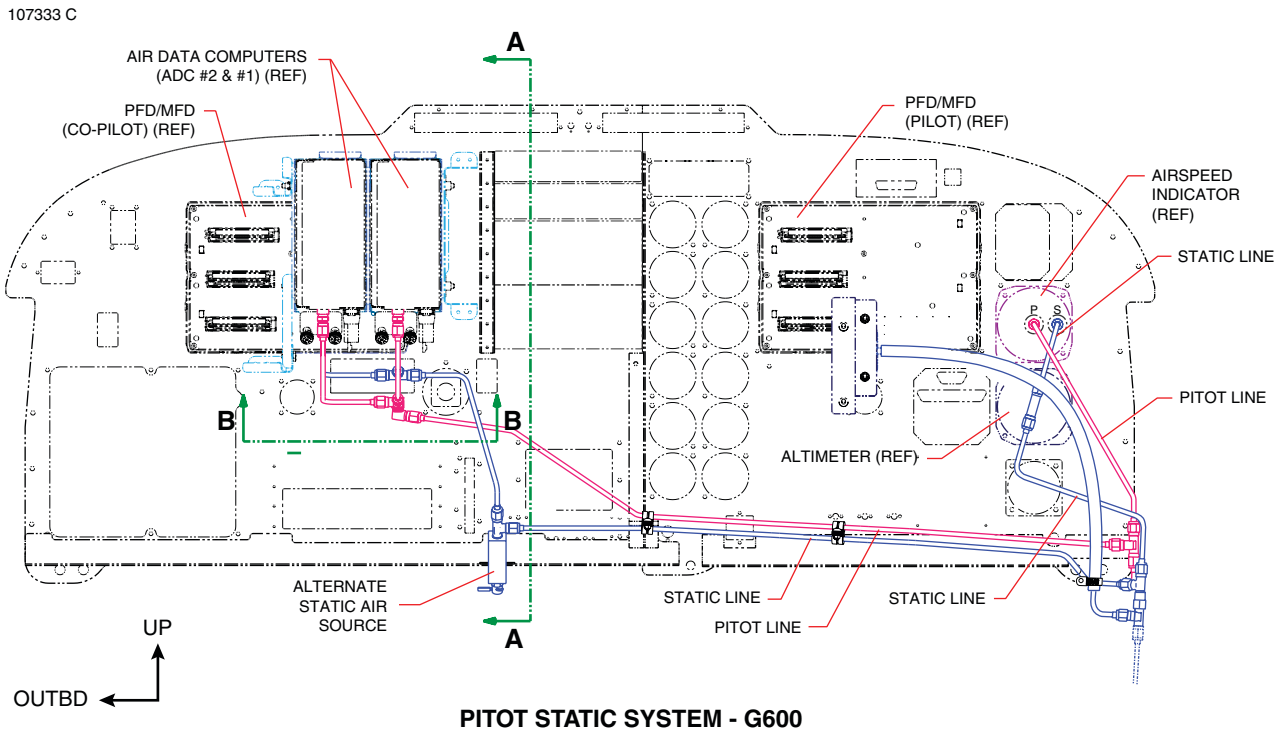


SINGLE-SIDED EFIS OPTION (MEGGITT)

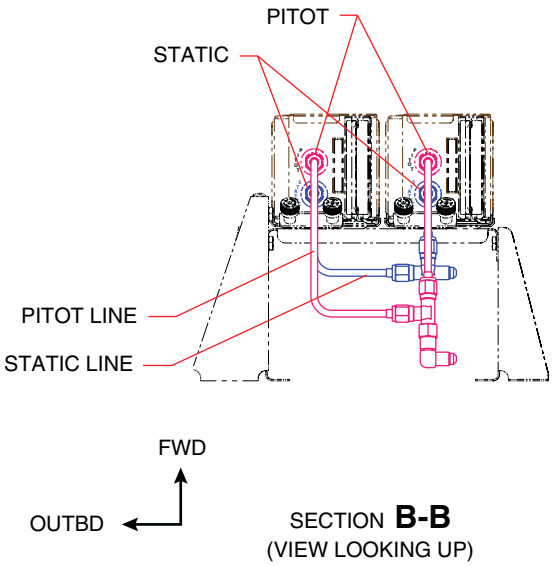


AVIDYNE EFIS OPTION

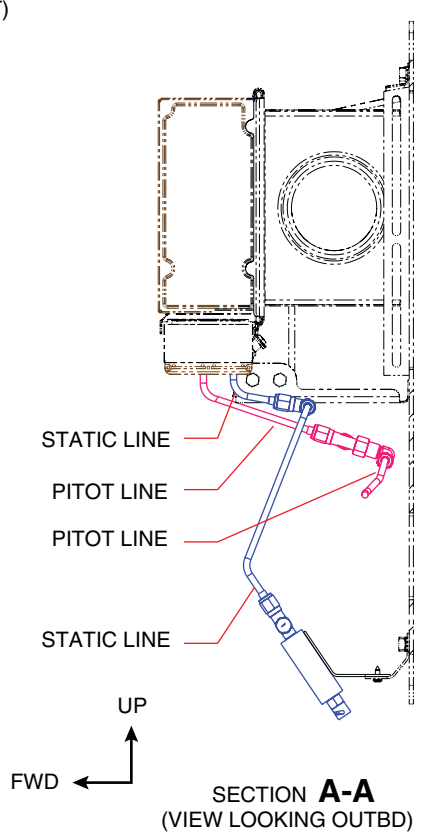
Pitot-Static System
Figure 1 (Sheet 2 of 5)



PITOT STATIC SYSTEM - G600
(VIEW LOOKING AFT)



SECTION **B-B**
(VIEW LOOKING UP)

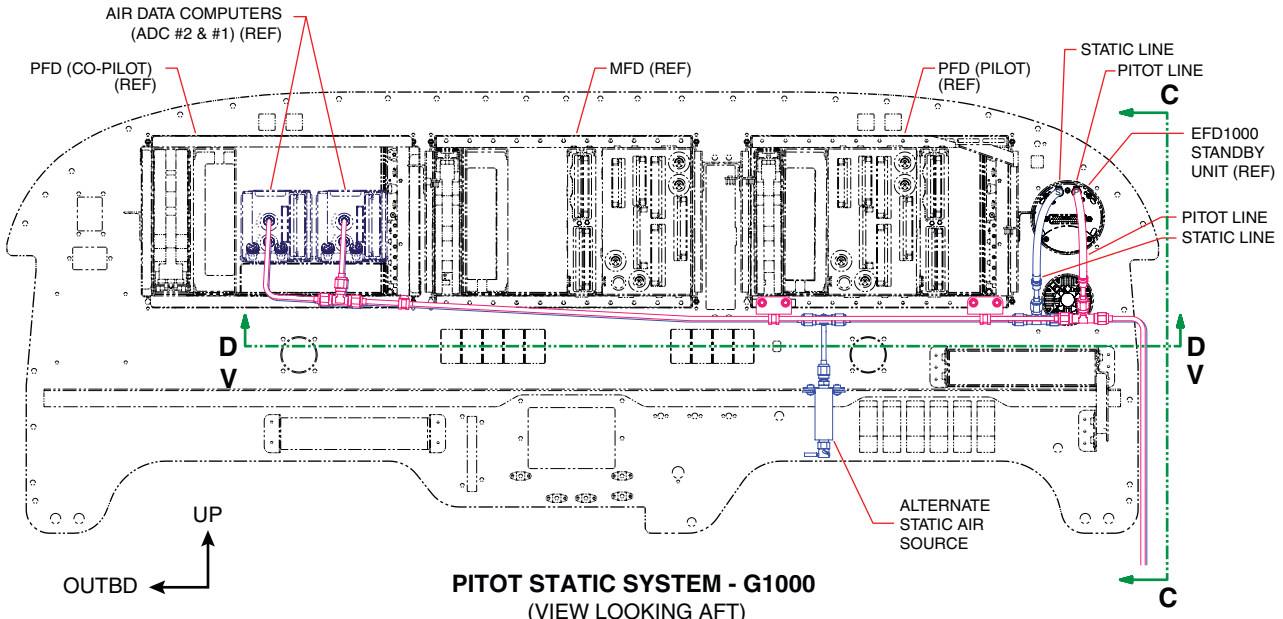


SECTION **A-A**
(VIEW LOOKING OUTBD)

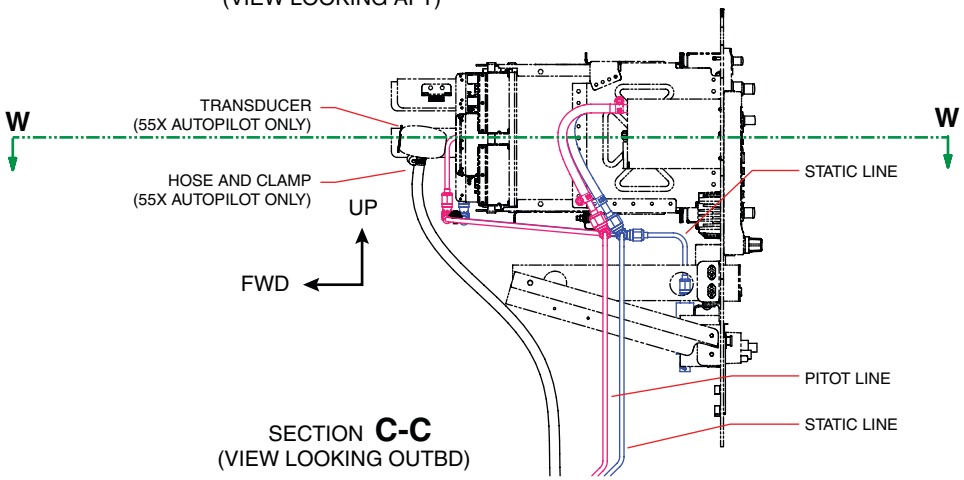
Effectivity
Seneca V
with Garmin G600

Pitot-Static System
Figure 1 (Sheet 3 of 5)

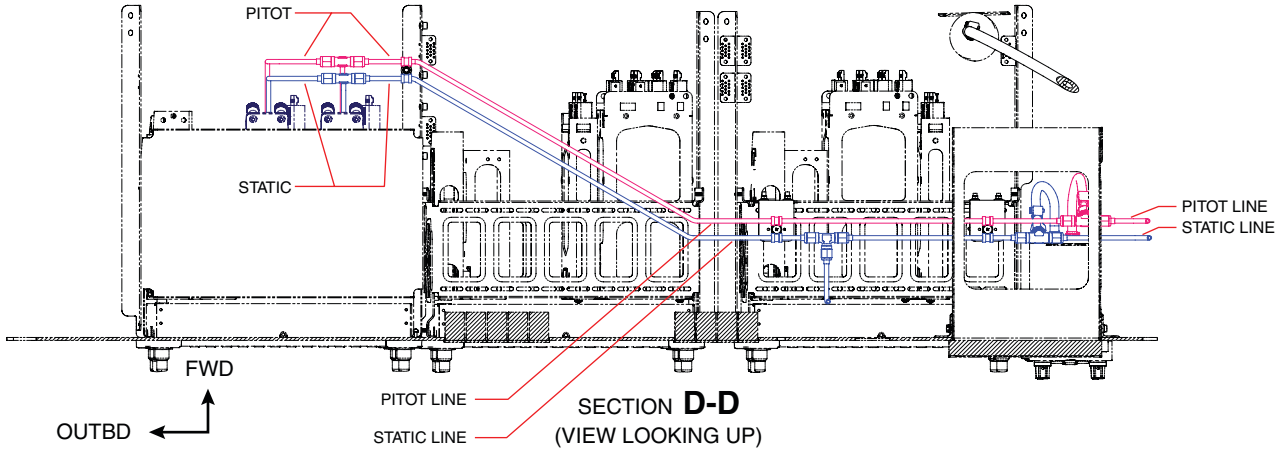
107260 D



PITOT STATIC SYSTEM - G1000
(VIEW LOOKING AFT)



SECTION **C-C**
(VIEW LOOKING OUTBD)

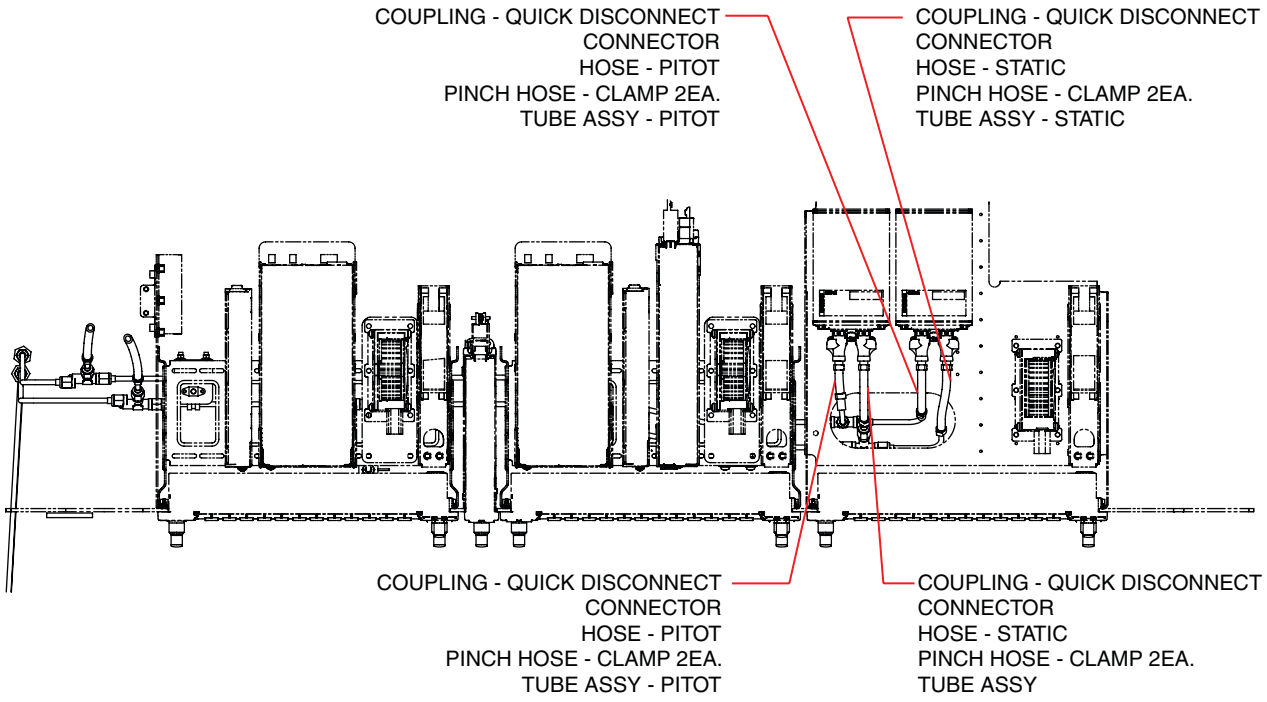


SECTION **D-D**
(VIEW LOOKING UP)

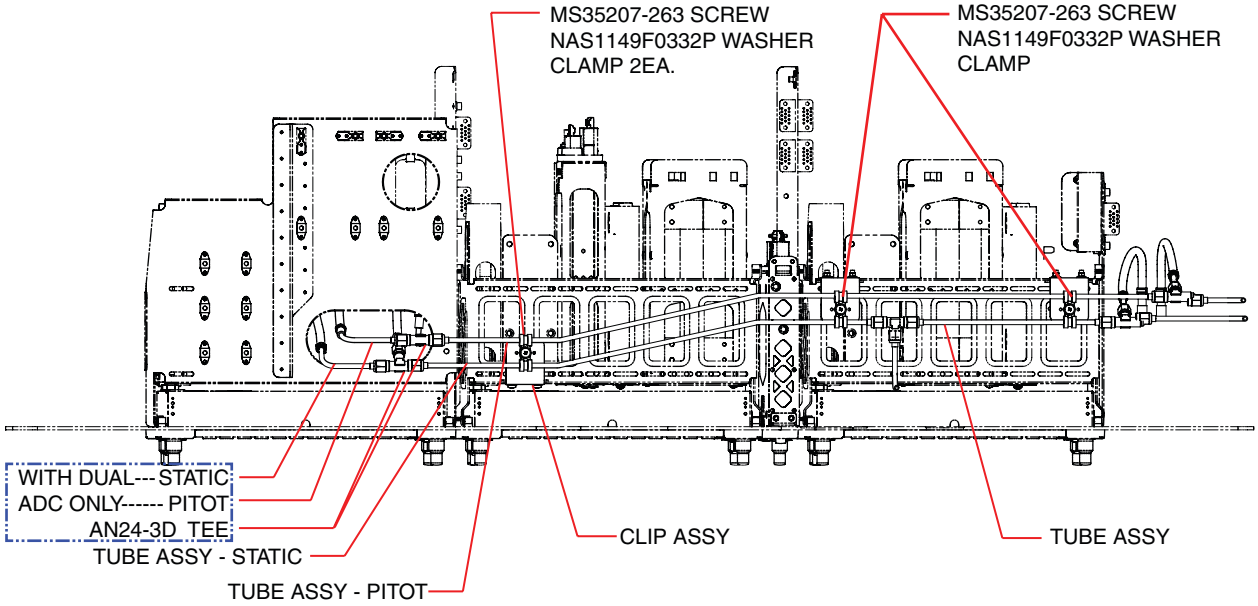
Pitot-Static System
Figure 1 (Sheet 4 of 5)

Effectivity
Seneca V
with Garmin G1000

107260 E 3.0



SECTION **W-W**
(VIEW LOOKING DOWN
ROTATED 90° CCW)



SECTION **V-V**
(VIEW LOOKING UP)

Effectivity
Seneca V
with Garmin G1000 NXi

Pitot-Static System
Figure 1 (Sheet 5 of 5)

CHART 1
TROUBLESHOOTING PITOT AND STATIC SYSTEM

Trouble	Cause	Remedy
Heating element inoperative.	Defective switch.	Replace the switch.
	Grounded or open circuit.	Check for continuity and repair.
	Defective heating element in pitot head.	Replace the pitot-static mast.
Circuit breaker keeps tripping.	Grounded wire.	Check for continuity and repair.
Instruments inoperative or erratic in operation.	Lines clogged.	Disconnect lines at instruments and blow out with low pressure air.
	Line leaks.	Check lines for loose connections at all connection points.

C. Test

(PIR-PPS60035, Rev. AD / FTP2001-12, Rev. F.)

CAUTION: FOR AIRPLANES EQUIPPED WITH GARMIN G1000 / G1000 NXi, ENTER "MAINTENANCE MODE" WHEN ACTIVATING THE GARMIN G1000 / G1000 NXi SYSTEM ON THE GROUND TO CONDUCT TESTS, INSPECTIONS, ETC. SEE MAINTENANCE MODE UNDER INTEGRATED AVIONICS SYSTEM - GARMIN G1000 IN 34-25-01 OR G1000 NXi IN 34-25-02.

This test requires a pitot/static test fixture (i.e. - Aerosonic Air Data Test Set - Model 90000-0168 or equivalent) and calibrated air source (i.e. - airspeed simulator) and should be performed at any time an instrument, fitting, line, pitot head, or static button is disconnected. The test should be performed prior to the next flight.

(1) Setup

(a) For airplanes equipped with Garmin G600, see "Airspeed Test" on page 342361 and "Altitude Test" on page 342362 before beginning this test.

(b) For airplanes equipped with Garmin G1000 / G1000 NXi IAS, see "GDC 74A Tests" on page 342501132 (G1000) or "GDC 72 Tests" on page 342502132 (G1000 NXi) before beginning this test.

(c) Ensure the lines and fittings are free of any entrapped moisture or restrictions.

(d) In airplanes with glass cockpits, supply external power to the airplane.

(e) Set the airplane altimeter to 29.92 In·Hg.

(2) Procedure

(a) Attach the test fixture to the pitot head. Align the holes in the fixture with the holes in the head.

(b) Attach the airspeed simulator hose to the pitot (pressure) port of the fixture.

(c) When equipped with the optional Meggitt EFIS, Avidyne Entegra, or Garmin G600; turn ON the PFD (or EADI).

(d) When equipped with Garmin G1000 / G1000 NXi, turn ON both PFD('s).

(e) Operate the simulator to obtain a reading of 75 knots on the airplane airspeed indicator(s).

(f) Check that the airspeed indicator needles (or PFD display(s)) follow in the same direction as the simulator airspeed indicator needle.

(g) Raise airspeed to 204 knots and wait 15 seconds to allow the airplane airspeed indicators to stabilize.

(h) Observe the simulator and airplane airspeed indicators for 15 seconds. If a leak is present, the indicator needles (PFD airspeed on G1000 / G1000 NXi) will move toward zero. Maximum acceptable leak rate is one (1) knot in 15 seconds.

(i) If a leak rate greater than one (1) knot in 15 seconds is present, check the fixture installation, hose connections, and pitot system lines and fittings. Repair the leak when found, then repeat steps (a) thru (g).

(j) When equipped with the optional Meggitt EFIS, Avidyne Entegra, or Garmin G1000 / G1000 NXi:

1) Operate the simulator on the pilot's PFD to indicate 140 knots on the simulator.

2) Verify that the pilot's PFD shows within three (3) knots of the simulator indication.

3) With Garmin G1000 / G1000 NXi only:

a) Turn OFF the pilot's PFD and turn ON the copilot's PFD, .

b) Repeat steps 1 and 2 for the copilot's PFD.

c) Turn OFF the copilot's PFD.

d) Turn ON the pilot's PFD.

4) Operate the simulator until the overspeed alarm (PFD visual and/or aural) first activates. Verify that the simulator indicates an airspeed of 204 to 207 knots.

5) Increase the simulator airspeed indication to 217 knots.

6) Decrease the simulator airspeed indication until the overspeed alarm (PFD visual and/or aural) first stops. Verify that the simulator indicates an airspeed of less than 204 knots.

7) With Garmin G1000 / G1000 NXi only:

a) Pull the PFD 1 circuit breaker.

b) Repeat steps 4, 5, and 6, for the copilot's PFD.

c) Reset all PFD circuit breakers.

(k) Remove the test fixture from the pitot head.

(l) Attach static test fixture to the static port and pitot test port of the aircraft. Tape over the other static button.

(m) Set the aircraft altimeter needles to read zero altitude. Operate the static simulator to cause the aircraft altimeter needles (PFD airspeed on G1000 / G1000 NXi) to read 1,000 feet altitude. Momentarily open the alternate static port. There should be a decrease in altimeter indication. If no change occurs the system is blocked and must be repaired prior to further testing.

(n) Increase altitude to 1,050 feet.

(o) Check that the aircraft altimeter shows an increase.

(p) Observe the aircraft altimeter. Loss of indicated altitude shall not exceed 100 feet in one minute.

(q) If a leak exceeds that tolerance, check the fixture installation, plumbing and fittings. Repair the leak when found and repeat the static system checks above.

(r) Remove the test fixture and tape from the static button.

(s) In airplanes with glass cockpits, remove external power from aircraft.

2. Pitot Head

A. Removal

The pitot head is located on the lower side of the left wing.

(1) Remove the four screws which secure the pitot tube to the mast assembly.

(2) Carefully pull the pitot tube from the mast assembly.

(3) Remove the hose from the elbow on top of the pitot tube.

(4) Disconnect the electrical leads at the connector.

B. Installation

(1) Reconnect the electrical leads to the heating elements.

(2) Install the pitot tube into the mast assembly.

(3) Secure the pitot tube to the mast assembly with the four screws.

(4) Reconnect the electrical leads to the gauge.

3. Vertical Speed Indicator

NOTE: In airplanes equipped with Avidyne Entegra IFDS or the Garmin Flight Display, this function is performed soley by the Primary Flight Display (PFD).

The vertical speed indicator measures the rate of changes in static pressure when the airplane is climbing or descending. By means of a pointer and dial, this instrument will indicate the rate of ascent or descent of the airplane in feet per minute. But due to the lag of the instrument, the aircraft will be climbing or descending before the instrument starts to read and the instrument will continue to read after the aircraft has assumed level flight. In rough air, this should not be considered a malfunction.

NOTE: If any connection in the pitot / static system is opened for maintenance, the entire system must be rechecked per Pitot / Static System “Test” on page 34108. With the optional Meggitt Single-side EFIS installed, see also Appendix 1.

A. Troubleshooting

See “Chart 2” on page 341011.

B. Removal and Installation

See 39-10-00, Face Mounted Instruments - Removal and Installation.

CHART 2
TROUBLESHOOTING VERTICAL SPEED INDICATOR

Trouble	Cause	Remedy
Pointer does not set on zero.	Aging of diaphragm.	Reset pointer to zero by means of setting screw. Tap instrument while resetting.
Pointer fails to respond.	Obstruction in static line.	Disconnect all instruments connected to the static line. Clear line.
	Water in static line.	Check individual instruments for obstruction in lines.
Pointer oscillates.	Leaks in static lines.	Disconnect all instruments connected to the static line. Check individual instruments for leaks. Reconnect instruments static line and test installation for leaks.
Vertical Speed indicates when aircraft is banked.	Water in static line.	Disconnect static lines blow out lines from cockpit out toward static vents.
Pointer indicates a descent when airplane is “slipped”	Static vent on side opposite slip is blocked.	Disconnect static line at aft “T” connection (“Figure 1” on page 34102, #5) and blow out line.
Pointer has to be set before every flight.	Instrument malfunction.	Replace instrument.
Pointer cannot be reset to zero.	Diaphragm distorted.	Replace instrument.
Instrument reads very low during climb or descent.	Case of instrument broken or leaking.	Replace instrument.

4. Sensitive Altimeter

NOTE: In airplanes equipped with Avidyne Entegra IFDS or the Garmin Flight Display, the primary altimeter is the Primary Flight Display (PFD).

The altimeter indicates pressure in feet above sea level. The indicator has three pointers and a dial scale. The longer pointer is read in hundreds of feet, the middle pointer in thousandths of feet and the short pointer in ten thousandths of feet. A barometric pressure window indicating inches of mercury is located on the right side of the indicator dial and is set by the knob located on the lower left corner of the instrument. On the left side of the indicator is a window which indicates pressure in millibars. The altimeter consists of a sealed diaphragm that is connected to the pointers through a mechanical linkage. The diaphragm mounting is made of bi-metallic temperature sensitive strips that compensate for variations from standard temperatures. The instrument case is vented to the static air system and as static air pressure decreases, the diaphragm expands, causing the pointers to move through the mechanical linkage. Altitude encoding altimeters are standard.

NOTE: If any connection in the pitot / static system is opened for maintenance, the entire system must be rechecked per Pitot / Static System “Test” on page 34108. With the optional Meggitt Single-side EFIS installed, see also Appendix 1.

A. Troubleshooting

See “Chart 3” on page 341013.

B. Removal and Installation

See 39-10-00, Face Mounted Instruments - Removal and Installation.

CHART 3
TROUBLESHOOTING ALTIMETER

Trouble	Cause	Remedy
Excessive scale error.	Improper calibration adjustment.	Replace instrument.
Excessive pointer oscillation.	Defective mechanism.	Replace instrument.
High or low reading.	Improper venting.	Eliminate leak in static pressure system and check alignment of airspeed tube.
Setting knob is hard to turn.	Wrong lubrication or lack of lubrication.	Replace instrument.
Inner reference marker fails to move when setting knob is rotated.	Out of engagement.	Replace instrument.
Setting knob set screw loose or missing.	Not tight when altimeter was reset.	Tighten instrument screw, if loose. Replace instrument, if screw is missing.
Cracked or loose cover glass.	Case gasket hardened.	Replace instrument.
Dull or discolored markings.	Age.	Replace instrument.
Barometric scale and reference markers out of synchronism.	Slippage or mating parts.	Replace instrument.
Altimeter sticks at altitude or does not change with change of altitude.	Water or restriction in static line.	Remove static lines from all instruments, blow line clear from cockpit to static vents.
Altimeter changes reading as aircraft is banked.	Water in static line.	Remove static lines from all instruments and blow line clear from cockpit cockpit to static vents.

5. Radar Altimeter

The King KRA-10A radar altimeter system may be installed as optional equipment. The system makes use of an indicator, on/off switch, transmitter, and antenna, and provides above ground level (AGL) altitude information from 20 feet up to a maximum of 2500 feet.

NOTE: On some installations, if the air conditioning and radar altimeter are both functioning at the same time, the actual AGL should be relied on for accuracy only up to 1500 feet.

The King KRA-10A system utilizes a KA-131 antenna, KRA-10 receiver/transmitter, and KI-250 indicator. For further information, contact Allied Signal Inc., Commercial Avionics Systems, Olathe, KS.

The receiver, as well as the converter (when required) for the King system, is mounted on a panel in the tail of the aircraft near Fuselage Station 222.437 and the antenna to the rear of that near Fuselage Station 229.0.

CAUTION: DO NOT ALTER THE CABLE OR CONNECTOR PROVIDED WITH THE ANTENNA. SYSTEM PERFORMANCE IS DEPENDENT ON PROPER CABLE LENGTH FROM THE ANTENNA TO THE RECEIVER-TRANSMITTER UNIT.

The antennas are skewed internally to permit mounting on an inclined surface. This requires that each be mounted in a specific manner. During removal, take note of the relationship of the cable attachment to the tail of the aircraft and install the new antenna in the same manner.

NOTE: The KI-250 indicator is a face-mounted instrument. See 39-10-00 for removal/installation.

A. King KA-131 Antenna

Refer to “Figure 2” on page 341015.

(1) Removal

(a) Support the antenna and remove the eight screws securing it to the belly of the aircraft.

(b) Carefully let the antenna drop from its opening in the belly. Take note of the cable attachment plug position relative to the tail of the aircraft.

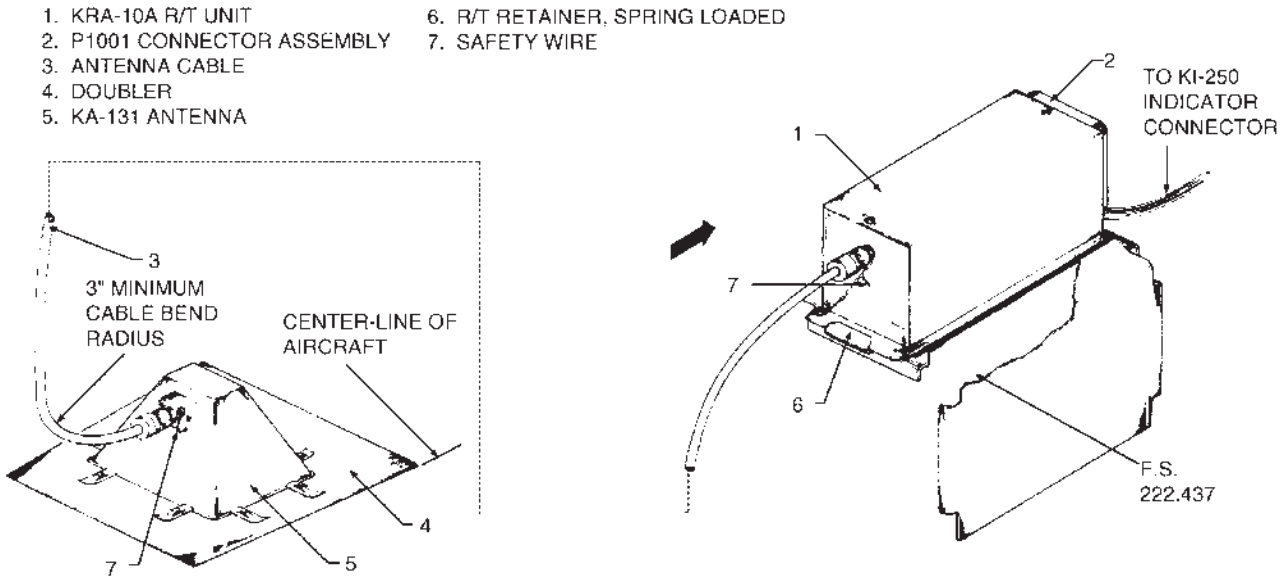
(c) Remove the safety wire and plug.

(2) Installation

(a) Install gasket in place on antenna ensuring metal portions of gasket will contact airframe for good bonding.

(b) Install antenna cable finger tight and secure with safety wire.

(c) With the cable attachment facing aft, position the antenna on the fuselage and install the eight screws.



King Radar Altimeter Installation
Figure 2

B. King KRA-10A Receiver/Transmitter (R/T) Unit

Refer to “Figure 2”.

CAUTION: BEFORE ENTERING THE AFT FUSELAGE SECTION, MAKE CERTAIN THE AIRCRAFT IS SUPPORTED AT THE TAIL SKID.

(1) Removal

(a) Remove the close-out panel at the rear of the baggage compartment.

(b) Remove the safety wire and unscrew the antenna cable from the KRA-10A R/T unit.

(c) Remove the screw securing the P1001 connector assembly to the KRA-10A R/T and remove the connector assembly from the R/T unit.

(d) Loosen the screw on the spring loaded R/T retainer and lift the R/T unit from the mounting rack.

(2) Installation

(a) Position the R/T unit on the mounting rack and secure by tightening the screw on the spring loaded R/T retainer.

(b) Attach the P1001 connector to the R/T unit and secure with screw.

(c) Install antenna cable finger tight and secure with safety wire.

(d) After checking security of installation, reinstall baggage compartment close-out panel.

6. Airspeed Indicator

NOTE: In airplanes equipped with Avidyne Entegra IFDS or the Garmin Flight Display, the primary airspeed indicator is the Primary Flight Display (PFD).

The airspeed indicator provides a means of indicating the speed of the airplane passing through the air. Airspeed indicators are sensitive pressure gauges which measure the difference between the pitot and static pressures, and present such difference in terms of indicated airspeed. This instrument has the diaphragm vented to the pitot air source and the case is vented to the static air system. As the airplane increases speed, the pitot air pressure increases, causing the diaphragm to expand. A mechanical linkage picks up this motion and moves the instrument pointer to the indicated speed. The instrument dial is calibrated in knots. Colored arcs and radial lines are used to mark the operating speed ranges necessary for safe operation of the airplane.

NOTE: If any connection in the pitot / static system is opened for maintenance, the entire system must be rechecked per Pitot / Static System “Test” on page 34108. With the optional Meggitt Single-side EFIS installed, see also Appendix 1.

A. Troubleshooting

See “Chart 4”.

B. Removal and Installation

See 39-10-00, Face Mounted Instruments - Removal and Installation.

CHART 4
TROUBLESHOOTING AIRSPEED TUBES AND INDICATOR

Trouble	Cause	Remedy
Pointers of stick instruments do not indicate properly.	Leak in instrument case or in pitot lines.	Check for leak and seal.
Pointer of instrument oscillates.	Defective mechanism.	Replace instrument.
Instrument reads high.	Pointer not on zero.	Replace instrument.
	Leaking static system.	Find leak and correct.
Instrument reads low.	Pointer not on zero.	Replace instrument.
	Leaking pitot system.	Find leak and correct.
	Pitot head not aligned correctly.	Realign pitot head.
Airspeed changes as aircraft is banked.	Water in pitot line or static line.	Remove lines from static instruments and blow out lines from cockpit to pitot head.

7. Outside Air Temperature (OAT)

A. Seneca IV

The OAT gauge is internally lit. The only maintenance required is replacing a faulty instrument or a bad light bulb. The gauge must be removed to replace the light bulb.

Removal and Installation

CAUTION: DURING INSTALLATION OF ALL OAT'S, ONLY FINGER TIGHTEN GAUGE AND EXTERNAL THREADED HEX TUBE (SUNSHIELD). OVER TORQUING WILL DAMAGE INSTRUMENT.

(a) Remove pilot's window close out panel.
(b) Disconnect power wire.
(c) Remove external sunshield.
(d) Unscrew instrument from support bushing. Note position of any washers or spacers.
(e) Install new gauge in reverse sequence. Check gauge position before tightening sunshield.
(f) Install pilot's window close out pane

B. Seneca V - Meggitt Single-side EFIS or Avidyne Entegra IFDS Option

S/N's 3449269 and up.

The OAT Probe is located under the rear facing seats below the floor at F.S.128.73.

NOTE: On Seneca V models, the outside air temperature gauge is displayed on the Digital Display Monitoring Panel or, with the Avidyne Entegra installed, on the MFD. See 77-40-00 for information.

Removal and Installation

(a) Remove rear seats.
(b) Remove center console.
(c) Remove floor panel.
(d) Remove probe.
(e) Installation is the reverse of the above steps.

C. Seneca V - Garmin G600 Flight Display System

S/N's 3449410 and up.

A GTP 59 OAT Probe, mounted to the bottom side of each wing, outboard of W.S.164.72, provides air temperature data to the PFD/MFD. The G600 installation also retains the OAT probe located under the rear facing seats below the floor at F.S.128.73 which provides an OAT display on the Digital Display Monitoring Panel.

NOTE: On Seneca V models with Garmin G600 Flight Display System, the outside air temperature is displayed on the PFD to the left of the HSI.

See Flight Display System – Garmin G600, in 34-23-00, for additional information.

D. Seneca V - Garmin G1000 Integrated Avionics System

S/N's 3449459, 3449467-3449508

A GTP 59 OAT Probe, mounted to the bottom side of each wing, outboard of W.S.164.72, provides air temperature data to the GDC 74A air data computers which provide information to the G1000 system.

See Integrated Avionics System – Garmin G1000, in 34-25-01, for additional information.

E. Seneca V - Garmin G1000 NXi Integrated Avionics System

S/N's 3449509 and up.

A GTP 59 OAT Probe, mounted to the bottom side of each wing, outboard of W.S.164.72, provides air temperature data to the GDC 74A air data computers which provide information to the G1000 NXi system.

See Integrated Avionics System – Garmin G1000 NXi, in 34-25-02, for additional information.

ATTITUDE AND DIRECTION

1. Attitude Deviation Indicators (ADI) - (Gyro Horizon)

NOTE: In airplanes equipped with Avidyne Entegra IFDS or the Garmin Flight Display, no air-driven ADI is installed. In those installations, this function is provided solely by the Primary Flight Display (PFD).

The ADI is essentially an air-driven gyroscope rotating in a horizontal plane and is operated by the same principle as the directional gyro. The pilot and copilot's (if installed) ADI's are King KI-256, or S-TEC ST-361 Flight Director Attitude Indicators that combine air-driven gyros with electrical pitch and roll inputs and outputs to the autopilot/flight director The gyroscopes rotate in a horizontal plane and are operated by vacuum provided by engine driven pumps. A bar across the face of the indicator represents the horizon and aligning the miniature airplane to the horizon bar simulates the alignment of the airplane to the actual horizon. Any deviation simulates the deviation of the airplane from the true horizon. The ADI is marked for different degrees of bank.

A. Troubleshooting

See "Chart 1" on page 34202.

B. Removal and Installation

See 39-10-00, Face Mounted Instruments - Removal and Installation.

2. Standby Attitude Indicator

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

NOTE: In airplanes equipped with Avidyne Entegra IFDS or the Garmin Flight Display, the primary Attitude Indicator is the Primary Flight Display (PFD).

With the optional Avidyne Entegra Integrated Flight Display System (IFDS) or the Garmin Flight Display installed; an electric standby attitude indicator is installed to the left of the Primary Flight Display (PFD).

Other than removing and replacing the unit itself (see 39-10-00), the only line replaceable part is the emergency power battery. For Avidyne equipped aircraft, the battery is located on the rear of the instrument's case. For Garmin G600 equipped aircraft, the battery is remotely located on the left, aft side of the forward bulkhead at F.S. 49.50.

Required periodic maintenance is listed in 5-20-00 and 5-30-00. Checkout, test procedures and Instructions for Continued Airworthiness are provided in Mid-Continent Instruments Manual No. 9015762.

CHART 1
TROUBLESHOOTING ATTITUDE DEVIATION INDICATOR

Trouble	Cause	Remedy
Bar fails to respond.	Observe vacuum gauge for insufficient vacuum.	If insufficient vacuum exist, check pump and tubing.
	Filter dirty.	Clean or replace filter.
	Defective instrument.	Replace gyro instrument.
Bar does not settle.	Insufficient vacuum.	Check line and pump. Adjust valve.
	Incorrect instrument.	Check part number.
	Defective instrument.	Replace.
Bar oscillates or shimmies continuously.	Instrument loose in panel.	Tighten mounting screws.
	Vacuum too high.	Adjust vacuum regulators.
	Defective mechanism.	Replace instrument.
Instrument does not indicate level flight.	Instrument not level in panel.	Loosen screws and level instrument.
	Aircraft out of trim.	Trim aircraft.
Bar high after 180° turn.	Normal, if it does not exceed 1/16 inch.	
Instrument tumbles in flight.	Observe vacuum gauge for low vacuum.	If vacuum is low, reset regulator.
	Dirty filter.	Clean or replace filter.
	Line to filter restricted.	Replace line.
	Plug missing or loose in instrument.	Replace or tighten plug.

3. Air-Driven Heading Indicator (Directional Gyro)

NOTE: In airplanes equipped with Avidyne Entegra IFDS or the Garmin Flight Display, no air-driven Heading Indicator is installed. In those installations, this function is provided solely by the Primary Flight Display (PFD).

The gyro-stabilized heading indicator is a flight instrument incorporating an air-driven gyro stabilized in the vertical plane. The gyro is rotated at high speed by lowering the pressure in the airtight case and simultaneously allowing filtered atmospheric air pressure to enter the instrument against the gyro buckets. Due to gyroscopic inertia, the spin axis tends to continue pointing in the same direction, even though the aircraft yaws, turns, pitches or rolls. This relative motion between the gyro and the instrument case is shown on the instrument dial which is similar to a compass card. The gyroscopic heading indicator has no sense of direction and must be set to the magnetic compass. The dial, when set to agree with the airplane's magnetic compass, provides a positive heading indication free from acceleration/deceleration and turning errors. However, precession forces applied to the gyro during turns may cause the gyro to "drift", and, upon completion of the turn, result in a difference in readings between the directional gyro and the magnetic compass, necessitating resetting the gyro. Even while maintaining a given heading, the gyro compass tends to precess (drift) due to internal friction, spin axis error, air turbulence and airflow. Therefore, the gyro should be checked against the magnetic compass, and reset as necessary, at least every 15 minutes. Maximum acceptable precession is 5° in 0:15 minutes.

Some heading indicators are limited to 55° of roll and pitch. Should these limits be exceeded, the gyro will "tumble". This is evidenced by a rapid spinning of the compass card. The gyro in a properly operating instrument can be reerected, after returning to straight and level flight, by caging the gyro and resetting it.

A. Troubleshooting

See "Chart 2".

B. Removal and Installation

See 39-10-00, Face Mounted Instruments - Removal and Installation.

CHART 2
TROUBLESHOOTING HEADING INDICATOR (DIRECTIONAL GYRO)

Trouble	Cause	Remedy
Excessive precession (drift) in either direction.	Setting error.	Review paragraph titled AIR DRIVEN HEADING INDICATOR above.
	Defective instrument.	Replace instrument.
	High or low vacuum. If vacuum is not correct, check for the following:	
	a. Relief valve improperly adjusted.	a. Adjust.
	b. Incorrect gauge reading.	b. Replace gauge.
	c. Pump failure.	c. Repair or replace.
	d. Vacuum line kinked or leaking.	d. Check and repair. Check for collapsed inner wall of hose.
Card spins during turn.	Gimbal limits of (55° bank or pitch) exceeded.	Recage gyro in level flight.
Card spins continuously.	Defective mechanism.	Replace.

4. Horizontal Situation Indicator (HSI)

The King Horizontal Situation Indicator (HSI) is installed as standard equipment on the pilot's instrument panel.

The HSI receives heading information from a remote electrically operated gyro installed in the tail section (Seneca IV) or nose section (Seneca V) of the airplane. When operating normally, the gyro is slaved to the Earth's magnetic field by the Magnetic Slaving Transmitter installed in the left wing tip. Thus, the remote gyro is continuously precessed to align itself with the Earth's magnetic field which, in effect, enables the HSI to provide the pilot with gyro stabilized magnetic heading information.

In the event of erroneous input from the Magnetic Slaving Transmitter, the gyro can be deslaved from the Earth's magnetic field by a switch located on the pilot's instrument panel. Once deslaved, the gyro is subject to precession. A second switch on the pilot's instrument panel permits resetting the gyro by driving it electrically. Maintenance of the HSI, and its related components, should be referred to an authorized avionics repair shop.

5. Magnetic Heading Systems

A. Flux Detector

See "Figure 1" on page 34205.

A flux detector installed in the wing tip of the left wing is used to provide heading data to the HSI/EHSI and, if installed, the ADAHRS.

CAUTION: PERFORM "MAGNETIC HEADING COMPENSATION / CALIBRATION" ON PAGE 34206 WHENEVER THE FLUX DETECTOR IS CHANGED.

(1) Removal

(a) Remove the left wing tip fairing to expose the flux detector.

(b) Disconnect the wiring harness(es) from the top of the flux detector to be removed.

(c) Unscrew and remove the three brass screws and washers and remove the flux detector.

(2) Installation

CAUTION: THE FLUX DETECTOR IS SECURED TO THE MOUNTING BRACKET WITH BRASS SCREWS. ENSURE ONLY BRASS SCREWS ARE USED WHEN REINSTALLING.

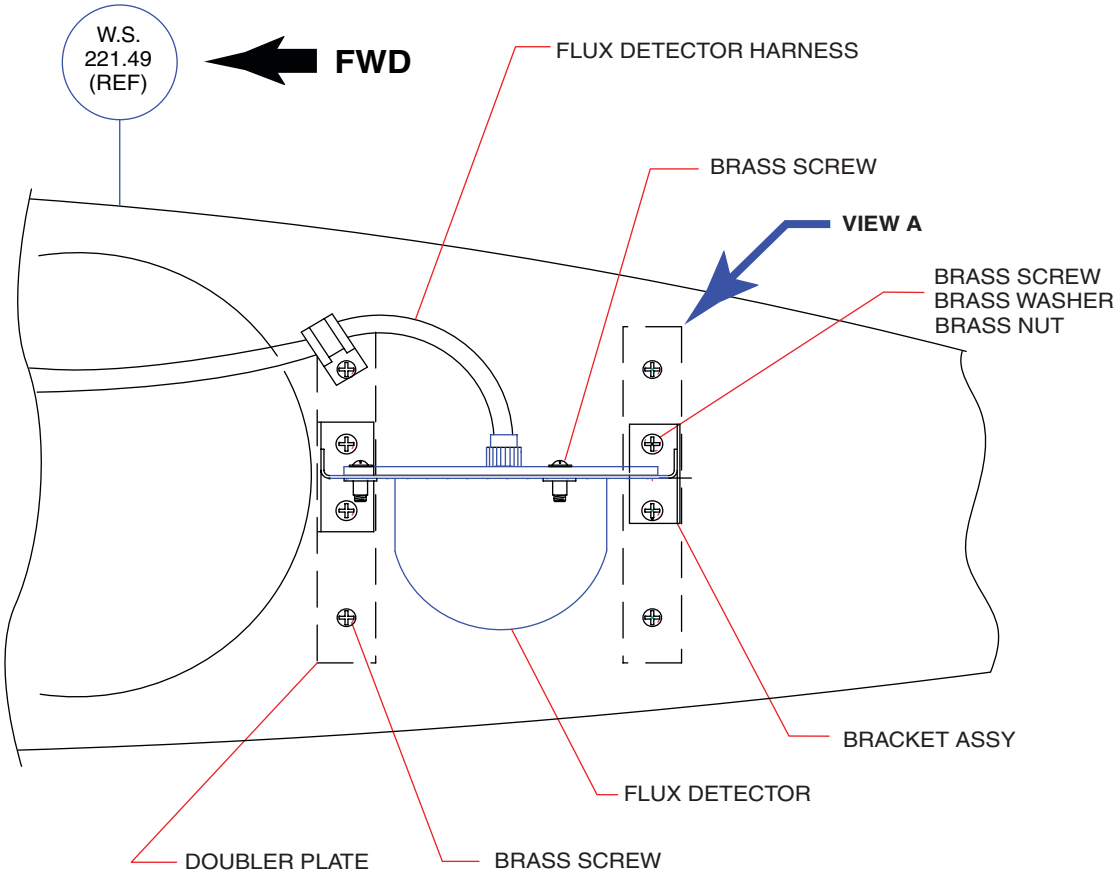
(a) Place the flux detector into position on its mounting bracket and secure with brass screws and washers (3 ea.)

(b) Connect wiring harness to the connector on top of the flux detector.

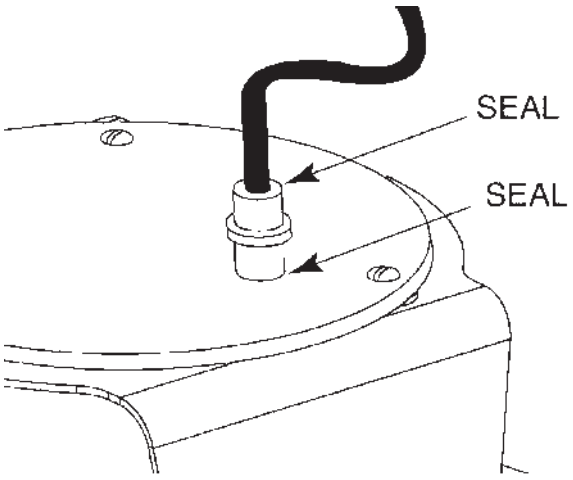
(c) Seal the connector backshell and base as shown in "Figure 1" on page 34205 with DOW Corning No. 4 Electrical Insulating Compound.

(d) Reinstall the Left wing tip.

NOTE: Ensure correct hardware is used when reinstalling wing tip over flux detector.



LOOKING INBOARD



VIEW A

Flux Detector Installation
Figure 1

B. Magnetic Heading Compensation / Calibration

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION SUPPLEMENTARY PUBLICATIONS.)

(1) Standard Installation (S/N's 3449001 thru 3449151 only)

Slaving compasses installed in these airplanes are part of the installed King/Allied Signal (now Honeywell) Autopilot System.

Honeywell (King/Allied Signal) technical support, parts support, and service publications can be obtained as described under Supplementary Publications, Vendor Publications, in the Introduction.

(2) Standard Installation S-TEC ST-180 (S/N's 3449152 thru 3449323 only.) (PIR-PPS60191, Rev. New.)

Accuracy of the entire heading system is dependent on the location of the flux detector and proper calibration. Accuracies of plus or minus one degree are possible when care is taken during installation and calibration. To obtain such results:

The flux detector must be positioned so that it points in the direction of aircraft flight; and, the north-south and east-west correctors must be adjusted to compensate for extraneous magnetic fields near the location of the flux detector.

(a) Required Equipment

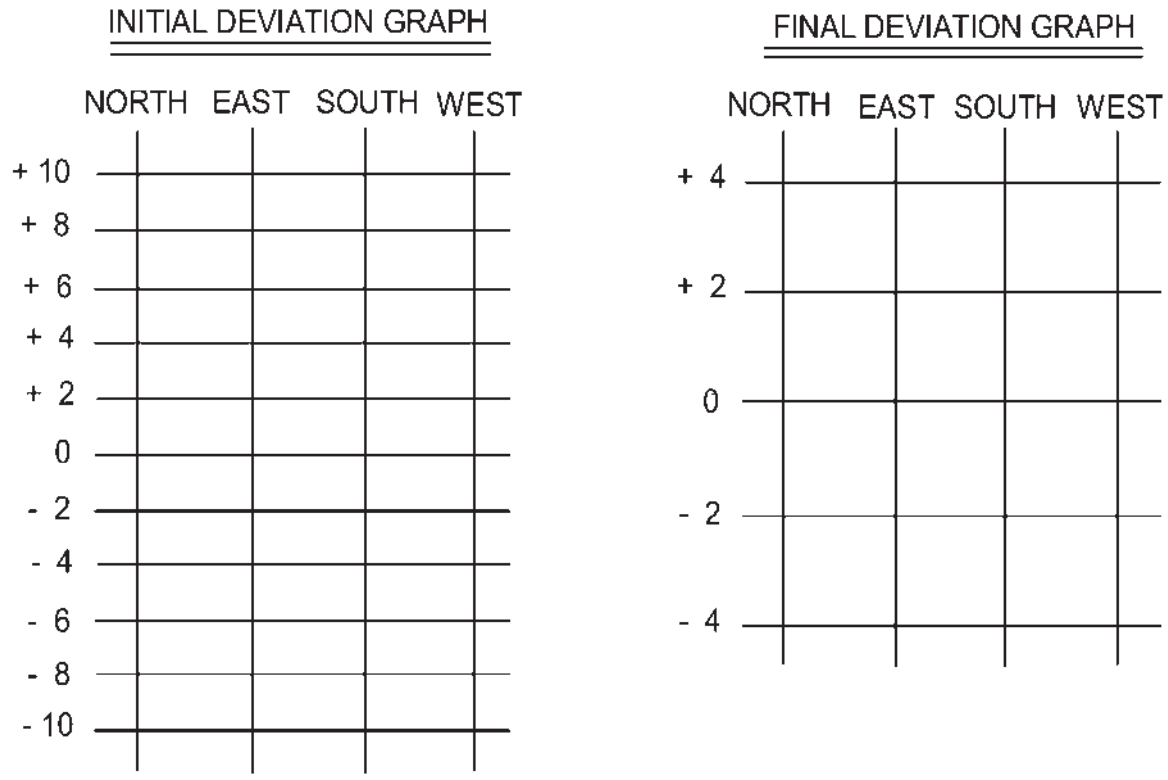
Instrument Flight Research Corp. RF signal generator NAV-40IL or equivalent. Characteristics of the signal generator include: Frequency ranges 108 to 118 MHz, 117 to 136 Mhz, 328 to 336 MHz; +1/-0.01% accuracy, output level continuously adjustable from 1 .0uV to 0.1V into a 50 ohm load; 50 ohm output impedance; and internal adjustable or stepped VOR, LOC, and GS modulation. The RF signal generator should be portable and convenient for use while sitting in the aircraft cockpit.

(b) Procedure

1) Apply power to the Model ST-180 HSI System. Allow at least three (3) minutes for the gyro to erect and synchronize.
2) Prior to actual alignment of the flux sensor, turn the aircraft to both north and east headings. Apply power to electrical equipment such as navigation and beacon lights and verify that the compass system is not affected.
3) Align the aircraft to an approximate magnetic north heading. On "Chart 3" on page 34207, record the actual magnetic heading and the HSI heading card reading.
4) Determine and record on "Chart 3" on page 34207 the deviation between the actual magnetic heading and the heading card heading. If the heading card reads high, the deviation is a plus.
5) Repeat steps (3) and (4) for east, south, and west headings. Record actual magnetic headings, heading card readings, and deviations on "Chart 3" on page 34207.
6) Plot deviations on the initial deviation graph in "Figure 2" on page 34207.
7) Realign the aircraft to north. Adjust the north-south corrector on the Slaving Panel, for one half of the difference between the north and south deviations. Record the new deviation for north and south on the initial deviation graph in "Figure 2" on page 34207.
8) Realign the aircraft to east. Adjust the east-west corrector on the Slaving Panel, for one half the difference between the east and west deviations. Record the new deviation for east and west on the initial deviation graph in "Figure 2" on page 34207.
9) If the pattern is not centered around zero, rotate the flux sensor clockwise to correct for minus deviations or counterclockwise for plus deviations. Plot final deviations on the final deviation graph in "Figure 2" on page 34207.

CHART 3
DEVIATION CHART

Approx. Magnetic Heading	Actual Magnetic Heading	Heading Card Reading	Deviation
North			
East			
South			
West			



ST-180 Deviation Graphs
Figure 2

10) The deviations should now center around the zero reference line of the graph. If the error exceeds the specified system error limits (± 3°), repeat the complete procedure.

(3) Standard Installation King KCS- 55A (S/N's 3449324 & up.) (PIR-PPS60081, Rev. C)

Before attempting to compensate compass place the aircraft in simulated flight conditions. Check to see that the doors are closed, flaps in retracted position, engine running, throttle set at 1000 rpm or low idle, and aircraft in level flight attitude. Aircraft master switch, alternator switch and all radio switches should be in the ON position. All other cockpit controlled electrical switches should be in the OFF position. Use a brass or other non-magnetic screwdriver to make adjustments to compensator screws.

(a) Flux Detector index error

1) Place the aircraft on a compass rose in each of the four cardinal headings.

2) At each cardinal heading, with the slaving control in the "Free Gyro" position, center the slave meter with the manual slave buttons on the slaving control, and record the (plus or minus) difference between the cardinal heading and the indicator heading.

3) Add the four differences found, then divide the result by four, this is the index error.

4) Loosen the mounting screws on the flux detector, and rotate the unit as required,to cancel the index error. (If the index error is +2°, set the indexing device at -2°).

5) Retighten the mounting screws.

(b) Slaving control compensator check and adjustment.

Check and adjust the slaving set upplace the aircraft in simulated flight conditions Engine running, Battery master on, Alternators on, All radios on, recognition lights on, Instrument panel lights on, Navigation, position & strobe lights on, Power the KCS-55A on (allow to synchronize), and Place the slaving control in the "Free Gyro" position.

1) Place the aircraft on the North compass rose heading. Rotate the indicator card to North using the manual slave buttons on the slaving control. With a non-magnetic screwdriver in the N/S opening on the slaving control, center the slave meter.

2) Place the aircraft on the East compass rose heading. Rotate the indicator card to East using the manual slave buttons on the slaving meter. With a non-magnetic screwdriver in the E/W opening on the slaving control, center the slave meter.

3) Place the aircraft on the South compass rose heading. Rotate the indicator card to South using the manual slave buttons on the slaving control. With a non-magnetic screwdriver in the E/W opening on the slaving control, adjust to remove one-half of the existing deviation shown by the slave meter.

4) Place the aircraft on the West compass rose heading. Rotate the indicator card to West using the manual slave buttons on the slaving control. With a non-magnetic screwdriver in the E/W opening on the slaving control, adjust to remove one-half of the existing deviation shown by the slave meter.

5) If deviations (difference between actual magnetic heading and what compass indicates on that particular heading) exceeds ± 10° on any heading:

a) Check to be sure no magnetic metals are near compass (tools, flashlights, pocket knives, wristwatches etc.)

b) Check to be sure screwdriver being used to make adjustments is either fiber or non-magnetic metal, such as brass.

6) When satisfied that errors in excess of 10° is fault of the instrument, replace instrument. After installing new instrument, repeat setup

7) After all adjustments are completed, the indicator shall read within ± 2° of any compass rose heading on which the aircraft is placed.

(4) Optional Meggitt Single-side EFIS Installation (PIR-PPS 60195 Rev. New.)

The Air Data & Attitude Heading Reference System (ADAHRS) unit mounted on the aft equipment shelf provides the other components of the Meggitt Magic EFIS system with magnetic heading data derived from the flux detector installed in the wing tip of the left wing. The ADAHRS unit should be swung (checked for changes in deviation) whenever the ADAHRS unit, flux detector, or either cabin recirculation blower is changed, and at least once a year.

Use the following procedure to compensate and calibrate the magnetic heading indication portion of the Meggitt Magic EFIS installation:

(a) Setup

1) Level the airplane within ± 1 degree laterally and longitudinally at the top surface of the seat rails.

2) Configure the airplane as follows:

Generator/ Alternator - ON	Avionics - ON
Instrument Panel Lights - Full Bright	LCD Displays - Full Bright
Day/Night Switch - Day	Navigation Lights - ON
Strobe Lights - ON	Pitot Heat - OFF
Windshield Heat - OFF	Air conditioning - OFF
Cabin Blower Fan - Low	Vent Blower - OFF

(b) ADAHRS Unit Alignment Procedure

1) Place the EHSI in the “TEST MODE” by applying power with the “MENU” button held in the depressed position until the initialization page is displayed.

2) Select ADAHRS test page 2 from the EHSI menu selections.

3) Allow the ADAHRS to complete the initialization period and display an attitude sphere on the EADI.

4) Verify that the pitch and roll errors, as displayed on the EHSI test page, are less than ± 0.5 degrees in either axis.

5) If required for ADAHRS leveling, install shims under the ADAHRS mounting feet and re-secure the unit to the mounting structure.

6) Repeat the alignment procedure until the minimum acceptable criteria is met.

(c) Compass System Calibration Procedure

1) Flux Detector Indexing

CAUTION: THE TEST CABLE PIPER P/N 101453-002 IS REQUIRED EQUIPMENT. NO OTHER TEST CABLE IS AUTHORIZED. SEE FIGURE 36 TO FABRICATE.

a) Install the ADAHRS test cable and verify that the ADAHRS test cable calibration switch is in the “NORMAL” (open) position.

b) Place the EHSI in the “TEST MODE” by applying power with the “MENU” button held in the depressed position until the initialization page is displayed.

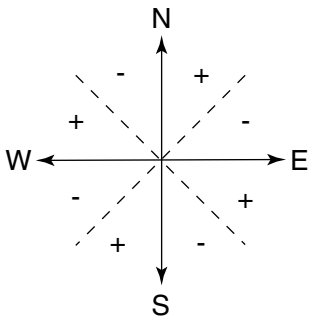
c) Select the ADAHRS interface test page 2 from the Navigation Display (ND) menu selection. The ADAHRS must be in the “INITIALIZATION” mode for the indexing procedure.

d) Place the ADAHRS test cable calibration switch in the “CAL” (closed) position. Ensure that the heading system flag comes into view on the EADI, replacing the heading scale tape and the digital readout.

NOTE: If no calibration has ever been entered into the ADAHRS memory, a heading flag will be in view at the time of the unit initial power-up.

e) Position the aircraft on the NORTH heading of the compass rose and record the indicated “MAGHDG” as displayed on the ND. Record the heading indication in “Chart 4” on page 342011. (This is the raw data output from the flux detector). Repeat this step for EAST, SOUTH, and WEST headings.

f) After recording the raw heading information, calculate and record the index heading errors in “Chart 4” on page 342011. Refer to “Figure 3” to calculate the errors. The following logic applies to calculating heading errors:



For N, if indication is +, then error is +
For N, if indication is -, then error is -
For E, if indication is more than 90, then error is +
For E, if indication is less than 90, then error is -
For S, if indication is - and less than 180, (i.e. -179) then error is +
For S, if indication is + and less than 180, (i.e. +179) then error is -
For W, if indication is - and less than 90, (i.e. -89) then error is +
For W, if indication is - and greater than 90, (i.e. -91) then error is -

Calculating Heading Error
Figure 3

Enter the errors for the four cardinal headings in “Chart 4”. Add the errors and enter the ADAHRS Calibration Harness sum in the “TOTAL ERROR” block in “Chart 4” on page 342011. Divide the Total Error by 4 and enter the result in the “INDEX ERROR” block in “Chart 4” on page 342011. This result is the “Index Error” of the flux detector.

g) If the index error is greater than one (1) degree, loosen the flux detector and adjust out the index error by rotating the flux detector.

h) Repeat steps (e) thru (g) until the index error is less than one (1) degree.

NOTE: Upon completion of the flux detector indexing procedure, apply torque stripe to each of the flux detector mounting screws.

i) This completes flux detector indexing, proceed to Heading System Calibration / Compensation, below.

2) Heading System Calibration / Compensation

a) Heading system calibration / compensation should be performed upon completion of flux detector indexing, above, and using the same aircraft configuration specified in setup, above.

b) Place the Navigation Display (ND) in the “NORMAL” or mission operational mode (normal power-up).

c) With the ADAHRS test cable installed, and the calibration switch in the "CALIBRATE" (closed) position, slowly taxi the aircraft in two (2) complete 360-degree turns. The direction of the turn does not matter; however, a minimum time of one (1) minute for each 360-degree turn (two (2) minutes minimum for the full 720 degrees) is required for the calibration process. The slow turn rate permits more data point samples for constructing the compensation curve. Make sure that the turns are continued until the heading flag is replaced by a normal heading indication.

d) Upon completing the two turns, the magnetic heading will automatically be displayed on both the EADI and EHSI. Place the ADAHRS test cable calibration switch in the "NORMAL" (open) position. Allow the compass system to stabilize for one (1) minute.

e) With the ADAHRS test cable calibration switch in the NORMAL (open) position, perform a complete heading system "swing". A settling time of two (2) minutes must be used at each heading to ensure heading stabilization. After two (2) minutes of settling, record the indicated heading against the known compass rose heading for all twelve compass rose points in "Chart 5". The heading system error should not exceed ± 3 degrees on any compass point.

CHART 4
INDEXING DATA

Aircraft S/N:	**Registration No.:**	
ND Error (±)	**ND Heading**	**Compass Rose**
		N
		E
		S
		W
	Total Error (N+E+S+W)	
	Index Error (Total Error ÷ 4)	

CHART 5
HEADING SYSTEM CALIBRATION

ND Heading	**Compass Rose**
	N
	30
	60
	E
	120
	150
	S
	210
	240
	W
	300
	330

(5) Optional Avidyne Entegra Installation

See “Primary Flight Display (PFD)” on page 34221 under Maintenance for the Avidyne Entegra Integrated Flight Display System.

(6) Optional Garmin G600 Flight Display System Installation

See “GRS/GMU Calibration Procedures” on page 342352 under LRU Test Procedures for Garmin G600 Flight Display System.

(7) Optional Garmin G1000 / G1000 NXi Integrated Avionics System Installation

See “GRS/GMU Calibration Procedures” on page 342501134 under LRU Test Procedures for the Garmin G1000 or “GRS/GMU Calibration Procedures” on page 342502134 for G1000 NXi.

C. Magnetic Compass

(1) Description

The magnetic compass is a self-contained instrument. This instrument has an individual light which is connected to the instrument lighting circuit. The compass correction card is located in the card holder mounted on the instrument.

Swing the compass as described under “Adjustment”, below, as follows:

(a) When the accuracy of the compass is suspected.

(b) After any cockpit modification or major replacement involving ferrous metal.

(c) Whenever a compass has been subjected to a shock; e.g., after a hard landing or turbulence.

(d) After aircraft has passed through a severe electrical storm.

(e) After lighting strike.

(f) Whenever a change is made to the electrical system.

(g) Whenever a change of cargo is likely to affect the compass.

(h) When aircraft’s area of operation is changed to a different geographic location with a major change in magnetic deviation. (e.g., from Miami, Florida to Fairbanks, Alaska.)

(i) After aircraft has been parked on one heading for over a year.

(j) When flux valves / magnetometers are replaced.

(2) Troubleshooting

See “Chart 6” on page 342013.

(3) Adjustment

Before attempting to compensate compass, every effort should be made to place the aircraft in simulated flight conditions; check to see that the doors are closed, flaps in retracted position, engine running, throttle set at cruise position and aircraft in level flight attitude. Aircraft master switch, alternator switch, and all radio switches should be in the ON position. All other cockpit controlled electrical switches should be in the OFF position.

(a) Set adjustment screws of compensator on zero. Zero position of adjusting screws is when the dot of the screw is lined up with the dot of the frame.

(b) Head aircraft on a magnetic North heading. Adjust N-S adjustment screw until compass reads exactly North.

(c) Head aircraft on a magnetic East heading and do the same as step (b), adjusting E-W adjusting screw.

(d) Head aircraft on a magnetic South heading and note resulting South error. Adjust N-S adjusting screw until one-half of this error has been removed.

(e) Head aircraft on magnetic West and do same as step (d), adjusting E-W adjustment screw.

(f) Head aircraft in successive magnetic 30° degree headings and record compass readings on appropriate deviation card. Deviations must not exceed +10° on any heading.

CHART 6
TROUBLESHOOTING MAGNETIC COMPASS

Trouble	Cause	Remedy
Excessive card error.	Compass not properly compensated.	Compensate instrument.
	External magnetic interference.	Locate magnetic interference and eliminate if possible.
Excessive card oscillation.	Insufficient liquid.	Replace instrument.
Card sluggish.	Weak card magnet.	Replace instrument.
	Excessive pivot friction or broken jewel.	Replace instrument.
Liquid leakage.	Loose bezel screws.	Replace instrument.
	Broken cover glass.	Replace instrument.
	Defective sealing gaskets.	Replace instrument.
Discolored markings.	Age.	Replace instrument.
Defective light.	Burned out lamp or broken circuit.	Check and replace lamp. Check continuity of wiring.
Card sticks.	Altitude compensating diaphragm collapsed.	Replace instrument.
Card does not move when compensating screws are turned.	The gears that turn compensating magnets are stripped.	Replace instrument.
Compass swings erratically. when radio transmitter is keyed.	Normal.	

6. Turn Coordinator

NOTE: In airplanes equipped with Avidyne Entegra IFDS, Garmin G600 or Garmin G1000 / G1000 NXi, this function is provided solely by the Primary Flight Display (PFD).

Unlike the conventional turn and slip indicator, the electrically operated gyroscope in the turn coordinator is canted. However, like conventional turn and slip indicator, it works on the principal of precession. By canting the gyro, the instrument not only measures rate of turn, but also measures rate of roll. With this indicator, if the aircraft is rolled right and left rapidly, the indicator will move, measuring the rate at which the airplane is rolled by indicating a turn in the direction of the roll. If the aircraft is held in a bank, and rudder is applied (such as when slipping), the needle indicator will come back to neutral, indicating no turn. The slip/skid portion of the indicator is a ball sealed in a curved glass tube filled with damping fluid. In the previous example, It would indicate the airplane is slipping. By utilizing rudder and aileron to establish the airplane in a desired rate of turn will eventually establish the airplane in a coordinated turn at the desired rate.

A. Troubleshooting

See “Chart 7”.

B. Removal and Installation

See 39-10-00, Face Mounted Instruments - Removal and Installation.

CHART 7
TROUBLESHOOTING TURN COORDINATOR

Trouble	Cause	Remedy
Instrument fails to respond when power is being applied to instrument.	Foreign matter lodged in instrument.	Replace instrument.
Incorrect sensitivity.	Out of calibration.	Replace instrument.
Incorrect turn rate.	Out of calibration.	Replace instrument.
Ball sticky.	Flat spot on ball.	Replace instrument.
Ball not in center when aircraft is correctly trimmed in wings level flight.	Instrument not level in panel.	Level instrument.
Instrument will not run.	No power to instrument.	Check appropriate circuit breaker on pilot’s circuit breaker panel. Check circuit and repair.
	Instrument malfunction.	Replace instrument.

MEGGITT SINGLE-SIDE EFIS OPTION

Electronic Flight Instrument System (EFIS) - Meggitt Single-side Option

Available in S/N's 3449269 and up.

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

A. Electronic Attitude Director Indicator (EADI)

i.e., Meggitt MAGIC EFIS Primary Flight Display (PFD)

The Primary Flight Display (PFD) is a microprocessor-based color, liquid crystal (LCD) display system. Information displayed includes Airspeed, Altitude, Attitude, Vertical Speed, Heading and Instrument Landing System data. The information is conveyed via predefined display formats. In addition, the PFD includes built-in test, performance and health monitoring functions and display brightness control. The PFD provides Gillam Code Altitude Output for the transponder (Mode C altitude).

Descriptions of the operator controls and basic operating modes and screens are in the appropriate Pilot's Operating Handbook Supplement.

NOTE: If any connection in the pitot / static system is opened for maintenance, the entire system must be rechecked per Appendix 1.

(1) Troubleshooting

See "Chart 1". See also Appendix 1.

CHART 1 (Sheet 1 of 3)
TROUBLESHOOTING EADI (I.E., MEGGITT PFD)

Trouble	Cause	Remedy
No display visible at power up.	+28 VDC power missing.	Check wiring & bus voltage.
	+28 VDC Return missing.	Check wiring.
	Failed PFD.	Replace PFD.
	EADI Display Down mode selected.	Check EADI Display Down switch.
Test mode visible at power up.	Stuck Baro Push button.	Check Baro Push button.
	PFD failed self test.	Replace PFD.
Display brightness does not change when PFD brightness control operated.	28 VDC lighting bus not connected or not functional.	Check wiring and 28 VDC lighting bus.
	Brightness control not functional.	Check Brightness control.
	Failed PFD.	Replace PFD.
Barometric correction does not change when BARO Rotary control is rotated.	PFD in test mode.	Ensure PFD in mission operational mode.
	Failed PFD.	Replace PFD.

CHART 1 (Sheet 2 of 3)
TROUBLESHOOTING EADI (i.e., MEGGITT PFD)

Trouble	Cause	Remedy
Barometric correction does not change to standard setting when Baro Push button is pressed.	PFD in test mode.	Ensure PFD in mission operational mode.
	Failed PFD.	Replace PFD.
Unable to enter test mode when Baro Push button is pressed.	Airspeed equal to or greater than 40 knots detected.	Ensure airspeed is less than 40 knots.
	Unit not powered off before Baro Push button is pressed.	Ensure unit is powered off before Baro Push button is pressed.
	Baro Push button pressed and held for insufficient time.	Ensure Baro Push button is pressed and held until initialization page displays.
	Failed PFD.	Replace PFD.
Test page does not change when Baro Rotary Control is rotated.	PFD in mission operational mode.	Ensure PFD in test mode.
	Failed PFD.	Replace PFD.
Power on self test status on Configuration Data page is indicated as FAIL.	Failed PFD.	Replace PFD.
Configuration discrete parity status on Configuration Data page is indicated as FAIL.	Parity input (Configuration Discrete 7) not wired correctly.	Check configuration discrete setting.
		Check wiring.
	Failed PFD.	Replace PFD.
Installation location on Configuration Data page incorrect.	Configuration discretes 0 and 1 not wired correctly.	Check configuration discrete setting.
		Check wiring.
	Failed PFD.	Replace PFD.
ADAHRS initialization screen is not removed from the display after 3 minutes have elapsed since ADAHRS power on.	ADAHRS not completing initialization.	Perform ADAHRS test.
Rad Alt indication and/or decision height annunciation are not displayed correctly.	Failed Radar Altimeter.	Perform Rad Alt test.
	Failed PFD.	Replace PFD.
Localizer/Glideslope and/or Backcourse/localizer sensing indications not displayed correctly.	Failed VOR radio.	Perform VOR radio test.
	Failed PFD.	Replace PFD.
Marker beacon indications not displayed correctly.	Failed marker beacon system.	Perform marker beacon system test.
	Failed PFD.	Replace PFD.

CHART 1 (Sheet 3 of 3)
TROUBLESHOOTING EADI (i.e., MEGGITT PFD)

Trouble	Cause	Remedy
Flight director bars not displayed correctly.	Failed Autopilot system.	Perform Autopilot system test.
	FD/AP switch not functional/wired correctly.	Check FD/AP switch and wiring.
	Failed PFD.	Replace PFD.
Full Field Red, Green, Blue, Black, and White pages are not displayed correctly.	Failed PFD.	Replace PFD.
ND Interface Data page shows valid data is not being transmitted.	Failed PFD.	Replace PFD.
ND Interface Data page shows valid data is not being received.	RS422 Receive A and B lines are not connected or are wired incorrectly.	Check wiring.
	Failed ND.	Replace ND.
ADAHRS Interface Data pages show valid data is not being received.	ARINC IN A and B lines are not connected or are wired incorrectly.	Check wiring.
	Failed ADAHRS.	Replace ADAHRS.
	Failed PFD.	Replace PFD.
Selecting the EADI Display Down switch to EADI Display Down does not remove the PFD display.	EADI Display Down switch not functional/wired correctly.	Check EADI Display Down switch and wiring.
	Failed PFD.	Replace PFD.
PFD screen format not displayed on the PFD when the EADI Display Down switch selected to normal.	EADI Display Down switch not functional/wired correctly.	Check EADI Display Down switch and wiring.
	Failed PFD.	Replace PFD.

(2) Basic Functional Test

CAUTION: IF, DURING THE FOLLOWING TEST, ANY COMPONENT DOES NOT PRODUCE THE EXPECTED RESULTS, THAT COMPONENT MUST NOT BE USED IN FLIGHT UNTIL THE PROBLEM IS CORRECTED.

(a) Before applying power, verify proper voltage inputs and ground connections as well as all system interfaces.

(b) Start the PFD in Test Mode by pressing and holding the Baro Push button while applying power until the Initialization page is shown. Verify that the Initialization page displays “Seneca Primary Flight Display” at the top of the screen and is replaced by the Configuration Data page after approximately five (5) seconds.

(c) Verify that the Configuration Data page Configuration Discrete Parity Status and Power On Self Test results both show results of Pass. If the result is FAIL, further diagnostic effort is required to determine if the failed indication is the result of improper installation, a faulty PFD, or both.

(d) Verify that the Configuration Data page displays the correct installation location (primary or secondary).

(e) Remove power from the PFD to terminate Test Mode. Re-apply power to the PFD and start the display in Mission Operational Mode.

(f) Verify that the PFD displays the Initialization page for approximately five (5) seconds prior to displaying the PFD Screen Format. If power has not been applied to the ADAHRS then the PFD screen format will be replaced by failure indications.

(g) Apply power to the Air Data Attitude Heading Reference System (ADAHRS) (i.e., AHRS).

(h) Verify that the PFD Screen Format is displayed (if the ADAHRS is performing initialization then the ADAHRS Initialization screen will replace the attitude sphere). If any of the failure indications are displayed refer to “Chart 1” on page 34211.

(i) Verify that the ADAHRS has completed initialization by observing that the attitude sphere replaces the ADAHRS Initialization screen. If the ADAHRS has not completed initialization within 180 seconds after application of power to the ADAHRS refer to “Chart 2” on page 34216.

(j) Apply power to the Navigation Display (ND) (i.e., EHSI).

(k) If installed, apply power to the Radar Altimeter. Perform a functional test on the Radar Altimeter system and verify that the PFD Rad Alt Indication and Decision Height Annunciation are displayed correctly.

(l) Apply power to the VOR radio. Perform a functional test on the VOR Radio system and verify that the PFD Localizer/Glideslope and Backcourse/Localizer Sensing Indications are displayed correctly.

(m) Apply power to the Marker Beacon system. Perform a functional test on the Marker Beacon system and verify that the PFD Marker Beacon indications are displayed correctly.

(n) Apply power to the Autopilot system. Place the Autopilot Master Switch in the Flight Director position and verify that the flight director bars are displayed correctly.

(o) Press the Baro Push button and verify that the barometric correction displays ‘STD’. Rotate the Baro Rotary Control clockwise and verify that the barometric correction increases. Rotate the Baro Rotary Control counter-clockwise and verify that the barometric correction decreases.

(p) On the overhead switch panel, decrease the Panel LCD’s dimmer switch and verify PFD brightness decreases.

(q) On the overhead switch panel, increase the Panel LCD’s dimmer switch and verify that the PFD brightness increases.

(r) Press the EADI Display Down switch. Verify that the PFD screen blanks and that the PFD screen format is displayed on the Navigation Display (ND) (i.e., EHSI) .

(s) Press the EADI Display Down switch again. Verify that the PFD screen format is displayed on the PFD and that the PFD screen format is removed from the ND.

(3) Removal and Installation

See 39-10-00.

B. Electronic Horizontal Situation Indicator (EHSI)

i.e., Meggitt MAGIC EFIS Navigation Display (ND)

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

The Navigation Display (ND) is, like the Meggitt Primary Flight Display (PFD), a microprocessor-based color, liquid crystal (LCD) display system. Information displayed includes Navigation Data, Autopilot Data, Radio Tuning, Airspeed, Altitude, Attitude, Vertical Speed, Heading and Instrument Landing System data. The information is displayed via a series of predefined display formats (HSI, ARC, MAP). In addition, the ND includes built-in test, performance and health monitoring functions and display brightness control.

Descriptions of the operator controls and basic operating modes and screens are in the appropriate Pilot's Operating Handbook Supplement.

(1) Troubleshooting

See "Chart 2" on page 34216. See also Appendix 1.

(2) Basic Functional Test

The following test procedure should be performed with the aircraft in a suitable position to receive valid VOR/ILS/GPS/DME signals. If this is not possible, then use the appropriate test sets to simulate the signals as required.

CAUTION: IF, DURING THE FOLLOWING TEST, ANY COMPONENT DOES NOT PRODUCE THE EXPECTED RESULTS, THAT COMPONENT MUST NOT BE USED IN FLIGHT UNTIL THE PROBLEM IS CORRECTED.

(a) Before applying power, verify proper voltage inputs and ground connections as well as all system interfaces.

(b) Start the ND in Test Mode by pressing and holding the MENU Push button while applying power until the Initialization page is shown. Verify that the Initialization page displays "Seneca Navigation Display" at the top of the screen and is replaced by the Configuration Data page after approximately five (5) seconds.

(c) Verify that the Configuration Data page Configuration Discrete Parity Status and Power On Self Test results both show results of Pass. If the result is FAIL, further diagnostic effort is required to determine if the failed indication is the result of a fault in the wiring, a faulty ND, or both.

(d) Verify that the Configuration Data page displays the correct installation location (primary or secondary).

(e) Remove power from the ND to terminate Test Mode. Re-apply power to the ND and start the display in Mission Operational Mode.

(f) Verify that the ND displays the Initialization page for approximately five (5) seconds prior to displaying the HSI Screen Format. If power has not been applied to the ADAHRS then the HSI screen format will display the heading failure indication.

CHART 2 (Sheet 1 of 4)
TROUBLESHOOTING EHSI (I.E., MEGGITT ND)

Trouble	Cause	Remedy
No display visible at power up.	+28 VDC power missing.	Check wiring & bus voltage.
	+28 VDC Return missing.	Check wiring.
	Failed ND.	Replace ND.
Display powers-up in EADI Display Down mode.	EADI Display Down mode selected.	Check EADI Display Down switch.
	Failed ND.	Replace ND.
Test mode visible at power up	Stuck MENU button	Check MENU button
	ND failed self test	Replace ND
Display brightness does not change when Panel LCD's dimmer switch operated	28 VDC lighting bus not connected or not functional	Check wiring and 28 VDC lighting bus
	Panel LCD's dimmer module not functional	Check/replace module
	Panel LCD's dimmer switch not functional	Check/replace switch.
	Failed ND	Replace ND
Menu does not appear when MENU button is pressed	Stuck MENU button	Check MENU button
	Failed ND	Replace ND
Menu cursor does not change when ↑ and/or ↓ buttons are pressed	Stuck ↑ and/or ↓ buttons	Check ↑ and ↓ buttons
	Failed ND	Replace ND
Unable to select menu options when SEL button is pressed	Stuck SEL button	Check SEL button
	Failed ND	Replace ND
Unable to enter Test mode when MENU button is pressed	Airspeed equal to or greater than 40 knots detected	Ensure airspeed is less than 40 knots
	Unit not powered off before MENU button is pressed	Ensure unit is powered off before MENU button is pressed
	MENU button pressed and held for insufficient time	Ensure MENU button is pressed and held until initialization page displays
	Failed ND	Replace ND
Test menu does not change when the ↑ and/or ↓ buttons are pressed	ND in mission operational mode	Ensure ND in test mode
	Stuck ↑ and/or ↓ buttons	Check ↑ and/or ↓ buttons
	Failed ND	Replace ND

CHART 2 (Sheet 2 of 4)
TROUBLESHOOTING EHSI (I.E., MEGGITT ND)

Trouble	Cause	Remedy
Unable to select Test pages when the SEL button is pressed	Stuck SEL button	Check SEL button
	Failed ND	Replace ND
Barometric setting does not change in EADI Display Down mode when the ↑ and/or ↓ buttons are pressed	Stuck ↑ and/or ↓ buttons	Check ↑ and ↓ buttons
	Failed ND	Replace ND
Selected CRS does not change when CSR knob rotated	Stuck CRS knob	Check CRS knob
	Failed ND	Replace ND
Selected HDG does not change when HDG knob rotated	Stuck HDG knob	Check HDG knob
	Failed ND	Replace ND
Selected CRS does not align with present heading when CSR knob pushed	Stuck CRS knob	Check CRS knob
	Failed ND	Replace ND
Selected HDG does not align with present heading when HDG knob pushed	Stuck HDG knob	Check HDG knob
	Failed ND	Replace ND
Failure flags present on display	Equipment supplying relevant data to ND failed	Check status of equipment supplying data to ND
	Failed ND	Check data on relevant ND test page
Power on self test status on Configuration Data page is indicated as FAIL	Failed ND	Replace ND
Configuration discrete parity status on Configuration Data page is indicated as FAIL	Parity input (Configuration Discrete 7) not wired correctly	Check configuration discrete setting
		Check wiring
	Failed ND	Replace ND
Installation location on Configuration Data page incorrect	Configuration discretes 0 and 1 not wired correctly	Check configuration discrete setting
		Check wiring
	Failed ND	Replace ND
PFD Interface Data Page shows valid data is not being received/transmitted	RS422 Receive and/or Transmit lines are not connected or wired correctly	Check wiring
	Failed ND	Replace ND
	Failed PFD	Replace PFD

CHART 2 (Sheet 3 of 4)
TROUBLESHOOTING EHSI (I.E., MEGGITT ND)

Trouble	Cause	Remedy
ADAHRS Interface Data pages show valid data is not being received	ARINC IN A and B Bus 1 lines are not connected or are wired incorrectly	Check wiring
	Failed ADAHRS	Replace ADAHRS
	Failed ND	Replace ND
NAV Interface Data Page shows valid data is not being Received	ARINC In A and B Bus 2 lines are not connected or are wired incorrectly	Check wiring
	Failed ND	Replace ND
	Failed Nav radio	Replace Nav radio
GPS Interface Data Pages show valid data is not being Received	ARINC In A and B Bus 4A and/or 4B lines are not connected or are wired incorrectly	Check wiring
	Failed ND	Replace ND
	Failed GPS Receiver	Replace GPS Receiver
Cross Side Input Interface Data Page shows valid data is not being Received	ARINC In A and B Bus 3 lines are not connected or are wired incorrectly	Check wiring
	Failed ND	Replace ND
	Failed cross-side ND	Test cross-side ND
Cross Side Output Interface Data Page shows valid data is not being Transmitted	ARINC Out A and B Bus 2 lines are not connected or are wired incorrectly	Check wiring
	Failed ND	Replace ND
NAV/GPS Output Interface Data Page shows valid data is not being transmitted	ARINC Out A and B Bus 1 lines are not connected or are wired incorrectly	Check wiring
	Failed ND	Replace ND
Analog Interface Data Page shows valid data is not being Received or Transmitted	Failed ND	Replace ND
	Failed Autopilot Input	Perform Autopilot Test
	Failed DME Input	Perform DME Test
	Failed Audio Panel Input	Perform Audio Panel Test
	Failed Rad Alt Input	Perform Rad Alt Test
	Failed ADF Input	Perform ADF Test
	Failed Autopilot outputs	Check ND inputs

CHART 2 (Sheet 4 of 4)
TROUBLESHOOTING EHSI (I.E., MEGGITT ND)

Trouble	Cause	Remedy
Discrete Interface Data Page shows valid data is not being Received or Transmitted	Failed ND	Replace ND
	Failed Autopilot Input	Perform Autopilot Test
	Failed DME Input	Perform DME Test
	Failed Rad Alt Input	Perform Rad Alt Test
	Failed DAU output	Check GPS inputs
	Failed Autopilot output	Check NAV inputs

(g) Apply power to the Air Data Attitude Heading Reference System (ADAHRS) (i.e., AHRS).

(h) Apply power to the Primary Flight Display (PFD) (i.e., EADI).

(i) Verify that the HSI Screen Format is displayed and the installation specific defaults as shown in “Chart 3” on page 342111 apply. If any of the failure flags are displayed refer to “Chart 2” on page 34216.

(j) Enable the Mission Operational Mode/Rev Mode menu by pressing the bezel mounted ‘MENU’ button. Verify that the menu is displayed correctly.

(k) Ensure that the Nav Radios are selected off. Use the bezel mounted “↑, ↓” and “SEL” buttons to navigate through the menu, selecting RMI1 to display VOR1 and RMI2 to display VOR2. Verify that the RMI1 and RMI2 second level menus are displayed correctly and the RMI1 and RMI2 source annunciations match the selections.

(l) Verify that the RMI1, RMI2 and CDI source annunciations are amber in color. Verify that the associated frequency annunciations are replaced with amber dashes. Verify that both RMI pointers and the CDI pointer are removed from the display.

(m) Apply power to the Nav radios. Select valid VOR frequencies on the Nav Radios and verify that the RMI1, RMI2 and CDI source and frequency annunciations are displayed in the correct colors (see “Chart 4” on page 342111). Verify that the ND displayed frequencies match the Nav radio frequencies (i.e. RMI1 frequency matches Nav Radio 1 and RMI2 frequency matches Nav Radio 2. The CDI frequency should match the currently selected CDI source frequency). Verify that both RMI pointers and the CDI pointer are displayed correctly.

(n) Select valid ILS frequencies on the Nav Radios and verify that the RMI1, RMI2 and CDI source annunciations display VOR1, VOR2 and ILS1/ILS2 (dependent on current selected CDI source) respectively in the correct colors (see “Chart 4” on page 342111). Verify that the ND displayed frequencies match the Nav radio frequencies (i.e. RMI1 frequency matches Nav Radio 1 and RMI2 frequency matches Nav Radio 2. The CDI frequency should match the currently selected CDI source frequency). Verify that both RMI pointers are removed and the CDI pointer is displayed correctly.

(o) Apply power to the GPS Receivers. Use the bezel mounted “↑ ,↓” and “SEL” buttons to navigate through the menu, selecting RMI1 and RMI2 to display GPS and CDI (verify that the CDI second level menu is displayed correctly) to display GPS (GPS1 or GPS2 will be available dependent on GPS source selection). Verify that the RMI1, RMI2 and CDI source annunciations reflect the selections.

(p) Refer to the GPS Installation Manual and display the self test page on the on-side GPS. Verify that the applicable ND parameters display the correct self test values as listed in the GPS Installation Manual during the display of the self test page.

(q) Verify that the GPS source annunciation displays the currently selected GPS source in white. Use the bezel mounted “↑, ↓” and “SEL” buttons to navigate through the menu, selecting the GPS source to the off-side GPS. Verify that the GPS second level menu is displayed correctly. Verify that the GPS source annunciation displays the currently selected GPS source in white.

(r) Refer to the GPS Installation Manual and display the self test page on the off-side GPS. Verify that the applicable ND parameters display the correct self test values as listed in the GPS Installation Manual during the display of the self test page.

(s) Press the EADI Display Down switch to select EADI Display Down mode and verify that the ND displays the PFD (i.e., EADI) screen format. Verify that the PFD screen blanks.

(t) If installed, apply power to the Radar Altimeter. Perform a functional test on the Rad Alt system and verify that the ND Rad Alt Indication and Decision Height Annunciation are displayed correctly.

(u) Perform a functional test on the VOR Radio system and verify that the ND Localizer / Glideslope and Backcourse / Localizer Sensing indications are displayed correctly.

CHART 3
NAVIGATION DISPLAY (ND) DEFAULT SETTINGS (MEGGITT)

SCREEN FORMAT	HSI
ELEMENT	**SETTING**
CDI	VOR1
RMI1	OFF
RMI2	OFF
RANGE	2.5NM
GPS SEL	GPS1
APORTS	Disabled
NAVAIDS	Disabled
WPTS	Disabled
NDB	Disabled
Selected Heading	Last Known Setting
Selected Course / Desired Track	Last Known Setting
Barometric Setting	Last Known Setting

CHART 4
NAVIGATION DISPLAY (ND) COLOR CODING (MEGGITT)

	Display Parameter	
Color	**Mission Operational Mode**	**EADI Display Down Mode**
White	A/C Symbol, Scales, Range Rings, Digital Readouts, RMI1 Pointer and Annunciation	Scales, Digital Readouts
Grey	N/A	Scale Tapes
A/C Symbol	N/A	Aircraft Symbol, Lubber Line
Amber	Failures and Annunciation's	Stall Warning, Failures and Annunciation's
Red	N/A	Airframe Limits
Sky	N/A	Sky
Ground	N/A	Ground and Radar Altimeter ground reference
Green	N/A	Baro Correction
Magenta	Deviation Bar, Sel HDG/CRS Bugs, CDI Pointer and Annunciation	Flight Director Command Bars and Deviation Pointers
Cyan	RMI2 Pointer and Annunciation	N/A

(v) Apply power to the Marker Beacon system. Perform a functional test on the Marker Beacon system and verify that the ND Marker Beacon indications are displayed correctly.

(w) Apply power to the Autopilot system. Place the Autopilot Master Switch in the Flight Director position and verify that the flight director bars are displayed correctly.

(x) Press the bezel mounted "↑" button once and verify that the barometric correction increases by 0.01in Hg/1hPa. Press and hold the bezel mounted "↑" button and verify that the barometric correction continuously increases in increments of 0.01in Hg/1hPa until the button is released or the maximum barometric correction is reached.

(y) Press the bezel mounted "↓" button once and verify that the barometric correction decreases by 0.01in Hg/1hPa. Press and hold the bezel mounted "↓" button and verify that the barometric correction continuously decreases in increments of 0.01in Hg/1hPa until the button is released or the minimum barometric correction is reached.

(z) Press the EADI Display Down switch to shift back to normal mode and verify that the HSI screen format is displayed on the ND. Verify that the PFD screen format displays on the PFD.

(aa) Use the bezel mounted "↑", "↓" and "SEL" buttons to navigate through the menu, selecting the Arc screen format. Verify that the Arc screen format is displayed correctly.

(ab) Use the bezel mounted "↑", "↓" and "SEL" buttons to navigate through the menu, selecting the Map screen format. Verify that the Map screen format is displayed correctly.

(ac) Use the bezel mounted "↑", "↓" and "SEL" buttons to navigate through the menu, selecting the HSI screen format. Rotate the CRS Rotary Control clockwise and verify that the selected CRS pointer rotates clockwise around the compass card and the selected course digital readout increases and mirrors the pointer heading. Rotate the CRS Rotary Control anti-clockwise and verify that the selected CRS pointer rotates anti-clockwise around the compass card and the selected course digital readout decreases and mirrors the pointer heading. Push the CRS Push button and verify that the selected course pointer and digital readout align to the current aircraft heading.

(ad) Rotate the HDG Rotary Control clockwise and verify that the selected HDG bug rotates clockwise around the compass card and the selected heading digital readout increases and mirrors the bug heading. Rotate the HDG Rotary Control anti-clockwise and verify that the selected HDG bug rotates anti-clockwise around the compass card and the selected heading digital readout decreases and mirrors the bug heading. Push the HDG Push button and verify that the selected heading bug and digital readout align to the current aircraft heading.

(ae) On the overhead switch panel, decrease the Panel LCD's dimmer switch and verify that the ND brightness decreases.

(af) On the overhead switch panel, increase the Panel LCD's dimmer switch and verify that the ND brightness increases.

(3) Removal and Installation

See 39-10-00.

C. Air Data & Attitude Heading Reference System (ADAHRS)

i.e., Attitude Heading Reference System (AHRS).

The ADAHRS unit is located on the aft equipment shelf behind the cabin rear closeout panel.

NOTE: When installing the ADAHRS unit; first, level the airplane per 8-20-00; and then, use NAS1149FN816P washers as required to level the ADAHRS unit.

NOTE: If any connection in the pitot / static system is opened for maintenance, the entire system must be rechecked per Appendix 1.

(1) Magnetic Heading Compensation / Calibration

Perform the Magnetic Heading Compensation/Calibration procedure in Magnetic Heading Systems, below, whenever the ADAHRS unit is installed or replaced.

(2) Ground Test Procedure

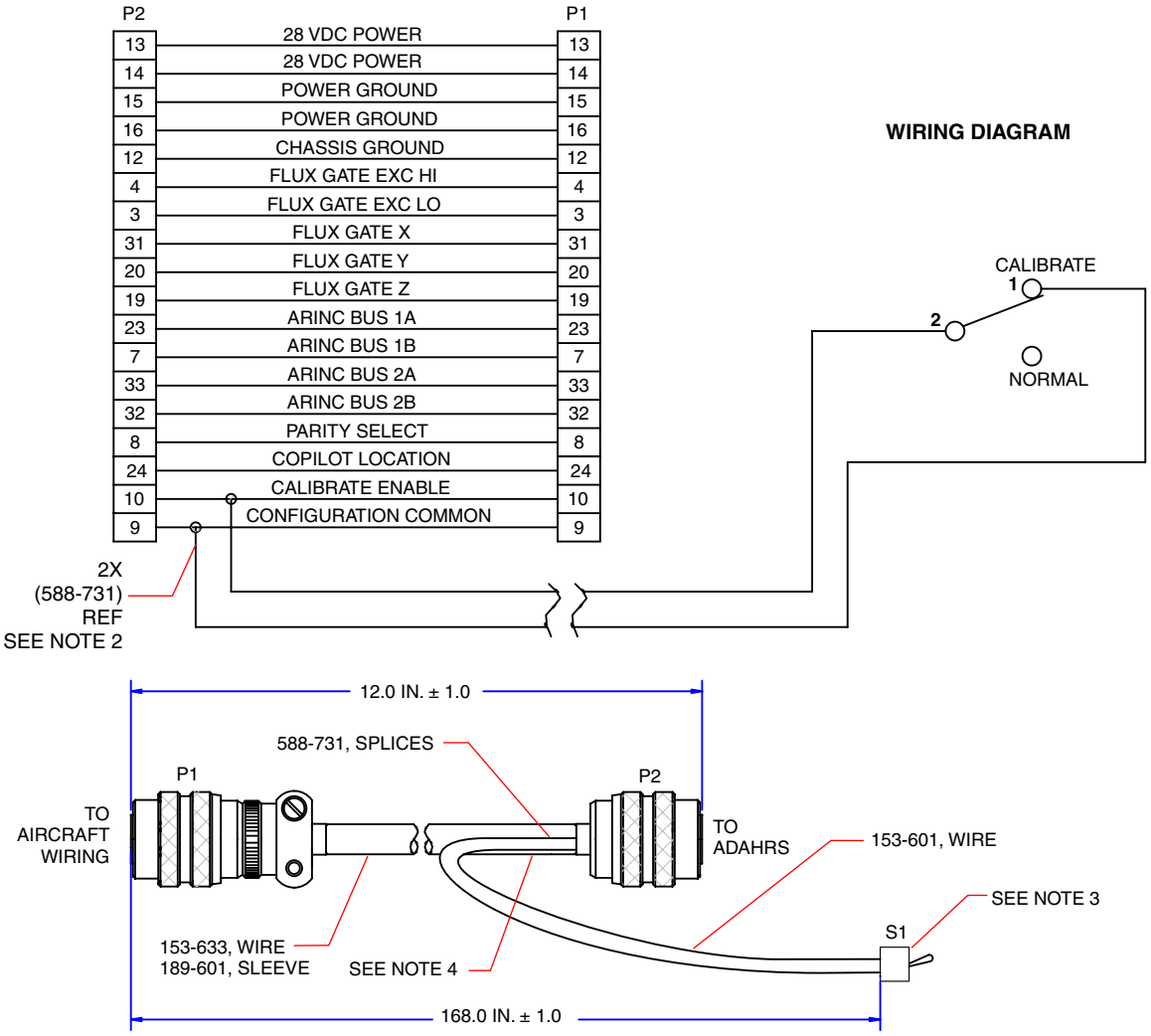
This test procedure (included as Appendix 1) verifies the integrity and accuracy of the air data and pitot-static systems and shows that these systems comply with the applicable airworthiness regulations. Perform this procedure:

(a) Each 24-months,

(b) To proof test the static system in accordance with each time any connections in static system are opened for maintenance, or

(c) Following installation or maintenance on the ADAHRS unit, the primary flight display (PFD), or the transponder Mode C interface.

NOTES:
1. COVER 153-633 WIRE WITH WHITE SHRINK SLEEVE 189-601.
2. USE RAYCHEM SPLICES, 588-731 TO CONNECT TO SWITCH TO WIRES SHOWN IN WIRING DIAGRAM.
3. LABEL SWITCH CALIBRATION POSITION WITH "CAL".
4. LABEL ASSEMBLY AS FOLLOWS "FOR CALIBRATION PURPOSE ONLY, "NOT APPROVED FOR PERMANENT INSTALLATION.



1	602-513	SWITCH, SPDT			MTL-106D							9
1	599-698	CONNECTOR, RECEPTACLE			MS27473T14B-35S							8
1	599-697	CONNECTOR, STRAIGHT PLUG			MS27508E14B35P							7
2	588-731	SPLICES			PS10025-16-2							6
A/R	189-601	SLEEVE, PTFE SHRINK, WHT			M23053/12-203-3							5
A/R	153-633	WIRE, PTFE, 22AWG WHT			M22759/11-22-9							4
A/R	153-601	WIRE, CABLE-EL			PMS-E0017-22-24							3
												2
X	-002	HARNESS ASSEMBLY-ADAHRS CALIBRATION				3400	M	1530				1

ITEM NO.	ITEM NAME	CODE NO. SPEC CONTROL DWG	MATERIAL	STOCK SIZE	MATERIAL SPECIFICATION	HEAT TREAT	FINISH SPECIFICATION	E-ITEM GROUP	ITEM TYPE	ITEM GROUP	NOTE	ZONE	POS. NO.

INSTL. ASSY, SUB ASSY, OR KIT

REVISION

			A	REV
			DC	BY
			6/16/03	DATE
				APPD
				REV
				BY
				DATE
				APPD

DRAFTSMAN	T TAYLOR	5/28/02
DESIGN ENGR		
CHECKER		
PROGRAM MGR		

SCALE | LAST SECTION OR VIEW
WT. | DRAWING SIZE D
TOL. .XXX = +/- .010 .XX = +/- .030 .X = +/- .10

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THE NEW PIPER AIRCRAFT, INC.
VERO BEACH, FLORIDA

MANUFACTURING PRACTICES - SEE PIPER REPORT V8-9

MANUFACTURING
MATERIALS

ADAHRS CALIBRATION HARNESS ASSEMBLY

REF. LAYOUT NO. (S)

SH 1 OF 1

101453 A

ADAHRS Calibration Harness
Figure 4

AVIDYNE ENTEGRA IFDS

Avidyne Entegra Integrated Flight Display System (IFDS)

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.

The Avidyne Entegra Integrated Flight Display System (IFDS) was installed as standard equipment in S/N's 3449311, 3449323, 3449326 thru 3449409, 3449411 thru 3449414, and 3449420.

A. Description

See "Figure 1" on page 34223 and "Figure 2" on page 34224.

This system uses two large 10.4-inch diagonal, high-resolution, sunlight-readable full color displays (PFD and MFD), to provide primary flight and engine information as well as a wide variety of other data. Standard primary flight instruments (i.e., airspeed, electric attitude indicator, and altimeter) provide redundancy.

The IFDS installation consists of the following components: Primary Flight Display (PFD), Multifunction Display (MFD), Data Acquisition Unit (DAU) and associated sensors, and Magnetometer/OAT Sensor Assembly.

B. Maintenance

The Instructions for Continued Airworthiness (ICA) published by Avidyne provide the necessary information for maintaining this system as installed in Piper airplanes, except as noted below.

NOTE: The Avidyne ICA documents cited below are included in Piper Tech Pubs subscriptions and are available from Avidyne.

(1) Primary Flight Display (PFD) (PIR-PPS60219, Rev. A/PPS60219-1, Rev. A.)

Use 700-00006-0XX PFD & 700-00011-0XX Mag/OAT ICA, Avidyne Document No. AVPFD-174, Revision 03, or later, with the following exceptions:

NOTE: Before attempting to set-up the PFD, ensure the GNS-430(s) / GNS-430W(s) have been configured per the "Post Installation Setup Procedure" on page 34501 under COM/NAV/GPS.

NOTE: In S/N's 3449301 and up, before attempting to set-up the PFD, ensure the GTX-330 is configured per the "Post Installation Setup Procedure" on page 345015 under Transponder, GTX 330.

NOTE: Whenever the PFD is replaced perform the setup / calibration procedures specified in Avidyne Document No. AVPFD-174 as modified below.

NOTE: If any connection in the pitot / static system is opened for maintenance, the entire system must be rechecked per Pitot / Static System "Test" on page 34108.

NOTE: The following PFD software part numbers were factory installed in these airplanes: (earlier versions may have been upgraded in the field)

530-00177-000 (i.e., Release 6.1 and earlier)

530-00183-000 (i.e., Release 6.2 and later) (EASA only)

530-00194-000 (i.e., Release 7.0)

(a) In para 5.1, for airplanes equipped with PFD Software 530-00194-000 or later, the "ADU Calibration" referenced in conjunction with the bi-annual altimeter check can be performed by following the prompts from the ADU Calibration page.

(b) In para 6, “Troubleshooting Information,” in the chart where it says “OAT (Option),” cross out “option.” The OAT is standard in the Piper installation.

(c) In para 7.2, “Primary Flight Display Installation,” and Figure 7, the standard Avidyne installation describes alignment and mating pins and retaining clips on the sides of the PFD. For the Piper installation, however, there is only a single alignment pin on the top of the PFD engaging a slot in the upper rear cross bracket. (See Figure 3, Sheet 1.)

(d) In para 7.5.1, apply power by turning the Battery Master Switch ON or by applying external power and turning the Radio Master Switch ON.

In airplanes equipped with PFD Software 530-00194-000 or later, when the countdown is finished and the system setup page appears, select the “System Info” tab and verify checksums have no “FAILED” or “PENDING” messages.

(e) In airplanes equipped with PFD Software 530-00194-000 or later, at the end of para 7.5.2.1, add the following:

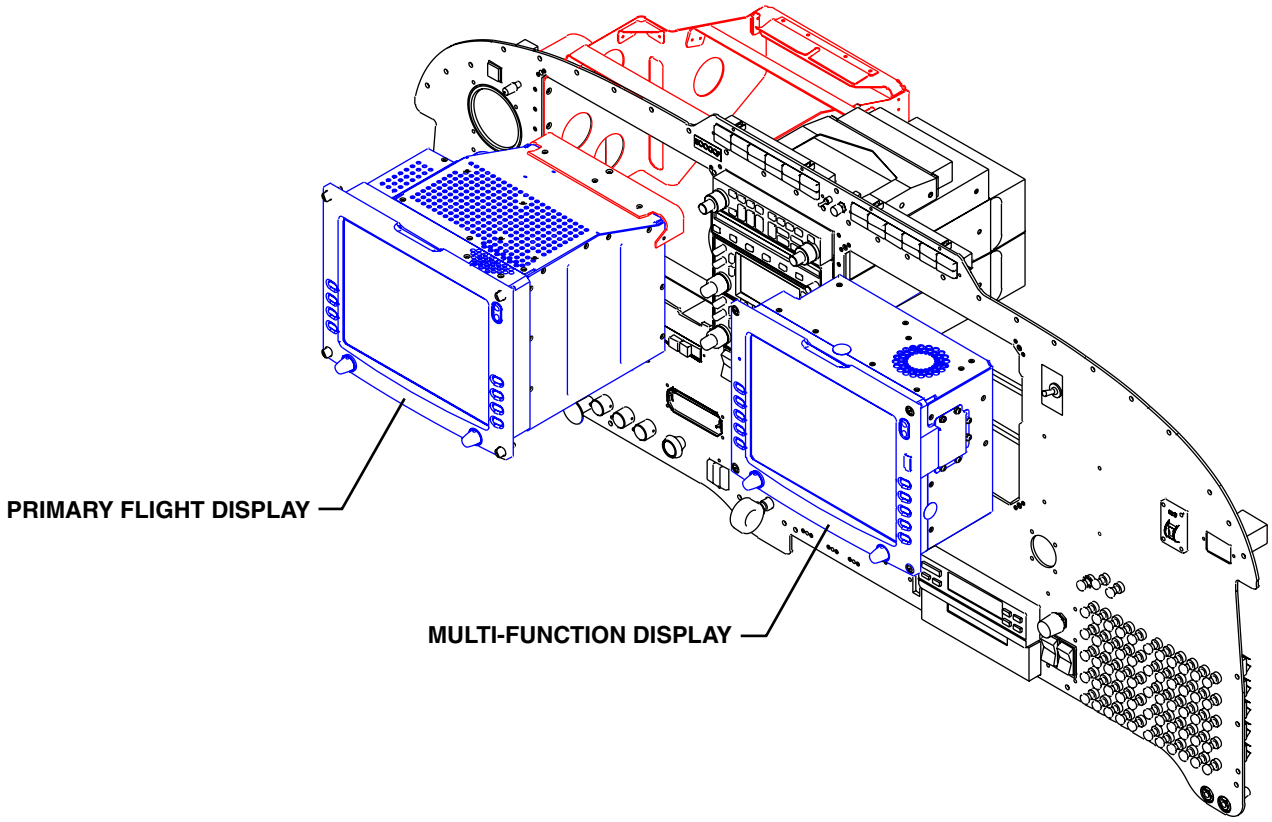
With 34-220T selected verify the following:

PFD Serial Number:	XXXXXXXXXXX
S/W Version ID:	XXXXXXXXXXX
Aircraft Make:	Piper
Model:	PA-34-220T
Pitch Offset:	0.0
Avionics:	
GPS 1:	Garmin GNS-430/530
GPS 2:	Garmin GNS-430/530
VHF 1:	Garmin GNS-430/530
VHF 2:	Garmin GNS-430/530
Autopilot:	S-TEC System 55x
Engine DAU1:	“Moritz DAU”
Engine DAU2:	“Moritz DAU”
ADF:	Not Installed
Radar Altimeter:	Not Installed
Copilot PFD:	Not Installed
Dimming Bus:	28 VDC

NOTE: If GPS 1, GPS2, VHF 1, or VHF 2 do not display the correct Garmin unit, proceed to para 7.7 in Avidyne Document No. AVPFD-174 as modified by step (h), below, and update the display. Then repeat step (e).

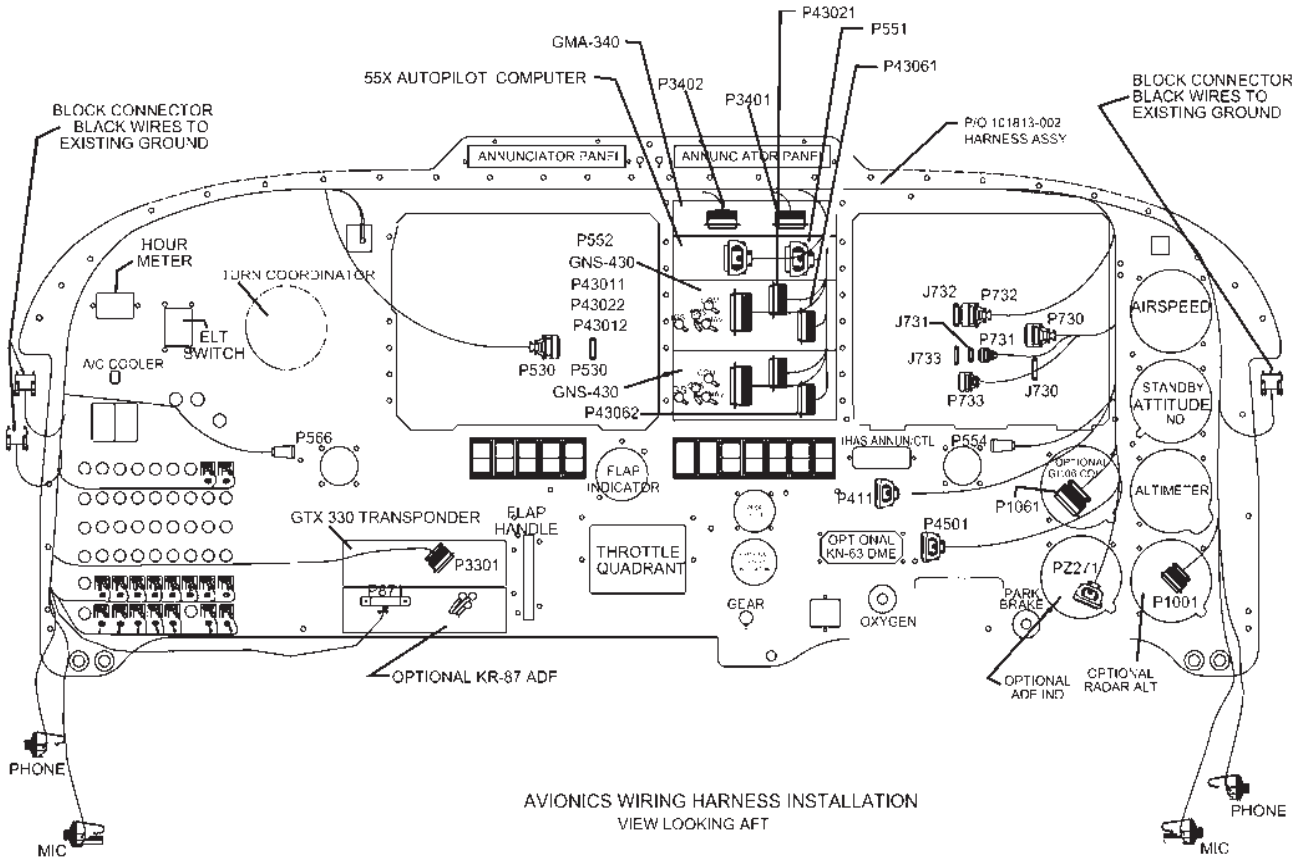
(f) Following para 7.5.3, add the following:

1) Rotate the lower left control knob to select the Display Set-up Page by highlighting the “Display” tab at the bottom of the screen.
2) Press the “Trim Ann” Line Select Key (L1) until “SHOW” is displayed.
3) Press the “A/P Annun” Line Select Key (L2) until “SHOW” is displayed.
4) Press the “V-Speed” Line Select Key (L3) until “SHOW” is displayed.
5) Press the “Horiz Marks” Line Select Key (R3) until “SHOW” is displayed.
6) Depress the “ARS” (Aircraft Reference Symbol) Line Select Key (R4) until “DELTA” is displayed.



Avidyne Entegra Installation
Figure 1

Effectivity
Seneca V
with Avidyne Entegra



AVIONICS WIRING HARNESS INSTALLATION
VIEW LOOKING AFT

Avidyne Entegra Wiring Installation
Figure 2

(g) If using para 7.6, add the following:

Verify that the Avionics Page Display indicates:

Aircraft Make:	Piper
Model:	PA34-220T
Sensor Suite:	
GPS 1:	GARMIN 430/530 on ARINC 1
GPS 2:	GARMIN 430/530 on ARINC 3
VHF 1:	GARMIN 430/530 on ARINC 2
VHF 2:	GARMIN 430/530 on ARINC 4
AutoPilot:	STEC 55X
Engine DAU:	INSTALLED
ADF:	NOT INSTALLED (even if an ADF is installed)
Copilot PFD:	NOT INSTALLED

(h) If using para 7.7, add the following:

Verify that the Avionics Page Display indicates:

GPS 1:	Garmin GNS-430/530
Inputs:	ARINC Rx1
Outputs:	ARINC Tx0
GPS 2:	Garmin GNS-430/530
Inputs:	ARINC Rx3
Outputs:	ARINC Tx0
VHF 1:	Garmin GNS-430/530
Inputs:	ARINC Rx2
VHF 2:	Garmin GNS-430-530
Inputs:	ARINC Rx4
Autopilot:	STEC 55X
Attitude Rate Source:	External
Attitude Selector:	Emulating STEC ST-360
Annunciators:	Emulating STEC ST-645
Horizontal NAS Data:	Emulating Bendix/King KCS 55A
Flight Director:	Analog
GPS Roll Steering:	Retransmitting on ARINC Tx0
Engine DAU 1:	Moritz DAU
Engine DAU 2:	Moritz DAU
ADF:	Not Installed
Radar Altitude:	Not Installed
Copilot PFD:	Not Installed
Dimming Bus:	28 V DC

(i) In para 7.7.1 in the “Main RS232 Configuration Page” table, for CHNL 3 under GNS-430 in slot # 2, both Input and Output should read “Crossfill” instead of “Off.”

(j) In para 7.7.3, add the following note: “An airfield compass rose may be used to align the airplane in lieu of a handheld sight compass.”

(2) Multifunction Display (MFD)

(PIR-PPS60218, Rev. A)

Use 700-00004-0XX-() Multifunction Display ICA, Avidyne Document No. AVMFD-167, with the following exceptions:

NOTE: Whenever the MFD is replaced, calibrate the Fuel Quantity Indicator as described under Seneca V - Avidyne Fuel Quantity Calibration in 28-40-00.

(a) In para 2, items 7 and 8 are standard in the Piper installation.

(b) Replace items 3 and 4 in paragraph 7.3.1, Maintenance Mode Access with the following:

1) At the prompt “Press any bezel key to continue”, press any LSK. This places the screen at the exceedance page. Press O.K. to acknowledge exceedances and to continue to the Initial Usable Fuel page. Depress the “Fuel Done” LSK (R4).

2) Rotate the right outer concentric knob clockwise until the AUX page is displayed.

(c) Replace paragraph 7.3.2.1, GAMA 429 Graphics Setup (EX5000 only), with the following:

1) From the Maintenance page, depress the LSK (L1) to access the GPS/FMS Setup page.

2) Using the right outer concentric knob to select the configuration field and the right inner concentric knob to select the desired configuration; configure the GPS/FMS Setup page for the following conditions:

Receiver 1:	GAMA 429 FORMAT
Port:	ARINC 429 RX 1
Speed:	LOW
Receiver 2:	GAMA 429 FORMAT
Port:	ARINC 429 RX 2
Speed:	LOW

3) Press the SAVE button.

(d) Replace paragraph 7.3.5.1, TAS (SkyWatch & Bendix King) Setup, with the following:

1) If equipped with SKY-497, configure the Traffic Setup page for the following conditions (otherwise not installed):

Sensor:	TAS
Port:	ARINC 429 TX 3
TAS Type:	SKYWATCH
External Controller Box:	NOT CHECKED

2) Press the SAVE button.

(e) Replace paragraph 7.3.4, Lightning Sensor Interface and Setup, with the following:

1) From the Maintenance Setup page depress the LSK (L3) to access the Lightning Setup page.

2) Using the right outer concentric knob to select the configuration field and the right inner concentric knob to select the desired configuration. Configure the Lighting Setup page for the following condition:

Sensor	WX-500
Operating Mode:	Weather
Port:	RS232-3-(Lightning Default)
Stabilization Type:	Use Map Heading/Track
Enable Lightning Ahead:	Box CHECKED
Antenna on Top:	Box NOT CHECKED

3) Press the SAVE button.

(f) Replace paragraph 7.3.10.1, Engine Sensor Setup, with the following:

1) From the Maintenance Page, depress the LSK (L4) to access the Traffic Setup page.

2) Using the right outer concentric know to select the configuration field and the right inner concentric knob to select the desired configuration. Configure the Engine Setup page for the following conditions:

Aircraft Model:	PA34-220T (Seneca V)
Left Engine Serial Port:	RS232 4
Right Engine Serial Port:	RS232 2
Vacuum System Installed:	Box CHECKED if aircraft is deice equipped, NOT CHECKED if aircraft is not deice equipped

3) Press the SAVE button.

4) When configuring a MFD for the first time only, the following setup box appears: “Select ACFT Model and Port(s).” Select “Resync.”

(g) Replace paragraph 7.3.8, Map Setup, with the following:

1) From the Maintenance Setup page, depress the LSK (R1) to access the MAP Setup page.

2) Using the right outer concentric knob to select the configuration field and the right inner concentric knob to select the desired configuration; configure the Map Heading Setup page for the following conditions:

Map Heading:	FMS/GPS

3) Press the SAVE button.

(h) After completing the previous procedures, complete the Aircraft Setup procedure as follows:

1) From the Maintenance page, depress the LSK (R2) to access the AIRCRAFT Setup page.

2) Using the right outer concentric knob to select the configuration field and the right inner concentric knob to select the desired configuration; configure the AIRCRAFT Setup page for the following conditions:

Narrowcast:	Quake SC
Port:	RS232 6 (Datalink Default)
Broadcast:	XM Radio
Port:	ARINC 429 RX 4
Dimming Bus Voltage:	No setting required
Brightest Dimming Voltage:	24.0 Volts
Darkest Dimming Voltage:	5.0 Volts

3) Press the SAVE button.

(i) See 28-40-00 for Fuel Quantity Indicator calibration.

(3) Data Acquisition Unit (DAU)

NOTE: Whenever the DAU is replaced, calibrate the Fuel Quantity Indicator as described under Seneca V - Avidyne Fuel Quantity Calibration in 28-40-00.

(a) Use 200-00041-000 DAU ICA, Avidyne Document No. AVSIU-011, with the following exceptions:

1) In para 6, in “Table 2 - DAU Pinout,” pins J1-2 and J1-21 have “No Connection” in the Piper installation.

2) In para 6, in “Table 4 - DAU Sensor Compatibility,” parameter “VAC” is not used in the Piper installation.

(b) Removal and Installation

The DAU’s are mounted on the forward side of the nacelle bulkhead assembly for each engine. Maintenance is on condition.

(4) Magnetometer/OAT Sensor Assembly

See “Figure 3” on page 34229.

The Magnetometer / OAT Sensor Assembly (Mag/OAT) is mounted on a wing access cover plate in the underside of the outboard left wing and supplies magnetic heading information to the Primary Flight Display (PFD). The cover plate - Mag/OAT sensor assembly is removed and installed as a unit.

NOTE: The OAT sensor probe is disconnected and not used.

(a) Removal

1) Remove eight (8) screws and support cover plate with your hand.

2) Drop cover plate down sufficient to reach inside and disconnect the wiring harness.

3) Remove cover plate - Mag/OAT sensor assembly.

(b) Installation

NOTE: Whenever the Mag/OAT is replaced perform the PFD setup / calibration procedures specified in Avidyne Document No. AVPFD-174 as modified by “Primary Flight Display (PFD)” on page 34221.

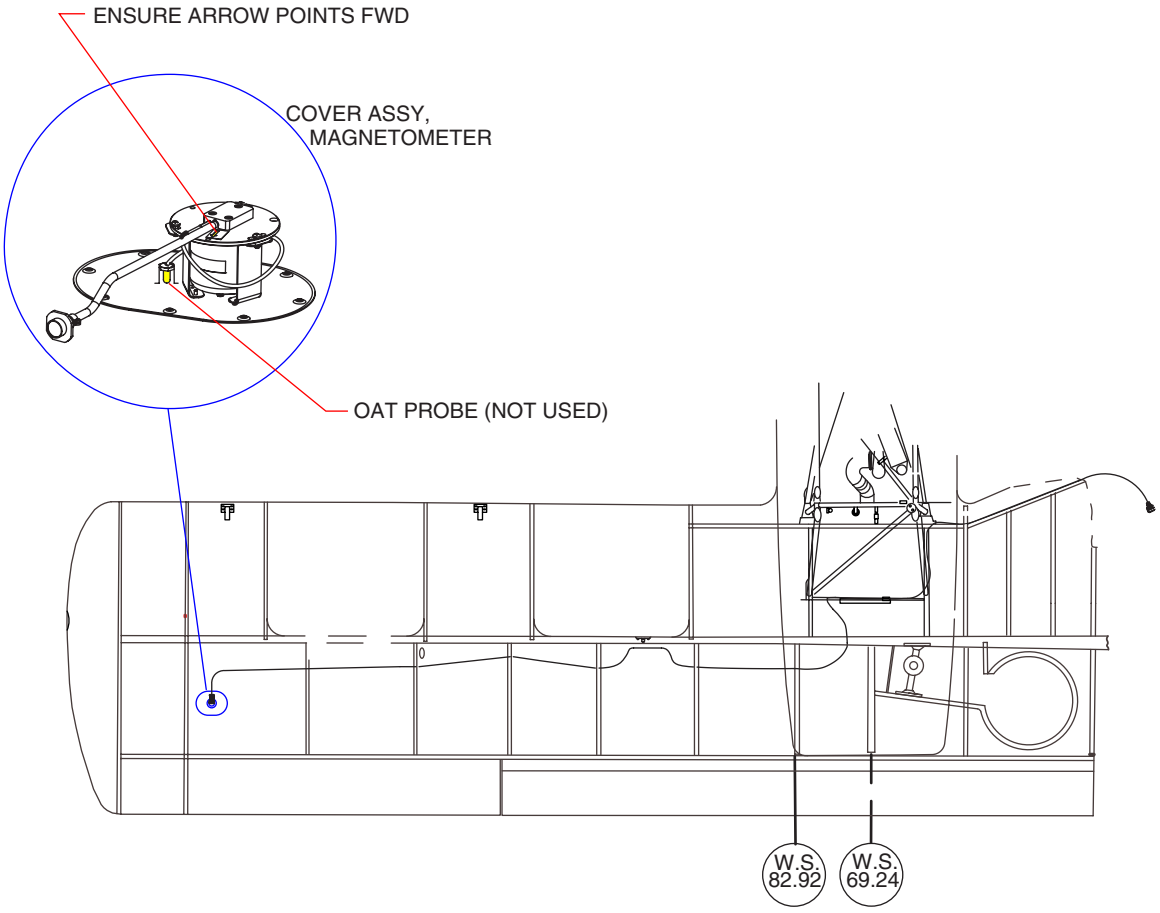
1) Connect the Mag/OAT sensor assembly wiring harness.

2) Position the cover plate - Mag/OAT sensor assembly in the access hole with the arrow on the magnetometer pointing forward.

3) Secure with screws (8).

C. Component Locator

See “Figure 1” on page 34223, “Figure 2” on page 34224, and “Figure 3” on page 34229.



Magnetometer / OAT Installation
Figure 3

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GARMIN G600

Flight Display System - Garmin G600

(PIR-190-00601-06, Rev. E.)

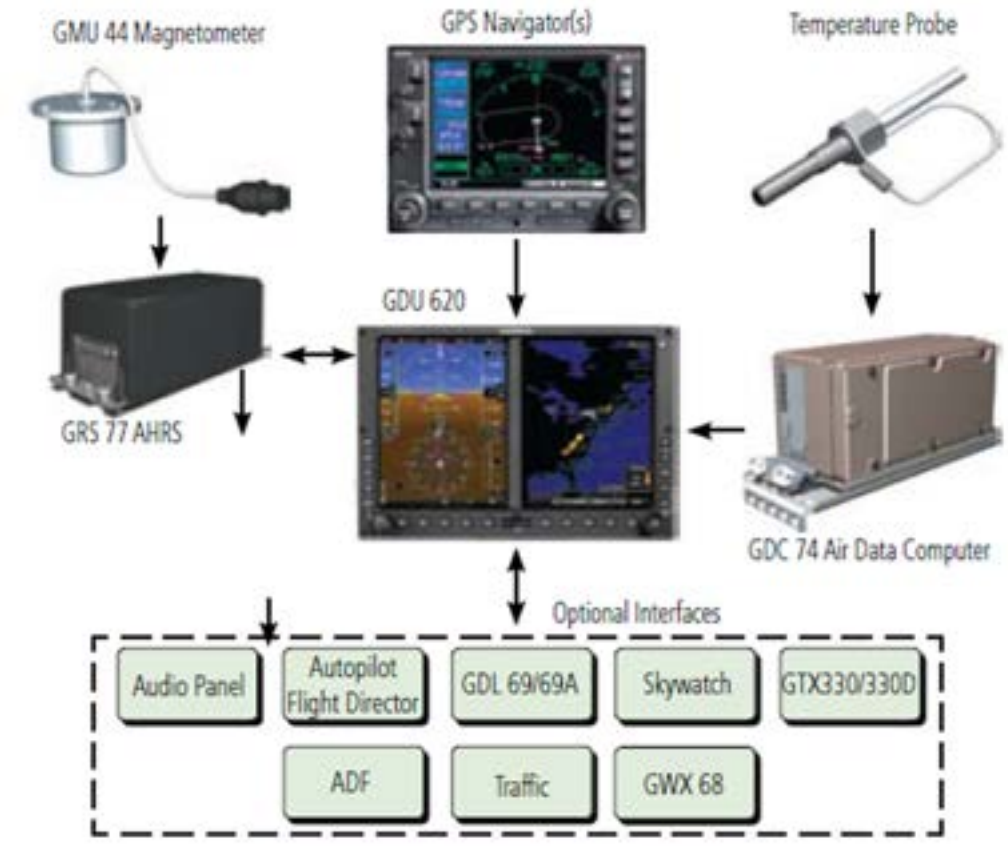
WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

A. Description

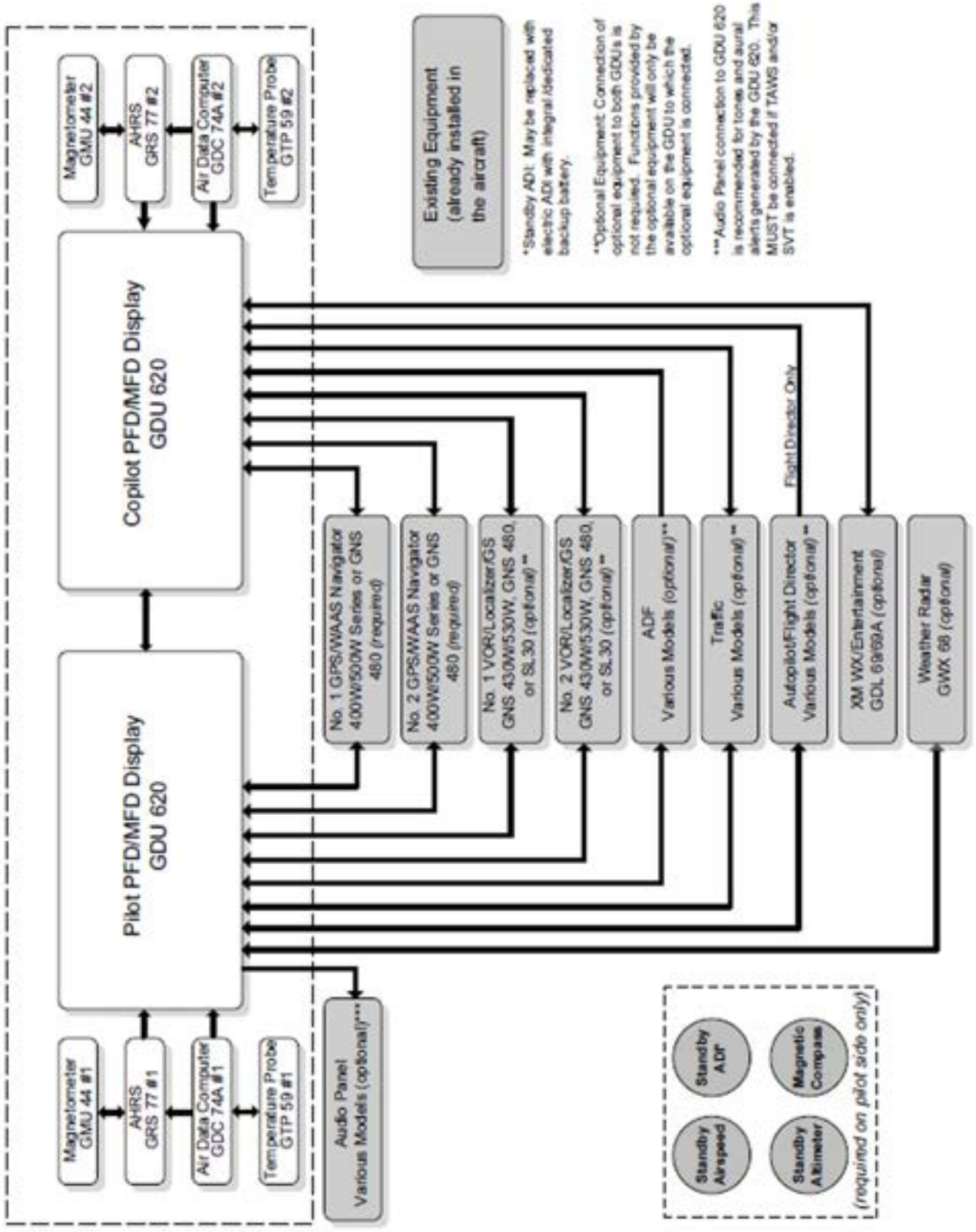
The Garmin G600 Flight Display System was installed as an option in S/N's 3449410, 3449415–3449419; and as standard equipment in 3449421–3449458, and 3449460–3449466. See "Figure 1".

The Garmin G600 PFD/MFD System consists of dual Garmin Display Units (GDU 620) which each house a Primary Flight Display (PFD) and Multi-Function Display (MFD). The system also includes a GRS 77 AHRS (Attitude and Heading Reference System), a GDC 74A ADC (Air Data Computer), a GMU 44 magnetometer, and GTP 59 outside air temperature probe for each GDU 620. Functions provided by the PFD system include display of attitude, heading, air data, and navigation information derived by GPS. Functions provided by the MFD system include display of mapping, terrain, and flight plan information. The G600 interfaces with other installed systems in the aircraft, including Garmin GNS series GPS/WAAS navigators, Garmin GDL 69 data link radios, the S-TEC System 55X Autopilot and various audio panels, traffic systems and ADF navigators.

See the latest revision of the Garmin G600 Pilots Guide for control and operation (190-00601-02, Rev. C or later).



Garmin G600 Displays
Figure 1



Garmin G600 System Communications
Figure 2

(1) Flight Instrumentation

See “Figure 2” on page 34232.

The GRS 77 AHRSs, GDC 74A Air Data Computers, GMU 44 Magnetometers, and GTP 59 Temperature Probes are responsible for providing the G600 system with aircraft attitude, heading, altitude, airspeed, vertical speed, and outside air temperature information, all displayed on the PFD/MFD.

The GRS 77 receives airspeed data from the GDC, and magnetic heading from the GMU. Using these external sources, combined with internal sensor data, the GRS accurately calculates aircraft attitude and heading. Primary data outputs from the GRS and GDC are sent directly to the PFD/MFD via ARINC 429.

(2) Navigation Systems

See “Figure 2” on page 34232.

The 400W Series GNS unit is a panel-mount WAAS/GPS navigator with a color moving map. Position and flight plan data are displayed on the GDU 620 MFD via RS-232 and ARINC 429 interfaces. GPS position information is also forwarded to the GRS 77 AHRS in order to ensure normal AHRS operation. The GNS 430W also provides LOC/GS information for display on the GDU 620 HSI via an ARINC 429 interface. COM and NAV audio is sent via digital audio to the GMA 340 Audio Panel.

(3) Optional Systems

See “Figure 2” on page 34232.

The GDU 620 can receive traffic information from various traffic sensors, including the Garmin GTX 330. Traffic information is received over an ARINC 429 interface, and optional control is provided using discrete(s). All aural traffic alerts are still generated by the traffic system.

The GDU 620 can receive data link information from the GDL 69/69A, which is an XM Satellite Radio data link receiver that receives broadcast weather data. The GDL 69A is the same as the GDL 69 with the addition of XM Satellite Radio audio entertainment. Weather data and control of audio channel and volume is displayed on the GDU 620 MFD, via a High-Speed Data Bus (HSDB) Ethernet connection. The GDL 69A is also interfaced to an audio panel for distribution of the audio signal.

The GDU 620 can receive bearing information from an ADF receiver. The ADF bearing information can be displayed on the GDU 620 HSI on a pilot-selectable bearing pointer.

(4) Component Details

See “Components” on page 342313, for detailed descriptions of the individual G600 system components/Line Replaceable Units (LRU’s).

B. Power Distribution

Distribution of power to the G600 system occurs on three aircraft electrical busses (see “Figure 3” on page 34234).

(1) Main Bus

The Main bus current from the battery is controlled by the Master switch on the overhead panel. Alternator power to the main bus is introduced by Alternator switches on the overhead panel. Three in-line diodes provide isolation in the event of a ground fault in one of the feeder lines. The three feeders are protected by two 90 amp and one 60 amp circuit breakers.

(2) Avionics Bus

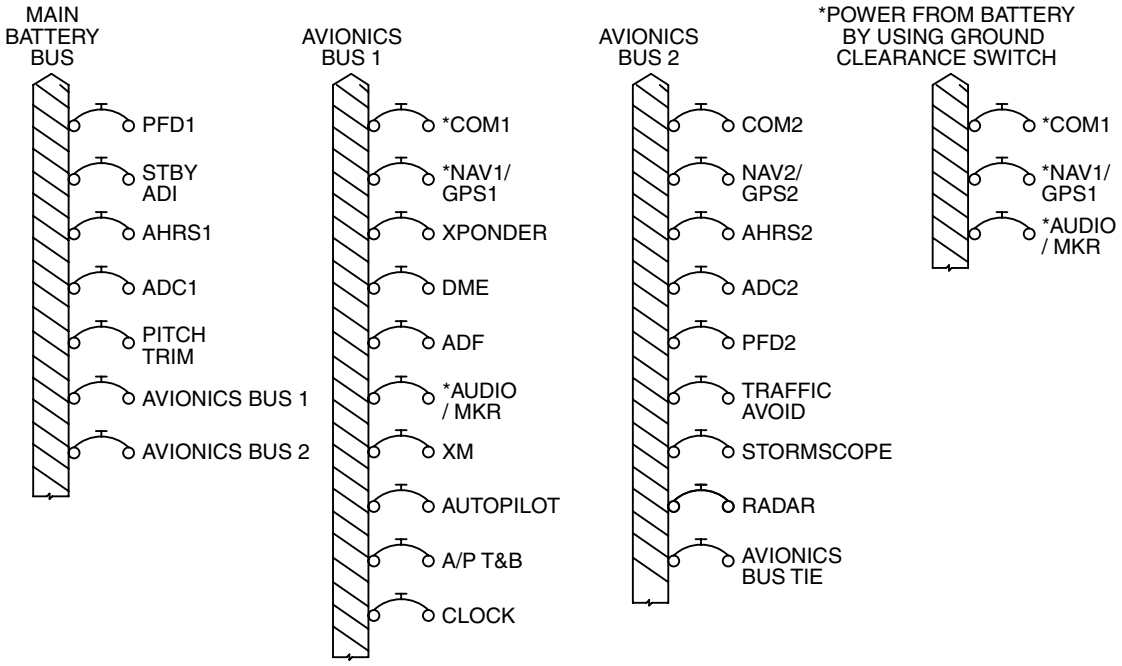
The two avionics busses are fed through independent contactors. The feeders to the contactors are protected by 25 ampere circuit breakers. When the Radio Master switch on the instrument panel is activated, both avionics contactors de-energize allowing current to flow to both avionics busses. Should the need arise, either avionics bus can be isolated by pulling the avionics bus BUS TIE circuit breaker and the appropriate avionics bus circuit breaker.

(3) Ground Clearance

The COM1 radio, NAV1 radio and Audio panel can be utilized by activating the Ground Clearance switch on the instrument panel. This permits use of this equipment without switching ON the Master switch

C. G600 System Controls and Operation

The G600 system controls are located on the PFD and MFD bezels. They are comprised of a PFD knob, MFD dual concentric knobs, bezel keys and soft keys. The soft keys are located along the bottoms of the displays below the soft key labels. The soft key labels shown depend on the soft key level or page being displayed. The soft keys can be used to select the appropriate soft key function. When a soft key is selected, its color changes to black text on gray background and remains this way until it is turned off, at which time it reverts to white text on black background. When a soft key function is disabled, the soft key label is subdued (dimmed). Soft keys revert to the previous level after 45 seconds of inactivity.



Component Power Sources
Figure 3

D. Troubleshooting

NOTE: If any connection in the pitot / static system is opened for maintenance, the entire system must be rechecked per Pitot / Static System “Test” on page 34108.

Troubleshoot the G600 system by first identifying, then isolating the specific failure to the responsible LRU. There are several indications that the G600 presents to the pilot or technician, showing overall system condition. A course of action should be determined based on the information presented on the display.

The 4th AUX group page on the MFD (see “Figure 4”) provides LRU health status by means of a green check or a red ‘X’. Also, LRU software versions are shown along with other database versions and dates. If a red ‘X’ is shown for an LRU, see the appropriate LRU troubleshooting chart for guidance.

Typically, the G600 Alerting System provides alerts/annunciations in conjunction with the information presented at the 4th AUX page.

CAUTION: THE RETURN-TO-SERVICE PARAGRAPH UNDER THE INDIVIDUAL LRU “COMPONENTS”, PROVIDES DETAILED INSTRUCTIONS ON EQUIPMENT CONFIGURATION AND RETURN-TO-SERVICE TESTING. ANYTIME A G600 COMPONENT OR LRU IS REMOVED, SWAPPED, OR REPLACED, THE TECHNICIAN MUST FOLLOW THE PROCEDURES GIVEN IN THE RETURN-TO-SERVICE PARAGRAPH TO ENSURE PROPER OPERATION OF THE SYSTEM.

Troubleshooting information is provided in “Chart 5” on page 34236 and “Chart 6” on page 34238. See also “LRU Test Procedures” on page 342352 and “Post-Installation Procedures” on page 342361.



System Status Page (MFD - AUX Group)
Figure 4

CHART 5 (Sheet 1 of 2)
TROUBLESHOOTING GDU 620 SYSTEMS

Problem	Cause	Solution
Unit does not power up – blank screen.	Improper wiring; circuit breaker open	Ensure power is properly wired to the GDU 620 and the circuit breaker is closed.
	Unit intensity turned down.	Ensure that unit is not in manual intensity control mode with the intensity turned down.
All expected configuration pages are not displayed.	An Installer Unlock Card is not inserted into the GDU 620.	Insert the Installer Unlock Card P/N 010-00769-60 into the bottom slot of the GDU 620 and cycle power.
The GDC OAT probe type shows up as UNKNOWN	The RS-232 connection to the GDC 74A is not working.	Ensure that the GDC 74A RS-232 connection to the GDU is properly wired, and ensure that the GDC 74A circuit breaker is closed.
When loading software, the LRU software is not being displayed on the SOFTWARE UPLOAD page.	The software loader card is installed in the bottom slot of the GDU 620.	Insert the loader card in the top slot and cycle power to the GDU.
	The software loader card contains no information.	Repeat the process for making the software loader card.
Configuration errors are displayed on power-up, before the GDU enters normal mode.	The configuration module has not been updated.	Update the configuration module – refer to Updating the Configuration Module.
Vertical GPS deviation is not displayed on the GDU 620.	For 430W Series units, the ARINC 429 vertical deviation labels are not being transmitted.	Enable Labels on the 430W unit ARINC 429 configuration page.
Unable to control the GPS course when in OBS mode.	The GPS navigator is not correctly configures as LNAV1/2 or SYS1/2.	Configure the ARINC 429 inputs/outputs for LNAV1 (SYS1) or LNAV2 (SYS2) based upon whether the navigator is GPS1 or GPS2.
Data is not being received from an ARINC 429 device. (valid data is being received on the 429 input port as shown on the GDU 620 PORT MONITORING page)	ARINC 429 bus hi and low are swapped.	Verify wiring.
	Wrong device is connected to port on GDU 620.	Use correct ports (refer to Chapter 91 - Wiring Diagrams for interconnect details).

CHART 5 (Sheet 2 of 2)
TROUBLESHOOTING GDU 620 SYSTEMS

Problem	Cause	Solution
Data is not being received from an ARINC 429 device. (no data is being received on the 429 input port as shown on the GDU 620 PORT MONITORING page)	On the transmitting LRU, the ARINC 429 transmitter speed is not set correctly. Wiring is not correct.	Set the ARINC 429 transmitter speed to correct speed. Check for continuity/shorts and correct as required.
Attitude and heading on GDU 620 red 'X' / GRS 77 resets during air data ground testing.	Attitude and heading errors/resets are possible if the air data tests are conducted indoors without a good GPS signal. With marginal or no GPS signals present, sudden changes in airspeed caused by using a pressure tester may result in attitude and heading errors and possibly cause the GRS 77 to reset. This occurs because the artificial changes in airspeed cause disagreement with the other sensor measurements internal to the GRS 77. This sensor disagreement will not occur in the normal conditions of flight.	This is expected behavior and no troubleshooting is required if this occurs. To reduce the chances of inducing attitude and heading errors/resets while conducting the air data tests, ensure that the G600 is receiving good GPS signals.
Heading red 'X' during air data ground testing	Invalidation of heading is possible if the air data tests are conducted indoors, due to typical magnetic anomalies, even with a good GPS signal.	This is expected behavior and no troubleshooting is required if this occurs.

CHART 6 (Sheet 1 of 4)
TROUBLESHOOTING GDU 620 ALERTS

Alert Text	Cause	Solution
AHRS1 GPS	AHRS is not receiving any GPS information.	Verify navigator is on, has GPS signal, not in self-test.
AHRS1/2 GPS	AHRS 1 operating exclusively in no-GPS mode.	Ensure that at least one GPS has acquired a valid position.
	AHRS is not receiving any GPS information.	If GDU 620 does not have a valid position, verify wiring between GDU and GPS receiver, and configuration of GDU 620 and GPS receiver.
		If GDU has a valid GPS position, verify wiring between GDU and GRS. Also verify time mark wiring.
AHRS1/2 GPS	AHRS1 is not receiving GPS information from GPS2.	Verify GPS2 unit operation.
AHRS1/2 GPS	AHRS1 is using the backup GPS information (GPS2)	Primary GPS nav data not forwarded to AHRS.
AHRS1/2 SRVC	AHRS magnetic field model should be updated. Appears on ground only.	Update GRS 77 IGRF model (current model is with aviation database).
AHRS1/2 TAS	AHRS is not receiving true airspeed from ADC.	GDC not powered up. Close ADC C/B.
		GDC not receiving input from GTP 59 OAT probe. Verify wiring is correct.
		ARINC 429 connection from GDC 74A to GRS 77 is not working. Verify wiring is correct.
CAL LOST	Registry reports that it has lost calibration data.	Contact Garmin Technical Support.
CNFG MISMATCH	GDU 1-2 airframe configurations disagree. GDU in normal mode has received updated crossfill tags for configuration registry entries.	Restart both GDUs in configuration mode to automatically synchronize settings.
CNFG MODULE	The configuration module is inoperative.	Verify wiring to configuration module Replace configuration module

CHART 6 (Sheet 2 of 4)
TROUBLESHOOTING GDU 620 ALERTS

Alert Text	Cause	Solution
DATA LOST	Pilot stored data was lost.	Recheck data and settings.
DIAG MODE	System is in Diagnostic Mode.	
FAN 1/2 FAIL	Fan 1/2 has reported 0 RPM when it was powered with a PWM duty cycle higher than or equal to 10%	
GDL69 CONFIG	The GDL 69 configuration information stored in the GDL 69 and the GDU 620 configuration module do not match.	With the GDU 620 in configuration mode, go to the GDL 69 page in the GDL page group. Verify that the SET and ACTIVE configuration settings are the same. If not, use the SET>ACTV soft key to copy the configuration settings from the GDU 620 into the GDL 69.
	The GDL 69 configuration was updated using another LRU (e.g. the GNS 400W).	Update the GDL 69 configuration using the GDU 620.
GDU CONFIG	This error appears whenever the GDU is replaced with a GDU that was configured for a different installation.	Cycle power to the GDU. This error automatically clears on the second power up with a different configuration module.
	Error in the configuration of the GDU 620.	
GDU (1/2) COOLING	Specific GDU has poor cooling, and power usage is being reduced.	Ensure fans on indicated GDU are functioning. Ensure fans on indicated GDU are not obstructed
GDU (1/2) DB ERR	Error in specific database, where GDU (1/2) DB denotes specific database.	
GDU (1/2) VOLTAGE	GDU supply voltage is below 12 VDC.	Increase the voltage above 12VDC.
GEO LIMITS	Location is too far north/south for GRS 77 magnetic compass.	
GPS(1/2) FAIL	No GPS1 or GPS2 data is available.	Ensure GPS (1/2) is turned on. Verify RS-232 wiring from the GPS to the GDU 620.

CHART 6 (Sheet 3 of 4)
TROUBLESHOOTING GDU 620 ALERTS

Alert Text	Cause	Solution
GPS(1/2) PPS Failure	This alert will be set if the PPS signal has not been received in more than 5 sec. If the unit is configured for dual GPSs then the side will be specified in the error.	Ensure GPS (1/2) is turned on. Verify 1PPS wiring from the GPS to the GDU 620.
GPS2 FPL USED	The GPS1 has failed and GPS2 is configured and operating.	
GWX CONFIG	The GWX 68 configuration information stored in the GWX 68 and the GDU 620 configuration module do not match.	With the GDU 620 in configuration mode, go to the GWX WEATHER RADAR page in the WXR page group. Press the SET>ACTV soft key to copy the GDU 620 configuration into the GWX 68.
HDG FAULT	AHRS magnetometer fault has occurred.	GRS 77 not receiving information from GMU 44. Verify wiring to GMU 44.
HDG LOST	Heading from the GRS77/ GMU 44 is not valid.	Caused by a local magnetic anomaly. No action required.
<LRU> SERVICE	Specific LRU should be serviced, where <LRU> denotes specific LRU.	Return indicated LRU to Garmin for service.
MANIFEST	GDU has received product data for an LRU that should have a manifest entry, but is not in the manifest.	Ensure the manifest is properly configured. Refer to Manifest Configuration for additional information.
	The LRU software P/N and version number in the manifest does not match the values being reported by that LRU.	Update the LRU software to match the manifest (refer to Software Loading) Update the manifest to match the LRU software (refer to Manifest Configuration)
NAV1 FAIL	No navigation receiver 1 data.	Switch to alternate GPS
NAV2 FAIL	No navigation receiver 2 data.	Switch to alternate GPS
SIMULATOR	The simulator mode is active.	Ensure P6202-36 is not grounded.

CHART 6 (Sheet 4 of 4)
TROUBLESHOOTING GDU 620 ALERTS

Alert Text	Cause	Solution
SVT DISABLED	Out of available terrain region. Location is beyond region covered by terrain database.	
	Terrain DB resolution too low. A 30 arc-second terrain database is being used.	Update the Supplemental Data card with the 9 arc-second terrain database.
SW MISMATCH	GDU software version strings do not match. Xtalk is off.	
TRAFFIC FAIL	Traffic data will no longer be displayed	The GDU 620 is not receiving traffic information from the traffic sensor. Verify wiring between GDU 620 and traffic sensor.
	The traffic information system has failed.	The GDU 620 is receiving information from the traffic sensor, but the information is indicating that the traffic sensor has failed. Troubleshoot traffic system.
TRK LOST	GPS1 TRK lost. HSI defaulted to GPS2 TRK.	
TRK TRAFFIC	Heading Lost. Traffic is now based on track.	See HDG errors.
WX ALERT	Possible severe weather ahead	This is a normal system alert. No installation action is required.
WX RADAR FAIL	Weather radar has failed.	Ensure that the weather radar is powered up. Verify wiring between GDU and weather radar. If the GWX 68 weather radar is routed through the GDL 69, ensure that the GDL 69 Ethernet ports are enabled.

E. Recommended Tools

The following tools (or equivalent) are recommended to perform various maintenance tasks on the G600 system.

(1) Calibrated voltmeter capable of measuring 0-32 Volts DC.
(2) Ground Power Unit (Capable of supplying 28 Vdc).
(3) #2 Phillips Screwdriver.
(4) 3/32nd inch Hex Tool.
(5) Standard sockets & wrenches (3/8”, 9/16”, and 13/16”).
(6) Calibrated torque wrench capable of measuring 0 – 70 in/lbs.
(7) Calibrated digital Level with 0.25 degrees of accuracy capability.
(8) Calibrated VHF NAV/COM/ILS ramp tester.
(9) Calibrated transponder ramp tester including Mode S capability for Mode S transponder equipped aircraft.
(10) Calibrated digital thermometer suitable for measuring ambient temperature.
(11) Air Data Test Set (ADTS) capable of simulating altitude up to the aircraft’s service ceiling.
(12) Headset/Microphone.
(13) Outdoor line-of-site to GPS satellite signals or GPS indoor repeater.
(14) Laptop with RS232 emulation software.
(15) WX-PA portable analyzer kit (If Stormscope is installed).
(16) Laser Square with a line accuracy of ± 3/32nd inches end-to-end at 15 feet perpendicular distance (or better) is optional, but recommended.
(17) Protractor tool.

F. Inspections

(1) Annual / 100 Hour Inspection

Each 12 months or 100 hours time-in-service, whichever comes first:

(a) Inspect all units for security of attachment.

(b) Inspect all knobs and buttons for legibility.

(c) Visually inspect each unit's wiring for chafing or wear at each termination.

(2) 1000 Hour Inspection

Each 1000 hours conduct the G600 Equipment Electrical Bonding Check in 51-80-00.

(3) 5 Year Update

Update GRS 77 (AHRS) Earth magnetic field model which is a part of Aviation Database. If the magnetic model is not up to date, the unit will issue an alert upon startup indicating the model has expired.

G. Components

(1) General

Maintenance of the G600 system components is "On Condition" only. The following provides basic information regarding unit removal, installation, and troubleshooting. Basic guidance on loading and configuring software is provided under Post Installation Set-up, below. When removing and/or replacing any G600 component, always ensure that aircraft power is off. Unplug any auxiliary power supplies.

Before removing any G600 LRU, it is required that the technician verify the LRU software serial numbers.

To check an LRU software serial number and/or version:

(a) Turn the large MFD knob to select the AUX Page Group.

(b) Turn the small MFD knob to select the System Status Page.

(c) Press the LRU soft key to highlight the first item in the LRU Info window.

(d) Turn the small MFD knob to scroll thru items in the LRU Info window in case more items are available than are displayed. If more items are available than can be displayed in the window, a scroll bar will show on the right side of the window.

(e) The System Status page of Aux mode shows the status of LRUs and the date of databases.

(f) The software serial number and version is also displayed in the LRU info window. Compare this to the latest configuration shown in the appropriate Required Equipment Lists (REL), "Chart 12" on page 342376.

NOTE: If a faulty LRU is not reporting its software version and serial number, check aircraft maintenance logs for last software version loaded and verify against the latest configuration.

(2) GDU 620 PFD/MFD

See “Figure 5”.

CAUTION: THE GDU 620 PFD AND MFD DISPLAYS USE A LENS COATED WITH A SPECIAL ANTI-REFLECTIVE COATING THAT IS VERY SENSITIVE TO SKIN OILS, WAXES AND ABRASIVE CLEANERS. CLEANERS CONTAINING AMMONIA WILL HARM THE ANTI-REFLECTIVE COATING. IT IS VERY IMPORTANT TO CLEAN THE LENS USING A CLEAN, LINT-FREE CLOTH AND AN EYEGLASS LENS CLEANER THAT IS SPECIFIED AS SAFE FOR ANTI-REFLECTIVE COATINGS.

(a) Description

The GDU 620 has dual VGA 6.5 inch LCD displays, each having 640 x 480 resolution. The left side of the GDU is a PFD and the right side is the MFD. The MFD shows a moving map, flight plan, weather, and more. The PFD shows primary flight information, in place of traditional pitot-static and gyroscopic systems and also provides an HSI for navigation.

Two Garmin GDU 620 displays are installed in the instrument panel. The GDU 620 units are configured as PFD/MFD1 and PFD/MFD2. The GDU 620 displays are located on either side of the avionics stack, with the stand-by instruments located to the left of the Pilot’s PFD/MFD. The GMA 340 Audio Panel is located above the two GNS 430W units.

(b) Troubleshooting

See “Chart 5” on page 34236 and “Chart 6” on page 34238.



Garmin G600 Displays
Figure 5

(c) Removal

NOTE: Whenever removing or replacing units, remove power from the LRU by removing aircraft power or opening the LRU circuit breaker.

1) Remove the six mounting screws from the bezel of the GDU 620.
2) Pull the GDU 620 far enough out from the instrument panel to access the three rear connectors
3) Disconnect the rear connectors.
4) Remove the GDU 620.

(d) Installation

1) Visually inspect the connectors to ensure that there are no bent or damaged pins. Repair any damage.
2) Connect the rear connectors, ensuring that each slide-lock is secured on both sides.
3) Set the GDU 620 into place.
4) Install the six mounting screws into the bezel of the GDU 620.

NOTE: The installation configuration settings are stored in the configuration module and will be retained when the GDU 620 is replaced with a new unit. User settings, such as map orientation preferences, are stored internally and will be lost when the GDU 620 is replaced with a new unit.

(e) Return to Service

1) Original Display Reinstalled.

If the removed display(s) are re-installed in their original positions, no software loading or configuration is required. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) New Repair or Exchange Display(s) Installed.

If a new, repaired or exchange GDU 620 is installed, the correct software must be loaded to the unit and properly configured. See “GDU 620 Software Load” on page 342335. This is followed by “MANIFEST Configuration” on page 342340 and “Configuration Module Update” on page 342345.

3) Configuration Module (1P6202 or 2P6202 harness connector) is Replaced.

If the GDU 620 Configuration Module is replaced, the GDU 620 will update the configuration module from its internally-stored settings when the UPDT CFG soft key is pressed. Verify configuration is correct. If the GDU 620 is replaced at the same time as the Configuration Module, then see New Repair or Exchange Display(s) Installed, above.

4) Test

After removing and reinstalling the GDU 620 per the instructions above, the following return-to-service check should be performed.

a) Power up the GDU 620 in configuration mode. Verify that the configuration settings match those recorded in the checkout log.
b) Power up the GDU 620 in normal mode. Verify that there are no red-Xs and that no alerts are present. If red Xs or alerts are present, troubleshoot using “Chart 5” on page 34236 and “Chart 6” on page 34238.

(3) GDC 74A Air Data Computer

See “Figure 6”.

(a) Description

The Garmin GDC 74A computers compile information from the pitot/static system and the GTP 59 outside air temperature (OAT) probes as digital air data computations to the G600 system. The GDC 74A provides pressure altitude, airspeed, vertical speed, and OAT information to the GDU 620, and GRS 77 using an ARINC 429 digital interface. The units are mounted behind the co-pilot GDU 620. Power is provided to #1 GDC 74A unit (ADC 1) from the main bus. The #2 GDC 74A (ADC 2) is powered by the #2 avionics bus.



Garmin GDC 74A Air Data Computer
Figure 6

NOTE: If any connection in the pitot / static system is opened for maintenance, the entire system must be rechecked per Pitot / Static System “Test” on page 34108.

(b) Troubleshooting

See “Chart 5” on page 34236 and “Chart 6” on page 34238.

(c) Removal

1) Remove the co-pilot PFD/MFD as described above.
2) Disconnect the pitot/static plumbing from the rear of the unit. Disconnect the single connector.
3) Loosen each thumbscrew on the hold-down clamp and remove the clamp.
4) Carefully remove the unit from its mount.

(d) Installation

1) Place the unit in the mounting tray.
2) Position the locking clamp and fasten using the thumbscrews.
3) Connect the pitot/static plumbing.
4) Inspect the connector and pins for damage. Repair any damage. Connect the connector to the unit.
5) Reinstall the co-pilot PFD/MFD as described above.

NOTE: If any connection in the pitot / static system is opened for maintenance, the entire system must be rechecked per Pitot / Static System “Test” on page 34108.

(e) Return to Service

1) Original GDC 74A is Reinstalled

No software or configuration loading is required if the removed GDC 74A is re-installed. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) New, Repaired or Exchange Unit GDC 74A is Installed

If a new, repaired or exchange unit GDC 74A is installed, the correct software must be loaded to the unit.

3) GDC 74A Configuration Module is Replaced

The GDC 74A must be configured if the GDC 74A configuration module has been replaced. Perform Config Module update.

4) Test

After removing and reinstalling the GDC 74A per the instructions above, the following return-to-service check should be performed.

a) Power up the G600 system with the GDU 620 in normal mode.

b) Verify that the GDU displays valid air data within approximately one minute.

c) Verify that no unexpected alerts are present. If alerts are present, troubleshoot using “Chart 5” on page 34236 and “Chart 6” on page 34238.

d) Perform a leak check of the pitot-static system and observe the airspeed, altitude, and vertical speed for proper operation.

(4) GTP 59 OAT Probe

See “Figure 7”.

(a) Description

The Garmin GTP 59 OAT Probe provides the GDC 74A with air temperature data. The OAT probe is mounted to the underside of each wing, outboard of W.S. 164.72.

(b) Troubleshooting

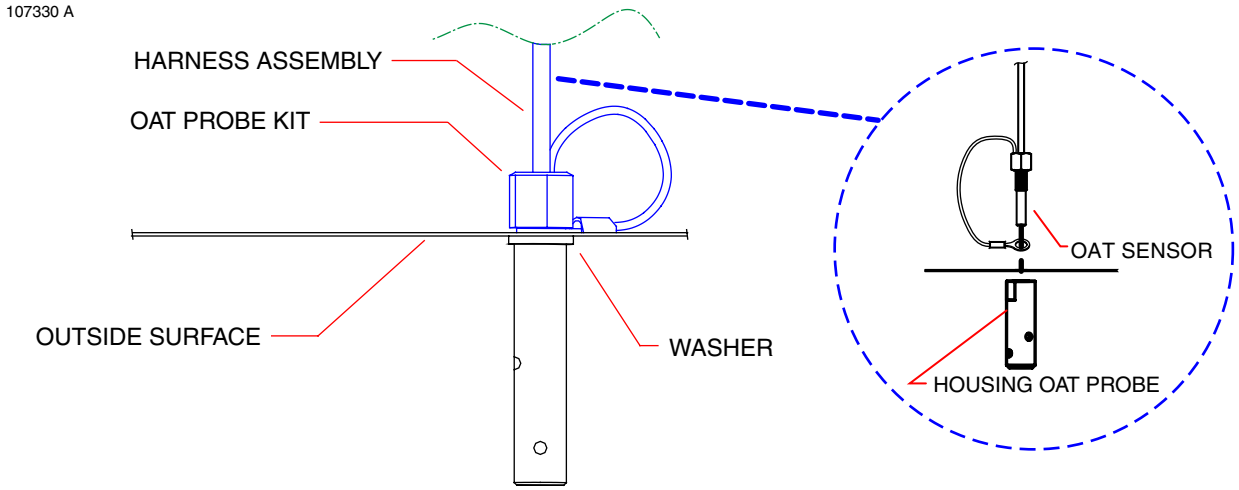
See “Chart 5” on page 34236 and “Chart 6” on page 34238.

(c) Removal

1) Remove access cover adjacent to the OAT probe.
2) Disconnect the connector.
3) Use an open-end wrench to hold the probe in place on the inside of the wing. On the outside of the wing, loosen the GTP 59 Housing and remove the GTP 59.

(d) Installation

1) Place the GTP 59 grounding strap O-ring connector over the OAT probe end and insert the probe through the wing from the inside. Secure with washer and housing on the outside of the wing. Use an open-end wrench to hold the GTP 59 on the inside and torque the housing on the outside to 100 ± 20 in·lbs.
2) Reconnect the connector to the airplane wiring harness.
3) Reinstall access cover adjacent to the OAT probe.
4) Verify ambient temperature OAT display on each PFD.



GTP 59 OAT Probe
Figure 7

(5) GRS 77 Attitude and Heading Reference System

See “Figure 8”.



GRS 77 Attitude and Heading Reference System
Figure 8

(a) Description

The Garmin GRS 77 AHRS units provide attitude and flight characteristics information to the G600 system. The units, mounted below the aft passenger seats, contain advanced tilt sensors, accelerometers, and rate sensors. The unit interfaces with the GDC 74A and GMU 44 Magnetometer and utilizes GPS signals from the GNS 430Ws. Actual attitude and heading information is sent using ARINC 429 digital interface to both GDU 620s. Power is provided to #1 GRS 77 unit (AHRS 1) from the main bus. The #2 GDC 74A (AHRS 2) is powered by the #2 avionics bus. The GRS 77 interfaces with and provides power to the GMU 44 Magnetometer. The GRS 77 supplies attitude and heading information directly to the GDU 620s.

NOTE: The GRS 77 uses an Earth magnetic field model which is updated every five years. The next update will be available from Garmin in: 2020 and every five years thereafter, as long as the GRS 77 remains a Garmin-supported product. The G600 system alerts the operator that the magnetic field database is out of date by issuing the message “AHRS SERVICE – AHRS Magnetic-field model needs update”. Garmin will distribute update instructions when updates are available.

(b) Troubleshooting

See “Chart 5” on page 34236 and “Chart 6” on page 34238.

(c) Removal

1) Remove aft passenger seats (See 25-10-00).
2) Disconnect the GRS 77 AHRS connector.
3) Remove the four Phillips thumbscrews with a screwdriver and set them aside.
4) Gently lift the GRS 77 from the mounting plate. (Avoid removing the mounting plate. If the mounting plate is removed, the GRS 77 must be re-calibrated upon installation.)

(d) Installation

1) Level the airplane per 8-20-00.
2) Verify GRS 77 bracket assembly is level within 3.0 degrees.
3) Place the GRS 77 on the mounting plate, ensuring the general orientation is correct.
4) Align forward direction of the GRS 77 in heading to within 1.0 degree of the aircraft heading.
5) Fasten the unit to the plate using the Phillips thumbscrews. Recommended torque is 22-25 inch pounds.
6) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage. Connect the connector to the GRS 77, ensuring that each slidelock is secured on both sides.
7) Reinstall aft passenger seats (See 25-10-00).

(e) Return to Service

1) Original GRS 77 is Reinstalled

If the original GRS 77 was reinstalled, then no software loading is required. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process. See “Chart 8” on page 342353 to determine whether recalibration is required.

2) New Repaired or Exchange GRS 77 is Installed

If the repaired or exchange GRS 77 was replaced with a new unit, then software must be loaded. See “Chart 8” on page 342353 to determine whether recalibration is required.

3) GRS 77 Configuration Module is Replaced

If the GRS 77 configuration module was replaced, no software loading is required. Continue to the “GRS/GMU Calibration Procedures” on page 342352.

4) Test

After removing and reinstalling the GRS 77 per the instructions above, the following return-to-service check should be performed.

a) Power up the G600 system with the GDU 620 in normal mode.

b) Verify that the GDU displays valid heading and attitude within approximately one minute.

NOTE: Heading can remain invalid if the magnetometer is near a large metal structure such as a hangar wall or if the magnetometer is close to a large ground power cart.

c) Verify that no unexpected alerts are present. If alerts are present, troubleshoot using “Chart 5” on page 34236 and “Chart 6” on page 34238.

(6) GMU 44 Magnetometer

See “Figure 9”.



GMU 44 Magnetometer
Figure 9

(a) Description

The GMU 44 provides horizontal and vertical magnetic field information to the GRS 77 AHRS. This allows heading to be calculated and provides assistance during AHRS alignment. The GMU 44 is mounted on the access cover under each wing between W.S. 193.18 and W.S. 208.33.

(b) Troubleshooting

See “Chart 5” on page 34236 and “Chart 6” on page 34238.

(c) Removal

1) Remove the access cover to expose the magnetometer.
2) Disconnect the GMU 44 harness from the wing harness.
3) Unscrew and remove the three brass screws securing the magnetometer to its mounting rack.
4) Carefully lift the GMU 44 from the rack.

(d) Installation

CAUTION: THE MAGNETOMETERS ARE SECURED TO THE MOUNTING BRACKET WITH BRASS SCREWS. ENSURE ONLY BRASS SCREWS ARE USED WHEN REINSTALLING.

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
2) Lower the GMU 44 into the rack and secure the plate with the three brass Phillips screws. Torque 12-15 inch pounds.
3) Connect GMU 44 harness to the wing harness.
4) Reinstall the access cover.

NOTE: Ensure correct hardware is used when reinstalling access covers attached to magnetometers.

(e) Return to Service

1) Original GMU 44 is Reinstalled

If the original GMU 44 was reinstalled, then no software loading is required. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.air as their software and configuration files are deleted during the repair testing process. See “Chart 8” on page 342353 to determine whether recalibration is required.

2) New, Repaired or Exchange GMU 44 is Installed

If the GMU 44 was replaced with a new, repaired or exchange unit, then software must be loaded. Continue to “GMU 44 Software Load” on page 342336. Then continue to “GRS/GMU Calibration Procedures” on page 342352.

3) Perform Return to Service “Test” on page 342320 under GRS 77.

(7) GDL 69A Data Link Unit

See “Figure 10”.

(a) Description

The GDL 69A provides XM Radio weather and music entertainment through means of a dedicated satellite data link. The GDL 69/69A is mounted in a mount rack on the upper left side of the fuselage aft of the cabin hatshelf and just forward of F.S. 220.164. Power to the GDL 69A is received from the #1 avionics bus. The GDL 69A sends weather data through the HSDB bus to the GDU 620 #2 (Copilot PFD/MFD). Digital audio is sent directly to the GMA 340 Audio Panel.

(b) Troubleshooting

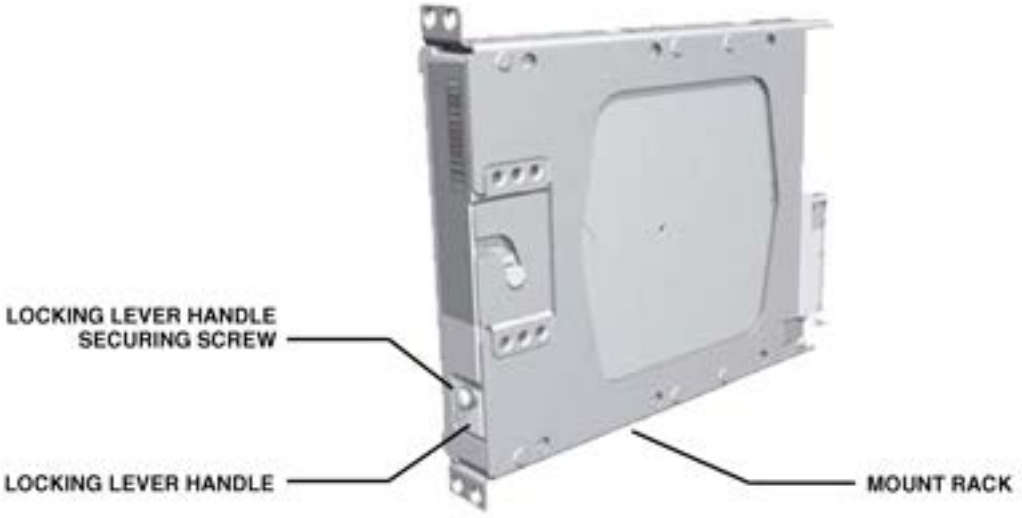
See “Chart 5” on page 34236 and “Chart 6” on page 34238.

(c) Removal

1) Remove the aft cabin hatshelf assembly.
2) Unlock the GDL 69A handle by unscrewing the Phillips screw.
3) Pull the handle upward to unlock the GDL 69A.
4) Gently remove the GDL 69A from its rack.

(d) Installation

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
2) Gently insert the GDL 69A into the rack. The handle should engage the dogleg track.
3) Press down on the handle to lock the unit into place.
4) Lock the handle to the GDL 69A body using the Phillips screw.
5) Reinstall the aft cabin hatshelf assembly.



GDL 69A Data Link Unit
Figure 10

(e) Return to Service

1) Original GDL 69A is Reinstalled

No software or configuration loading is required if the removed GDL 69A is re-installed. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process. Continue to the “GDL 69A Test” on page 342359

2) New Repaired or Exchange GDL 69A is Installed

If a new repaired or exchange GDL 69A is installed, the correct software and configuration files must be loaded to the unit, then the XM Satellite Radio subscription must be reactivated. Then continue to the “GDL 69A Test” on page 342359.

(8) GWX 68 Weather Radar

See “Figure 11”.

(a) Description

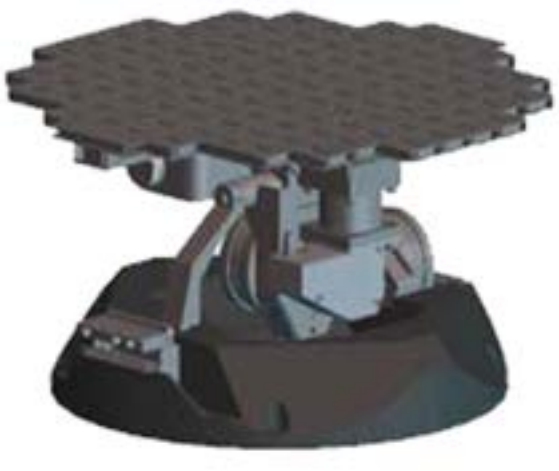
The GWX 68 Airborne Weather Radar provides weather radar data output to the GDU 620. The GWX 68 is mounted in the nose cone assembly, forward of the nose gear doors. Power to the GWX 68 is received from the avionics No. 2 bus. Data received from the GWX 68 is routed to the Pilot PFD/MFD via high-speed data bus (Ethernet).

(b) Troubleshooting

See “Chart 5” on page 34236 and “Chart 6” on page 34238.

(c) Removal

NOTE: Take note of the number, type and location of any shims and washers during removal.



GWX 68 Weather Radar
Figure 11

1) Disconnect backshell assembly (P400) from top of unit.
2) Use a 3/16” hex drive tool to remove each of the two mounting screws, washers and nuts connecting the unit to the upper beam assembly.
3) Use a #3 Phillips bit to remove two mounting screws which connect the upper beam to each side clip assembly. Remove the upper beam assembly.
4) Use a 3/16” hex drive tool to remove each of the two mounting screws connecting the unit to the lower bracket assembly.
5) Remove GWX 68 by rotating unit back over the bracket assembly.

(d) Installation

NOTE: Install shims and washers of same number and type in the same locations noted during removal.

1) Inspect connector for damaged pins.
2) Hold unit flush with the radar mount, ensuring that the connector is at the top.
3) Reattach and tighten each of the two mounting hex head screws to the lower bracket assembly.
4) Reattach and tighten each of the two mounting phillips head screws which connect the upper beam to each side clip assembly.
5) Reattach and tighten each of the two mounting hex head screws, washers and nuts connecting the unit to the upper beam assembly.
6) Connect backshell assembly (P400) to top of unit.

(e) Return to Service

1) Original GWX 68 Reinstalled

No software or configuration loading is required if the removed GWX 68 is re-installed. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process. Continue to the “GWX 68 Post-Installation Test” on page 342366.

2) New Repaired or Exchange GWX 68 Installed

If a new repaired or exchange GWX 68 is installed, the correct software and configuration files must be loaded to the unit. See “GWX 68 Software Load” on page 342337 and then continue to the “GWX 68 Post-Installation Test” on page 342366.

(9) Configuration Module

See “Figure 12”.

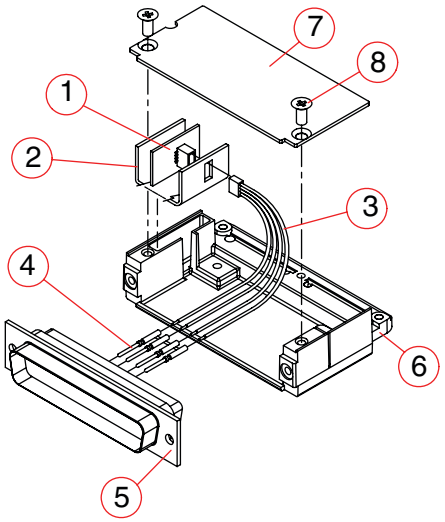
The configuration module is located in the backshell of the connector it is terminated to.

(a) Removal

1) Disconnect connector from LRU.
2) Remove 2 screws (8) from cover (7) and remove cover.
3) Unplug connector from configuration module (1).
4) Remove configuration module.

(b) Installation

1) Inspect connector for damaged pins (4).
2) Place configuration module (1) in position.
3) Insert connector into configuration module (1).
4) Assembly of the connector is the reverse of disassembly.



Configuration Module
Figure 12

(c) Return to Service

1) If a GRS 77 AHRS Configuration Module is replaced:

All three GRS 77/GMU 44 calibration procedures must be performed. See “Chart 8” on page 342353.

2) If a GDC 74A Configuration Module is replaced:

Configuration settings must be reloaded to the GDC 74A. See “GDC Configuration” on page 342344.

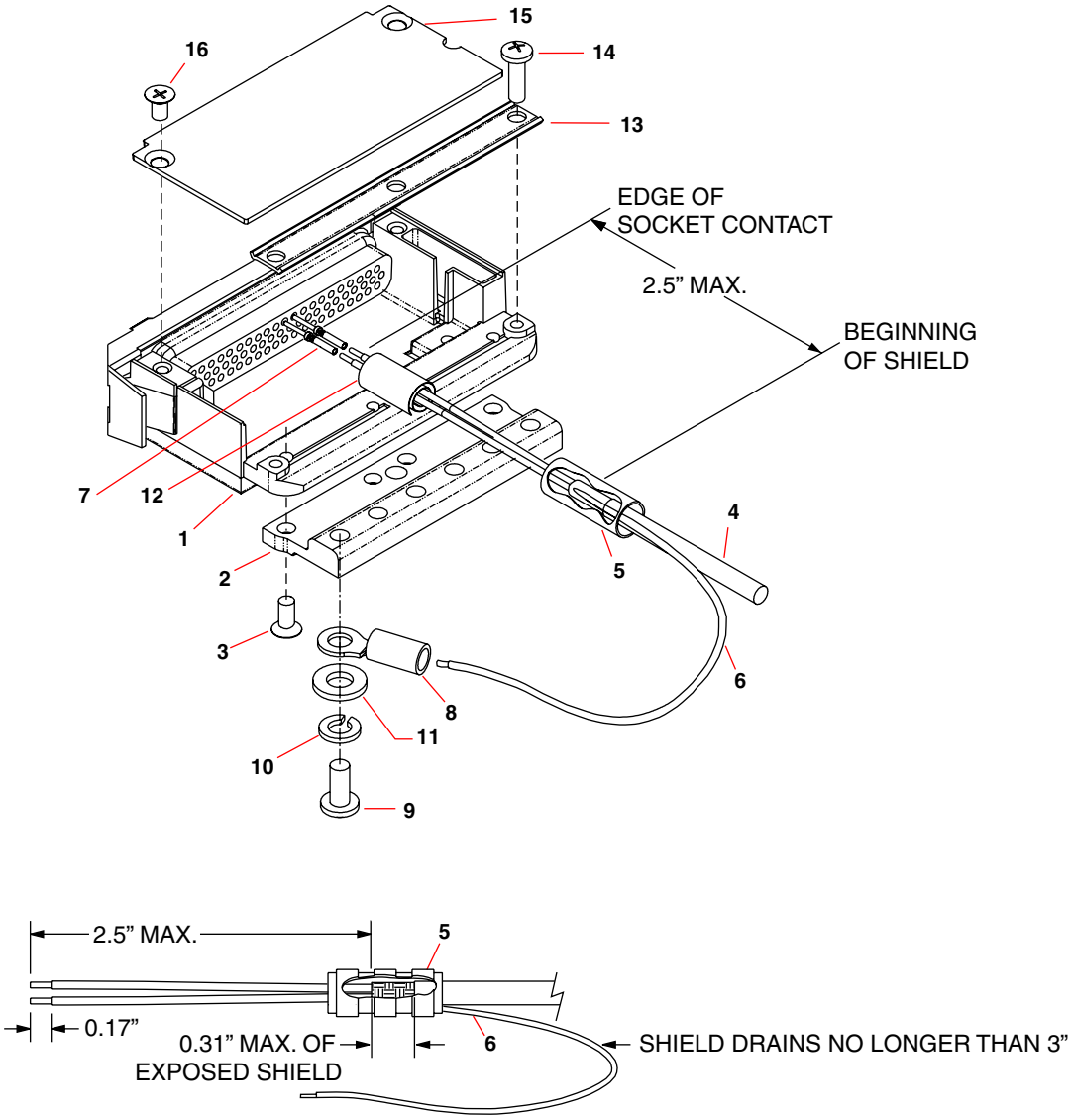
3) If the GDU 620 Configuration Module is replaced:

a) Start the G600 system in configuration mode.
b) Go to the Configuration Upload Page on the PFD.
c) Press the UPDT CFG softkey.

(10) Shield Block Installation

See “Figure 13” on page 342328.

Most G600 connectors employ a shield block grounding system to provide necessary ground reference to shielding and/or transducers.



1. BACKSHELL
2. SHIELD BLOCK
3. SCREW
4. MULTIPLE CONDUCTOR SHIELDED CABLE (2–CONDUCTOR DEMONSTRATED HERE)
5. DRAIN WIRE SHIELD TERMINATION (METHOD OPTIONAL)
6. WIRE, INSULATED
7. PINS
8. RING TERMINAL, #8, INSULATED
9. SCREW
10. SPLIT WASHER
11. FLAT WASHER
12. SILICON FUSION TAPE
13. STRAIN RELIEF
14. SCREW
15. COVER
16. SCREW

Shield Block Installation to Backshell
Figure 13

H. Configuration Mode Overview

(Garmin Doc. No. 190-00601-04, Rev. D.)

CAUTION: THE CONFIGURATION MODE CONTAINS CERTAIN PAGES AND SETTINGS THAT ARE CRITICAL TO AIRCRAFT OPERATION AND SAFETY. THESE PAGES ARE PROTECTED AND CANNOT BE MODIFIED, UNLESS THE TECHNICIAN IS PROPERLY AUTHORIZED AND EQUIPPED. HOWEVER, MOST PROTECTED PAGES ARE VIEWABLE TO ALLOW SYSTEM AWARENESS FOR TROUBLESHOOTING.

The Configuration Mode allows the G600 system to be configured for a particular installation. As part of the initial configuration, the GDU 620 functions must be enabled/disabled as desired, external data sources must be configured, and miscellaneous options must be set up. To do this an Installer Unlock Card (P/N 010-00769-60) must be inserted, and the unit must be started in configuration mode by pressing and holding the ENT button while applying power.

When in configuration mode the left display has only one page that is used to display status and product information. The right display has multiple pages (see “Chart 7”) that are used to configure and checkout the installation of the GDU 620.

For this dual GDU 620 installation, each GDU 620 must be set up individually.

NOTE: To access all of the configuration screens shown herein, an Installer Unlock Card P/N 010-00769-60 must be inserted in the bottom card slot prior to applying power to the GDU 620.

NOTE: For a complete description and breakdown of each Configuration Mode page, see Section 5 in the GDU 620 Installation Manual, Garmin P/N 190-00601-04.

CHART 7
CONFIGURATION MODE NAVIGATION

System Page Group		
1. System Configuration	4. Manifest Configuration	7. Audio Alert Configuration
2. Airframe Configuration	5. SW Upload	8. ARINC Post Config
3. Lighting Configuration	6. Feature Configuration	9. RS Post Config
GRS Page Group		
1. AHRS / Air Data Input	2. GRS / GMU Calibration	
GDC Page Group		
1. GDC Configuration		
GDL Page Group		
1. GDL 69 Config		
FSC Page Group		
1. Autopilot Configuration	3. Autopilot Test	
2. Flight Director		
DIAG Page Group		
1. Port Monitoring	3. Outputs	
2. Input Monitoring	4. Error Log	

(1) GDU 620 System Status Page (Left Side)

The PFD System Status Page (“Figure 14”) is used as information reference only; it is not configurable.

(A) DEVICES ONLINE Window:

1) Green Light: The device is configured as Present and valid data is being received as expected from the device.

2) Red Light: The device is configured as Present and invalid or no data is being received from the device.

3) No Light (Black): The device is configured as Not Present.

(b) LRU and DATA Windows:

To navigate within the LRU window, push the PFD knob (left side knob) to activate the cursor. Turn the PFD knob to select an LRU. Once an LRU is highlighted, information about it will be displayed in the DATA window.



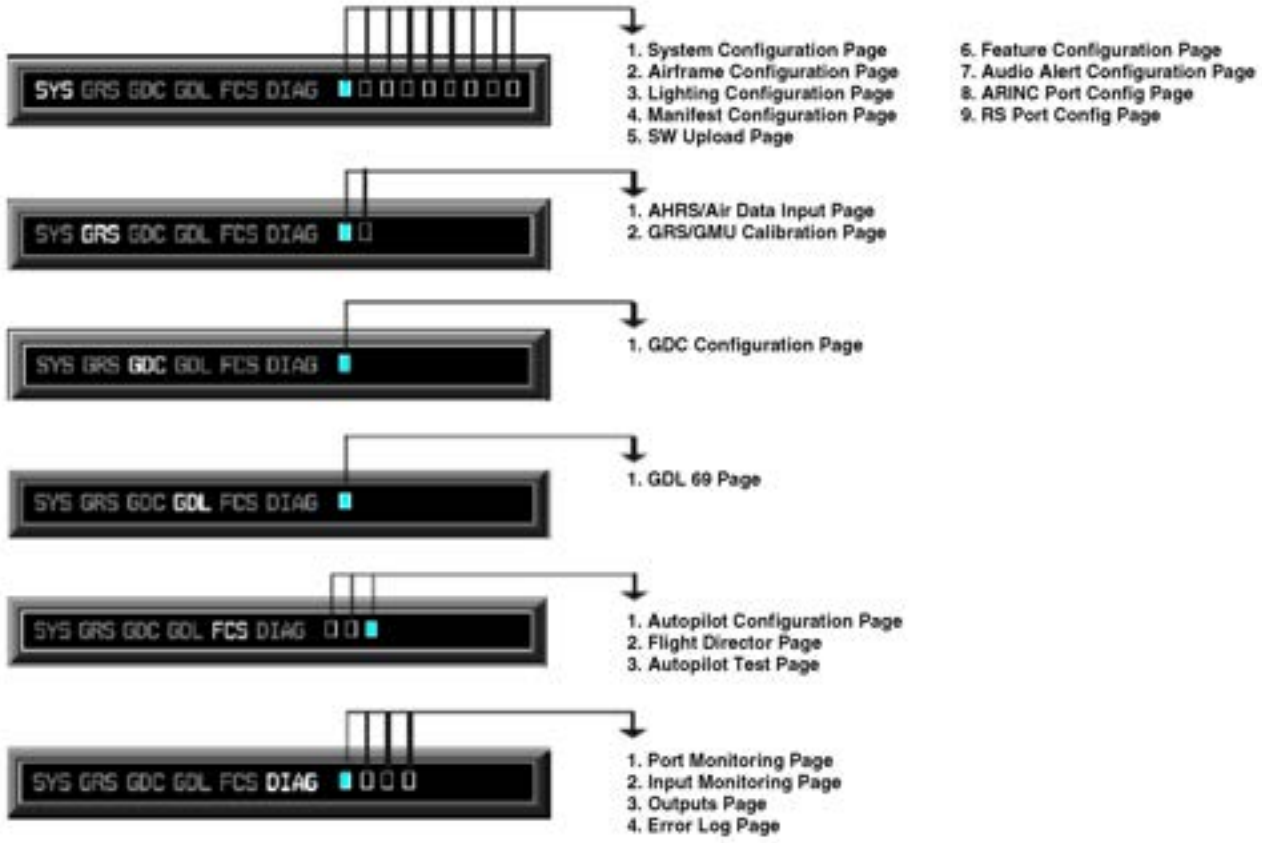
System Status Page (Configuration Mode)
Figure 14

(2) GDU 620 System Setup and Checkout Pages (Right Side)

The MFD knob (right hand dual concentric knob) is the primary control for the GDU 620 MFD.

- To change page groups (“Figure 15”) and cycle through different configuration screens, rotate the large MFD knob. The small MFD knob will change pages within the group.
- To activate the cursor for a page, press the small MFD knob directly in, as one would push a regular button.
- To cycle the cursor through different data fields, rotate the large MFD knob.
- To change the contents of a highlighted data field, rotate the small MFD knob. This action either brings up a menu with options for a particular field, or in some cases allows the operator to enter data for the field.
- To confirm a selection, press the ENT button.
- To cancel a selection, press the small MFD knob in or press the CLR key. Pressing the small MFD knob in again will deactivate the cursor.

Some configuration pages have commands or selections that are activated by the GDU 620 soft keys. If a soft key is associated with a command, that command will be displayed directly above the key. A grayedout soft key shows a command that is unavailable. A highlighted soft key shows the current active selection.



Configuration Page Navigation
Figure 15

(3) Configuration File Descriptions

(a) Configuration Files are divided into groups, and are only displayed at the System Upload page during the software/configuration loading process.

(b) Configuration files contain preset selections for input/output channels, aircraft-specific settings, and LRU-specific settings. The following list describes each of the configuration files:

SYSTEM Configures the G600 to expect GDUs, GPS/NAVs, GRSs, GDCs and optional interfaces.

AIRFRAME Contains data such as airspeed parameters, airframe sensor limitations, fuel tank parameters and alerting system settings that tailor a G600 PFD/MFD to the PA34-220T.

LIGHTING Contains items such as input source and levels, backlight output levels, minimum level, response time, slope and offset settings that tailor a G600 PFD/MFD to the PA34-220T.

MANIFEST Loads a manifest checklist of all software part numbers and versions associated with an approved system configuration. The G600 system performs a software check between each LRU's reported version and the version contained on the manifest. If an inequality is detected for an LRU, this LRU is then excluded from the GDU 620 and a manifest alert is triggered to the operator.

FEATURE Configures optional features (ChartView, Altitude Preselect, SVT, Weather Radar or TAWS) to be enabled/disabled.

AUDIO ALERT Configures audio alert volume levels are set for the PA34-220T.

GDC Configure GDC 74A air data values for the PA34-220T.

GDL_69 Configures gains and cable loss for the PA34-220T installation.

GWX Configures parameters as appropriate for the PA34-220T radar installation.

I. G600 Software Information

NOTE: The following sections provide a detailed description of loading all G600 software and configuration files, which may be excessive for individual LRU removal and replacement. If removing and replacing individual LRUs, refer to “Components” on page 342313, for the necessary steps.

(1) G600 Software Loader Card

CAUTION: IT IS CRITICAL THAT THE TECHNICIAN INSTALLS THE CORRECT SOFTWARE IMAGE PART NUMBER WHEN SERVICING THE G600 SYSTEM. APPROVED SOFTWARE DEFINITIONS FOR EACH CONFIGURATION ARE DEFINED IN THE REQUIRED EQUIPMENT LIST (REL) “CHART 12” ON PAGE 342376.

All software and configuration files were certified by Garmin as part of the FAA-approved Type Design data. Approved software definitions for each configuration are defined in the Required Equipment Lists (REL), “Chart 12” on page 342376.

G600 software and configuration files are controlled via the approved software image part number listed in the REL. This software image is loaded into the G600 using a software loader card. The installer shall create this software loader card by downloading the approved software image in accordance with the procedure provided immediately following.

CAUTION: BE CAUTIOUS WHEN USING SOFTWARE LOADER CARDS DURING MAINTENANCE. THE G600 SYSTEM IMMEDIATELY INITIALIZES THE CARD UPON POWER-UP. ON-SCREEN PROMPTS MUST BE GIVEN CAREFUL ATTENTION IN ORDER TO AVOID POTENTIAL LOSS OF DATA. ALWAYS READ THROUGH PROCEDURES GIVEN IN THIS SECTION, BEFORE ATTEMPTING TO USE THE SOFTWARE LOADER CARDS.

(2) Software Loader Card Creation

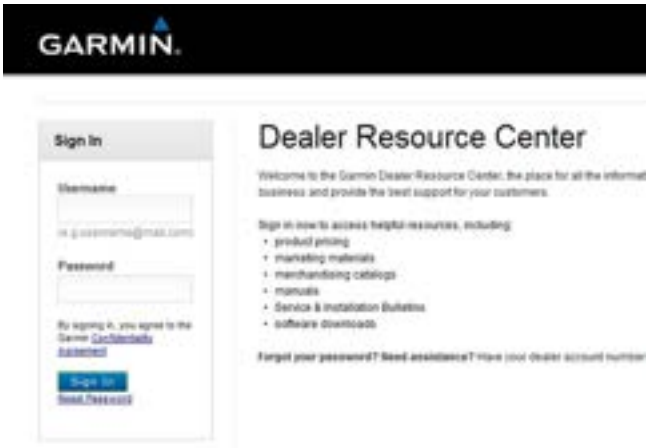
The following procedure requires a PC with an SD card reader. Piper recommends SanDisk® SDDR-99 and SDDR-93 card readers. Additionally, only SanDisk brand 16GB SD cards are recommended.

The software image is an executable self-extracting file which builds the correct file structure onto an SD card for use loading software to the G600. To obtain the current file follow the procedures outlined below.

NOTE: In order to create a loader card, the installer completing these procedures must be an authorized Piper or Garmin service center to gain access to the necessary data via the Garmin website.

(a) Go to https://www.garmin.com and click on the ‘Dealer Resource Center’ link in the lower right hand portion of the home page. Enter username and password. See “Figure 16”.

(b) Select Support and then Software Downloads.



Dealer Login Screen
Figure 16

(c) Search for the appropriate image version (i.e., “942.xx” from the appropriate “Chart 12” on page 342376) in the “Search by Keyword” field.

(d) Save the file to a known location on the PC.

(e) Run the executable file that was downloaded. A window will appear to confirm software installation.

(f) Insert the SD card into the SD card reader and select the Next button in the pop-up window.

(g) In the pop-up window, select the drive that corresponds to the inserted SD card and then click the Next button.

NOTE: The selected drive will be completely erased. Click the Next button to acknowledge any warnings that appear. Another pop-up window will appear to monitor loader card creation progress. After the card is successfully completed, another pop-up window will appear.

(h) Select the Finish button.

(i) Stop the card reader using Windows stop device function.

(j) Eject the SD card from the card reader.

(k) Apply a label to the software loader card identifying the aircraft, G1000 SFTW and the appropriate software image part number as shown below:

NOTE: See appropriate “Chart 12” on page 342376 for software image part number.

PA-34-220T
G600 SFWT
006-x942-xx

(3) Handling a Software Download Card

The SD memory cards used for the Software Download Card are manufactured to the specifications of the SD Card Association for use in portable devices. As such they are reasonably robust devices that do not require special handling instructions. However, the memory cards are not unbreakable and reasonable precautions regarding their use and handling should be used including the following guidelines:

(a) Do no bend an SD memory card or subject it to violent impact.

(b) Do not leave SD cards in locations where they may be exposed to moisture or high temperatures.

(c) Avoid exposing the card to static electricity or electro-magnetic interference.

J. Software Load Files

(PIR-PPS55028, Rev. F.)

The G600 LRUs come pre-loaded with software. However, to ensure that the latest software is loaded it is recommended that software from a current G600 Software Loader Card be loaded into each LRU. For dual GDU 620 installations the software loading procedures below must be carried out on each GDU.

CAUTION: DO NOT INTERRUPT POWER TO THE AIRCRAFT AT ANY TIME DURING THIS PROCEDURE. INTERRUPTION OF POWER COULD RESULT IN DAMAGE TO THE EQUIPMENT.

NOTE: To Select/Deselect the cursor in fields, press the small MFD knob. The large MFD knob changes Groups, and the small MFD knob changes Pages within a Group.

(1) Definition

Software files are defined by part number and version number in the Required Equipment List (REL), “Chart 12” on page 342376, in this section. Each G600 LRU reports the software version it currently contains to the user in two places:

(a) Normal System Mode: The AUX – SYSTEM STATUS page lists each LRU and the reported software version.

(b) Configuration Mode: The SYSTEM STATUS page (SYSTEM page group) reports more detailed LRU information, including software version, part number, and LRU status. Software files are loaded to LRUs from the SYSTEM UPLOAD page in configuration mode.

(2) GDU 620 Software Load

(a) Apply external power to the aircraft. Switch Main power and Avionics power ON. Ensure that the Avionics tie circuit breaker is pushed in.

(b) Pull circuit breakers labeled PFD1 and PFD2.

(c) Insert Software Loader Card P/N 010-00768-00 into the top card slot of GDU #1.

(d) Insert Installer Unlock Card P/N 010-00769-60 into the bottom card slot of GDU #1.

(e) While holding the ENT key, restore power by closing the PFD1 circuit breaker.

(f) Release the ENT key when the words “INITIALIZING SYSTEM” appear in the upper center of the PFD/MFD.

(g) Press the ENT key at the following prompt:

“DO YOU WANT TO UPDATE SYSTEM FILES?”

“PRESS ENT KEY FOR YES OR CLR KEY FOR NO”

“NO WILL BE ASSUMED IN 11 SECONDS”

(h) The following item is displayed:

“UPDATING SYSTEM FILES”

“DO NOT TURN OFF THE POWER”

(i) New software is loaded to the GDU #1. When complete, the following prompt is displayed:

“UPDATED XX FILES SUCCESSFULLY!” (where XX equals file quantity)

“PRESS ANY KEY TO CONTINUE.”

“CONTINUING IN 7 SECONDS.”

(j) Press any softkey. The GDU #1 starts in configuration mode.

(k) Repeat steps (c) thru (i) on GDU #2, using PFD2 circuit breaker.

(3) GRS 77 Software Load

(a) Go to the SOFTWARE UPLOAD page (“Figure 17”) on GDU #1 MFD using the MFD knob.

(b) Highlight the **GRS 77** software file. Ensure that the **GRS** is shown in the LRU window.

(c) Press the LOAD soft key.

(d) Select YES and press the ENT key to acknowledge the **“BEGIN FILE UPLOAD?”** prompt.

(e) New software is loaded to GRS #1. Monitor loading progress.

(f) After loading is complete, press ENT key to acknowledge the **“UPLOAD COMPLETED”** prompt.

(g) Check the SUMMARY window to verify that the upload of the GRS system file and FPGA file were “COMPLETED”.

(h) Repeat steps (a) thru (g) for GRS #2 using GDU #2 (PFD2 circuit breaker).

(4) GMU 44 Software Load

(a) Go to the SOFTWARE UPLOAD page (“Figure 17”) on GDU #1 MFD using the MFD knob.

(b) Highlight the **GMU 44** software file. Ensure that the **GMU** is shown in the LRU window.

(c) Press the LOAD soft key.

(d) Select YES and press the ENT key to acknowledge the **“BEGIN FILE UPLOAD?”** prompt.

(e) New software is loaded to GMU #1. Monitor loading progress.

(f) After loading is complete, press ENT key to acknowledge the **“UPLOAD COMPLETED”** prompt.

(g) Check the SUMMARY window to verify that the upload of the GMU system file and FPGA file were “COMPLETED”.

(h) Repeat steps (a) thru (g) on GMU #2 using GDU #2 (PFD2 circuit breaker).

(5) GDC 74A Software Load

(a) Go to the SOFTWARE UPLOAD page (“Figure 17”) on GDU #1 MFD using the MFD knob.

(b) Highlight the **GDC 74A** software file. Ensure that the **GDC** is shown in the LRU window.

(c) Press the LOAD soft key.

(d) Select YES and press the ENT key to acknowledge the **“BEGIN FILE UPLOAD?”** prompt.

(e) New software is loaded to GDC #1. Monitor loading progress.

(f) After loading is complete, press ENT key to acknowledge the **“UPLOAD COMPLETED”** prompt.

(g) Check the SUMMARY window to verify that the upload of the **GDC** system file and FPGA file were “COMPLETED”.

(h) Repeat steps (a) thru (g) on GDC #2 using GDU #2 (PFD2 circuit breaker).



Software Upload Page
Figure 17

(6) GDL 69A Software Load

(a) Apply external power to the aircraft. Switch Main power and Avionics power ON. Ensure that the Avionics tie circuit breaker is pushed in.

(b) Pull circuit breakers labeled PFD1 and PFD2.

(c) Insert Software Loader Card P/N 010-00768-00 (GDL 69A) into top card slot of GDU #1.

(d) Insert Installer Unlock Card P/N 010-00769-60 into bottom card slot of GDU #1 and GDU #2.

(e) While holding the ENT key, restore power by closing the PFD1 and PFD2 circuit breakers.

(f) On the PFD side of display, verify Datalink box is green.

(g) Go to the SOFTWARE UPLOAD page (“Figure 17” on page 342336) on GDU #1 MFD using the MFD knob.

NOTE: GDL 69/69A Software Version 3.20 or later **must** be installed in the unit before GDL 69 Series Software Version 4.01 can be correctly installed.

(h) Highlight the **GDL 69** software file in the file list window. Ensure that the **GDL 69** is shown in the LRU window.

(i) Press the LOAD soft key.

(j) Select YES and press the ENT key to acknowledge the **“BEGIN FILE UPLOAD?”** prompt.

(k) New software is loaded to **GDL**. Monitor loading progress.

(l) After loading is complete, press ENT key to acknowledge the **“UPLOAD COMPLETED”** prompt.

(m) Go to another page and then back to the SOFTWARE UPLOAD page. Verify **GDL 69** software version displayed in the LRU window (may take several seconds to update display).

(n) Remove power from GDU #1, GDU #2 and GDL by pulling PFD1, PFD2 and XM circuit breakers.

(o) Restore power to GDUs and GDL by closing PFD1, PFD2 and XM circuit breakers, starting the GDUs in normal mode.

(p) Go to AUX page group on GDU #1 and GDU #2 and select the System Status page. Verify GDL 69 software field displays 4.01.00 or later.

(7) GWX 68 Software Load

(a) Apply external power to the aircraft. Switch Main power and Avionics power ON. Ensure that the Avionics tie circuit breaker is pushed in.

(b) Pull circuit breakers labeled PFD1 and PFD2.

(c) Insert Software Loader Card P/N 010-00768-00 (GWX 68) into top card slot of GDU #1.

(d) Insert Installer Unlock Card P/N 010-00769-60 into bottom card slot of GDU #1 and GDU #2.

(e) While holding the ENT key, restore power by closing the PFD1 and PFD2 circuit breakers.

(f) Go to the SOFTWARE UPLOAD page (“Figure 17” on page 342336) on GDU #1 MFD using the MFD knob.

(g) Highlight the **GWX 68** software file. Ensure that the **GWX** is shown in the LRU window.

(h) Press the LOAD soft key.

(i) Select YES and press the ENT key to acknowledge the **“BEGIN FILE UPLOAD?”** prompt.

(j) New software is loaded to **GWX**. Monitor loading progress.

(k) After loading is complete, press ENT key to acknowledge the **“UPLOAD COMPLETED”** prompt.

(l) Check the SUMMARY window to verify that the upload of the **GWX** system file and FPGA file were “COMPLETED”.

K. System Configuration

CAUTION: THE CONFIGURATION MODE CONTAINS CERTAIN PAGES AND SETTINGS THAT ARE CRITICAL TO AIRCRAFT OPERATION AND SAFETY. THESE PAGES ARE PROTECTED AND CANNOT BE MODIFIED, UNLESS THE TECHNICIAN IS PROPERLY AUTHORIZED AND EQUIPPED. HOWEVER, MOST PROTECTED PAGES ARE VIEWABLE TO ALLOW SYSTEM AWARENESS FOR TROUBLESHOOTING.

To configure the GNS 430W, see the “Post Installation Setup Procedure” on page 34501 under COM/NAV/GPS. If the optional GI-106 Indicator or DME units are installed, perform configuration procedures provided in “VOR/LOC/GS CDI Config” on page 34506. If the optional Stormscope unit is installed, perform configuration procedures provided in “Stormscope Config” on page 34507.

For configuration of the GTX 330, see the “Post Installation Setup Procedure” on page 345015 under Transponder, GTX 330.

(1) GDU 620 Initial Setup

NOTE: To access all of the configuration screens shown herein, an Installer Unlock Card P/N 010-00769-60 must be inserted in the bottom card slot prior to applying power to the GDU 620.

Go to the SYSTEM CONFIGURATION page in the SYS page group of GDU #1. The System Configuration Page (“Figure 18”) allows the installers to select what LRUs are present and not present in a given installation.

Activate the cursor, select the system using the large MFD knob, and then press the ENT button. This will activate or deactivate the highlighted system. For activated systems, use the small knob to select the appropriate LRU from the dropdown TYPE menu, and then press ENTER to select the LRU.

Continue to activate or deactivate each LRU, as required.

A green light means that the LRU is configured as ‘present’. No light (black) means that the LRU is configured as ‘not present’ (the TYPE field will be blank in this case).

Deactivate the cursor when all interfacing systems have been configured.

NOTE: Only one traffic feature can be enabled in an aircraft. TIS is installed if TAS is not installed.

Use the small knob to move to the next page. Repeat the process using GDU #2.

INTERFACING SYSTEMS Window:

INTERFACING SYSTEM	PRESENT	TYPE
Cross-side GDU:	YES	GDU 620
AHRS:	YES	GRS 77
ADC:	YES	GDC 74
GPS1:	YES	GNS 430W
GPS2:	YES	GNS 430W
NAV1:	YES	GNS 430W
NAV2:	YES	GNS 430W
ADF:	YES (If installed)	KR 87
+Superflag,	YES (If ADF installed)	
Traffic (TIS):	YES (If installed)	GTX 330
Traffic (TAS):	YES (If installed)	KTA 870
+External Control	NO	
Data Link:	YES (If XM installed)	GDL 69A
Weather Radar:	YES (If WXR installed)	GWX 68
Adapter:	NO	



System Configuration Page
Figure 18

(2) MANIFEST Configuration

Perform the following steps on GDU #1 and GDU #2:

(a) Apply electrical power to Avionics, GDU running in configuration mode.

(b) Ensure that all devices configured “Present”, per “GDU 620 Initial Setup” on page 342338, are reported with valid status, as shown in the DEVICES ONLINE window of the SYSTEM STATUS page on the PFD.

NOTE: GWX 68 is valid on GDU#1 only.

NOTE: Traffic (TAS) will not be valid Present until TAS Configuration is completed.

NOTE: Weather Radar will not be valid Present until GWX 68 Configuration is completed.

NOTE: Data Link will not be valid Present until GDL 69/69A Configuration is completed.

(c) Select the MANIFEST CONFIGURATION page in the SYS page group. The manifest will be empty on first use.

(d) Press the MANIFEST soft key to automatically populate the manifest.

CAUTION: FAILURE TO HAVE ALL LRU’S POWERED WITH A VALID SOFTWARE LOAD WILL RESULT IN AN INCOMPLETE LIST OF SOFTWARE PART NUMBERS AND VERSION NUMBERS BEING STORED IN THE MANIFEST. IF AN INCOMPLETE LIST OF SOFTWARE PART NUMBERS OR VERSION NUMBERS IS STORED IN THE MANIFEST, THE MANIFEST SOFT KEY CAN BE PRESSED AGAIN TO REFLECT THE CURRENT INFORMATION.

(e) Verify that the manifest contains a complete list of all applicable software part numbers and version numbers.

NOTE: The GDU 620 maintains a list of software part numbers and version numbers (the “manifest”) for the G600 system. Each time the G600 system is powered up, the LRUs report their software information to the GDU 620. If the information reported by the LRU does not match the information stored in the manifest, the GDU 620 will not use the data provided by that LRU and will provide a MANIFEST alert.

(3) Configuration Module Interface Check

NOTE: This procedure is to be performed once on each GDU, prior to any Feature Activation Procedures (see below). Any features that are activated prior to having a fully functional configuration module will be disabled when the configuration module interface is corrected. The activation card that was originally used will not work to re-activate the feature if this occurs.

Perform the following steps on GDU No.1 and GDU No. 2:

(a) Electrical power applied to Avionics, GDU running in configuration mode.

(b) Select the SOFTWARE UPLOAD page.

(c) Press the UPDT CFG soft key.

(d) Press ENT to confirm and update the configuration module.

(e) Verify that “Config Module Update Complete” is displayed with no error messages.

NOTE: If error messages are displayed, verify that the configuration module is correctly installed.

(4) GDU 620 Software Load Confirmation

(a) Ensure that each GDU is in configuration mode.

(b) On the PFD, go to the System Status Page (“Figure 19”) and activate the cursor at the LRU window.

(c) Highlight each of the listed items in the LRU window and, using “Chart 11” on page 342375, verify that the software part number and software version for each item matches those shown in the Required Equipment List (REL), “Chart 12” on page 342376.

NOTE: GWX 68 is valid on GDUNo. 1 only.

NOTE: Weather Radar will not be valid Present until GWX 68 Configuration is completed.

NOTE: Data Link will not be valid Present until GDL 69/69A Configuration is completed.

NOTE: Use each GDU to read the units that interface to it, since all units cannot be read from a single GDU.

NOTE: If any software version and/or part number does not match, or is not successfully loaded, do not continue with post-installation procedures. Re-load software as required. Any previously run setup procedures for an LRU must be performed again after loading software on that LRU.

(d) De-activate the cursor.



System Status Page (Left Side - PFD)
Figure 19

(5) AIRFRAME Configuration

Go to the AIRFRAME CONFIGURATION (“Figure 20”) page in the SYS page group. This page allows the installer to configure the PFD airspeed, altitude and vertical speed tape scales and appearance, MFD display, and GDL69 Audio. Perform the following steps on GDU #1 and GDU #2.

NOTE: All PFD settings must match what is in the aircraft flight manual (AFM) or the pilot’s operating handbook (POH).

Activate the cursor and use the MFD knob to select following configurable items.

(a) PFD Window:

1) DIS. SPD, changes the units displayed on the airspeed tape.
Selection: Nautical (NM, KT).

2) ALT. VS, changes the units displayed on the altitude and vertical speed tapes.
Selection: Feet (FT, FPM).

3) VS TAPE RANGE, changes the range of the vertical speed tape.
Selection: +/- 2000 FPM. Both Alt Tape Range and IAS Tape Range are informational only.

4) ADI Roll Pointer, changes which pointer moves to indicate bank angle.
Selection: Ground.



Airframe Configuration Page
Figure 20

(b) MFD Window:

1) DIS. SPD, changes the distance and speed units displayed on the MFD. Selection: Nautical (NM, KT).

2) ALT. VS, changes the altitude and vertical speed units on the MFD. Selection: Feet (FT, FPM).

3) AIRCRAFT ICON, changes the appearance of the ownship icon on the MFD map. Selection: Low Wing.

(c) AIRSPEEDS (PFD) Window: The following settings affect the markings on the airspeed tape. Refer to the AFM or POH to determine the appropriate settings for this installation.

ITEM	VALUE
Vs0	61
Vs1	67
Vfe	113
Vno	164
Vne	204
GLIDE	0
Vr	0
Vx	83
Vy	88
Vle	OFF
VMCA	66
Vyse	88

(d) Deactivate the cursor to move to the next page.

(6) LIGHTING Configuration

(a) Select the LIGHTING CONFIGURATION page in the SYS page group.

(b) Select the following parameters to the values defined below.

BRIGHTNESS WINDOW	DISPLAY	KEY
INPUT SOURCE	28V DC	28V DC
SOURCE INPUT LEVEL	Not Selectable	Not Selectable
BACKLIGHT OUT LEVEL	Not Selectable	Not Selectable
MINIMUM LEVEL	5	10
RESPONSE TIME	2	2
SLOPE	40	50
OFFSET	5	10
PHOTOCELL LEVEL	Not Selectable	N/A
PHOTO TRANSITION %	20	N/A
PHOTO SLOPE	50	N/A
PHOTO OFFSET	50	N/A

(7) AUDIO ALERT Configuration

NOTE: Perform the following steps on GDU #1 only.

(a) Select the AUDIO ALERT CONFIGURATION page in the SYS page group.

(b) Set the following parameters to the values defined below:

ITEM	VALUE
VOLUME ADJUST	0 dB
VOICE	FEMALE

NOTE: To set value, place cursor on item and rotate small right knob.

(c) Select sample message from the Audio Message type list and then select the PLAY field. Press ENT to play the message.

(d) Verify the audio alert is clearly audible.

(8) TERRAIN-SVT Setup

(a) Select the TERRAIN-SVT SETUP page in the SYS page group.

(b) If TAWS is NOT installed, in the TERRAIN ALERTING CONFIGURATION window, verify and set the values defined below:

ITEM	VALUE
CONFIGURATION	TERRAIN-SVT
EXT TAWS	NOT INSTALLED

NOTE: If TAWS is installed, skip this step and move on to the Airport Criteria window.

(c) In the AUDIO CONFIGURATION window of the TERRAIN-SVT SETUP page group, set the following parameters to the values defined below:

ITEM	VALUE
RTC-CAUTION	Caution Terrain Terrain
RTC-WARNING	Warning Terrain Terrain
ROC-CAUTION	Caution Obstacle Obstacle
ROC-WARNING	Warning Obstacle Obstacle
ITI-CAUTION	Caution Terrain Terrain
ITI-WARNING	Warning Terrain Terrain
IOI-CAUTION	Caution Obstacle Obstacle
IOI-WARNING	Warning Obstacle Obstacle

NOTE: If TAWS is installed, skip this step.

(d) In the AIRPORT CRITERIA window of the TERRAIN-SVT SETUP page group, set the following parameters to the values defined below:

ITEM	VALUE
RNWY SURFACE	HARD/SOFT
MIN LENGTH	1250

(9) GDC Configuration

(a) Select the GDC CONFIGURATION page in the GDC page group.

(b) Activate the cursor and select the GTP 59 OAT probe type.

(c) Perform a Configuration Module Update.

(10) Configuration Module Update

NOTE: This procedure is to be performed after any configuration procedure is completed and prior to power-down of each GDU.

(a) GDU #1 and/or GDU #2 running in configuration mode. Perform the following steps on GDU #1 and/or GDU #2.

(b) Select the SOFTWARE UPLOAD page.

(c) Press the UPDT CFG soft key.

(d) Press ENT to confirm and update the configuration module.

(e) Verify that “Config Module Update Complete” is displayed with no error messages.

NOTE: If error messages are displayed following completion of configuration module update, troubleshoot the configuration module for faults.

(11) Optional G600 Feature Activation Procedures

NOTE: In most cases, in order to enable/disable features, an Installer Unlock Card (P/N 010-00769-60) is required. However, it is possible to view the status of all features without using an Installer Unlock Card.

All optional features are activated using the FEATURE CONFIGURATION page of the GDU 620.

CAUTION: PRIOR TO ENABLING ANY OPTIONAL FEATURES IT IS RECOMMENDED THAT AT LEAST ONE CONFIGURATION MODULE UPDATE BE SUCCESSFULLY COMPLETED (REFER TO CONFIGURATION MODULE INTERFACE CHECK) TO VERIFY CORRECT FUNCTIONALITY OF THE CONFIGURATION MODULE INTERFACE. ANY OPTIONAL FEATURES THAT ARE ENABLED WITHOUT A FULLY FUNCTIONAL CONFIGURATION MODULE WILL BE DISABLED WHEN THE CONFIGURATION MODULE INTERFACE IS CORRECTED. THE ENABLEMENT CARD THAT WAS USED ORIGINALLY WILL NOT WORK TO RE-ENABLE THE FEATURE IN THIS CASE.

(a) Basic Feature Activation

1) Electrical power applied to Avionics, GDU #1 and GDU #2 running in configuration mode by holding the ENT key while closing the PDF circuit breakers.

2) Select the FEATURE CONFIGURATION page from the SYS page group.

3) Set Synthetic Vision to ENABLED on each GDU, “Enabled” will appear in the ACTIVE column.

4) If Weather Radar installed, set Weather Radar to ENABLED on each GDU, “Enabled” will appear in the ACTIVE column.

5) Set GAD 43 Adapter to DISABLED on each GDU. “Disabled” will appear in the ACTIVE column.

6) Perform a Configuration Module Update.

(b) Altitude Preselect Activation (Optional)

Perform this procedure only if the optional Autopilot is installed. When the optional Altitude Preselect feature is enabled, the GDU 620 can provide selected altitude information to the autopilot, allowing the autopilot to capture the selected altitude.

NOTE: Altitude preselect is activated only on GDU #1. Do not attempt to activate this feature on GDU #2.

1) Pull circuit breaker for PFD1.
2) Apply power to Avionics.
3) On GDU #1, insert the Installer Unlock Card into the bottom card slot and Altitude Preselect Enablement Card P/N 010-00769-52 into the top card slot.
4) Enter configuration mode on the GDU #1 by holding the ENT key while closing the PDF1 circuit breaker.
5) Go to the FEATURE CONFIGURATION page in the SYS page group.
6) Activate the cursor and change the SET value for “Altitude Preselect” to Enabled. Press ENT to confirm your selection.
7) Press ENT to acknowledge the prompt and activate the Altitude Preselect feature.
8) When the Altitude Preselect feature is activated, “Enabled” will appear in the ACTIVE column.

(c) Chartview Activation (Optional)

The GDU 620 can display Jeppesen charts using the optional ChartView feature, which must be activated.

NOTE: Each Chartview Enablement Card can only activate the Chartview feature on one GDU display. A different Chartview Enablement Card must be used for each GDU. Ensure cards are properly labeled for each GDU.

1) Pull circuit breakers for PFD1 and PFD2.
2) Apply power to Avionics.
3) On GDU #1, insert the Installer Unlock Card into the bottom card slot and Chartview Enablement Card P/N 010-00769-53 into the top card slot.
4) Enter configuration mode on the GDU #1 by holding the ENT key while closing the PDF1 circuit breaker.
5) Go to the FEATURE CONFIGURATION page in the SYS page group.
6) Activate the cursor and change the SET value for “Charts” to ChartView. Press ENT to confirm your selection.
7) Press ENT to acknowledge the prompt and activate the ChartView feature.
8) When the ChartView feature is activated, “ChartView” will appear in the ACTIVE column.
9) Repeat steps 3) thru 8) for GDU #2.
10) Perform a Configuration Module Update.

NOTE: Navigation or chart data must not be programmed on the ChartView Enablement Card.

(d) TAWS Activation (Optional)

When the optional TAWS feature is enabled, the GDU 620 will provide Class B TAWS functionality.

NOTE: A single TAWS Enablement Card for dual GDU 620 installations (P/N 010-00769-56) will enable TAWS-B functionality on both GDU 620s in a dual GDU 620 installation.

1) Pull circuit breakers for PFD1 and PFD2.
2) Apply power to Avionics.
3) On GDU #1, insert the Installer Unlock Card into the bottom card slot and TAWS Enablement Card P/N 010-00769-56 into the top card slot.
4) Enter configuration mode on the GDU #1 by holding the ENT key while closing the PDF1 circuit breaker.
5) Go to the FEATURE CONFIGURATION page in the SYS page group.
6) Activate the cursor and change the SET value “TAWS” to Enabled. Press ENT to confirm your selection.
7) Press ENT to acknowledge the prompt and activate the TAWS feature.
8) When the TAWS feature is activated, “Enabled” will appear in the ACTIVE column. Deactivate cursor.
9) Select the TAWS-B SETUP page in the SYS page group.
10) Verify “TAWS-B” is displayed as the configuration in the TERRAIN ALERTING CONFIGURATION window.
11) In the AUDIO CONFIGURATION window of the TERRAIN-SVT SETUP page group, set the following parameters to the values defined below:

ITEM	VALUE
RTC-CAUTION	Caution, Terrain (2x)
RTC-WARNING	Warning Terrain (2x), Pull-Up (2x)
ROC-CAUTION	Caution, Obstacle (2x)
ROC-WARNING	Obstacle (2x), Pull-Up (2x)
ITI-CAUTION	Terrain Ahead (2x)
ITI-WARNING	Terrain Ahead, Pull-Up (2x)
IOI-CAUTION	Obstacle Ahead (2x)
IOI-WARNING	Obstacle Ahead, Pull-Up (2x)
PDA-CAUTION	Too Low - Terrain
NCR-CAUTION	Don’t Sink
EDR-CAUTION	Sink Rate
EDR-WARNING	Pull-Up
VCO-500	Five Hundred

12) Perform Configuration Module Update on GDU.
13) Pull PFD1 circuit breaker and remove TAWS enable card from GDU #1.
14) Repeat steps 3) thru 13) for GDU #2.

(12) GDL 69/69A Configuration

NOTE: Perform this procedure only if XM option is installed.

Perform the following steps on GDU #1 and GDU #2:

(a) Apply electrical power to Avionics, GDU running in configuration mode.

(b) Select the GDL 69 CONFIGURATION page in the GDL page group.

NOTE: Press ENT to save the entries made on the page.

(c) Activate the cursor and adjust the Cable Loss to 5 dB.

(d) Adjust the Antenna Gain to 27 dB.

NOTE: DO NOT DISABLE Ethernet port 2. Doing so will cease communication between the system and the GDL 69A.

(e) Select ENABLE for Ethernet Port 2, DISABLE for the remaining ports.

(f) Select the SET>ACTV soft key control (lower left of GDL 69 configuration page) to save the GDU configuration module settings to the GDL.

(g) Perform a Configuration Module Update.

(13) AFCS Configuration

NOTE: Perform this procedure only if Autopilot is installed.

Perform the following steps on GDU #1 and GDU #2:

(a) Apply electrical power to Avionics, GDU running in configuration mode.

(b) Go to the FEATURE CONFIGURATION page in the SYS page group. Ensure that the Altitude Preselect and SVT options are Enabled.

NOTE: Altitude Preselect is enabled only on GDU #1.

(c) Select the AUTOPILOT CONFIGURATION page in the FCS page group.

(d) Activate the cursor and press ENT to set the Autopilot to PRESENT.

(e) Move the cursor to the TYPE field and set the value to “King KI 525”.

NOTE: Perform steps (f) and (g) on GDU #1 only.

(f) Move the cursor and highlight Altitude Preselect. Press ENT to set the Altitude Preselect to PRESENT.

(g) Move the cursor to the TYPE field and set the Altitude Preselect type value to “S-TEC 55X”.

(h) De-activate cursor on both GDUs.

(i) Select the FLIGHT DIRECTOR page in the FCS page group.

(j) Activate the cursor and highlight Flight Director. Press ENT to set the flight director to PRESENT.

(k) Move the cursor to the TYPE field and set the Flight Director type value to “S-TEC 55X”.

(l) Move the cursor to the SVT FD DISPLAY field and set the value to ENABLE.

(m) Perform a Configuration Module Update.

(14) GWX 68 Configuration

(a) Aircraft level in pitch axis (See 8-20-00).

(b) Radar dish parked at center of sweep and zero tilt.

(c) Using digital level, measure angular error of weather radar dish in pitch axis by holding the level against the surface of the radar dish. Note the error to the nearest 0.10 degree.

(d) Apply electrical power to Avionics, GDU #1 running in configuration mode.

(e) Select the GWX WEATHER RADAR CONFIGURATION page in the WXR page group.

(f) Enter pitch trim value to compensate for the angular errors recorded in step (c).

NOTE: Pitch trim can be adjusted in 0.05 degree increments, between +/- 4.00 degrees.

(g) Perform a Configuration Module Update.

(h) Select the SET>ACTV soft key (lower left of page) to write the GDU #1 configuration data to the GWX 68.

(i) Ensure AHRS #1 alignment is complete.

(j) Select the GWX WEATHER RADAR CONFIGURATION page in the WXR page group.

(k) Verify that there are no failed status indications in the STATUS window.

(l) Re-set electrical power to aircraft, GDU running in normal mode.

(m) Perform several sweeps with radar, adjust radar tilt and then re-set tilt to zero.

(n) Radar dish parked at center of sweep and zero tilt.

(o) Using digital level, measure angular error of weather radar dish in pitch axis by holding the level against the surface of the radar dish. Note the errors to the nearest 0.10 degree.

(p) Verify that the pitch error in the radar dish has been eliminated.

(15) TAS Configuration

NOTE: Perform this procedure only if TAS option is installed.

NOTE: Setup of TAS configuration requires use of the TAS DIAG software provided by Honeywell. Refer to the KTA-870 Installation manual, App. D, regarding usage of TAS DIAG software.

(a) Connect the Com 1 port on the laptop to the TDIAG connector on the aircraft.

(b) Start the TAS DIAG software on the laptop.

(c) Select the “Config Mod” menu item.

(d) Select “Initialize Config Mod”.

(e) When the “Initialize Configuration Module” dialogue box appears, select “Yes”.

(f) When the “Configuration Module” is available, select “OK”.

(g) When the “Modify Config Module Strap Data” dialogue box appears, the “Straps Page 1” tab will be presented in the dialogue box. Make the following selections and then proceed to “Straps Page 2”:

STRAPS PAGE 1	SELECTION	STRAPS PAGE 2	SELECTION
Heading Type	ARINC 429 Data	Intruder File Protocol	ARINC 735
Attitude Type	ARINC 429 Data	Airborne Functional Test	Allow
Controller Type	Discrete	Display Intruder Limit	30 Intruders
Baro Altimeter Type	ARINC 706/429	Heading Sense Valid	Ignore
Airborne Headphones	100mW	On-Ground Display Mode	Standby
Ground Headphones	100mW	Display Valid Input Usage	Ignore
Radio Altimeter Type	Not installed	ATE Outputs	Disable
		RS-422 Outputs	Disable

(h) When “Straps Page 2” is completed, select “OK”.

(i) When asked to write data to the config module, select “OK” and TAS will reboot.

(j) After the TAS reboots, select “OK”. The Traffic Configuration is complete.

(16) Navigation Database Aircraft Loading Procedures

(a) G600 Navigation Database Aircraft Loading Procedure

The GDU 620 utilizes a database stored on an SD memory data card for easy updating and replacement. The Navigation database may be updated by simply inserting an updated Navigation database update card into the top SD card slot in the front panel in the GDU 620. The actual database is downloaded into the unit, so the card can be removed after the update. Each card will only update one system.

1) Remove power from the GDUs by pulling the PFD1 and PFD2 circuit breakers.

2) Apply electrical power to Avionics.

3) Install FlyGarmin database SD card (010-00769-70) in the top slot of GDU #1.

4) Restore power to the GDU #1 by closing the PFD1 circuit breaker.

5) The following item is displayed in the upper left corner of the PFD:

“DO YOU WANT TO UPDATE THE AVIATION DATABASE?”

“PRESS CLR KEY FOR NO AND ENT KEY FOR YES”

“YOU HAVE 30 SECONDS BEFORE NO IS RETURNED”

6) Press the ENT key to confirm the database update. The following prompts are displayed:

“UPDATING AVIATION DATABASE”

“UPDATED x FILES SUCCESSFULLY!”

NOTE: The quantity “x” is variable, depending on database version.

7) After the update completes, the GDU starts in normal mode.

8) Verify that the expiration date of the aviation database is shown as valid.

9) Repeat steps 2) thru 8) for the GDU #2 using PFD2 circuit breaker.

NOTE: Wait for the GPS receiver to obtain date and time.

(b) GNS 430 Navigation Database Aircraft Loading Procedure

The GNS 430 utilizes a database stored on an SD memory data card for easy updating and replacement. The Navigation database may be updated by simply inserting an updated Navigation database update card into the left SD card slot in the front panel in the GNS 430.

1) Remove power from Avionics. Perform the following steps on GNS #1 and GNS #2.

2) With the label facing left, insert the aviation database card into the left slot on front panel of the GNS until fully seated, then depress tab.

3) Apply electrical power to Avionics.

4) The GNS will go through a normal start-up sequence.

5) On the start-up screen, verify that the expiration date of the aviation database is shown as valid.

NOTE: Wait for the GPS receiver to obtain date and time.

(17) Supplemental Databases

(a) Terrain Database Loading Procedures

NOTE: Garmin Field Service Engineering, resident at Piper, will be responsible for supplying updates for the 010-00769-43 Supplemental SD card.

1) Pull the PFD1 and PFD2 circuit breakers.
2) Install Supplemental SD card P/N 010-00769-43 in the bottom slot of each GDU.
3) Push in the PFD1 and PFD2 circuit breakers.
4) On each startup screen, verify that the following databases are shown as valid:
   a) Terrain
   b) Airport Terrain
5) On each startup screen, verify that a valid cycle is shown for the following databases:
   a) Obstacle
   b) Safetaxi

   NOTE: Wait for the GPS receiver to obtain date and time.

(b) JEPPS ChartView Loading Procedure

The GDU 620 utilizes a database stored on an SD memory data card for easy updating and replacement. The ChartView database may be updated by simply inserting an updated Supplemental SD card into the bottom SD card slot in the front panel in the GDU 620. The Supplemental SD card remains in the GDU for normal operation.

1) Pull the PFD1 and PFD2 circuit breakers.
2) Apply electrical power to Avionics.
3) Install Supplemental SD card (010-00769-43) in the bottom slot of GDU #1.
4) Restore power to the GDU #1 by closing the PFD1 circuit breaker.
5) On the startup screen, verify that a valid cycle is shown for the Chartview database.

   NOTE: Wait for the GPS receiver to obtain date and time.

6) Repeat steps 3) thru 5) for the GDU #2 using PFD2 circuit breaker.

(18) Supplemental Databases Check Procedures

CAUTION: THE DATABASES ON THE SUPPLEMENTAL DATA (SD) CARD ARE LOCKED TO SPECIFIC GDU 620 INSTALLATIONS. THE FIRST TIME THE SUPPLEMENTAL DATA CARD IS INSERTED INTO A DISPLAY IT BECOMES "LOCKED" TO THAT PARTICULAR AIRCRAFT INSTALLATION AND WILL NOT WORK IN OTHER INSTALLATIONS.

(a) Remove power from the PFD/MFD by pulling the PFD circuit breaker.
(b) Ensure that a Supplemental SD card is inserted in the bottom SD slot of the GDU 620.
(c) Turn on power to the G600 system.
(d) Verify all self-tests pass on the main startup screen.

NOTE: Databases that have expired will be displayed in yellow text on the MFD startup screen. Databases will also be displayed in yellow until a valid GPS Position has been acquired.

(e) Verify the expiration on the Terrain Database.
(f) If activated, verify the expiration of the ChartView database.

L. LRU Test Procedures

(1) GRS/GMU Calibration Procedures

See “Chart 8” to determine when various calibrations are required. The following calibration procedures are provided for the GRS 77 and GMU 44:

- AHRS Pitch/Roll Offset Calibration - Procedure A
- Magnetometer Calibration - Procedure B
- Compass Swing
- Heading Offset Compensation
- AHRS Engine Run-Up Vibration Test - Procedure C
- Magnetometer Interference Test

When ready to perform the procedures, shut the PFD/MFD’s off by pulling the PFD1 and PFD2 circuit breakers. Restart all displays in configuration mode. Follow the steps given for each procedure on-screen at the GRS/GMU CALIBRATION page. Note that the CALIBRATE command cannot be selected and activated until the installer acknowledges all required steps have been carried out by pressing the ENT key on each step.

(2) GRS 77 AHRS Pitch/Roll Offset Calibration - Procedure A

NOTE: An Installer Unlock Card (P/N 010-00769-60) is required in order to run the Pitch/Roll Offset Compensation Procedure.

This procedure must be performed for both GRS 77 units installed in the aircraft. The procedure must be carried out with the engine off.

(a) Level the aircraft to within ±0.25° of zero pitch and zero roll using a digital level. (Refer to 8-20-00.)

(b) Pull circuit breakers for PFD1 and PFD2.

(c) Apply power to Avionics.

(d) On GDU #1, insert the Installer Unlock Card into the bottom card slot.

(e) Enter configuration mode on the GDU #1 by holding the ENT key while closing the PDF1 circuit breaker.

(f) Go to the GRS Page Group and select the GRS/GMU Calibration (“Figure 21”) page.

(g) Initiate the AHRS Ground Pitch/Roll Aircraft Level compensation mode by performing the following steps:

1) Activate the cursor and select **PITCH/ROLL OFFSET**.
2) Press the ENT key.
3) Follow the checklist items displayed on the MFD and press the ENT key as each item is completed or confirmed.
4) When the CALIBRATE field is blinking, press the ENT button to begin the procedure.
5) After several seconds, a new checklist appears in the lower half of the MFD. Press the ENT key as each item is confirmed. When the CONFIRM AIRCRAFT IS LEVEL field is blinking, press the ENT key to continue.

(h) The result of the pitch/roll offset compensation is displayed on the MFD. If successful, the AHRS records the required pitch and roll offsets, informs the operator of a successful conclusion and returns to normal operation.

(i) Press the ENT key on the MFD to conclude this procedure for GRS1.

(j) Repeat steps (d) thru (i) for GRS2 using GDU #2. PFD2 circuit breaker is closed in step (e).

(k) Perform “Configuration Module Update” on page 342345.

CHART 8
G600 - GRS 77 AHRS/GMU 44 MAGNETOMETER CALIBRATIONS

Required GRS/GMU Calibrations			
Condition	Calibrations Required		
	Procedure A: GRS 77 Pitch/Roll Offset	Procedure B: GRS/GMU Magnetic Calibration	Procedure C: Engine Run-up Vibration Test
GRS 77 AHRS was removed and/or replaced. The mounting tray was NOT removed and the mounting tray bolts were NOT loosened.	None Required. Continue to GRS/GMU Test section.		
GRS 77 AHRS was removed and/or replaced. The mounting tray WAS removed and/or mounting tray bolts WERE loosened.	X	X	X
GRS 77 AHRS Configuration Module was replaced.	X	X	X



GRS/GMU Calibration Page
Figure 21

(3) GRS 77/GMU 44 Magnetometer Calibration - Procedure B

NOTE: The Magnetometer Calibration Procedure must be carried out at a site that is free of magnetic disturbances. If it is unsure whether the site is 'clean', the technician should verify that the site is 'clean' by following the guidance provided. The technician may skip the Compass Rose Evaluation Procedure if the site condition is acceptable.

NOTE: This procedure needs to be performed on a compass rose that has been validated by a site evaluation with regards to magnetic disturbances.

NOTE: Performing this procedure clears any previously stored heading offset values.

NOTE: An Installer Unlock Card (P/N 010-00769-60) is required in order to run the Magnetometer Calibration procedure.

(a) Start the aircraft engines following the procedures referenced in the PA34-220T POH. Apply power to Avionics.

(b) After aircraft engine startup, taxi the aircraft to a properly calibrated compass rose.

(c) At the compass rose, align the aircraft to a heading of magnetic north (±5°).

(d) Ensure an Installer Unlock Card is inserted in the bottom slot of both GDUs.

(e) Restart both GDUs in configuration mode by holding the ENT key on each GDU while closing the associated circuit breaker.

NOTE: For this dual GDU 620 installation, the following calibration procedure will be performed on GDU #1 and GDU #2 simultaneously.

(f) On each GDU, select the GRS Page Group on the MFD, then select the GRS/GMU Calibration page.

(g) On each GDU, activate the cursor and highlight the SELECT PROCEDURE window and, using the MFD small knob, select MAGNETOMETER.

(h) On each GDU, press the ENT button.

(i) On each GDU, use the cursor to highlight the BEFORE CALIBRATION window.

(j) On each GDU, follow the checklist items displayed on the MFD and press the ENT key as each one is completed or confirmed. When the CALIBRATE field is blinking, press the ENT key to begin the procedure.

(k) The MFD display advises the operator when to turn the aircraft, when to stop, and when to turn again.

NOTE: When performing simultaneous calibration of both magnetometers, only follow the turn prompts from GDU #1.

(l) Upon instruction to turn, taxi the aircraft in a right turn. After approximately 25° to 30° of turn from the last heading, the MFD display advises the operator to stop the aircraft.

NOTE: Due to the difficulties in executing smooth, accurate turns, the MFD may incorrectly interpret a station and instruct to "HOLD POSITION" prior to full completion of a 30° turn. If this scenario is encountered, it is best for the operator to ignore the "HOLD POSITION" command and instead use outside references to complete the approximate 30° turn. Instead of using the MFD instruction to turn as a real-time indication of when to turn, simply judge the 30° (±5°) turn increments of the aircraft by using the compass rose radials. Dwelling at these 30° increments for the time recommended by the MFD should result in successful calibration.

(m) The MFD guides the operator to turn and then hold the aircraft at increments of 30 degrees around a complete 360 degree circle.

NOTE: Due to high winds or excessive airframe vibration, the operator may encounter a condition where the MFD restarts the 18-second countdown without full completion of the previous countdown. If this is encountered more than once for a given station, the operator should begin turning to the next station (approximately 30°). A minimum of two (2) successful stations per quadrant is required, where a successful station is a full 18-second countdown followed by instruction to move. Ensure that at least two (2) stations per quadrant are completed. Thus, it may sometimes be required to dwell at a station after a countdown restart. A maximum of 20 stations is allowed for the entire calibration procedure. If too many countdown restarts are encountered, the calibration will fail with the message, “TOO MANY STATIONS.”

(n) When this procedure is annunciated as successfully completed, press the ENT key for that MFD. This must be performed on both GDUs.

(o) If both GMU #1 and GMU #2 calibrations fail, repeat this procedure. If only one GMU calibration fails, repeat this procedure only for the failed side.

NOTE: This procedure should be repeated up to three (3) times prior to taking subsequent action.

(4) Compass Swing

NOTE: This procedure needs to be performed on a compass rose that has been validated by a site evaluation with regards to magnetic disturbances.

NOTE: The Magnetometer Calibration procedure must be completed as described above prior to performing this procedure.

(a) Aircraft engines running. All electrical and Avionics equipment powered and functioning normally.

(b) On the compass rose, align the nose of the aircraft to a heading of magnetic north (±5°).

(c) Perform a compass swing of the aircraft on the compass rose, while recording the heading displayed on PFD1, PFD2, and the standby compass for each of the compass rose headings in the table below.

NOTE: Use the standby compass values to complete the compass calibration card.

ROSE HEADING (A)	PFD1 VALUE (B)	PFD1 ERROR (A-B)	PFD2 VALUE (C)	PFD2 ERROR (A-C)	STANDBY COMPASS
360°	_____	_____	_____	_____	_____
30°	_____	_____	_____	_____	_____
60°	_____	_____	_____	_____	_____
90°	_____	_____	_____	_____	_____
120°	_____	_____	_____	_____	_____
150°	_____	_____	_____	_____	_____
180°	_____	_____	_____	_____	_____
210°	_____	_____	_____	_____	_____
240°	_____	_____	_____	_____	_____
270°	_____	_____	_____	_____	_____
300°	_____	_____	_____	_____	_____
330°	_____	_____	_____	_____	_____

(d) Calculate the heading error by subtracting the displayed value (B) for GDU #1 from the actual value (A), and the displayed value (C) for GDU #2 from the actual value (A), for each heading in the table.

NOTE: Each of the following steps must be completed independently for each AHRS. PFD1 errors apply to AHRS1, and PFD2 errors apply to AHRS2.

(e) If all heading errors are between -3° and +3° inclusive, the AHRS installation is acceptable and the Heading Offset Compensation procedure is not required.

(f) If at least one heading error is greater than 3 degrees, but all heading errors are between -5° and +5° inclusive, the Heading Offset Compensation procedure must be performed.

(g) If any heading error is greater than 5 degrees, the installation must be physically modified.

(5) Heading Offset Compensation

NOTE: The Magnetometer Calibration procedure must be completed as described above prior to performing this procedure.

NOTE: Necessity of application of this procedure is determined by the results of the Compass Swing procedure above. Apply this procedure to AHRS1 using GDU1 and to AHRS2 using GDU2.

NOTE: This procedure needs to be performed on a compass rose that has been validated by a site evaluation with regards to magnetic disturbances.

NOTE: An Installer Unlock Card (P/N 010-00769-60) is required in order to run the Magnetometer Calibration procedure.

(a) Aircraft engines running. All electrical and Avionics equipment powered and functioning normally.

(b) Ensure an Installer Unlock Card is inserted in the bottom slot of the GDU.

(c) Restart the GDU in configuration mode by holding the ENT key while closing the associated circuit breaker.

(d) On the GDU, select the GRS Page Group on the MFD, then select the GRS/GMU Calibration page.

(e) Select the HEADING OFFSET procedure and press the ENT button.

(f) Follow the checklist items displayed on the MFD and press the ENT key as each one is completed or confirmed. When the CALIBRATE field is blinking, press the ENT key to begin the procedure.

(g) Align the aircraft on the compass rose as instructed on the MFD. The procedure uses headings of 360, 90, 180, and 270 degrees. Press the ENT key to confirm each heading.

(h) When the procedure is successfully completed, the CALIBRATION SUCCESSFUL message will flash. Press the ENT key on the GDU to conclude the procedure.

(6) AHRS Engine Run-Up Vibration Test - Procedure C

(a) Aircraft engines running. All electrical and Avionics equipment powered and functioning normally.

(b) GDU #1 and GDU #2 running in configuration mode, perform the following steps on each GDU.

(c) On the GDU, select the GRS Page Group on the MFD, then select the GRS/GMU Calibration page.

(d) Initiate the AHRS engine run-up vibration test procedure by performing the following steps:

1) Select the ENGINE RUN-UP TEST procedure and press the ENT key.

2) Follow the checklist items displayed on the MFD, and press the ENT key as each one is completed or confirmed. When the CALIBRATE field is blinking, press the ENT key to begin the procedure.

(e) The MFD display instructs the operator to gradually increase power from idle to full throttle and back to idle over a period of 1–2 minutes.

(f) When the engine run-up is completed and the engine is back to an idle setting, press the ENT key to confirm completion of the procedure. When this is done, the TEST COMPLETE field stops blinking.

(g) The MFD informs the operator if the installation has passed or failed the vibration test. If the test fails, the specific measurements causing the failure are identified and associated numeric values are displayed on the MFD.

NOTE: Should a failure occur, the technician may perform the Engine Run-up test up to three (3) times successively before corrective action must be taken. If the test does not pass after three attempts, then the installation should not be considered reliable until the source of the vibration problem is identified and remedied. In the event of repeated failure of the engine run-up test, record the values that are reported to be out of range for future reference.

The following are potential causes for failure of the engine run-up test:

1) Vibration motion of GRS and/or GMU caused by neighboring equipment and/or supports.

2) Mounting screws and other hardware for GRS and/or GMU not firmly attached.

3) GRS connector not firmly attached to unit.

4) Cabling leading to GRS or GMU not firmly secured to supporting structure.

5) An engine / propeller that is significantly out of balance.

(h) Press the ENT key on the MFD to conclude this procedure.

(7) Calibration Procedure / Magnetometer Interference Test

NOTE: An Installer Unlock Card is not required in order to run the Magnetometer Interference Test.

NOTE: External power cart and other aircraft need to be as far away from the test aircraft as possible.

A magnetometer interference test is available for troubleshooting and/or verifying a magnetically 'clean' installation of the GMU 44. This test exercises various devices on the aircraft that could potentially affect the magnetic field as measured by the GMU 44 (examples include navigation lights, flap motor, etc).

NOTE: This test is used to validate that no electronic device interferes with the operation of the GMU 44 magnetometer. It is highly recommended that this test be performed after installation or maintenance of electrical components on the aircraft and/or for troubleshooting the GMU 44.

(a) All electrical and Avionics equipment powered and functioning normally.
   1) NAV lights OFF
   2) Anti-collision lights OFF
   3) Wingtip Landing lights OFF
   4) Pull ADC1 CB
   5) Stall Warn CB OFF
   6) Aileron centered.
   7) Flaps retracted to zero (clean).

(b) Restart GDU #1 in configuration mode by holding the ENT key while closing the PFD1 circuit breaker.

(c) On the GDU #1, select the GRS Page Group, then select the GRS/GMU Calibration page.



Magnetometer Interference Test
Figure 22

(d) Select MAG INTERFERENCE TEST (“Figure 22” on page 342358) and press the ENT key.

(e) Follow the checklist items displayed on the MFD, and press the ENT key as each one is completed or confirmed.

NOTE: The third item on the checklist instructs the operator to “prepare a detailed test sequence with precise start and stop times for exercising all electronic devices”. The list of relevant electronic devices are given in “Chart 9” on page 342360. Begin test with flaps retracted, flight controls in a neutral position, all lights selected OFF.

(f) When the CALIBRATE field is blinking, press the ENT key to begin the procedure. A CALIBRATION PROCEDURE window will appear. Have a stopwatch ready to begin recording the elapsed time.

(g) The operator should carry out the actions called for in the prepared test sequence.

NOTE: All actions are carried out in the order and at the precise elapsed time as specified in the prepared test sequence.

(h) After the “Begin test sequence” message appears, wait approximately five seconds and perform each of the actions listed in “Chart 9” on page 342360.

(i) When the test is complete, wait approximately five seconds and press the ENT key to confirm completion of the test. When this is done, the TEST COMPLETE annunciation stops blinking.

(j) The MFD will display if the corresponding AHRS installation has passed or failed the magnetometer interference test.

NOTE: If the test fails, record the three maximum deviation values and their corresponding timestamps. A maximum deviation value greater than 5.0 milligauss in either the X or Y axes, or greater than 8.0 milligauss in the Z axis indicates a problem that must be resolved. Compare the corresponding timestamps during the exceedance with the test sequence to identify which action produced the problem and contact Piper.

(k) Repeats steps (b)–(j) for AHRS2 using GDU2. Reset power using PFD2 circuit breaker.

(l) Press the ENT key on the MFD to conclude this procedure.

(8) GDL 69A Test

(a) If an XM Satellite Radio subscription has already been activated for the GDL 69A, then power up the G600 and go to the AUX – XM INFORMATION page and verify that the GDL 69A is working properly.

(b) If the GDL 69A is replaced (new unit), the owner must re-activate the subscription, using the new GDL 69A’s Radio ID number(s).

(c) As a final operations check, make sure there are no MANIFEST errors shown for the GDL 69A.

CHART 9
MAGNETOMETER INTERFERENCE TEST

Elapsed Time (since start of test (minutes:seconds))	Action
0:00	Test begins (calibration - no activity)
0:20	Calibration ends
0:30	Navigation lights ON
0:40	Navigation lights OFF
0:50	Anti-collision lights ON
1:00	Anti-collision lights OFF
1:10	Wingtip Landing lights ON
1:20	Wingtip Landing lights OFF
1:30	ADC ON (1)
1:40	ADC OFF (1)
1:50	Aileron full left
2:00	Aileron centered
2:10	Aileron full right
2:20	Pitot Heat ON (2)
2:30	Pitot Heat OFF (2)
2:40	Flaps 10 degrees
2:50	Flaps 25 degrees
3:00	Flaps 40 degrees (full)
3:10	Flaps 25 degrees
3:20	Flaps 10 degrees
3:30	Flaps clean
3:40	Air Cond & Fan ON (condenser door open)
3:50	Air Cond & Fan OFF (condenser door closed)
4:00	End of Test

(1) Use ADC1 when testing AHRS1, ADC2 when testing AHRS2.
(2) Only required when testing AHRS1.

M. Post-Installation Procedures

(1) Air Data Power-Up Test

(a) Ensure PFD1, PFD2, ADC1, and ADC2 circuit breakers are pushed in.

(b) Apply electrical power to Avionics.

(c) Verify all self-tests pass on both GDU main startup screens.

NOTE: Passage of self-test is indicated by a lack of flags on the PFD and a lack of alert messages in the alert window on the MFD.

(d) Verify that the OAT readouts on GDU1, GDU2, and DDMP are equivalent to ambient temperature.

NOTE: Ambient temperature should be verified with the aircraft removed from direct sunlight or other sources of radiant heat. A variance of ±3 degrees Celsius is normal.

(2) Airspeed Test

Perform this test using a pitot-static test set.

NOTE: When conducting airspeed tests it is possible to induce attitude and/or heading errors on the GDU 620 display. This is normal operation.

(a) Apply electrical power to Avionics.

(b) Increase airspeed until PFD1 airspeed pointer is equivalent to the values noted below.

(c) Verify that the airspeed on PFD1 is equivalent to the airspeed on the pitot static test set within the tolerance specified in the values noted below.

(d) Verify that the airspeed on PFD2 and the standby airspeed indicator are equivalent to the airspeed on PFD1 within the tolerance specified in the values noted below.

(e) Repeat steps (b) thru (d) for each of the values noted below.

AIRSPEED VALUE	PFD1 ERROR TOLERANCE	PFD2 AND STBY ERROR TOLERANCE
VSO (61 knots) Bottom of white arc	±5 knots	±5 knots
VMCA (66 knots) Lower red radial	±4 knots	±5 knots
VS1 (67 knots) Bottom of green arc	±4 knots	±5 knots
VYSE (88 knots) Blue radial	±3 knots	±5 knots
VFE (113 knots) Top of white arc	±2 knots	±5 knots
VNO (164 knots) Bottom of yellow arc	±2 knots	±5 knots
VNE (204 knots) Upper red radial	±2 knots	±5 knots

(f) Perform the following pitot system leak check:

1) Set test set to VNE (204 knots).

2) Wait 15 seconds to allow the aircraft airspeed indicator to stabilize.

3) Observe the simulator and aircraft airspeed indicators for 15 seconds. If a leak is present, the indicator needles will move toward zero. Maximum acceptable leak rate is one knot in 15 seconds.

4) If a leak rate of greater than one knot in 15 seconds is present, check fixture installation, hose connections, and pitot system lines and fittings. Repair leak when found, and repeat steps (b) thru (f).

(g) Decrease airspeed until PFD1 airspeed pointer is equivalent to the values noted below.

(h) Verify that the airspeed on PFD1 is equivalent to the airspeed on the pitot static test set within the tolerance specified in the values noted below.

(i) Verify that the airspeed on PFD2 and the standby airspeed indicator are equivalent to the airspeed on PFD1 within the tolerance specified in the values noted below.

(j) Repeat steps (g) thru (i) for each of the values in the table below.

AIRSPEED VALUE	PFD1 ERROR TOLERANCE	PFD2 AND STBY ERROR TOLERANCE
180	±2 knots	±5 knots
150	±2 knots	±5 knots
120	±2 knots	±5 knots
100	±2 knots	±5 knots
80	±3.5 knots	±5 knots
50	±5 knots	±5 knots

(3) Altitude Test

This procedure constitutes compliance with FAR 91.411 and Part 43, Appendix E.

NOTE: Pitot-static test equipment is required for this procedure.

NOTE: When conducting altitude tests it is possible to induce attitude and/or heading errors on the GDU-620 display. This is normal operation.

(a) Select primary static source.

(b) Apply electrical power to Avionics.

(c) Select 29.92 In·Hg using baro-set control.

(d) Record the indicated altitude reading on each altimeter.

NOTE: This includes altimeters on PFD1, PFD2, and Standby altimeter.

(e) Scale Error and Transponder

1) Using test equipment, modify the static pressure to obtain an altitude of -1000 feet (as indicated on test equipment).

2) Hold altitude for at least 1 minute, but not more than 10 minutes.

3) Verify that the difference between test equipment altitude and each altimeter does not exceed the values defined in the table below.

NOTE: This includes altimeters on PFD1, PFD2, and Standby altimeter.

4) Verify that the difference between the indicated altitude on PFD1 and the Transponder altitude is <125 feet.

5) Using test equipment, modify the static pressure for each of the altitudes in the table below and repeat steps 2) thru 4).

NOTE: Perform “Mode S Transponder Ground Tests” on page 345021 in parallel with this step.

NOTE: Rate of change not to exceed 20,000 feet per minute between altitudes. Begin step 1) under “(f) Hysteresis”, below, no more than 15 minutes after completing maximum altitude in table.

Altitude	Equivalent Pressure (In·Hg)	Tolerance (feet)
-1,000	31.018	± 20
0	29.921	± 20
500	29.385	± 20
1,000	28.856	± 20
1,500	28.335	± 25
2,000	27.821	± 30
3,000	26.817	± 30
4,000	25.842	± 35
6,000	23.978	± 40
8,000	22.225	± 60
10,000	20.577	± 80
12,000	19.029	± 90
14,000	17.577	± 100
16,000	16.216	± 110
18,000	14.942	± 120
20,000	13.750	± 130
22,000	12.636	± 140
25,000	11.104	± 155

(f) Hysteresis

1) Using test equipment, decrease altitude to 14,000 feet (as indicated on test equipment).

NOTE: Rate of altitude decrease to be between 5,000–20,000 feet per minute until reaching 17,000 feet, then at a rate of 3,000 feet per minute.

2) Hold altitude for 5 minutes (but not more than 15 minutes).

3) Verify that the difference between test equipment altitude and each altimeter does not exceed ±75 feet.

NOTE: This includes altimeters on PFD1, PFD2, and Standby altimeter.

4) Using test equipment, decrease altitude to 10,000 feet (as indicated on test equipment).

NOTE: Rate of altitude decrease to be 3,000 feet per minute.

5) Hold altitude for 1 minute (but not more than 10 minutes).

6) Verify that the difference between test equipment altitude and each altimeter does not exceed ±75 feet.

NOTE: This includes altimeters on PFD1, PFD2, and Standby altimeter.

(g) Leak Test

1) Select airport ambient barometric pressure using the baro-set control.

2) Using test equipment, apply a vacuum on the aircraft static system to obtain a reading of 1,050 feet on the standby altimeter. Turn off the vacuum pump and wait for a period of 1 minute.

3) Verify that the loss of indicated altitude does not exceed 100 feet.

(4) Audio System Post-Installation Test

(a) Apply electrical power to Avionics.

(b) Verify Pilot and Co-Pilot intercom operation (transmit and receive).

(c) Select PA.

(d) Press pilot's Mic key, speak into pilot's headset, and verify audio is heard over the cockpit speaker.

(e) Press co-pilot's Mic key, speak into co-pilot's headset, and verify audio is heard over the cockpit speaker.

(5) Marker Beacon Test

NOTE: This procedure requires the use of a Marker Beacon RF test set.

(a) Apply electrical power to Avionics.

(b) Select Marker audio by pressing the MKR/Mute button.

(c) Apply 400 Hz signal from test set.

(d) Verify Outer marker audio is audible in pilot headset and Blue marker lamp illuminated.

(e) Apply 1300 Hz signal from test set.

(f) Verify Middle marker audio is audible in pilot headset and Amber marker lamp illuminated.

(g) Apply 3000 Hz signal from test set.

(h) Verify Inner marker audio is audible in pilot headset and White marker lamp illuminated.

(6) Audio System Post-Interior Installation Test

(a) Apply electrical power to Avionics.

(b) Connect hand microphone to pilot's Mic jack.

(c) Verify interphone operational using hand mic.

(d) Verify Passenger intercom operation at all passenger jacks (transmit and receive).

(e) Verify MP3 player input is functional from passenger 1 station (stereo operation).

NOTE: Passenger music input cannot be heard by the pilots.

(7) GNS 430W COM/NAV/GPS Post-Installation information

See "Post Installation Setup Procedure" on page 34501.

(8) Autopilot Post-Installation information

See 22-10-00, S-TEC System 55 / 55X, Post-Maintenance Operational Checkout, Garmin G600 Option.

(9) GTX 330 Transponder Post-Installation information

See "Post Installation Setup Procedure" on page 345015.

(10) Traffic Information System Post-Installation information

See "Post Installation Setup Procedure" on page 345015.

(11) Chartview Post-Installation Test

ChartView must be enabled using a ChartView Enable Card, and a current ChartView database. Reference “Chartview Activation (Optional)” on page 342346 for enabling procedures.

NOTE: The required ChartView databases are subscription-based and are to be procured by the aircraft owner directly from Jeppesen.

(a) Ensure Charts database has been loaded per ChartView Loading Procedure.
(b) Apply electrical power to Avionics.
(c) On GDU #1, select Flight Plan page group using large MFD knob.
(d) Verify selection of Charts page is provided via the small MFD knob.
(e) Press the SELECT soft key and enter KATL.
(f) Select an approach chart from the list.
(g) Verify the selected chart is displayed.
(h) Repeat steps (b) thru (g) using GDU #2.

(12) TAWS Post-Installation Test

(a) Ensure that terrain data base cards P/N 010-00769-43 are installed in the bottom card slot in each GDU.
(b) Apply electrical power to Avionics.
(c) Wait for the G600 system to initialize (attitude, heading, airspeed, altitude and GPS position become valid).
(d) On the MFD, go to the TAWS-B map in the Map Page Group.

NOTE: If the TAWS-B map is not available (i.e., the terrain map is labeled “TERRAIN PROXIMITY” or “TERRAIN – SVT”) then the TAWS option has not been enabled.

(e) Press the MNU key, select “Test TAWS” option, and press the ENT key to initiate the TAWS test.
(f) After the TAWS self-test has completed (5-10 seconds), verify that “TAWS System Test Okay” is heard over the cockpit speaker.

NOTE: The aural message “TAWS System Failure” will be annunciated if the TAWS system is NOT functioning properly. If an audio message is not heard following the test, then a fault exists and the TAWS capability must be considered non-functional.

(13) SVT Post-Installation Checkout

NOTE: Synthetic Vision requires a valid GPS signal.

(a) Ensure that database cards P/N 010-00769-43 are installed in the bottom card slot in each GDU.
(b) Apply electrical power to Avionics.
(c) Wait for the G600 system to initialize (attitude, heading, airspeed, altitude and GPS position become valid).
(d) On the GDU #1, press the PFD soft key, followed by the SYN VIS soft key.
(e) Press the SYN TERR soft key to turn on synthetic vision.

NOTE: It may take up to a minute for the synthetic terrain data to be displayed on the PFD. Until SVT is active, the horizon display will be the standard blue over brown.

(f) Verify that there are no SVT failure alerts.
(g) Verify that synthetic terrain data is displayed on the PFD.
(h) Repeat steps (d) thru (g) using GDU #2.

(14) GWX 68 Post-Installation Test

NOTE: Before energizing the equipment, be sure microwave radiation safety precautions including both fuel and personnel safety considerations have been observed. These include clearing all personnel to an area beyond the maximum permissible exposure level (MPEL) boundary. The MPEL for the GWX 68 is 11 feet.

(a) Apply electrical power to Avionics. Wait for AHRS alignment to complete.
(b) On both GDUs, select the WEATHER RADAR page in the WX page group
(c) On GDU #1, press the MODE soft key and then select the STANDBY mode and wait for the Warm-Up to complete.
(d) Select the WEATHER soft key.
(e) Verify that the GWX 68 completes full sweeps without contacting aircraft structure or wiring.
(f) Select maximum Tilt angle UP.
(g) Verify that the GWX 68 completes full sweeps without contacting aircraft structure or wiring.
(h) Select maximum Tilt angle DOWN.
(i) Verify that the GWX 68 completes full sweeps without contacting aircraft structure or wiring.
(j) Press the MNU key and then press ENT to go into the Radar Test Mode.
(k) Verify that a valid test pattern is shown on GDU #1.
(l) Verify that stabilization is ON (STAB ON is displayed in the upper right corner of the MFD).
(m) Using the MODE soft key, set the WXR mode to OFF.

(15) Stormscope Post-Installation Test

(a) Apply electrical power to Avionics. Wait for AHRS to align.
(b) On GNS #1 and GNS #2, Select the NAV page.
(c) Turn the small right knob to select the Lightning (LTNG) Page.
(d) On GNS #1, press MENU to display the Page Menu.
(e) Turn the small right knob to select “Self-Test?” and press ENT.
(f) Verify the Self-Test is displayed as passed on GDU #1 and GDU #2.
(g) On GNS #1, select Demo mode. Wait until several strikes are displayed.

NOTE: Demo mode is not needed if there is sufficient lightning activity in the local line of sight area around the aircraft. Operate the Stormscope in normal mode.

(h) Select the Stormscope clear switch on instrument panel.
(i) Verify Stormscope data is cleared.
(j) Select the STORMSCOPE TEST page.
(k) Verify that the heading shown on the GNS is equivalent to the heading on GDU #1.

(16) Stormscope Antenna Test

This procedure assumes familiarity with the set-up and operation of the WX-PA Stormscope test set.

(a) Apply electrical power to Avionics. Wait for AHRS to align.

(b) On GNS #1, Select the NAV page.

(c) Turn the small right knob to select the Lightning (LTNG) Page.

(d) Set the WX-500 to STRIKE mode; 100 NM range.

(e) Select the Continuous Out mode displayed on the WX-PA menu and press MENU/ENTER.

(f) Select the bottom mount antenna configuration on the WX-PA keyboard.

(g) Set a 90 degree bearing and range of 120 NM.

(h) Use the F1 and F2 keys to adjust range and F3 and F4 keys to adjust heading.

NOTE: The WX-500 will plot data at one-half the range selected on the WX-PA.

(i) On the WX-PA, press MENU/ENTER to start the test.

(j) Verify the following:

- Strike indications displayed ±10 degrees of value selected on test set (i.e., setting 90 degrees, strikes should be indicated between 80-100 degrees).
- Strike indications displayed at range 20% of value selected on test set (i.e., setting 120 NM, strikes should be indicated between 48-72 NM, which is ±20% of 60NM).

(k) Verify after 20 seconds of operation that the strike counter reads 580 ± 40.

(l) Repeat steps (i) thru (k) using the settings in “Chart 10”.

(m) On the WX-PA, after testing for all ranges and bearings indicated, press 2ND, then MENU/ENTER to return to the main menu.

CHART 10

Bearing	Range
90	75
90	30
180	120
180	75
180	30
270	120
270	75
270	30
360	120
360	75
360	30

(17) TAS Post-Installation Test

NOTE: This procedure requires use of the TAS DIAG software provided by Honeywell. Refer to the KTA-870 Installation manual, App. D, regarding usage of TAS DIAG software.

(a) Aircraft on jacks, per 7-10-00.

(b) Pull the TRAFFIC AVOID circuit breaker.

(c) Apply electrical power to Avionics.

(d) Select the Traffic Map page on GDU #1.

(e) Verify that TAS FAIL is displayed in the upper left corner of the Traffic Map.

(f) Close the TAS circuit breaker. Wait for TAS to initialize.

(g) Verify that the amber TAS FAIL is not displayed in the upper left corner, and NO DATA (amber) is not displayed over the ownship symbol.

(h) On the upper left corner of the Traffic Map page, verify that the status of the traffic system is either TAS Standby or TAS operating.

NOTE: TIS should not be displayed.

(i) Verify the Heading Status on TAS DIAG indicates VALID.

(j) With a heading NOT 360 degrees (North), compare the heading indication on GDU #1 with the Magnetic Heading readout displayed via TAS DIAG on the laptop.

(k) Verify the TAS DIAG readout is equivalent to GDU #1 heading display.

(l) Verify the Encoder Altitude Status on TAS DIAG indicates VALID.

NOTE: Precautions must be taken to insure that the altitude interface check does not interfere with other TCAS/TAS equipped aircraft. Shielding the transponder and TAS antennas is the best way to prevent interference with other aircraft.

(m) Select several different barometric settings on GDU #1 while comparing the barometric altitude indication on GDU #1 with the Encoder Altitude readout displayed via TAS DIAG on the laptop.

(n) Verify the TAS DIAG readout is within 50 feet of GDU #1 barometric altitude display.

(o) Swing landing gear between extended and retracted positions.

(p) Verify the TAS DIAG status for gear extension matches actual gear position.

(q) Press the STANDBY soft key and OPERATE soft key to change the TAS operating mode. It may take several seconds for the TAS to change modes.

(r) Verify that the TAS operating mode can be changed from GDU #1.

(s) Put the traffic system in Standby mode.

(t) Select a range of at least 5 NM on the Traffic Map page.

(u) Press the MENU key and press ENT to initiate a traffic system self test.

(v) Verify the following self-test is displayed on the Traffic Map page:

- A Traffic Advisory (yellow circle) at 9 o'clock, 2 NMI 200 feet below, and climbing.
- A proximity traffic (solid white diamond) at 1 o'clock, 3.6 NMI, 1000 feet below, and descending.
- A non-threat traffic (open white diamond) at 11 o'clock, 3.6 NMI, 1000 feet above, and level.

(w) Verify the aural message "TCAS System Test Okay" is clearly audible from the pilot's headset.

(x) Repeat steps (b) thru (n) for GDU #2.

(y) Ensure WOW signal to TAS is airborne.

(z) Select a TAS operating mode to OPERATING.

(aa) Switch WOW signal to on ground.

(ab) Verify that the TAS mode switches to STANDBY.

(18) TAS Ramp Test Mode

NOTE: Precautions must be taken to ensure that this test does not interfere with other TCAS/TAS equipped aircraft.

(a) Disable the WOW logic and put the aircraft systems into the “AIRBORNE” mode.

(b) Landing gear logic in the “EXTENDED” or DOWN position.

(c) Select an active TAS and Transponder mode.

(d) Using TASDIAG, enable the “Ramp Tester Mode” under the “File” drop down menu. This will place the own aircraft at 50,000 ft. and place the TAS into a test mode (as seen on the display).

(e) Using the IFR-201 Ramp Test Set: Block the lower TAS antenna from the test signal.

(f) Create a scenario to simulate an approaching Mode C transponder equipped aircraft beginning at 10 NM, at an altitude of 50,500 feet, at a closure rate of +450 KTS, and an altitude rate of 0 FPM.

(g) Verify that the intruder symbol corresponds to the scenario in bearing, distance and relative altitude.

NOTE: If the installed system is functioning properly, the intruder symbol will begin as an open white diamond with relative altitude tag of + 05. The bearing should be from the direction of the test antenna. The intruder will become proximity traffic at 5 NM; The white diamond will be filled. Finally, the intruder becomes a TA (traffic advisory) and the symbol will change to a yellow circle.

(h) Block the upper TAS antenna from the test signal and unblock the lower antenna.

(i) Repeat steps (f) thru (g), verifying equivalent behavior with the exception of relative bearing information in the Traffic Map display. The Traffic target should be shown as a no bearing target.

(j) After completion of ramp test, disconnect TAS DIAG and Cycle power on the TAS.

(19) TAS Bearing Accuracy Test

NOTE: Whenever checking the TAS bearing accuracy, the test antenna must be within line-of-sight of the directional antenna (top TAS antenna). The test antenna should be elevated slightly above the height of the top TAS antenna. The test set antenna must remain 50-75 feet from the top TAS antenna throughout the test.

(a) Position the aircraft outside away from hangers and other aircraft.

(b) Disable the WOW logic to put the aircraft systems in the “AIRBORNE” mode.

(c) TAS landing gear logic in “EXTENDED” or DOWN position.

(d) Create a scenario with the TCAS ramp test set with an intruder at +6,000 ft relative altitude and a relative bearing from aircraft to test set of 0 degrees.

(e) Verify that the TAS bearing error is ≤ 30 degrees.

NOTE: TAS bearing error is the difference between the relative bearing from TAS directional antenna to test set antenna and the relative bearing shown on the Traffic Map page.

(f) Move the TCAS test set antenna 30 degrees clockwise relative to the TAS directional antenna and repeat step (e).

(g) Repeat step (f) for complete 360 degree coverage.

NOTE: Perform the remainder of this procedure only if the relative bearing check fails.

(h) Ensure WOW to TAS is airborne.

(i) Using the IFR-201 Ramp Test Set, create a scenario to simulate a Mode C transponder equipped aircraft at 2NM, at an altitude of 50,000 feet, and an altitude rate of 0 FPM. Initiate this intruder scenario at the 0° heading.

(j) Using the TAS DIAG program from a laptop, select “Bore Sight Top Antenna” from the “Config Mod” menu.

(k) Select the “Ramp Tester with Target altitude at 50,000” radio button and click OK. The procedure will take a few seconds to complete.

(l) The procedure is successful if “Antenna Calculations Complete” is displayed in the pop-up message box. When successful, skip to step (r).

NOTE: When the Bore Sight procedure is successfully completed, the Delta 1 and Delta 2 offsets for the selected antenna will be stored in the configuration module. TAS will automatically shut down and restart after the procedure has been completed to begin using the new antenna offsets.

(m) If TAS DIAG reports “Bore Sight Failed. Cable calibration failed”, check cable connections from TAS to the antenna. Check TAS fault LED to verify that there are no fatal failures and retry the procedure.

NOTE: Cable calibration occurs automatically during normal operation. If the unit is unable to complete the cable calibration, then a bore sight calibration will not be performed.

(n) If TAS DIAG reports “Bore Sight Failed. The bearing of the reply is outside of the Bore Sight Window”, check antenna orientation. Check Ramp Tester placement and retry the procedure.

NOTE: If too many target bearing readings are outside the ± 45° window, then the bore sight will fail.

(o) If TAS DIAG reports “Bore Sight Failed. The power of the received reply is below the Bore Sight power minimum”, check the Ramp Tester set up and retry the procedure.

NOTE: This error is reported when the power level of the Ramp Tester replies is too low to accurately determine the bearing.

(p) If TAS DIAG reports “Bore Sight Failed. Not enough replies are being received”, Check Ramp Tester operation and placement and retry the procedure.

NOTE: This error will be reported when at least 30 Delta 1 and 2/swap A and B replies are not received.

(q) If TAS DIAG reports “Bore Sight Failed. The Antenna calculations could not be written to the configuration module”, check the wiring and the connection to the configuration module and retry the procedure. Or, try re-initializing the configuration module and retry the procedure.

NOTE: This error will be reported if the bore sight procedure completes, but the antenna Delta offsets cannot be saved to the configuration module.

(r) After the bore sight procedure has successfully completed, repeat steps (d) thru (e) at + 30°, 0°, and -30° relative bearing between top TAS antenna and TCAS test set antenna to confirm that the bore sight procedure has resolved the problem.

(20) ADF Post-Installation Test

(a) Apply electrical power to Avionics.

(b) Select ADF mode on ADF receiver.

(c) On GDU #1, press the PFD soft key on the PFD. Repeatedly press the BRG 1 soft key until ADF is displayed as the source for the No. 1 bearing pointer.

(d) Verify that the ADF window is not invalid (no red ‘X’).

(e) Using an ADF test set, generate a bearing to a station.

(f) Tune the ADF receiver to the same frequency as the test set.

(g) Verify that the ADF bearing pointer indicates the correct relative bearing (±3 degrees).

(h) Verify correct display for three other bearings, at 90° increments.

(i) Tune the ADF receiver to an unused frequency and verify that the bearing pointer is removed from the PFD.

(j) Select ANT mode on ADF receiver.

(k) Verify that bearing pointer display is parked at the 90 degree position relative to the nose of the aircraft.

(l) Repeat steps (b) thru (g) and (i) thru (j) using GDU #2.

(m) Select ADF as a receive input on the audio panel.

(n) Select BFO mode on ADF receiver.

(o) Tune the ADF to local ADF frequency.

(p) Verify that the Morse code identifier for the tuned station is audible from the pilot headset.

(21) DME Post-Installation Test

(a) Apply electrical power to Avionics.

(b) Ensure Transponder and TAS (if installed) are in Standby mode.

(c) Select NAV1 on the KDI-572.

(d) Using the DME test set, verify the DME is tune paired to NAV 1 selected frequency.

(e) Select NAV2 on the KDI-572, using a different frequency.

(f) Using the DME test set, verify the DME is tune paired to NAV 2 selected frequency.

(g) Select Hold on the KDI-572. Change the tuned frequency on NAV2.

(h) Verify the DME remains at the held frequency.

(i) Input slant range on test set.

(j) Verify the slant range displayed on KDI-572 is equivalent to test set.

(k) Select 50% reply on test set. Change slant range on test set.

(l) Verify range displayed on KDI-572 is equivalent to test set.

(m) Select active mode for transponder and TAS.

(n) Verify that DME functionality is not affected by transponder/TAS.

(o) Select DME as a receive input on the audio panel.

(p) Using test set Ident function, verify DME receive audio is functional.

N. Return to Service Procedure

WARNING: IT IS ESSENTIAL THAT THE SOFTWARE VERSIONS BE CHECKED AND MATCHED AGAINST THE LISTED VERSIONS IN THE REQUIRED EQUIPMENT LISTS (REL). SEE "CHART 12" ON PAGE 342376. SOFTWARE CONFIGURATION IS A CRITICAL PART OF THE G600 OPERATION AND MUST BE VERIFIED BEFORE RETURNING AN AIRCRAFT TO SERVICE.

(1) Display System Status

After reinstalling any G600 LRU, verify correct LRU software part numbers and versions against the current numbers and versions shown in the Required Equipment Lists (REL). See "Chart 12" on page 342376.

The G600 system status is tested while operating in the normal mode. If the system is in configuration mode, restart the displays by cycling the PFD1 and PFD2 circuit breakers to start the display in the normal mode.

In the normal operating mode, the MFD System Status page displays a list of LRUs in the LRU INFO window. This AUX group page provides LRU health status by means of a green check or a red 'X'. Data fields that are invalid have large red X's in the STATUS column. A valid field does not display a red X. Allow the displays to initialize for approximately one minute.

Also, LRU software versions are shown along with other database versions and dates. If a red 'X' is shown for an LRU, see the appropriate LRU troubleshooting chart for guidance.

(a) Apply electrical power to Avionics, GDU #1 running in normal mode.

(b) Observe the MFD power-up screen as shown in "Figure 23" on page 342375. Verify that there are no active ALERTS of ADVISORIES.

(c) On the SYSTEM STATUS PAGE (see "Figure 24" on page 342375) in the AUX page group, for the components listed in "Chart 11" on page 342375, verify a green check for status and that a valid serial number (not dashes) appears. Verify the LRU software version matches those shown in the Required Equipment Lists (REL), "Chart 12" on page 342376.

NOTE: If any software version and or part number does not match, or is not successfully loaded, do not continue with post-installation procedures. Re-load software as required. Any previously run setup procedures for an LRU must be performed again after loading software on that LRU.

(d) Press the DBASE soft key to move the cursor to the DATABASE window and verify that each database is current.

(e) Repeat steps (b) thru (d) using GDU #2.

(2) Flight Test

A flight test is recommended after the installation is complete to ensure satisfactory performance of the G600.

(3) VHF COM Tests

Verify the communications capability on both the high and low ends of the VHF COM spectrum.

(4) VOR/ILS TESTS

Select a VOR channel within a 40 nautical mile range. Listen to the VOR audio and verify that no electrical interference such as magneto noise is present. Check the tone identifier filter operation. Fly inbound or outbound on a selected VOR radial and check for proper LEFT / RIGHT, TO / FROM flag indications on the CDI. Check the VOR accuracy.

(5) MAINTENANCE RECORDS

After conducting the function check flight, the airplane may be returned to service. Record the following information in the appropriate airplane maintenance logs:

(a) Part number of the loader card used to perform software loading or software updates.

(b) Any other applicable information related to the maintenance work performed on the airplane.

CHART 11
G600 - RETURN TO SERVICE CHECK LIST

System	Status	SW Ver. OK	System	SW Ver. OK
GDU 1	__	_____		
GDU 2	__	_____		
GDL69A (If installed)	__	_____		
GWX 68 (If installed)	__	_____	GWX FPGA	_____
GDC	__	_____	GDC FPGA	_____
GMU	__	_____	GMU FPGA	_____
GRS	__	_____	GRS FPGA	_____



MFD Initial Power Up Page
Figure 23



System Status Page (Normal Mode)
Figure 24

CHART 12
REQUIRED EQUIPMENT LISTS (REL)

NOTE: Use the version appropriate for the software installed in your airplane.

G600 VER. 942.11
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version
GDU-620	006-B0498-12	3.02
GRS-77	006-B0223-0A	2.12
GDC-74A	006-B0261-12	3.02
GMU-44	006-B0224-00	2.01
GDL-69A	006-B0317-14	3.20
GWX-68	006-B0266-10	2.11
G600 Config File	006-B0942-11	0942.11

G600 VER. 942.22
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version
G600 Config File	006-B0942-22	0942.22
GDU-620	006-B0498-32	5.02
GDC-74A	006-B0261-18	3.08
GDC-74A FPGA	006-C0055-00	01.05
GMU-44	006-B0224-00	2.01
GMU-44 FPGA	006-C0048-00	2.00
GRS-77	006-B0223-22	3.02
GRS-77 FPGA	006-C0049-00	02.00
GDL-69A	006-B1023-01	4.01
GWX-68	006-B0266-10	2.11
GWX-68 FPGA	006-C0042-06	3.01

CHART 12
REQUIRED EQUIPMENT LISTS (REL)

NOTE: Use the version appropriate for the software installed in your airplane.

G600 VER. 942.40
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version
G600 Config File	006-B0942-40	0942.40
GDU-620	006-B0498-32	6.11
GDC-74A	006-B0261-18	3.09
GDC-74A FPGA	006-C0055-00	01.05
GMU-44	006-B0224-00	2.01
GMU-44 FPGA	006-C0048-00	2.00
GRS-77	006-B0223-22	3.03
GRS-77 FPGA	006-C0049-00	02.00
GDL-69A	006-B1023-01	4.01
GWX-68	006-B0266-10	2.11
GWX-68 FPGA	006-C0042-06	3.01

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ASPEN AVIONICS EVOLUTION BACKUP DISPLAY (EBD)

Aspen Avionics Evolution Backup Display (EBD)

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

A. Description

The Aspen Avionics EBD system is an electronic backup display for the Garmin G1000 / G1000 NXi Integrated Avionics Systems. The EBD contains an internal ADAHRS that is used to provide attitude and heading data for the display. The Evolution Backup Display (EBD) as installed by Piper is connected to the emergency power buss and is powered by the emergencey battery to provide a minimum of 30 minute operation in the event of power loss under all foreseeable operating conditions. In addition, the EBD contains an internal battery which will provide an additional nominal 30 minute operation in the event emergency battery exhaustion.

Additional equipment is normally installed in support of the displays, including the Remote Sensor Module (RSM) and Configuration Module (CM).

The EBD system is configured as a basic system in which the EBD provides display of attitude, airspeed, altitude, direction of flight, vertical speed, turn rate, and turn quality. The system provides display of pilot-selectable indices (“bugs”), and annunciations to increase situational awareness and enhance flight safety.

NOTE: The terms EFD1000, EFD, and EBD are used interchangeably.

B. Operation

Whenever indicated airspeed is invalid or below 30 KIAS the EBD will power up and power down with the application or removal of external power. A message is presented during the normal power down sequence to enable the pilot to abort the shutdown and switch to internal battery.

When IAS is greater than 30 KIAS and the input volatage drops below 25.6V the EBD will automatically switch to it’s internal battery (e.g., Aircraft charging system failure.).

C. Troubleshooting

See “Chart 1” on page 34242.

D. Inspections

(1) Annual

All units, brackets, installation hardware and wiring of the EBD system should be checked as defined below during annual inspection. Items found to be defective should be repaired or replaced prior to returning the aircraft to service.

(a) EBD Inspection

Inspect the EBD for damage and operation. Visually inspect the EFD wiring for damage, chafing, or excessive wear. The EBD braided bonding strap should be checked for proper termination at the EBD and aircraft grounding point to maintain HIRF and Lightning compliance.

Verify ≤ 3 milliohms from EBD ground stud to airframe ground. The installation of the EBD should be inspected for corrosion on the EBD and the structure it is mounted on. The fasteners should be inspected for tightness and general condition.

(b) EBD Internal Battery Test

Perform the EBD Internal Battery Test under Ground Test Procedure.

CHART 1 (Sheet 1 of 5)
TROUBLESHOOTING - EBD

Initialization Page Tests

Test	Cause	Remedy
IOP.	a) Fail.	a) Replace EFD1000.
	b) System reboots after IOP test.	b) Replace EFD1000.
Config Module (CM).	a) Fail.	a) Check CM wiring. Replace CM.
	b) Wrong CM version.	b) Install correct SW version CM.
	c) System reboots after CM Test.	c) v2.0 or v2.1 display installed with a v2.2 CM. Install correct CM or EFD.
	d) Displays “Initializing” for more than 20 seconds.	d) CM unplugged or miswired.
RSM.	Fail (x).	Check RSM to EFD wiring for shorts or opens. Repair or replace RSM. Repair or replace EFD1000.
IMU.	Fail.	Replace EFD1000.
ADC.	Fail.	Replace EFD1000.
ADAHRS.	a) Fail.	a) Replace EFD1000.
	b) “Initializing” for more than 3 minutes.	b) Remove Pitot and Static line from back of EFD and reboot. If problem still exists then replace the EFD1000. If problem clears then repair Pitot or Static obstruction/kink.
	c) “Initializing” for more than 3 minutes with a RSM Fail above.	c) Repair RSM wiring or replace RSM.
NOTE: With software version 2.3 and later it is possible to advance the system beyond a failed Initialization Screen by holding the MENU key and line select keys #1 and #2 simultaneously for 3–4 seconds. The Diagnostic screen may then be accessed in the installation menu for further troubleshooting.		

CHART 1 (Sheet 2 of 5)
TROUBLESHOOTING - EBD

System Troubleshooting

Trouble	Cause	Remedy
Display does not power on. (Note: there can be up to a 20 second delay from the application of power to a visible display.)	a) EBD missing A/C power.	a) Check EBD circuit breaker, EBD on/off switch on panel, wiring, and A/C battery voltage > 11.5 volts.
	b) EBD may have been improperly shut down.	b) Switch unit off using “REV” button or “SHUT DOWN” command from Main Menu page 6.
	c) EBD missing A/C ground.	c) Check wiring to EFD1000.
	d) EBD is defective.	d) Repair or replace EFD1000.
Display does not power off. (Note: EBD will switch to internal battery if airspeed is greater than 30kts.)	a) Airspeed is above 30kts.	a) Normal operation.
	b) EBD may have been switched to internal battery.	b) Switch unit off using “REV” button or “SHUT DOWN” command from Main Menu page 6.
	c) EBD may have been improperly shut down.	c) Hold “REV” button for 20 seconds or unplug EBD internal battery for 3 seconds.
	d) EBD is defective	d) Repair or replace EBD.
“CONFIG MODULE LINK FAIL” message.	a) Configuration Module unplugged or miss wired.	a) Check CM plug and wiring from EBD to CM.
	b) Configuration module defective.	b) Repair or replace CM.
	c) EBD defective.	c) Repair or replace EBD..
“RSM LINK FAIL” message.	a) RSM to EBD communication lost.	a) Check RSM to EBD wiring for shorts or opens.
	b) RSM failed.	b) Repair or replace RSM.
	c) EBD failed.	c) Repair or replace EBD
“WRONG CONFIG MODULE” message.	EBD is at one software level and config module is at a different software level.	Convert config module per appropriate service bulletin.

CHART 1 (Sheet 3 of 5)
TROUBLESHOOTING - EBD

System Troubleshooting - continued		
Trouble	**Cause**	**Remedy**
ALTIMETER, AIRSPEED, VSI FAIL (RED-X).	a) Air data sensor has not had sufficient warm-up time.	a) Allow up to 20 minutes at temps below -20ºC for flags to clear.
	b) Pitot/static lines reversed.	b) Connect pitot line to “P” port and static line to “S” port on EBD.
	c) Air data sensor failed.	c) Repair or replace EBD.
Vmo Fail.	Vmo configured incorrectly.	Re-check MOL programming.
ATTITUDE FAIL or DIRECTION FAIL (RED-X). (Note: Attitude flags could take up to 3 minutes to clear at temps below -20 ºC).	a) AHRS sensor has not completed initialization.	a) Allow up to 3 minutes for AHRS to initialize.
	b) RSM failed/data missing.	b) Check RSM to EBD wiring. Repair or replace RSM.
	c) Pitot and/or Static lines crossed, unplugged, or blocked.	c) Correct pitot/static plumbing. issue.
	d) EBD is defective.	d) Repair or replace EBD.
ATTITUDE FAIL and DIRECTION FAIL associated with “CHECK PITOT HEAT” message.	In Flight, Normal if pitot blockage due to ice or other.	Use pitot heat or check pitot system for blockage.
CROSS CHECK ATTITUDE message (yellow) (also see sluggish or poor AHRS (ADI) performance below)	a) If it occurred on system start.	a) RESET AHRS.
	b) Normal after abrupt maneuvers on ground or in air.	b) RESET AHRS.

CHART 1 (Sheet 4 of 5)
TROUBLESHOOTING - EBD

System Troubleshooting - continued

Trouble	Cause	Remedy
"ERRONEOUS CALIBRATION VALUES" message during RSM Cal or Excessive Heading errors in one quadrant, or errors that are higher than actual in some quadrants and lower than actual in other quadrants.	a) RSM is tilted more than: ±4º to the aircraft longitudinal axis; ±10º to the lateral (wings level) axis; ±10º to the zero pitch axis.	a) Shim RSM to within limits stated.
	b) Poor RSM calibration.	b) Re-run RSM calibration at constant rate turns on flat ground.
	c) RSM calibrated too close to buildings or ferrous objects.	c) Re-run RSM calibration away from buildings and other ferrous objects.
	d) Ferrous hardware used to mount RSM.	d) Only non-ferrous screws, nuts, washers may be used on RSM.
	e) Airframe or external magnetic interference.	e) Check for magnetized areas on airframe close to RSM. Verify no ferrous hardware is near RSM. Degauss magnetized area(s).

CHART 1 (Sheet 5 of 5)
TROUBLESHOOTING - EBD

System Troubleshooting - continued		
Trouble	**Cause**	**Remedy**
Sluggish or Poor AHRS (ADI) performance. Poor AHRS performance in steep bank turns. Sluggish compass card. (Note: May or may not be associated with "Cross Check Attitude" message.	a) RSM magnetic interference.	a) Survey RSM location using handheld compass. Move the compass around the location looking for needle deflection. There should be no more than 2º of compass needle movement within an area 18" x 18" around the location. If excessive needle movement is found, degauss the affected area.
	b) RSM has become magnetized.	b) With power removed from EBD system degauss RSM and general area.
	c) "Pitch Attitude Trim" or "Panel Tilt Pitch Compensation" adjustment made without performing a subsequent RSM Calibration.	c) Perform an RSM Calibration.
	d) Pitot and/or Static line connections at EBD blocked, kinked, or unplugged.	d) Check pitot/static connections and plumbing for blockage. Check IAS and ALT sensor per Ground Test Procedure.
	e) Normal after abrupt maneuvers.	e) Perform AHRS Reset.
Excessive Heading Lead / Lag during or after turns (>7°).	Magnetic Interference.	Check for RSM magnetic interference as in "Sluggish or Poor AHRS (ADI) performance", above. If that does not correct the issue, contact Piper Customer Service.

(c) RSM Inspection

Visually inspect the RSM for damage and wear on the bonding strap. RSM wiring should be checked for damage, chafing, or excessive wear. Verify RSM bonding from the ground stud to access plate ground is ≤ 3 milliohms to maintain HIRF and Lightning compliance. The RSM installation should be inspected for corrosion on the RSM and the wing access plate. The installation should be inspected for cracks and loose or damaged fasteners.

(d) Configuration Module Inspection

Visually inspect the Configuration Module(s) for damage. Visually inspect the Configuration Module wiring for damage, chafing, or excessive wear.

(2) Each Twenty-Four (24) Calendar Months

Ensure compliance with the requirements of FAR 43, Appendix E, and FAR 91.411. Use procedures in Aspen Avionics Tech Note for Air Data Calibration, Tech Note 2009-09, Rev. C or later, Document # 991-00023-001, to test and inspect the static pressure system and altimeters.

(3) Each Three Years or 2,200 Hours, Whichever Comes First

Replace the EBD internal battery per EBD Battery Replacement under Components, EBD Display, below.

E. Components

(1) EBD Display

The EBD is a digital system that consists of a high resolution 6” diagonal color LCD display, user controls, photocell and Micro SD data card slot. The rear portion of the EBD includes a non-removable electronics module which contains a full air data computer, attitude heading reference system, power supplies, backup battery, and dual processor electronics. Also on the rear of the unit, a fan is provided to cool the backlight and electronics.

The EBD mounts to the front surface of the instrument panel. The electronics module and cooling fins on the back are sized to fit into instrument panel holes. A recess-mount bracket mounts the displays nearly flush with the instrument panel.

The instrument is installed in a vertically oriented pair of instrument openings, without interfering with the surrounding instruments. The EBD contains a microSD card port and reader at the bottom of the display bezel. When authorized, software updates and system upgrades can be installed using the card port. The port is also used for database information in the EBD MFD.

The EBD is a digital system and supports the ARINC 429 digital interface. The PFD connects directly to the interfaced equipment.

(a) Removal

Verify power is off. Carefully insert a flat blade screw driver into the locking mechanism on the top center of the EFD. While gently prying pull back the top of the EBD and extract from bracket. Remove nut securing braided ground strap to EBD. Remove pitot and static quick connectors (EBD only) by pulling back outer spring loaded locking sleeve while unplugging connectors. To remove 44 pin D-sub connector unscrew both jackscrews fully and pull connector straight back.

(b) Installation

Verify power is off. Install 44 pin D-sub connector and tighten jackscrews until connector is fully seated. Install pitot and static lines (EBD only) to back of EFD by firmly pressing the fitting until fully seated (pitot and static quick connectors are keyed and cannot be crossed). Gently pull on connector to ensure proper connection. Connect braided bonding strap to EBD with nut. Insert bottom of EBD into bracket and pivot top forward until it locks into place on bracket.



Evolution Backup Display (EBD)
Figure 1

(c) Return to Service

1) Verify all system interfaces are functional.

2) Verify proper bonding per “Bonding Check” on page 342413.

3) Perform “System Leak Test” on page 342431.

(d) EBD Battery Replacement

EBD battery replacement must only be performed by a properly certified individual or facility. Remove EFD from panel as above. Remove two screws on each end of the football shaped cover plate on rear of the EBD. Unplug electrical connector and slide battery out of EFD. Install new battery in EBD then connect battery plug. Replace cover plate and tighten the cover screws. Tighten to 12 in·lbs. Reinstall and test EFD as above.

(e) EBD Knob Replacement Procedure

The knob(s) are removed by gripping the knob with the fingers of your hand and then pulling straight off the shaft using moderate force. Install the new knob P/N 520-00020-001 by pushing the knob on to the shaft with the fingers of your hand until it bottoms out. After knob replacement power up the system and verify the knob can turn freely and that the “push to enter” function operates properly by accessing the HDG bug or CRS pointer function.

(2) Remote Sensor Module (RSM)

The RSM is required for each EBD and connects directly to the display. It physically resembles a traditional GPS antenna.

The RSM contains all of the sensors that must be remotely located from the EBD display unit. It is mounted in a magnetically quiet environment. The RSM is installed internally on an access cover on the underside of the aircraft right wing.

The RSM is powered by the EBD through a shielded wire harness and contains the Magnetic “flux” sensors.

(a) Removal

CAUTION: DO NOT USE A MAGNETIC TIPPED SCREW DRIVER WHEN REMOVING AND REPLACING THE RSM.

1) Verify power is off.
2) Remove screws (8 ea.) securing access cover to wing skin. Take care to support access cover as it becomes detached from wing.
3) Disconnect RSM electrical connector from wing wiring harness and remove RSM and access cover from airplane.
4) To remove RSM from mounting bracket, first disconnect ground wire from mounting bracket stud. Then remove four (4) non-ferrous screws, washers, and nut from RSM and separate RSM from mounting bracket taking care to feed wiring pigtail through hole in bracket.

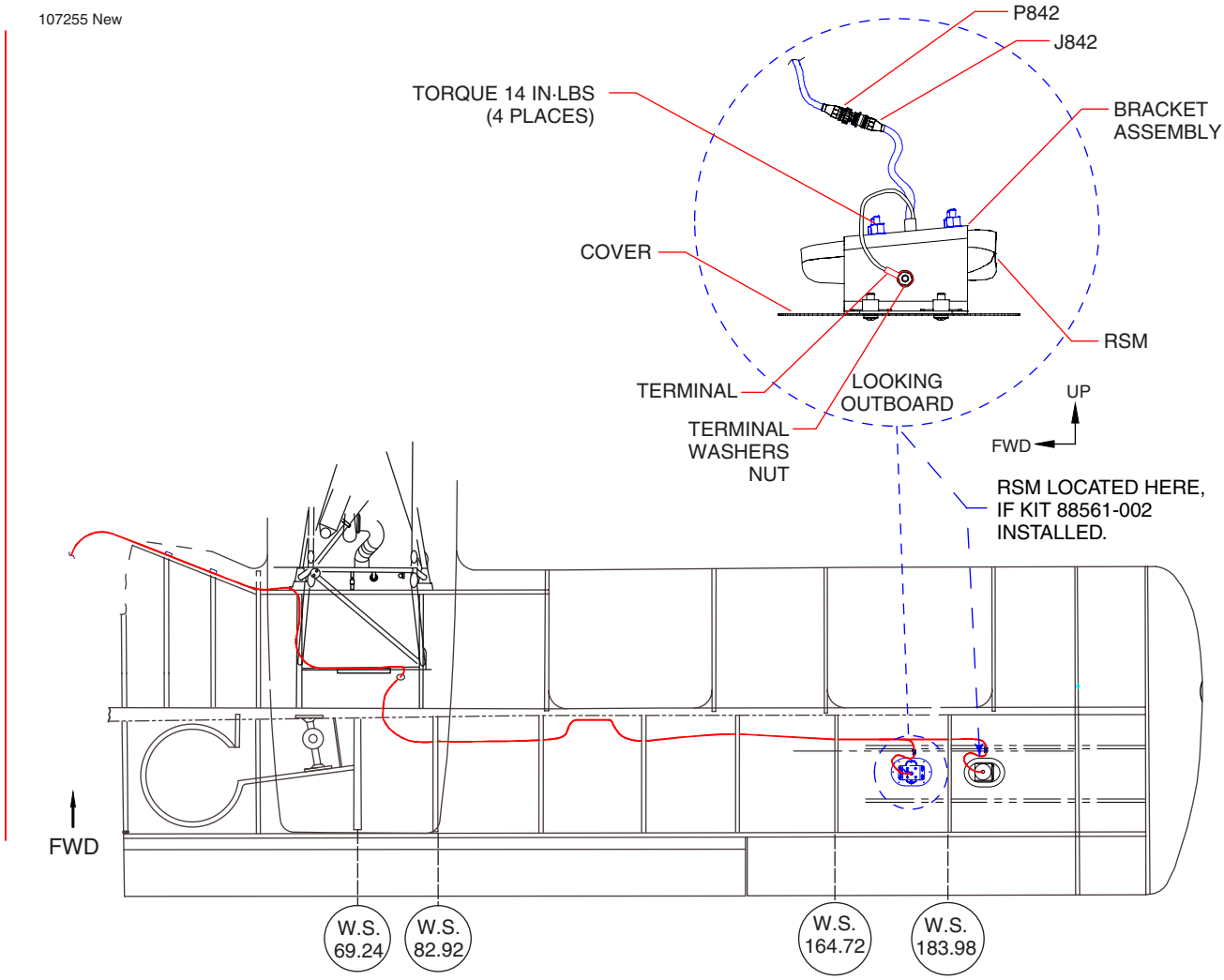
(b) Installation

CAUTION: DO NOT USE A MAGNETIC TIPPED SCREW DRIVER WHEN REMOVING AND REPLACING THE RSM.

1) Verify power is off.
2) If RSM has been separated from its mounting bracket, feed wiring pigtail through hole in bracket and place RSM in bracket with flat end forward (see arrow on mounting bracket). Secure with four (4) non-ferrous screws, washers, and nut. Torque as specified in “Figure 2”.
3) Secure the RSM ground wire to stud on mounting bracket.
4) Connect RSM electrical connector to wing wiring harness.
5) Verify proper bonding per Bonding Check under Post Installation Setup Procedure, below.
6) Place RSM and access cover in position at wing skin opening.
7) Secure access cover to wing skin with non-ferrous screws (8 ea.)

(c) Return to Service

Perform “RSM Calibration” on page 342424.



RHS LOOKING DOWN

RSM Installation
Figure 2

(3) Configuration Module (CM)

The Configuration Module retains system configuration settings and calibration data. The Configuration Module connects to the EBD through a short fabricated harness and is fastened to the main wiring bundle of the display.

Each display has an associated Configuration Module that retains that display's aircraft specific configuration, calibration data and user settings. This permits the display hardware to be replaced without re-entering the installation settings or re-calibrating the EBD.

(a) Removal

Verify power is off. Cut the two (2) cable ties affixing the CM to the PFD wiring harness. Unplug the Molex connector by pressing down on the locking tab and gently pulling the connector from the module.

(b) Installation

Verify power is off. Plug the Molex connector into the module until it clicks. Cable tie the module to the PFD wiring harness.

(c) Return to Service

1) Configure unit per "Installation Menu – Unit Configuration" on page 342414.

2) Perform "RSM Calibration" on page 342424.

F. Post Installation Setup Procedure

(PIR-107821, Rev. B.)

(1) Configuration and Equipment Checkout

The following Test Equipment will be required to complete the remaining steps in the ground test procedure:

- Pitot Static Test Set
- Digital Multimeter
- Electrical Bonding Tester

(2) Bonding Check

(a) Verify EBD mounting bracket is bonded to instrument panel with no greater than 3 milliohms resistance.

(b) Verify RSM base plate is bonded to access cover with no greater than 3 milliohms resistance.

(3) Initial System Turn On

(a) Remove power by pulling applicable circuit breakers.

(b) Install the EBD, RSM, and Configuration Module, if required.

(c) Push in all applicable circuit breakers and apply power.

(d) Verify the EBD displays the initialization page after 5 seconds.

NOTE: AHRS Flags may take up to 3 minutes to clear. Airspeed and Altitude flags may take up to 20 minutes to clear at temperatures below -20ºC.

(4) System Configuration

Configure the EBD system prior to running the ground test procedure. The configuration pages are accessed through the EBD display using the MENU button and the lower Right Control Knob labeled MODE/SYNC.

(a) Main Menu Access

The Main Menu operation is accessed by pushing the “MENU” button. See EBD Pilot Guide for Main Menu operation.

(b) Menu Navigation

When no fields are enabled for editing, rotating the right control knob clockwise advances to the next menu page and counterclockwise advances to a previous menu page.

Editable menu items are displayed in white text on a blue background, non-editable menus items are green text on a blue background while grey text on a blue background is disabled from editing.

(c) Edit Mode

Pushing the line select key adjacent to an editable field enables the associated field for editing. The field turns magenta when enabled and the right control knob reads “Edit Value”.

When the field is enabled for editing rotating the right control knob will adjust the value. Pushing the right control knob or the adjacent line select key will exit from the editable field.

(d) Main Menu Configuration

The Main Menu consists of pages that are pilot selectable. Refer to the EBD Pilot Guide for Main Menu configuration if necessary.

(e) Approved Software

Installation of approved software can be verified as follows:

1) Push the “Menu” key.
2) Rotate the right control knob clockwise until the system status page (“Figure 3”) appears.
3) Compare displayed software versions with “Chart 2” on page 342415.

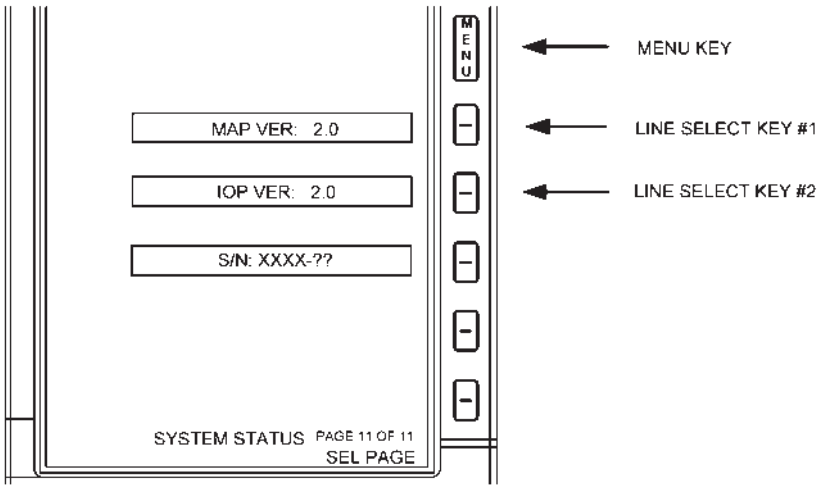
(f) Installation Menu – Unit Configuration

The Installation Menu is entered from the Main Menu’s “SYSTEM STATUS PAGE” (last page). Simultaneously push and hold the MENU key, Line Select Key #1 and Line Select Key #2 for 3 seconds while the airspeed is below 30 units.

Whenever the warning message in “Figure 4” on page 342415 is displayed, pressing either control knob shall advance the Installation menu.

To exit the Installation Menu at any time press the MENU button. All data will be saved as displayed. The system will reboot and “INITIALIZING” will appear on the display for approximately 40 seconds.

Configure the EBD system per the following instructions.



Example Installation Menu Access
Figure 3

CHART 2
APPROVED SOFTWARE

Identifier	Version	Part Number
MAP IOP	2.6.2 (1) 2.0.5	302-00007-015 302-00013-005 (PIR-107757-002, Rev. A.)
MAP IOP	2.6.3 (2) 2.0.5	302-00007-015 302-00013-005 (PIR-107757-002, Rev. B.)
MAP IOP	2.9 (3) 2.1.1	302-00007-026 302-00013-007 (PIR-107757-002, Rev. D.)
MAP IOP	2.10 (4) 2.2	302-00007-200 302-00013-200 (PIR-107757-002, Rev. E.)
MAP IOP	2.10.2 (5) 2.2.1	302-00013-201 B 302-00013-201 A (PIR-107757-002, Rev. F.)
MAP IOP	2.10.2 2.2.1	302-00007-202 C 302-00013-201 A (PIR-107757-002, Rev. G.)

(1) Factory installed in S/N's 3449459, 3449467 through 3449504. All of these S/N's can be upgraded, see Piper Vendor Service Publication 248.

(2) Factory installed through December 2017.

(3) Factory installed December 2017 and later.

(4) Factory installed circa February 2020.

(5) Factory installed circa April 2020.



Installation Menu Warning
Figure 4

(g) EBD Installation Menu Configuration

Installation Date: _____

Aircraft Model: _____ EBD S/N: _____

Aircraft Type: _____ RSM S/N: _____

Aircraft S/N: _____ ACU S/N: _____

CM S/N: _____

1) INSTALLATION MENU PAGE - IAS CONFIG A

Feature	Options	Actual Setting
MOL TYPE	1,2,3,4,5,6	1
MOL PARAM SEL	See instructions in table below	V_{ne}
MOL PARAM VALUE	See instructions in table below	204
MOL MARKER/BAND	BARBER POLE, REDLINE	REDLINE
MACH DSPL TYPE	ALTITUDE, SPEED	SPEED

2) INSTALLATION MENU PAGE – IAS CONFIG B - Set Speed Bands

Feature	Options	Actual Setting
OVERSPEED ALERT	DISABLE, ENABLE	DISABLE
Vno	0 to 450	164
Vfe	0 to 450	113
Vs	0 to 450	67
Vso	0 to 450	61

3) INSTALLATION MENU PAGE – IAS CONFIG C - Set Speed Markers

Feature	Options	Actual Setting
Vyse	0 to 450	88
Vmc	0 to 450	66
Triangle	0 to 450	0
Not Used		
Not Used		

4) INSTALLATION MENU PAGE - IAS CONFIG D - Rotorcraft Only

5) INSTALLATION MENU PAGE - IAS CONFIG E - Color of Airspeed Tape Speed Bands

Feature	Options	Actual Setting
SPD Band 2	YELLOW, CLEAR	YELLOW
SPD Band 3	GREEN, WHITE, CLEAR	GREEN
SPD Band 4	WHITE, CLEAR	WHITE
Not Used		
Not Used		

6) INSTALLATION MENU PAGE – IAS CONFIG F

Feature	Options	Actual Setting
IAS UNITS	KTS, MPH	KTS
TAPES	UNLOCKED, LOCK OFF, LOCK ON	LOCK ON
VSPD EDIT	UNLOCKED, LOCKED	LOCKED
Not Used		
Not Used		

7) INSTALLATION MENU PAGE - IAS CONFIG G

Feature	Options	Actual Setting
IAS DISPLAY	DISABLE, ENABLE	DISABLE
ALT DISPLAY	DISABLE, ENABLE	DISABLE
BARO DISPLAY	DISABLE1, ENABLE 1	ENABLE 1
VSI DISPLAY	ALWAYS ON, ALWAYS OFF, AUTO	ALWAYS ON
Not Used		

8) INSTALLATION MENU PAGE – NAV SETUP A

Feature	Options	Actual Setting
GPS / NAV #1	NONE,A,B,C,D,E,F,G,H,I,J,K,L,M,P,Q,R,S	NONE
GPS / NAV #2	NONE,A,B,C,D,E,F,G,H,I,J,K,L,M	NONE

9) INSTALLATION MENU PAGE – NAV SETUP B

Feature	Options	Actual Setting
429 IN PORT 1	NONE, GPS1, VLOC1, VLOC1+ACU, GPS1+ACU, GPS1+ACU+VLOC1, GPS2, VLOC2, VLOC2+ACU, GPS2+ACU, GPS2+ACU+VLOC2	NONE
429 IN PORT 2	Same as IN PORT 1 options	NONE
429 IN PORT 3	Same as IN PORT 1 options	NONE
429 IN PORT 4	Same as IN PORT 1 options	NONE
429 IN PORT 5	NONE, TRAFFIC	NONE

10) INSTALLATION MENU PAGE – NAV SETUP C

Feature	Options	Actual Setting
429 OUT PORT SPEED	HIGH, LOW	LOW
429 IN PORTS 1 & 2 SPEED	HIGH, LOW	LOW
429 IN PORTS 3 & 4 SPEED	HIGH, LOW	LOW
ADF CONFIG	NONE, 1, 2	NONE
NAV #2 POSITION SOURCE	GPS1, GPS2	GPS1

11) INSTALLATION MENU PAGE – RS232 PORT CONFIG A

Feature	Options	Actual Setting
232 IN PORT 0	NONE, RSM, GPS TYPE 4, GPS TYPE 5, DFC A/P	NONE
232 IN PORT 1	NONE, GPS TYPE 1, GPS TYPE 2, GPS TYPE 3, GPS TYPE 4, GPS TYPE 5, XM, DFC A/P	NONE
232 IN PORT 2	NONE, GPS TYPE 1, GPS TYPE 2, GPS TYPE 3, GPS TYPE 4, GPS TYPE 5, DFC A/P	NONE
232 IN PORT 3	NONE, WX500, GPS TYPE 4, GPS TYPE 5, DFC A/P	NONE
232 IN PORT 4	NONE, EBD MFD, EBD PFD, EFD500 MFD	NONE

12) INSTALLATION MENU PAGE – RS232 PORT CONFIG B

Feature	Options	Actual Setting
232 IN PORT 5	NONE, EBD MFD, EBD PFD, EFD500 MFD	NONE
232 OUT PORT 0	NONE, RSM, GPS TYPE 4, GPS TYPE 5, DFC A/P 1, DFC A/P 2	NONE
232 OUT PORT 1	NONE, XM, ADC TYPE 1, ADC TYPE 2, GPS TYPE 4, GPS TYPE 5, DFC A/P 1, DFC A/P 2	NONE
232 OUT PORT 2	NONE, EFD, ADC TYPE 1, ADC TYPE 2, GPS TYPE 4, GPS TYPE 5, DFC A/P 1, DFC A/P 2	NONE
232 OUT PORT 3	NONE, WX500, ADC TYPE 1, ADC TYPE 2, GPS TYPE 4, GPS TYPE 5, DFC A/P 1, DFC A/P 2	NONE

13) INSTALLATION MENU PAGE – RS232 CONFIG C

This page is grayed out and not configurable.

14) INSTALLATION MENU PAGE – ACU CONFIG A

Feature	Options	Actual Setting
Digital Nav Tx Speed	HIGH. LOW	LOW
Digital Nav Rx Speed	HIGH, LOW	LOW

15) INSTALLATION MENU PAGE – MISC CONFIG A

Feature	Options	Actual Setting
RSM Orientation	TOP, BOTTOM (Inverted orientation)	BOTTOM
RSM GPS Enable	DISABLE, ENABLE, INTERCOM	DISABLE
RSM GPS USAGE	EMER ONLY/ MODE 2	EMER ONLY
RAD ALT CONFIG	NONE, TYPE 1, TYPE 2, TYPE 3, TYPE 4, TYPE 5, TYPE 6, TYPE 7	NONE
WIND DISPLAY	DISABLE, ENABLE>=30KIAS, ENABLE>=40KIAS, ENABLE>=50KIAS, ENABLE>=60KIAS, ENABLE>=70KIAS, ENABLE>=80KIAS, ENABLE>=90KIAS	DISABLE

16) INSTALLATION MENU PAGE – MISC CONFIG B

Feature	Options	Actual Setting
ELEC SYSTEM	14 VOLT, 28 VOLT	14 VOLT
EFD BATTERY CONFG:	INTERNAL, REMOTE	REMOTE*
ATTITUDE REF SYMBOL ADJ:	-5.0 to +5.0 degrees	0
PANEL TILT PITCH ADJ	-10.0 to +20.0 degrees	0
PANEL ROLL ADJ	-2.0 to +2.0 degrees	0

17) INSTALLATION MENU PAGE – MISC CONFIG C

Feature	Options	Actual Setting
COMPOSITE PHASE (VOR1,2)	(0,0) (180,0) (0,180) (180,180)	0,0
GPSS GAIN:	0.5 to 2.0 (0.1 increments)	1.0
CRS SDI	COMMON, NAV 1/2	COMMON
OBS DISPLAY	DISABLE, ENABLE	DISABLE

18) INSTALLATION MENU PAGE – MISC CONFIG D

Feature	Options	Actual Setting
OAT SOURCE	NONE, RSM, PROBE, INTERCOM	NONE
OAT BIAS	0 to -8 degrees	

19) INSTALLATION MENU PAGE – RSM CALIBRATION

The following menu will be used in the next section to calibrate and validate the magnetometer in the RSM. Heading errors of up to +/- 6.0º can be calibrated at 30º increments beginning with North. Follow “RSM Calibration” on page 342424.

Feature	Options
START CALIBRATION	Press to initiate
ACCEPT CALIBRATION?	Press to ACCEPT Cal
REJECT CALIBRATION?	Press to REJECT Cal
HDG SEL	030° to 360° (in 30° increments)
HDG ADJ	-6.0 to +6.0 (degrees)

Use the following table to record the HDG ADJ values used to bring the compass heading values into specification:

Rsm Calibration	Page 12	Options	Actual Setting
HDG SEL: 030	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	_____
HDG SEL: 060	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	_____
HDG SEL: 090	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	_____
HDG SEL: 120	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	_____
HDG SEL: 150	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	_____
HDG SEL: 180	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	_____
HDG SEL: 210	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	_____
HDG SEL: 240	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	_____
HDG SEL: 270	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	_____
HDG SEL: 300	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	_____
HDG SEL: 330	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	_____
HDG SEL: 360	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	_____

20) INSTALLATION MENU PAGE – ACCEL BIAS CAL
This page is for Factory Calibration only.

21) INSTALLATION MENU PAGE – WX-500

Feature	Options	Actual Setting
HEAD TYPE	NONE, DISPLAY, CONTROL	NONE
SYSTEM DATA	PAGE1, PAGE2, PAGE3, PAGE4	
STRIKE TEST	Press to Initiate	
NOISE MONITOR	Press to Initiate	
ANTENNA MOUNT	TOP, BOTTOM	

22) INSTALLATION MENU PAGE – NETWORK PAGE
This menu is for diagnostic purposes only.

Feature	Options
IP ADDR	0-255
SUBNET MASK	0-255
GATEWAY	0-255
PORT	0-9999
Not Used	

Notes: Repeated presses of top 3 line select keys will select one of 4 selectable fields

23) INSTALLATION MENU PAGE - DIAGNOSTICS

This menu is for diagnostic purposes only. See “Figure 5”.

NOTE: Operational Time and Flight Time are stored in the configuration module of the aircraft and these times are cumulative for the software installation not the EBD unit itself.

24) EXITING / SAVING DATA

To exit the Installation Menu press the MENU button. All data will be saved as it was displayed on each page. The system will reboot and “INITIALIZING” will appear on the display for approximately 40 seconds. Normal operation continues.



Diagnostics
Figure 5

(h) IAS Configuration

1) Definitions

V_{ne}: Never Exceed speed (beginning of red band)

V_{no}: Maximum Structural Cruise speed (beginning of yellow band).

V_{fe}: Maximum Flap Extend speed (top of white band)

V_s: No Flap Stall speed (bottom of green band)

V_{so}: Full Flap Stall speed (bottom of white band)

V_{yse}: Single Engine best rate of climb (blue marker)

V_{mc}: Single Engine minimum control speed (red marker)

◁ Initial Flap Extension Speed

2) Use the following table to fill in Configuration Menu Page 1:

INDICATED AIRSPEEDS (IAS) CONFIGURATION

IAS Setting (KIAS)	Band Color	Band Range	Description	Pass
V_{ne} = 204	Red	>V_{ne}	Red arc displayed at all speeds above V_{ne}	_____
V_{no} = 164	Yellow	V_{no} - V_{ne}	Yellow arc extending from V_{no} to V_{ne}	_____
V_s = 67	Green	V_s - V_{no}	Green arc extending from V_s to V_{no}	_____
V_{fe} = 140	White	V_{so} - V_{fe}	Top of White arc	_____
V_{so} = 61	White		Bottom of White Arc	_____
V_{yse} = 88	Blue Marker	= V_{yse}	Blue Marker at V_{yse}	_____
V_{mc} = 66	Red Marker	= V_{mc}	Red Marker at V_{mc}	_____
◁ = 140	Triangle (White)	= ◁	White triangle at initial flap extension airspeed	_____

3) ATTITUDE REF SYMBOL ADJ

This is used to make small changes to the pitch attitude reference mark on the EBD should the aircraft fly consistently slightly nose high or nose low in cruise trim. It is not necessary to perform the “RSM Calibration” on page 342424 after making changes to this setting.

4) EXITING / SAVING DATA

To exit the Installation Menu press the MENU button. All data will be saved as it was displayed on each page. The system will reboot and “INITIALIZING” will appear on the display for approximately 40 seconds. Normal operation continues.

(5) RSM Calibration

(a) Calibration Overview

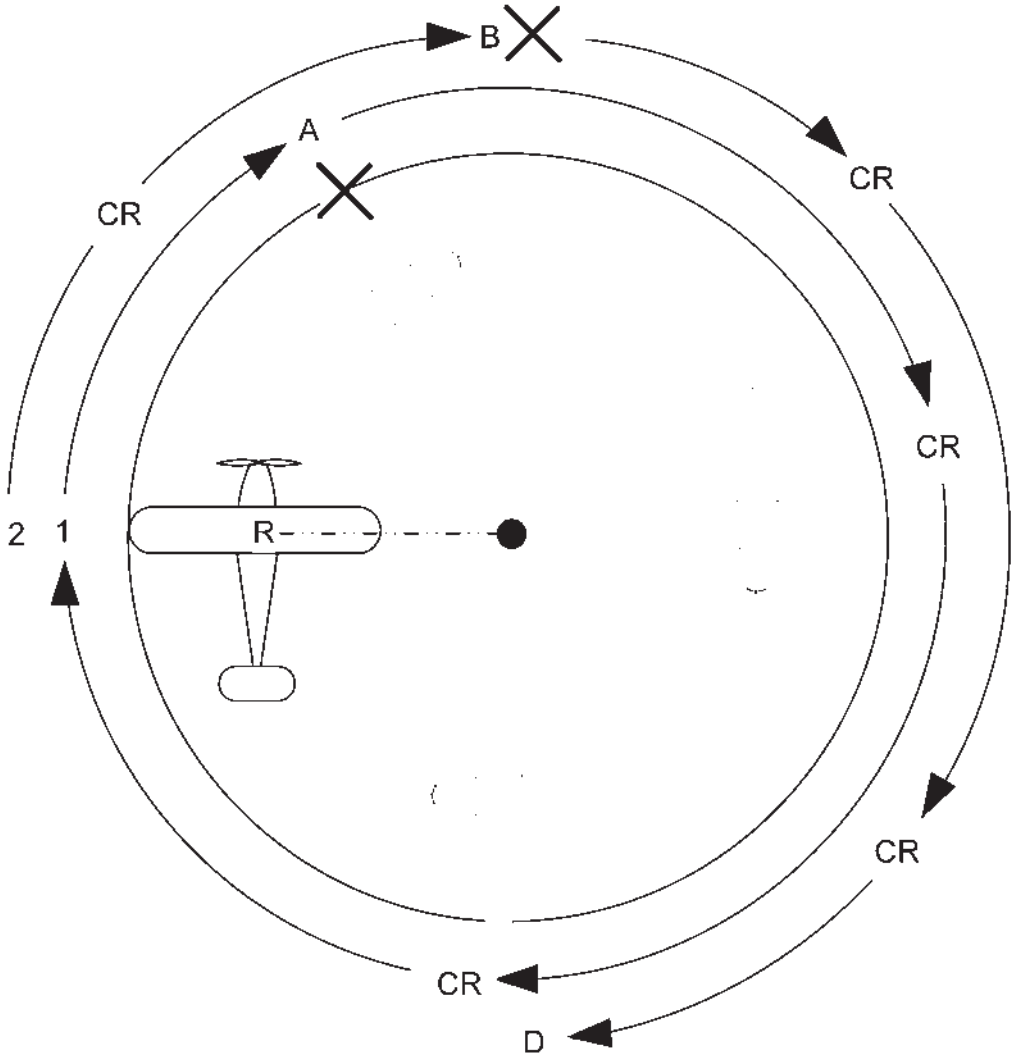
The Remote Sensor Module must be calibrated by performing a compass swing in the aircraft for any maintenance activities that could affect RSM accuracy.

Such activities might include but are not limited to the replacement of the RSM, replacement of the Configuration Module, installation of a mechanical or electrical device in the vicinity to the RSM, installation of an appliance that might generate a magnetic interference.

An overview of the RSM Calibration procedure is as follows (see “Figure 6” on page 342425):

CAUTION: WHEN A CALIBRATION IS INITIATED, THE AIRCRAFT MUST BE TURNED AS DESCRIBED BELOW. IF THE CALIBRATION IS INITIATED AND THEN ACCEPTED WITHOUT MOVING THE AIRCRAFT, AN ERRONEOUS CALIBRATION WILL BE WRITTEN TO THE CONFIGURATION MODULE, POTENTIALLY RESULTING IN FAILURE TO INITIALIZE.

- A heading will be checked to verify the reasonableness of EBD heading prior to calibration.
- The aircraft will be taxied to a magnetically quiet and level area at least 200ft from metal buildings and clear of metal grates, manhole covers and rebar within the concrete. A Compass Rose is ideal for this procedure.
- The aircraft can start from any heading.
- With engines running, all electrical equipment operating, and the aircraft stationary the RSM CAL page will be entered and Start Calibration will be initiated (see “Figure 7” on page 342426).
- After a 15 second countdown timer the pilot/operator will begin to taxi the aircraft in a circle (cw or ccw) with the radius of approximately twice the length of the aircraft’s wing as viewed from the cockpit (≈ 30ft).
- The aircraft will be taxied under its own power at a constant rate around a circle until a 60 second timer elapses. The aircraft must not stop until the timer has exhausted.
- At the completion of the 60 seconds the aircraft will have made at least a 450° circle (360° + 90°) to approximately two complete circles (720°).
- At the end of the 60 second timer four headings about 90° apart will be checked against a calibrated heading source (i.e., site compass, compass rose).
- If EBD heading is acceptable then the calibration is Accepted.
- If the EBD heading is not within tolerance then it is Rejected and the calibration procedure is re-run.
- After the calibration is accepted headings are checked using a calibrated reference (i.e., a sight compass) every 30° (starting from North) to verify that the heading accuracy is within ±4°.
- Heading can be calibrated every 30° if necessary.



● = Imaginary pivot point
R = Radius to pivot point – approximately equal to wingspan (≈30ft)
1 = Starting Point
A = Established Constant Rate turn ≈ 10 seconds after 1
CR = Constant Rate Turn must continue until end of 60sec timer
2 = 360° turn complete – continue at Constant Rate – Do not stop
B = 60sec timer must not reach Zero before this point
D = Accept/Reject CAL message occurs between B & D

RSM Calibration Graphic
Figure 6

(b) Calibration Procedure

The RSM calibration routine is accomplished using the Installation Menu "RSM CALIBRATION" menu page. See "Installation Menu – Unit Configuration" on page 342414 for instructions on entering the INSTALLATION MENU.

On the "RSM CALIBRATION" menu page the current calibrated heading (to the nearest 0.1 degrees) will be continuously displayed adjacent to the "CAL HDG:" menu field.

NOTE: Verify CAL HDG is within 25 degrees of a calibrated compass source prior to starting the RSM Calibration. This is a rough check of the hard and soft iron effects (magnetic interference) in the vicinity of the RSM. Should the CAL HDG be greater than 25 degrees then the RSM location should be re-surveyed.

With aircraft stationary at (POSITION 1) "Figure 6" on page 342425 press the "START CALIBRATION" line select key. The annunciation shown in "Figure 8" on page 342427 will be displayed with a countdown timer that begins with 15 secs and counts down to 0 secs.

CAUTION: DO NOT PRESS "ACCEPT CALIBRATION" WITHOUT MOVING THE AIRCRAFT IN THE PROCEDURE BELOW AS CORRUPTION TO THE CONFIGURATION MODULE COULD OCCUR.



RSM Calibration Page
Figure 7

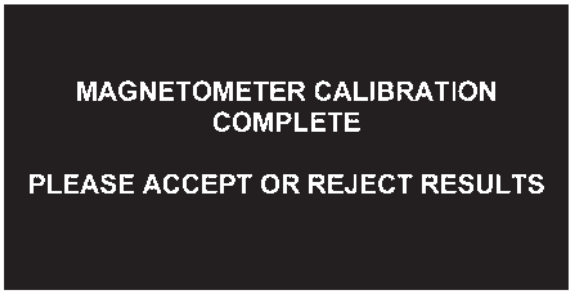


Calibration in Process
Figure 8



Aircraft Turning
Figure 9

When the menu of “Figure 9” is displayed immediately begin taxiing the aircraft clockwise or counter-clockwise at a constant rate of no faster than 1 turn every 30 seconds. About half normal taxi speed or a brisk walking speed is about right. Approximately 10 seconds after initial movement (POSITION A “Figure 6” on page 342425) the aircraft should be taxiing at a constant rate (CR) throughout the rest of the procedure. When the countdown timer is reached between one and a quarter turns (450º) (POSITION B “Figure 6” on page 342425) and two turns (720º) (POSITION D “Figure 6” on page 342425) should have been completed. While turning the aircraft do not stop the aircraft until the end of the 60 second timer and “Figure 10” appears.



Accept/Reject Results
Figure 10

If you find that the timing of the turns was not right such that “Magnetometer Calibration Complete” message occurs between B & D of “Figure 6” on page 342425, then REJECT the results and re-run the procedure.

At the end of the calibration routine the “ACCEPT CALIBRATION?” and “REJECT CALIBRATION?” menu options will be enabled.

NOTE: If the message “Erroneous Calibration Values” is displayed then magnetic interference exists in the vicinity of the calibration area or the RSM is mounted in a magnetically noisy area. Try the calibration process again in a flat magnetically quiet area. If the message is displayed again the RSM location must be surveyed for magnetic interference.

To determine whether to ACCEPT or REJECT the results check four headings approximately 90° apart against a known good heading source (i.e., aircraft compass, sight compass, compass rose).

If the headings are within ±10° then press ACCEPT and use the Heading Offset Adjustment, below, to align each heading value to actual.

NOTE: Typically with a clean RSM location the heading values will be within 3 degrees of actual heading. If they are outside this value the RSM mounting area should be rechecked for magnetic contamination (internal interference) or the calibration can be rerun in a new location (external interference) to see if the values are improved. RSMs tilted to the upper end of the limit may not be able to achieve the 3 degree or less of error without using the Heading Offset Adjustment).

Pressing the “ACCEPT CALIBRATION” selection shall accept the calibration results, display the annunciation shown in “Figure 11” for 5 seconds, and return the “RSM CALIBRATION” menu page to its initial state.

Pressing the “REJECT CALIBRATION” selection shall reject the calibration results. Reject the results if the calibration was poor or a previously stored calibration has better heading accuracies.



Results Accepted
Figure 11



Results Rejected
Figure 12

(c) Heading Offset Adjustment

When the calibration routine is complete and the results have been accepted the CAL HDG value will be displayed.

Position the aircraft so that it is at a heading of 30° as verified by a sight compass or other calibrated means. Press the line select key next to “HDG SEL” and then turn the right knob until HDG SEL: 030° is displayed. Now press the line select key next to “HDG ADJ” and turn the right knob until the CAL HDG of Figure 96 displays exactly 030.0° as shown in “Figure 14” or as close to 030° as possible within a ±4° tolerance. The HDG ADJ field will display the amount of correction (+/- 6.0 degrees) that was required. Press the HDG ADJ line select key to accept the setting.



Calibration Heading before adjustment
Figure 13

Now repeat the process above for all other headings in 30° increments from 60° to 360°. Press the MENU key to exit the Installation Menu. Continue with heading accuracy tests below.

NOTE: In some aircraft, prop wash and wind during ground operations can create inconsistent pressures in the pitot-static system. The pressures can affect the ADAHRS, resulting in small pitch and heading perturbations. Before reading the aircraft headings for the purposes of calibration, ensure the attitude solution has stabilized and is not influenced by external winds and pressures. The disturbances normally settle out over a period of 15 to 60 seconds. Idle power or temporarily selecting the alternate static source can sometimes eliminate the effect.



Calibration Heading after adjustment
Figure 14

(d) Heading Accuracy Test

As a final check, position the aircraft on the headings shown in the **HEADING ACCURACY TEST** table below and verify heading (viewed on HSI display) is within +/- 4 degrees using a calibrated heading source (i.e., sight compass, compass rose).

If any heading is outside ±4° then rerun the Calibration Procedure and or Heading Offset adjustment.

HEADING ACCURACY TEST

Calibrated Heading Source	Tolerance	Actual EFD Heading	Calibrated Heading Source	Tolerance	Actual EFD Heading
30	+/- 4	_____	210	+/- 4	_____
60	+/- 4	_____	240	+/- 4	_____
90	+/- 4	_____	270	+/- 4	_____
120	+/- 4	_____	300	+/- 4	_____
150	+/- 4	_____	330	+/- 4	_____
180	+/- 4	_____	360	+/- 4	_____

(e) Heading Interference Test

With aircraft engine(s) running monitor current aircraft heading on EBD and exercise flight controls stop to stop including flaps and any electric trim tabs. Verify the heading (viewed on HSI display) does not change on the EBD by more than 2°. If movement of flight controls causes more than a 2° heading change then it may be necessary to degauss the flight controls including the cables. A handheld degausser can be found at most audio and video stores.

- Operate all electrical and environmental equipment including:
- Blowers, fans, heaters, air conditioner
- Deice boots, fuel pump(s), backup vacuum pumps
- Landing, logo, NAV lighting
- Operate pulse equipment – transponder, WX radar, DME
- Key all VHF communication radios.
- Operate autopilot so that all servos run (roll, pitch, yaw, trim)

If the operation of any electrical system causes the heading to change by more than 2° the RSM wiring may need to be relocated away from the offending system. The offending system may also have a bonding issue to the airframe that needs to be corrected.

Run engine(s) from idle to take off power and verify that the heading does not change by more than 2°. Prop wash at higher engine RPMs may cause a heading shift, try an alternate static source if this is a issue.

This completes all RSM calibration and heading tests.

(6) Ground Test Procedure

CAUTION: DO NOT EXCEED THE AIRCRAFT'S MAXIMUM AIRSPEED, ALTITUDE, OR VERTICAL SPEED AT ANYTIME DURING THE TESTING. DAMAGE COULD RESULT TO THE PRE-EXISTING AIRCRAFT INSTRUMENTS.

NOTE: When changing indicated airspeed or altitude on the ground using pitot/static test equipment, changes in the AHRS display of pitch, roll and heading will result, possibly accompanied by a CROSS CHECK ATTITUDE annunciation. This behavior is the result of the Kalman Filter algorithms employed in the EBD attitude solution. These changes in pitch, roll or heading are normal and do NOT indicate a system failure. The integrated nature of the EBD AHRS algorithms is such that AHRS performance can only be properly evaluated during flight or ground maneuvers.

(a) Check the following items for proper operation:

1) Airspeed Tape, Altitude Tape
2) AHRS Sensor
3) EMI Test
4) Flight Control Interference Check

(b) Indicated Airspeed Display

Use the Indicated Airspeeds (IAS) Configuration table under IAS Configuration to record the aircraft speed settings from the Aircraft Flight Manual in the IAS Setting column. Set the Pitot/Static test set for 5000 ft above field elevation. Increase airspeed to Vne and check all Speed Bands and Speed Markers listed in table.

(c) Altitude Display

With the Pitot/Static tester still set for 5000 ft above field elevation and with BARO Set to 29.92 in. Hg. on the EFD1000, verify altitude tape displays altitude within ±40ft of the calibrated test set altitude.

(d) System Leak Test

Perform pitot-static system leak "Test" on page 34108 or set the Pitot Static Test Set to 1000 ft above field elevation and without additional pumping for a period of 1 minute the aircraft static system should not lose more than 100 ft of altitude.

(e) AHRS Sensor Test

Verify that correct aircraft attitude information is presented on the Attitude Indicator portion of the EFD1000. The Flags may take up to 5 minutes to clear when the ambient temperature is below -20° C. Typically the attitude solution will be available in less than 3 minutes.

(f) EBD Internal Battery Test

Test the internal back-up battery in the EBD every 12 months. Replace the battery if it fails the following operational test.

1) Setup

This test must be run at room temperature approximately 25º C.

2) Procedure

a) Turn on the EBD.

b) Press MENU Key.

c) Select POWER SETTINGS page from the Main Menu.

d) Press the BATTERY line select key.

e) BAT LEVEL IN --.-- will be displayed for a short period of time as battery capacity is being measured. This could take up to 10 minutes if the ambient temperature is below 0º C.



f) Once the capacity is measured ON BAT XX% REM will be displayed.



g) The “ON BAT” indication must read a minimum of 80% to continue. If the battery capacity is below 80% then the battery should be charged by returning the EBD to external power. The battery will charge as long as the EBD is turned on and external power is supplied.

h) With the battery displaying greater than 80% charge set a timer for 30 minutes. After the 30 minute time has elapsed the EBD must still be operating on battery. If the internal battery will not supply the minimum 30 minutes operating time or fails to charge above 80%, replace the battery and return the failed battery to Aspen Avionics.

i) Instructions for battery replacement are under “EBD Battery Replacement” on page 34249.

j) Switch the EBD back to external power and recharge the internal battery to 80% or greater prior to release to service.

(g) EMI Test

Monitor the EBD for Flags, Red-X's, Red Slashes, heading changes, altitude changes, airspeed changes, attitude changes or any error messages while performing the following Test:

1) Transmit on all Comm radios for 20 seconds each at 118.000MHz, 126.900MHz, and 136.950MHz.
2) Turn on all transponders, DMEs, Wx Radar, and all other pulse type equipment for 20 seconds each.
3) Operate all aircraft lighting including position lights, strobe lights, navigation lights, and all other forms of lighting for 20 seconds each.
4) Operate all environmental equipment including fans, air conditioning, heaters, and all other forms of environmental control equipment for 20 seconds each.
5) Operate Fuel pump(s), deice boots, windshield heat, prop heat, flaps.
6) Operate autopilot, servos, yaw damper, and all electric trim servos.
7) Operate engine(s) and verify no interference.

(h) Flight Control Interference Check

CAUTION: VERIFY CONTROL SURFACES ARE FREE AND CLEAR.

With the EBD mounted in the instrument panel push the control column (yoke or stick) all the way forward (nose down) and verify there is sufficient clearance between the EFD1000, and their knobs, and the control column. With the control column fully forward move it from lock to lock (full right to full left) and verify there is sufficient clearance between all EFD1000, and their knobs, and the control column.

(7) Post Installation Flight Check

CAUTION: ONLY PERFORM FLIGHT TEST IN DAY VFR CONDITIONS WITH AN APPROPRIATELY RATED PILOT FOR THE AIRCRAFT BEING FLOWN.

(a) Basic ADI Flight Checks

Fly the aircraft in straight and level flight and verify that the ADI roll indication is level with reference to the horizon. Observe the Slip Indicator is centered under the Roll pointer and adjust rudder trim if available to center.

1) Make a coordinated 30 degree banked turn to the right and verify that the ADI roll indication is correct with reference to the horizon.
2) Make a coordinated 30 degree banked turn to the left and verify the ADI roll indication is correct with reference to the horizon.
3) Pitch the aircraft up 10 degrees and verify the ADI pitch indication is correct with reference to the horizon.
4) Pitch the aircraft down 10 degrees and verify the ADI pitch indication is correct with reference to the horizon.

(b) Basic HSI/DG Flight Checks

1) Make a 180 degree coordinated turn to the right and verify that the compass scale and numerical heading indication correctly track the aircraft heading during the turn.
2) Make a 180 degree coordinated turn to the left and verify that the compass scale and numerical heading indication correctly track the aircraft heading during the turn.
3) Then turn from West to North (30° Angle of Bank) and, using an outside reference, roll out to a northerly heading. (In the Southern Hemisphere, also turn West to South)
4) Immediately after the aircraft rolls out, record the heading indication.
5) Maintain the same heading by outside visual reference. There may be some movement of the heading indicator as the heading system stabilizes.
6) When the heading stabilizes, record the heading again. Then perform the test from East to North (In the southern Hemisphere, also turn east to South).
7) If the difference between the heading on rollout and the heading after stabilizing is more than 7°, refer to “Chart 1” on page 34242.

(c) Document Test Flight

This completes the flight test. If everything was satisfactory then document the completion of the Test Flight in aircraft log book in accordance with FAR 91.407(b).

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INTEGRATED AVIONICS SYSTEM - GARMIN G1000

Integrated Avionics System - Garmin G1000

S/N's 3449459, 3449467–3449508.

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

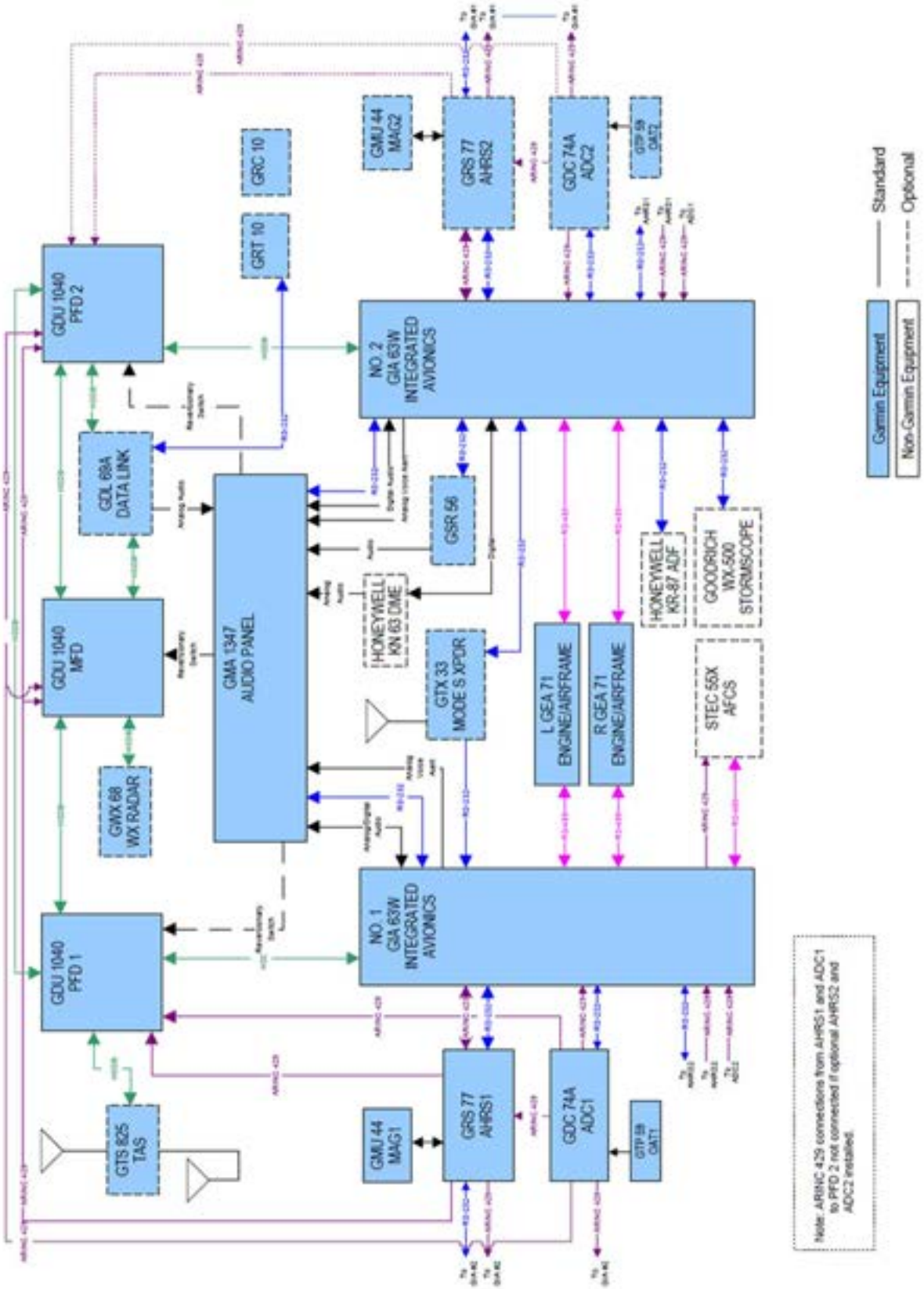
A. Description

The Garmin G1000 Integrated Avionics System provides a fully integrated avionics suite that interfaces with a Garmin GFC 700 Autopilot (or, in earlier installations, a S-TEC System 55X Autopilot).

It consists of three (3) displays, two (2) GDU 1040s dedicated as primary flight displays (PFD1/PFD2) and a GDU 1040 multi-function display (MFD). The PFD displays are located on either side of the MFD, with the stand-by instruments located to the left of the Pilot's PFD (PFD1). Functions provided by the Integrated Avionics System include display of attitude, heading, navigation, traffic (optional), air data, engine and airframe status, and a moving map display with position derived by GPS. In addition to display functions, GPS navigation, VHF/Com, VOR/ILS and transponder functions are provided by the G1000 system. See "Figure 1".



Garmin G1000 Displays
Figure 1



Electrical/Avionics Component Block Diagram
Figure 2

See the latest revision of the Garmin G1000 Pilot's Guide for control and operation. See Vendor Publications under Supplementary Publications in the Introduction.

See "Figure 2" on page 3425012.

(1) Flight Instrumentation

The GRS 77 AHRS(s), GDC 74A Air Data Computer(s), GMU 44 Magnetometer(s), and GTP 59 Temperature Probe(s) are responsible for providing the G1000 system with aircraft attitude, heading, altitude, airspeed, vertical speed, and outside air temperature information, all displayed on the PFD (data is displayed on the MFD in reversionary mode only).

Primary data outputs from the GRS and GDC are sent directly to the PFD1 via ARINC 429. Secondary data paths connect the GRS and GDC to the MFD and PFD2. Additional communications paths connect the GRS and GDC to both GIA 63W units.

The GRS 77 receives GPS data from both GIAs, airspeed data from the GDC, and magnetic heading from the GMU. Using these three external sources, combined with internal sensor data, the GRS accurately calculates aircraft attitude and heading.

(2) Engine Indicator System

The GEA 71 (L/R) provides engine/airframe data to the G1000 system. The unit interfaces to transducers. Analog data is received from the transducers and is converted to a digital signal by the GEA 71. Digital information is then sent through the primary RS-485 serial path to the #1 GIA 63W. From the GIA, data is sent through the HSDB connection to the PFD, then on to the MFD for display. A backup data path from the GEA to the #2 GIA 63W, then on to the MFD, exists in the event the primary path fails.

(3) Communications/Navigation Systems

The GIA 63W IAUs contain VHF COM, VHF NAV, and GPS receivers. COM and NAV audio is sent via digital audio to the GMA 1347 Audio Panel. The GMA 1347 Audio Panel controls the display reversionary mode.

GPS information is sent to the GRS 77 AHRS and to both displays for processing.

The GTX 33 Mode S Transponder communicates with both GIAs in a single transponder installation. Transponder data is sent from the GIAs to the PFD where control and operation occurs.

The #1 GIA outputs analog HSI signals to the S-TEC 55X, along with ARINC 429 roll steering, and altitude data, and digital selected altitude data.

(4) Components Details

See "Components" on page 34250137, for detailed descriptions of the individual G1000 system components/Line Replaceable Units (LRU's).

B. Power Distribution

Distribution of power to the G1000 system occurs on nine aircraft electrical busses. See "Figure 3" on page 3425014 and "Figure 4" on page 3425015.

(1) Main Bus

The Main bus current from the battery is controlled by the Master switch on the overhead panel. The battery feed is protected by a 250 amp current limiter (fuse). Alternator power to the main bus is introduced by Alternator switches on the overhead panel. The two Alternator feeds are protected by two 90 amp circuit breakers.

(2) Essential Buses

The two Essential busses are fed through the Main bus and are protected by 60-amp circuit breakers.

(3) Non-Essential Buses

The two Non-Essential busses are fed through the Main bus and are protected by 35-amp circuit breakers.

(4) Avionics Buses

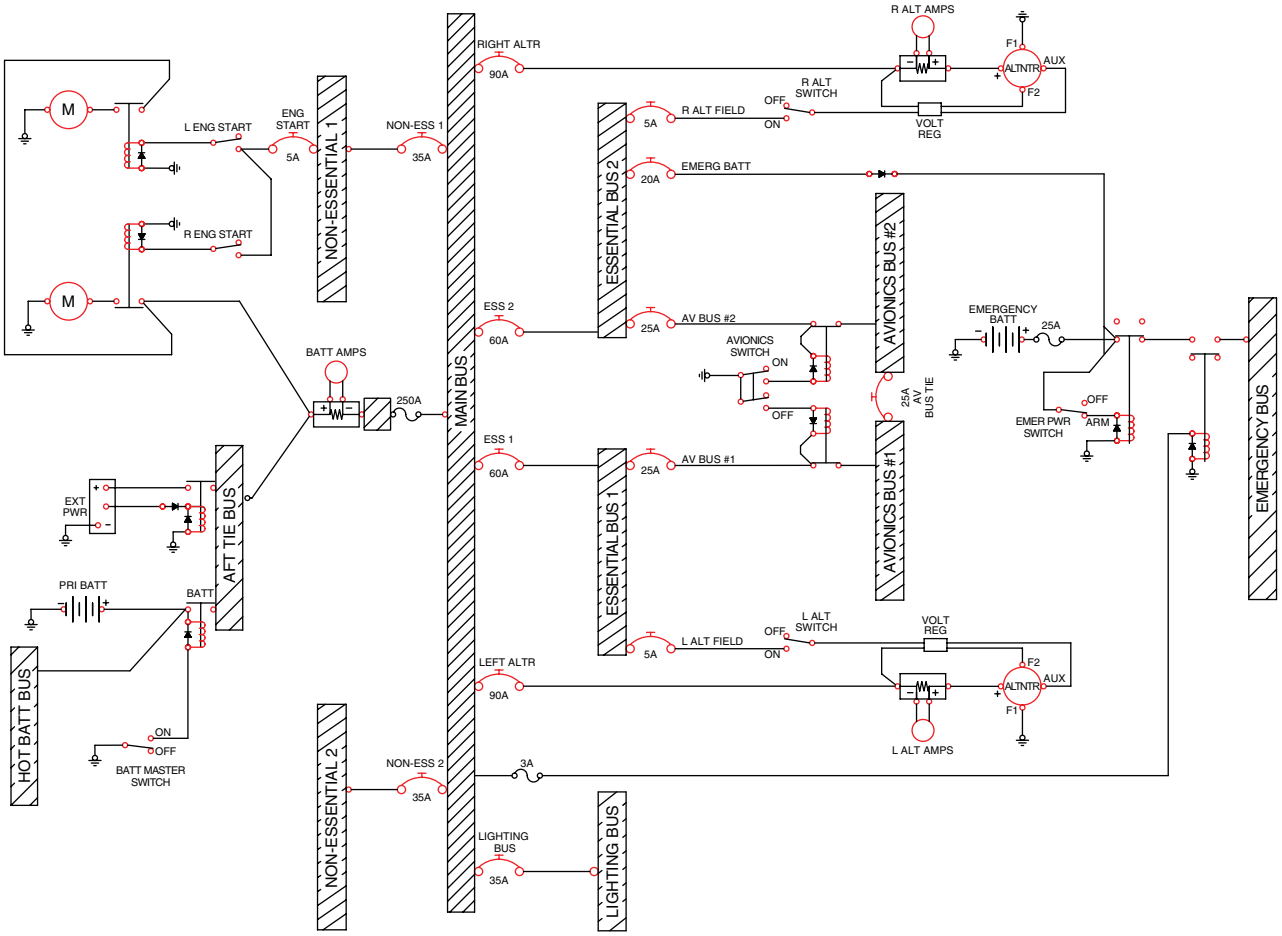
The two avionics busses are fed through independent essential busses which are protected by 25-amp circuit breakers. The busses are interconnected by a 25-amp bus tie and controlled by the avionics master switch. When the AVIONICS switch on the overhead switch panel is pressed, both avionics contactors (normally closed) relax allowing current to flow to both avionics busses. Should the need arise, either avionics bus can be isolated by pulling the avionics bus BUS TIE circuit breaker and the appropriate avionics circuit breaker.

(5) Lighting Bus

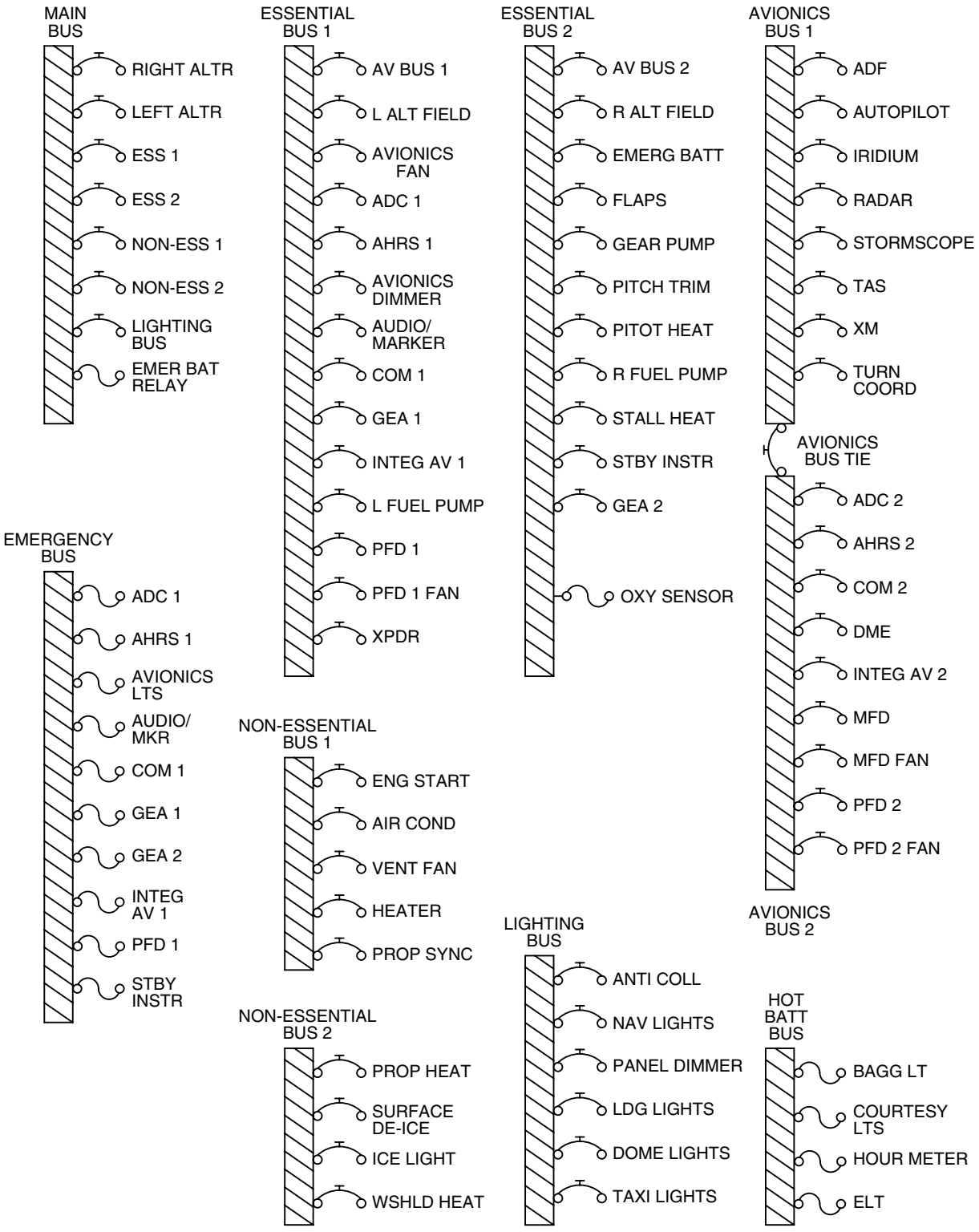
The Lighting bus is fed through the Main bus and is protected by a 35-amp circuit breaker.

(6) Emergency Bus

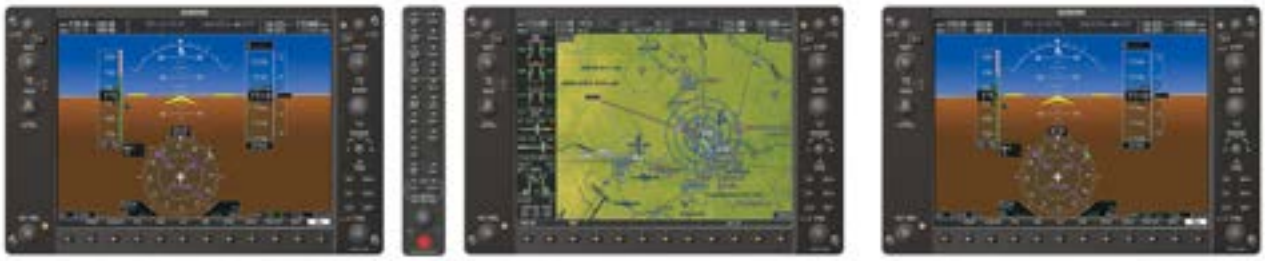
The Emergency bus exists as a backup power bus to the G1000 system. When the EMER switch is turned to the ARM position, power must be removed from the Main bus by turning the battery master switch to the “OFF” position. At the same time, the relay connecting the Emergency bus to the emergency battery is closed, providing emergency power to the connected equipment.



Bus Feeds
Figure 3



Component Power Sources
Figure 4



Normal Mode
Figure 5



Reversionary Mode #1 PFD Fail
Figure 6



Reversionary Mode MFD Fail
Figure 7

C. G1000 System Controls and Operation

(1) System Controls

The G1000 system controls are located on the PFD/MFD bezels and above the respective PFD/MFD displays. Indicators above the PFD, MFD, NAV, and COM keys are illuminated when their respective control mode(s) are selected. The unit is in MFD control mode by default on system power-up. NAV/COM radio tuning can be accomplished in either PFD or MFD control mode. The appropriate frequency box on the selected display is outlined by a light blue selection box, which flashes for a few seconds to indicate Control Unit activity. Selection of a different display control or radio tuning mode results in cancelation of the previous radio tuning mode.

(2) System Operation

The displays are connected together via multiple data busses, thus allowing for high-speed communication. Each GIA 63W is connected to the on-side PFD. This section discusses normal and reversionary modes of operation as well as the various AHRS modes of the G1000 system.

(a) Normal Operation (see “Figure 5” on page 3425016)

1) PFD

In normal mode, the PFDs present graphical flight instrumentation (attitude, heading, airspeed, altitude and vertical speed), thereby replacing the traditional flight instrument cluster. The PFDs also offer control for COM and NAV frequency selection.

2) MFD

In normal mode, the right portion of the MFD displays a full-color moving map with navigation information, while the left portion of the MFD is dedicated to the Engine Indication System (EIS). The MFD also offers control for COM and NAV frequency selection.

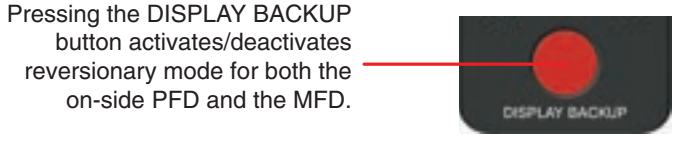
(b) Reversionary Mode (see “Figure 6” on page 3425016 and “Figure 7” on page 3425016).

NOTE: The G1000 system alerts the pilot when backup paths are utilized by the LRUs.

Reversionary mode is a mode of operation in which all important flight information is presented identically on at least one of the remaining displays. Transition to reversionary mode should be straight-forward for the pilot, for flight parameters are presented in the same format as in normal mode. In the event of a PFD failure, the G1000 system automatically switches to reversionary (backup) mode. In reversionary mode, all important flight information is presented on the remaining display(s) in the same format as in normal operating mode.

- PFD1 failure – MFD enters reversionary mode; PFD2 remains in normal mode.
- MFD failure – PFD1 automatically switches to reversionary mode; PFD2 remains in normal mode.
- PFD2 failure – PFD1 and the MFD remain in normal mode.

Each display may be put into reversionary mode by manually pressing the DISPLAY BACKUP button at the bottom of the audio panel (see “Figure 8”). Pressing this button again deactivates reversionary mode.



Display Backup Button
Figure 8

D. Maintenance Mode

CAUTION: ENTER “MAINTENANCE MODE” WHEN ACTIVATING THE GARMIN G1000 SYSTEM ON THE GROUND TO CONDUCT TESTS, INSPECTIONS, ETC.

The “Hobbs” Meter is activated by the GIA 63W Integrated Avionics Units and will run when one of the following is true, depending on how it was configured:

(1) the engine is running (Option A); or

(2) the airplane is inflight (Option B); or

(3) the GIA 63W No. 1 is ON (Option C).

To prevent needless accumulation of “hours” (i.e., to disable the “Hobbs” Meter) on the ground a Maintenance Mode has been implemented in the Garmin G1000 software.

Enter “Maintenance Mode” before exiting the splash screen at system startup by pressing SoftKeys 1, 2, 3, 1 in that order. The “MAINT MODE ON” advisory will then be displayed while the system remains in maintenance mode.

E. Troubleshooting

Troubleshoot the G1000 system by first identifying, then isolating the specific failure to the responsible LRU. There are several indications that the G1000 presents to the pilot or technician, showing overall system condition. A course of action should be determined based on the information presented on the display.

The 5th AUX group page on the MFD (“Figure 9” on page 3425019) provides LRU health status by means of a green check or a red ‘X’. Also, LRU software versions are shown along with other database versions and dates. If a red ‘X’ is shown for an LRU, see the appropriate LRU troubleshooting chart for guidance.

Typically, the G1000 Alerting System provides alerts/annunciations in conjunction with the information presented at the 5th AUX page.

CAUTION: RETURN TO SERVICE PARAGRAPHS UNDER INDIVIDUAL “COMPONENTS” ON PAGE 34250137 PROVIDE DETAILED INSTRUCTIONS ON EQUIPMENT CONFIGURATION AND RETURN TO SERVICE TESTING REQUIRED ANYTIME A G1000 COMPONENT OR LRU IS REMOVED, SWAPPED, OR REPLACED, THE TECHNICIAN MUST FOLLOW THESE PROCEDURES TO ENSURE PROPER OPERATION OF THE SYSTEM.

Troubleshooting information is provided as follows:

(1) Line Replaceable Unit (LRU) Failures
See “Chart 1” on page 34250112.

(2) Engine / Airframe Sensor Failures
See “Chart 2” on page 34250117.

(3) Data Path Failures
See “Chart 3” on page 34250121.

(4) GDU 1040
See “Chart 7” on page 34250140 and “Chart 8” on page 34250142.

(5) GMA 1347
See “Chart 9” on page 34250149 and “Chart 10” on page 34250150.

(6) GIA 63W
See “Chart 11” on page 34250153 and “Chart 12” on page 34250155.

(7) GEA 71
See “Chart 13” on page 34250163.
(8) GTX 33
See “Chart 14” on page 34250165.
(9) GDC 74A
See “Chart 15” on page 34250168.
(10) GRS 77
See “Chart 16” on page 34250172 and “Chart 17” on page 34250173.
(11) GMU 44
See “Chart 16” on page 34250172 and “Chart 17” on page 34250173.
(12) GDL 69/69A
See “Chart 18” on page 34250179 and “Chart 19” on page 34250180.
(13) GSR 56
See “Chart 20” on page 34250182.
(14) GWX 68
See “Chart 21” on page 34250185.
(15) GTS 825
See “Chart 22” on page 34250187.
(16) Software Configuration
See “Chart 23” on page 34250195.



System Status Page (AUX Group) (Typical)
Figure 9

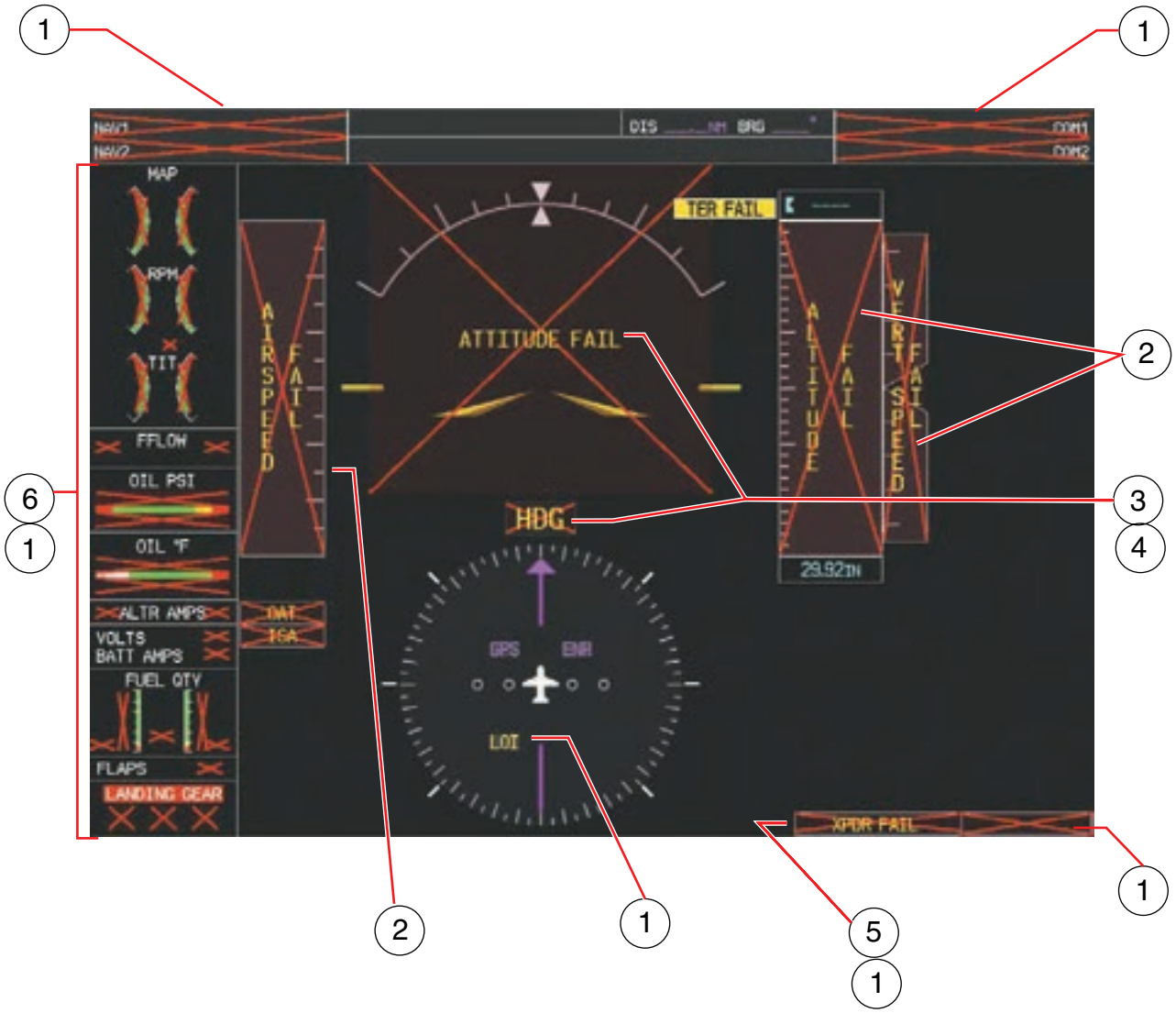
(17) Software Configuration and Status Verification

(a) Each PFD and MFD running in configuration mode.

(b) Select the “Systems Status Page”.

1) Verify that the following are valid. (Green annunciations indicate that the system is healthy): MFD1, PFD1, PFD2, GIA1, GIA2, GDL 69 (optional), GTS (optional), and GWX (optional).

2) Rotate the small FMS knob to change to the “System Configuration Page” and verify that the following are valid. (Indicated by green annunciations): MFD1, PFD1, PFD2, GIA1, GIA2, GDL 69 (optional), GTS (optional), and GWX (optional).



1. GIA 63W Integrated Avionics Units
2. GDC 74A Air Data Computer
3. GRS 77 AHRS
4. GMU 44 Magnetometer
5. GTX 33 Transponder
6. GEA 71 Engine Airframe Units

System Fail (Typical)
Figure 10

CHART 1 (Sheet 1 of 5)
LRU TROUBLESHOOTING

System Failure Troubleshooting

The following table provides basic troubleshooting guidance for LRU failures.

Invalid Data Field	Associated LRU(s)	Solution
Low Speed Awareness Band permanently displayed 	GDU	• With aircraft weight on wheels and the G1000 in Configuration Mode, check that the **RH GEAR ON GROUND** discrete in indication on GIA I/O CONFIGURATION page is illuminated. • If indication is illuminated Green, switch GIA1 and GIA2, to verify location of problem: ✓ If problem is resolved, replace defective unit. ✓ If indication is not illuminated, check for wiring faults between the GIAs and aircraft squat switch system.
Low Speed Awareness Band permanently inhibited 	GDU	• With aircraft weight off wheels and the G1000 in Configuration Mode, check that the **RH GEAR ON GROUND** discrete in indication on GIA I/O CONFIGURATION page is not illuminated. • If indication is not illuminated Green, switch GIA1 and GIA2, to verify location of problem: ✓ If problem is resolved, replace unit. ✓ If indication is illuminated, check for wiring faults between the GIAs and aircraft squat switch system.
NAV1 & COM1 	GIA1	• Check configuration settings for GIA1 and PFD1. • Swap GIA1 and GIA2, to verify location of problem: ✓ If problem follows GIA1, replace GIA1. • Check Ethernet interconnect from GIA1 to PFD1. ✓ If problem persists, replace PFD.
NAV2 & COM2 	GIA2	• Check configuration settings for GIA2 and PFD2. • Swap GIA1 and GIA2, to verify location of problem: ✓ If problem follows GIA2, replace GIA2. • Check Ethernet interconnect from GIA2 to PFD2. ✓ If problem persists, replace PFD.

CHART 1 (Sheet 2 of 5)
LRU TROUBLESHOOTING

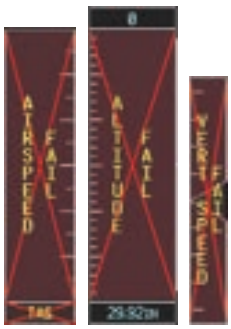
Invalid Data Field	Associated LRU(s)	Solution
GPS DR 	GIA1 or GIA2	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Check GPS1 and GPS2 signal strength on AUX page 4. Refer to GIA GPS section if GPS can not acquire a position lock for troubleshooting. • Check corresponding GPS antenna and cable for faults. • Check Ethernet interconnect between the PFD1 to GIA1 or PFD2 to GIA2 for faults. • Swap GIA1 and GIA2, to verify location of problem: ✓ If problem follows the GIA, replace the GIA. ✓ If problem persists, replace the PFD1 or PFD2.
XPDR FAIL 	GTX 33	• Check GTX 33 configuration settings for GIA1 and GIA2. • Check GTX 33/GIA1 interconnect. • Check GTX 33/GIA2 interconnect. • Perform a SET>ACTV configuration reset on the GTX Config page for each GTX, and verify the aircraft registration is present. • Replace GTX 33. ✓ If problem persists, replace GIA.
TAS FAIL AIRSPEED FAIL ALTITUDE FAIL VERT SPEED FAIL 	GDC 74B	• Inspect GDC 74B pitot/static plumbing integrity. • Inspect pitot/static ports and associated equipment for faults. • For TAS failure, also check GTP 59 OAT probe. • Check GDC 74B configuration settings for the PFDs, MFD, GIA1, and GIA2. ✓ If PFDs, MFD, and GIA configuration settings are correct, replace the GDC configuration module. ✓ If problem persists, replace the GDC 74B.
OAT 	GTP 59	• Check OAT probe wiring and connectors for faults. • Check GDC config module wiring for damage, replace if any is found. • Replace GDC config module • Replace GDC 74B with a known good unit: ✓ If problem persists, replace the GTP 59

CHART 1 (Sheet 3 of 5)
LRU TROUBLESHOOTING

Invalid Data Field	Associated LRU(s)	Solution
HDG FAIL 	GRS 77 & GMU 44	• Ensure metal objects (tool boxes, power carts, etc.) are not interfering with the magnetometer and aircraft is not in hangar, near other buildings, parked over metal drainage culverts or on hard surfaces that may contain steel reinforcements. • Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Cycle power after moving aircraft away from metal objects to determine if metal objects were the source of the interference. Allow up to five minutes for the heading to reinitialize. • Perform a Magnetometer Interference Test to check for interference from onboard electrical system components (e.g. NAV lights). Pay particular attention to any new electrical devices that have been installed since the aircraft was new. Correct any discrepancies that do not allow this test to pass before continuing. • Ensure GRS 77 and GMU 44 connectors are secure. • Check the wiring and any inline connectors between the GRS and GMU for faults. • Recalibrate the GMU 44. • Load configuration files to the PFD1, PFD2, GIA1, and GIA2. ✓ If problem persists replace the GMU 44. ✓ If problem persists, replace the GRS 77.
	GRS 77 & GMU 44	• If this message persists longer than five minutes, perform AHRS calibration procedures as described later in this chapter.

CHART 1 (Sheet 4 of 5)
LRU TROUBLESHOOTING

Invalid Data Field	Associated LRU(s)	Solution
ATTITUDE FAIL 	GRS 77	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Cycle GRS 77 power to restart initialization. • Ensure GRS 77 connecter is secure and proper wire harness strain relief is provided. • Ensure the GRS 77 is fastened down tightly in it's mounting rack and that the mounting rack is not loose (CAUTION - do not loosen the mounting rack hardware to the airframe shelf or the aircraft will need to be re-leveled and the PITCH/ROLL OFFSET procedure performed). • Ensure GPS has acquired at least four satellites, has a 3D navigation solution, and a DOP of less than 5.0. This is particularly important for an ATTITUDE FAIL that appears during ground operation only. • Perform an Engine Run-Up Test to check if engine vibration is causing the GRS 77 to go offline. • Load configuration files to the PFD, MFD, GIA1, and GIA2. • Calibrate the GRS 77 (Pitch/Roll Offset and Magnetometer Calibration). • Replace GRS 77. ✓ If problem persists replace GRS 77 configuration module. ✓ Contact Garmin Aviation Product Support if condition continues after replacing the GRS 77 and config module for additional assistance.

CHART 1 (Sheet 5 of 5)
LRU TROUBLESHOOTING

A DME or ADF failure is represented by the following red X's. The PFD displays function as a control head for the remote-mounted Honeywell KR 87 ADF and KN 63 DME.

Invalid Field	Sensor	Solutions
 	Honeywell ADF	• Ensure that GIA 63 #2 is properly functioning. • Reload the ADF option configurations. • Check for proper operation of the ADF receiver. Ensure that the receiver is receiving power. • Check ADF – GIA2 interconnect.
 	Honeywell DME	• Ensure that GIA 63 #2 is properly functioning. • Reload the DME option configurations. • Check for proper operation of the DME receiver. Ensure that the receiver is receiving power. • Check DME – GIA2 interconnect.

CHART 2 (Sheet 1 of 3)
TROUBLESHOOTING ENGINE / AIRFRAME SENSOR FAILURES

Invalid Data Field	Associated LRU(s)	Solution
Engine/Airframe Sensors (All Invalid) 	GEA 71 & GIA 63W	• If software was loaded to a new GDU display, be sure that the user settings for the replaced display were cleared. Clear user settings by pressing the CLR key on the replaced display while applying power to it. Acknowledge the on-screen prompt by pressing the ENT key or the right-most softkey. • Check for GEA related Alert messages on the PFD. Correct any Alerts concerning software or configuration errors by reloading software or configuration as noted. • Verify GEA internal power supply, configuration, and calibration status in configuration mode. ✓ The internal power supply, configuration and calibration boxes should be green. If they are red, replace the GEA 71. • Verify internal, external, and reference voltages listed in the Main Analog and I/O A Analog boxes are not dashed out (does not include Aircraft Power 1 and 2). ✓ If any voltages are dashed out, replace the GEA. • Ensure the GEA is online (green checkmark on the AUX – SYSTEM STATUS page). ✓ If GEA is not online verify unit is receiving power at the rack connector. ✓ Check the GIA/GEA interconnects for faults. ✓ Reload configuration files to both GIA's and the GEA. • If problem persists, replace the GEA 71.

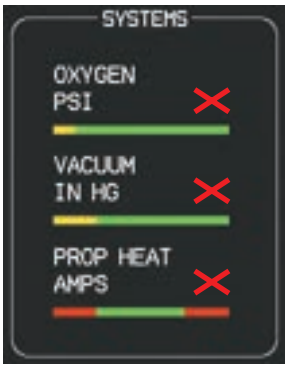
CHART 2 (Sheet 2 of 3)
TROUBLESHOOTING ENGINE / AIRFRAME SENSOR FAILURES

Engine/Airframe Instrument Failures

The following table provides guidance for troubleshooting individual engine/airframe sensor failures. Be sure to also follow previous guidance given for the GEA 71. The technician should troubleshoot to isolate the fault by checking sensor-to-GEA wiring, replacing the suspect sensor, and finally by replacing the GEA 71. Replace one part at a time. Check for correct operation of the sensors and GEA 71 after any part has been replaced. Refer to G1000 in 91-00-00 Wiring Diagrams.

Invalid Field	Sensor	Possible Solutions (for applicable engine/system)
	Manifold Pressure	• Check manifold pressure transducer wiring. • Replace manifold pressure transducer wiring. • Replace GEA 71.
	Propeller RPM	• Check engine tachometer sensor wiring. • Replace engine tachometer sensor. • Replace GEA 71.
	Turbine Inlet Temperature	• Check TIT probe wiring. • Replace TIT probe sensor. • Replace GEA 71.
	Fuel Flow	• Check fuel flow transmitter wiring. • Check fuel flow transmitter - GEA wiring. • Check power input to fuel flow transmitter. • Replace fuel flow transmitter. • Replace GEA 71.
	Oil Pressure	• Check oil pressure sensor – GEA wiring. • Replace oil pressure sensor. • Replace GEA 71.
	Oil Temperature	• Check oil temperature sensor – GEA wiring. • Replace oil temperature sensor. • Replace GEA 71.
	Fuel Quantity	• Check Fuel Senders - #1 GIA 63W wiring. • Check #1 GIA 63W - #1PFD wiring. • Check MFD - #1 PFD connections. • Replace Fuel Senders left or right • Replace #1 GIA 63W • Replace #1 PFD • For more details see 28-00-00.

CHART 2 (Sheet 3 of 3)
TROUBLESHOOTING ENGINE / AIRFRAME SENSOR FAILURES

Invalid Field	Sensor	Possible Solutions (for applicable engine/system)
	Amp/Volts	• Check Alt amp sensor - GEA 71 wiring. • Replace Alt amp sensor. • Replace GEA 71. • Check main bus - GEA 71 wiring. • Check emergency bus - GEA 71 wiring. • Replace GEA 71
	Flaps	• Check Potentiometer - GEA 71 wiring. • Replace Potentiometer. • Replace GEA 71.
	Oxygen	• Check oxygen pressure transducer wiring. • Replace oxygen pressure transducer sensor. • Replace GEA 71.
	Vacuum (w/Deice - Opt)	• Check vacuum sensor - GEA 71 wiring. • Replace vacuum sensor. • Replace GEA 71.
	Prop Heat (w/Deice - Opt)	

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(18) Backup Communications Path Checks

The G1000 system architecture is designed with redundant communication ports for several LRUs so that critical information can continue to be displayed in the event of an equipment or wiring failure. Of most importance is flight attitude, heading, and air data information. The GRS 77 and GDC 74A each have four separate ARINC 429 data lines which are all capable of sending data to the displays. The GEA 71, GTX 33, and GMA 1347 each have two redundant serial communication paths for the same purpose. See “Figure 2” on page 3425012 for a basic G1000 block diagram depicting this architecture. When troubleshooting, refer to the G1000 Wiring Diagrams listed in Chapter 91.

(19) Failed Path Messages

The following message indicates there is a data path connected to the GDU (PFD1, PFD2, or MFD) or the GIA 63W (1 or 2) that has failed.

FAILED PATH – A data path has failed. Check configuration mode.

The FAILED PATH message is triggered by a timeout of any one digital channel. The channels that are checked are listed on these pages in config mode:

(a) GDU RS-232 / ARINC 429 CONFIG (PFD1/2, and MFD).

(b) GIA RS-232 / ARINC 429 CONFIG (GIA1 and GIA2).

(c) GIA RS-485 CONFIGURATION (GIA1 and GIA2).

NOTE: Once the FAILED PATH message has been triggered, it will remain on the list of messages until the next power cycle. This latching was implemented so that for intermittent failures, the message would remain at the end of the flight (to alert maintenance crew). Also, this keeps the crew from having to acknowledge message repeatedly in the case of intermittent failures.

The box next to each channel indicates the current status of the channel per the below:

Red = data path is known to be failed.

Black = data path status is unknown.

Green = data path is known to be good.

The applicable data paths can be verified by viewing the configuration mode pages listed in “Chart 3” through “Chart 6” on page 34250133. See “Chart 25” on page 342501101 to navigate to a particular configuration page.

CHART 3
TROUBLESHOOTING - DATA PATH FAILURES

Failure Message	Cause	Remedy
FAILED PATH – a data path has failed.	A communications data path status is not being received by the G1000.	Determine which data path has failed - check configuration mode. Check wiring continuity for the failed path. Swap or replace the affected LRU.

CHART 4 (Sheet 1 of 3)
FAILED PATH MESSAGES, CONFIG PAGE (PFD1/2, AND MFD)

GDU RS-232 / ARINC 429 CONFIG Page
PFD 1 ARINC 429

Channel	LRU	Indicator	Status
IN 1	GRS 77 #1	GREEN	PFD1/GRS 77 #1 data path is functioning correctly.
		RED	PFD1/GRS 77 #1 data path is not functioning correctly. • Verify GRS77 #1 status is “OK” using the System Status page on the MFD. If it is not, correct condition before proceeding, reference GRS troubleshooting sections. o Swap GRS1 and GRS2 (no reconfiguration required) to confirm if the problem is in the original GRS1. o Replace original GRS1 if box turns green after swapping units. • Load PFD1 configuration file. • Swap PFD1 and PFD2 to confirm if the problem is in the original PFD1. o Replace original PFD1 if box turns green after swapping displays. • Swap GRS1 and GRS2 (no reconfiguration required) to confirm if the problem is in the original GRS1. o Replace original GRS1 if box turns green after swapping units. Check the PFD1/GRS 77 #1 interconnect wiring for faults.
		BLACK	PFD1/GRS 77 #1 data path functionally is unknown. Reload PFD1 configuration file.
IN 2	GDC 74 #1	GREEN	PFD1/GDC 74 #1 data path is functioning correctly
		RED	PFD1/GDC 74 #1 data path is not functioning correctly. • Verify GDC 74 #1 status is “OK” using the System Status page on the MFD. If it is not, correct condition before proceeding, reference GDC troubleshooting sections. • Load PFD1 and GDC 74 #1 configuration files. • Swap PFD1 and PFD2 to confirm if the problem is in the original PFD1. o Replace original PFD1 if box turns green after swapping displays. • Check the PFD1/GDC 74 #1 interconnect wiring for faults. Replace GDC 74 #1 if problem remains.
		BLACK	PFD1/GDC 74 #1 data path functionality is unknown. Reload PFD1 configuration file.

CHART 4 (Sheet 2 of 3)
FAILED PATH MESSAGES, CONFIG PAGE (PFD1/2, AND MFD)

PFD 2 ARINC 429

Channel	LRU	Indicator	Status
IN 1	GRS 77 #2	GREEN	PFD2/GRS 77 #2 data path is functioning correctly.
		RED	PFD2/GRS 77 #2 data path is not functioning correctly. • Verify GRS77 #2 status is “OK” using the System Status page on the MFD. If it is not, correct condition before proceeding, reference GRS troubleshooting sections. o Swap GRS1 and GRS2 (no reconfiguration required) to confirm if the problem is in the original GRS1. o Replace original GRS1 if box turns green after swapping units. • Load PFD2 configuration file. • Swap PFD2 and PFD1 to confirm if the problem is in the original PFD2. o Replace original PFD2 if box turns green after swapping displays. • Swap GRS1 and GRS2 (no reconfiguration required) to confirm if the problem is in the original GRS1. o Replace original GRS1 if box turns green after swapping units. • Check the PFD2/GRS 77 #2 interconnect wiring for faults.
		BLACK	PFD2/GRS 77 #2 data path functionality is unknown. Reload PFD2 configuration file.
IN 2	GDC 74 #2	GREEN	PFD2/GDC 74 #2 data path is functioning correctly.
		RED	PFD2/GDC 74 #2 data path is not functioning correctly. • Verify GDC 74 #2 status is “OK” using the System Status page on the MFD. If it is not, correct condition before proceeding, reference GDC troubleshooting sections. • Load PFD2 and GDC 74 #2 configuration files. • Swap PFD2 and PFD1 to confirm if the problem is in the original PFD2. o Replace original PFD2 if box turns green after swapping displays. • Check the PFD2/GDC 74 #2 interconnect wiring for faults. Replace GDC 74 #2 if problem remains.
		BLACK	PFD2/GDC 74 #2 data path functionality is unknown. Reload PFD 2 configuration file.

CHART 4 (Sheet 3 of 3)
FAILED PATH MESSAGES, CONFIG PAGE (PFD1/2, AND MFD)

MFD ARINC 429

Channel	LRU	Indicator	Status
IN 1	GRS 77 #1	GREEN	MFD/GRS 77 #1 data path is functioning correctly.
		RED	MFD/GRS 77 #1 data path is not functioning correctly. • Verify GRS77 #1 status is "OK" using the System Status page on the MFD. If it is not, correct condition before proceeding, reference GRS troubleshooting sections. o Swap GRS1 and GRS2 (no reconfiguration required) to confirm if the problem is in the original GRS1. o Replace original GRS1 if box turns green after swapping units. • Load MFD configuration file. • Swap MFD and PFD2 to confirm if the problem is in the original MFD. o Replace original MFD if box turns green after swapping displays. • Swap GRS1 and GRS2 (no reconfiguration required) to confirm if the problem is in the original GRS1. o Replace original GRS1 if box turns green after swapping units. Check the MFD/GRS 77 #1 interconnect wiring for faults.
		BLACK	MFD/GRS 77 #1 data path functionally is unknown. Reload MFD configuration file.
IN 2	GDC 74 #1	GREEN	MFD/GDC 74 #1 data path is functioning correctly
		RED	MFD/GDC 74 #1 data path is not functioning correctly. • Verify GDC 74 #1 status is "OK" using the System Status page on the PFD. If it is not, correct condition before proceeding, reference GDC troubleshooting sections. • Load MFD and GDC 74 #1 configuration files. • Swap MFD and PFD2 to confirm if the problem is in the original MFD. o Replace original MFD if box turns green after swapping displays. • Check the MFD/GDC 74 #1 interconnect wiring for faults. Replace GDC 74 #1 if problem remains.
		BLACK	MFD/GDC 74 #1 data path functionality is unknown. Reload MFD configuration file.

CHART 5 (Sheet 1 of 8)
FAILED PATH MESSAGES, GIA RS-232 / ARINC 429 CONFIG PAGE (GIA1 AND GIA2)

GIA1 RS-232

Channel	LRU	Indicator	Status
CHNL 1	GDC 74 #1	GREEN	GIA1/GDC 74 #1 data path is functioning correctly.
		RED	GIA1/GDC 74 #1 data path is not functioning correctly. • Load GIA1 and GDC 74 #1 configuration files. • Swap GIA1 and GIA2, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA1. o Replace original GIA1 if box turns green after swapping units. • Check the GIA1/GDC 74 #1 interconnect wiring for faults. Replace GDC 74 #1 if problem remains.
		BLACK	GIA1/GDC 74 #1 data path functionality is unknown. Reload GIA1 configuration files.
CHNL 4	GRS 77 #2 (output only)	BLACK	GIA1/GRS 77 #2 output data path is not monitored. A black box is normal.
CHNL 5	GTX 33	GREEN	GIA1/GTX 33 data path is functioning correctly.
		RED	GIA1/GTX 33 data path is not functioning correctly. • Load GIA1 and GTX 33 configuration files. • Swap GIA1 and GIA2, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA1. o Replace original GIA1 if box turns green after swapping units. • Check the GIA1/GTX 33 interconnect wiring for faults. Replace GTX 33 if problem remains.
		BLACK	GIA2/GTX 33 data path functionality is unknown Reload GIA1 configuration files.

CHART 5 (Sheet 2 of 8)
FAILED PATH MESSAGES, GIA RS-232 / ARINC 429 CONFIG PAGE (GIA1 AND GIA2)

GIA1 RS-232

Channel	LRU	Indicator	Status
CHNL 6	GRS 77 #1	GREEN	GIA1/GRS 77 #1 data path is functioning correctly.
		RED	GIA1/GRS 77 #1 data path is not functioning correctly. • Load GIA1 configuration files. • Swap GIA1 and GIA2, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA1. o Replace original GIA1 if box turns green after swapping units. • Swap GRS1 and GRS2 (no reconfiguration required) to confirm if the problem is in the original GRS1. o Replace original GRS1 if box turns green after swapping units. Check the GIA1/GRS 77 #1 interconnect wiring for faults.
		BLACK	GIA1/GRS 77 #1 data path functionality is unknown. Reload GIA1 configuration files.
CHNL 7	GMA 1347	GREEN	GIA1/GMA 1347 data path is functioning correctly.
		RED	GIA1/GMA 1347 data path is not functioning correctly. • Load GIA1 and GMA 1347 configuration files. • Swap GIA1 and GIA2, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA1. o Replace original GIA1 if box turns green after swapping units. Check the GIA1/GMA 1347 interconnect wiring for faults.
		BLACK	GIA1/GMA 1347 data path functionality is unknown. Reload GIA1 configuration files.

CHART 5 (Sheet 3 of 8)
FAILED PATH MESSAGES, GIA RS-232 / ARINC 429 CONFIG PAGE (GIA1 AND GIA2)

GIA1 ARINC 429

Channel	LRU	Indicator	Status
IN 5	GDC 74A #1	GREEN	GIA1/GDC 74A #1 data path is functioning correctly.
		RED	GIA1/GDC 74A #1 data path is not functioning correctly. • Check PFD Alert Window for GIA or GDC configuration or software error messages. Correct any errors before proceeding. • Verify GDC 74A status is OK using the AUX – SYSTEM STATUS page on the MFD. If it has a Red-X, troubleshoot why the GDC is offline before proceeding. • Swap GIA1 and GIA2, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA1. – Replace original GIA1 if box turns green after swapping units. • Check the GIA1/GDC 74A #1 interconnect wiring and unit connector pins for faults. • Replace GDC 74A #1.
		BLACK	GIA1/GDC 74A #1 data path functionality is unknown. Reload GIA1 configuration files.
IN 6	GRS 77 #1	GREEN	GIA1/GRS 77 #1 data path is functioning correctly.
		RED	GIA1/GRS 77 #1 data path is not functioning correctly. • Check PFD Alert Window for GIA or GRS configuration or software error messages. Correct any errors before proceeding. • Verify GRS77 status is OK using the AUX - SYSTEM STATUS PAGE on the MFD. If it has a Red-X, troubleshoot why the GRS is offline before proceeding. • Ensure GRS77 connecter is secure and proper wire harness strain relief is provided. • Swap GIA1 and GIA2, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA1. – Replace original GIA1 if box turns green after swapping units. • Check the GIA1/GRS 77 #1 interconnect wiring and unit connector pins for faults. Replace GRS 77 #1.
		BLACK	GIA1/GRS 77 #1 data path functionality is unknown. Reload GIA1 configuration files.

CHART 5 (Sheet 4 of 8)
FAILED PATH MESSAGES, GIA RS-232 / ARINC 429 CONFIG PAGE (GIA1 AND GIA2)

RESERVED

CHART 5 (Sheet 5 of 8)
FAILED PATH MESSAGES, GIA RS-232 / ARINC 429 CONFIG PAGE (GIA1 AND GIA2)

GIA2 RS-232

Channel	LRU	Indicator	Status
CHNL 1	GDC 74 #2	GREEN	GIA2/GDC 74 #2 data path is functioning correctly.
		RED	GIA2/GDC 74 #2 data path is not functioning correctly. • Load GIA2 and GDC 74 #2 configuration files. • Swap GIA2 and GIA1, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA1. o Replace original GIA2 if box turns green after swapping units. • Check the GIA1/GDC 74 #2 interconnect wiring for faults. Replace GDC 74 #2 if problem remains.
		BLACK	GIA2/GDC 74 #2 data path functionality is unknown. Reload GIA2 configuration files.
CHNL 2	GRS 77 #1 (output only)	BLACK	GIA2/GRS 77 #1 output data path is not monitored. A black box is normal.
CHNL 5	GTX 33	GREEN	GIA2/GTX 33 #2 data path is functioning correctly.
		RED	GIA2/GTX 33 #2 data path is not functioning correctly. • Load GIA2 and GTX 33 #2 configuration files. • Swap GIA2 and GIA1, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA1. o Replace original GIA2 if box turns green after swapping units. • Check the GIA2/GTX 33 #2 interconnect wiring for faults. Replace GTX 33 #2 if problem remains.
		BLACK	GIA2/GTX 33 #2 data path functionality is unknown Reload GIA2 configuration files.

CHART 5 (Sheet 6 of 8)
FAILED PATH MESSAGES, GIA RS-232 / ARINC 429 CONFIG PAGE (GIA1 AND GIA2)

GIA2 RS-232

Channel	LRU	Indicator	Status
CHNL 6	GRS 77 #2	GREEN	GIA2/GRS 77 #2 data path is functioning correctly.
		RED	GIA2/GRS 77 #2 data path is not functioning correctly. • Load GIA2 configuration files. • Swap GIA2 and GIA1, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA2. o Replace original GIA2 if box turns green after swapping units. • Swap GRS2 and GRS1 (no reconfiguration required) to confirm if the problem is in the original GRS2. o Replace original GRS2 if box turns green after swapping units. Check the GIA2/GRS 77 #2 interconnect wiring for faults.
		BLACK	GIA2/GRS 77 #2 data path functionality is unknown. Reload GIA2 configuration files.
CHNL 7	GMA 1347	GREEN	GIA2/GMA 1347 data path is functioning correctly.
		RED	GIA2/GMA 1347 data path is not functioning correctly. • Load GIA2 and GMA 1347 configuration files. • Swap GIA2 and GIA1, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA2. o Replace original GIA2 if box turns green after swapping units. Check the GIA2/GMA 1347 interconnect wiring for faults.
		BLACK	GIA2/GMA 1347 data path functionality is unknown. Reload GIA2 configuration files.

CHART 5 (Sheet 7 of 8)
FAILED PATH MESSAGES, GIA RS-232 / ARINC 429 CONFIG PAGE (GIA1 AND GIA2)

RESERVED

CHART 5 (Sheet 8 of 8)
FAILED PATH MESSAGES, GIA RS-232 / ARINC 429 CONFIG PAGE (GIA1 AND GIA2)

GIA2 ARINC 429

Channel	LRU	Indicator	Status
IN 5	GDC 74A #2	GREEN	GIA2/GDC 74A #2 data path is functioning correctly.
		RED	GIA2/GDC 74A #2 data path is not functioning correctly. • Check PFD Alert Window for GIA or GDC configuration or software error messages. Correct any errors before proceeding. • Verify GDC 74A status is OK using the AUX – SYSTEM STATUS page on the MFD. If it has a Red-X, troubleshoot why the GDC is offline before proceeding. • Swap GIA2 and GIA1, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA2. – Replace original GIA2 if box turns green after swapping units. • Check the GIA2/GDC 74A #2 interconnect wiring and unit connector pins for faults. • Replace GDC 74A #2.
		BLACK	GIA2/GDC 74A #2 data path functionality is unknown. Reload GIA1 configuration files.
IN 6	GRS 77 #2	GREEN	GIA2/GRS 77 #2 data path is functioning correctly.
		RED	GIA2/GRS 77 #2 data path is not functioning correctly. • Check PFD Alert Window for GIA or GRS configuration or software error messages. Correct any errors before proceeding. • Verify GRS77 status is OK using the AUX - SYSTEM STATUS PAGE on the MFD. If it has a Red-X, troubleshoot why the GRS is offline before proceeding. • Ensure GRS77 connecter is secure and proper wire harness strain relief is provided. • Swap GIA2 and GIA1, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA2. – Replace original GIA2 if box turns green after swapping units. • Check the GIA2/GRS 77 #2 interconnect wiring and unit connector pins for faults. Replace GRS 77 #2.
		BLACK	GIA2/GRS 77 #2 data path functionality is unknown. Reload GIA2 configuration files.

CHART 6 (Sheet 1 of 2)
FAILED PATH MESSAGES, GIA RS-485 CONFIG PAGE (GIA1 AND GIA2)

GIA RS-485 CONFIGURATION Page

GIA1 RS-485

Channel	LRU	Indicator	Status
CHNL 1	GEA1	GREEN	GIA1/GEA1 data path is functioning correctly.
		RED	GIA1/GEA1 data path is not functioning correctly. • Check PFD1 Alert Window for GIA or GEA configuration or software error messages. Correct any errors before proceeding. • Verify GEA1 status is OK using the AUX - SYSTEM STATUS PAGE on the MFD. If it has a Red-X, troubleshoot why the GEA is offline before proceeding. • Swap GIA1 and GIA2, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA1. • Replace original GIA1 if box turns green after swapping units. • Swap GEA1 and GEA2, reconfigure both GEA's to their new locations to confirm if the problem is in the original GEA1. • Replace original GEA1 if box turns green after swapping units. • Check the GIA1/GEA1 interconnect wiring and connector for faults.
		BLACK	GIA1/GEA1 data path functionality is unknown. Reload GIA1 configuration files.
CHNL 2	GEA2	GREEN	GIA1/GEA2 data path is functioning correctly.
		RED	GIA1/GEA2 data path is not functioning correctly. • Check PFD1 Alert Window for GIA or GEA configuration or software error messages. Correct any errors before proceeding. • Verify GEA2 status is OK using the AUX - SYSTEM STATUS PAGE on the MFD. If it has a Red-X, troubleshoot why the GEA is offline before proceeding. • Swap GIA1 and GIA2, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA1. • Replace original GIA1 if box turns green after swapping units. • Swap GEA1 and GEA2, reconfigure both GEA's to their new locations to confirm if the problem is in the original GEA2. • Replace original GEA2 if box turns green after swapping units. • Check the GIA1/GEA2 interconnect wiring and connector for faults.
		BLACK	GIA1/GEA2 data path functionality is unknown. Reload GIA1 configuration files.

CHART 6 (Sheet 2 of 2)
FAILED PATH MESSAGES, GIA RS-485 CONFIG PAGE (GIA1 AND GIA2)

GIA2 RS-485

Channel	LRU	Indicator	Status
CHNL 1	GEA1	GREEN	GIA2/GEA1 data path is functioning correctly.
		RED	GIA2/GEA1 data path is not functioning correctly. • Check PFD1 Alert Window for GIA or GEA configuration or software error messages. Correct any errors before proceeding. • Verify GEA1 status is OK using the AUX - SYSTEM STATUS PAGE on the MFD. If it has a Red-X, troubleshoot why the GEA is offline before proceeding. • Swap GIA1 and GIA2, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA2. • Replace original GIA2 if box turns green after swapping units. • Swap GEA1 and GEA2, reconfigure both GEA's to their new locations to confirm if the problem is in the original GEA1. • Replace original GEA1 if box turns green after swapping units. • Check the GIA2/GEA1 interconnect wiring and connector for faults.
		BLACK	GIA2/GEA1 data path functionality is unknown. Reload GIA2 configuration files.
CHNL 2	GEA2	GREEN	GIA2/GEA2 data path is functioning correctly.
		RED	GIA2/GEA2 data path is not functioning correctly. • Check PFD1 Alert Window for GIA or GEA configuration or software error messages. Correct any errors before proceeding. • Verify GEA2 status is OK using the AUX - SYSTEM STATUS PAGE on the MFD. If it has a Red-X, troubleshoot why the GEA is offline before proceeding. • Swap GIA1 and GIA2, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA2. • Replace original GIA2 if box turns green after swapping units. • Swap GEA1 and GEA2, reconfigure both GEA's to their new locations to confirm if the problem is in the original GEA2. • Replace original GEA2 if box turns green after swapping units. • Check the GIA2/GEA2 interconnect wiring and connector for faults.
		BLACK	GIA2/GEA2 data path functionality is unknown. Reload GIA2 configuration files.

F. Recommended Tools

The following tools (or equivalent) are recommended to perform various maintenance tasks on the G1000 system.

(1) Calibrated voltmeter capable of measuring 0-32 Volts DC.
(2) #2 Phillips Screwdriver.
(3) 3/32nd inch Hex Tool.
(4) Calibrated digital Level with 0.25 degrees of accuracy capability.
(5) Calibrated VHF NAV/COM/ILS ramp tester.
(6) Calibrated transponder ramp tester including Mode S capability for Mode S transponder equipped aircraft.
(7) Air Data Test Set (ADTS) capable of simulating altitude up to the aircraft's service ceiling.
(8) Headset/Microphone.
(9) Outdoor line-of-site to GPS satellite signals or GPS indoor repeater.
(10) Ground Power Unit (Capable of supplying 28 Vdc).
(11) Calibrated Flight Control Cable Tension Meter or equivalent.
(12) Servo Mount Slip Clutch Adjustment Tool – Garmin P/N T10-00110-01 and a 2 Amp, 24 V, DC Power Supply.
(13) Calibrated digital thermometer suitable for measuring ambient temperature.
(14) Laptop with RS232 emulation software.
(15) M22885/108T8234 extraction tool.
(16) An 0.060" 6-Spline wrench.
(17) WX-PA portable analyzer kit (If Stormscope is installed).
(18) Calibrated torque wrench capable of measuring 0–70 in·lbs.
(19) Standard sockets & wrenches (3/8", 9/16", and 13/16").

G. Inspections

(1) Annual / 100 Hour Inspection

Each 12 months or 100 hours time-in-service, whichever comes first: perform the “Avionics Cooling Fan Fail Annunciation Test” on page 342501122.

(2) 500 Hour Inspection

Perform the following test of the PFD1, MFD, PFD2 and AVIONICS cooling fans:

(a) Turn OFF all electrical components and avionics systems.

(b) Verify the master switch is ON.

(c) Pull the PFD1 FAN, MFD FAN, PFD2 FAN and AVIONICS FAN circuit breakers.

(d) One at a time, push IN each of the four FAN circuit breakers. Verify airflow from each fan by motor sound.

(e) After checking each fan, turn OFF master switch and push IN all four FAN circuit breakers.

(3) 1000 Hour Inspection

Each 1000 hours time-in-service visually inspect each unit listed below for corrosion, damage, or other defects. Replace any damaged parts as required. Inspection may require the temporary removal of a unit or units to gain access to connectors.

(a) GDU 1040 PFD & GDU 1040 MFD

Remove the PFD & MFD. Inspect the mounting surface and connector for corrosion, heavy oxidation, or other damage.

(b) GMA 1347 Audio Panel

Inspect the GMA 1347 unit, rack, and connectors for corrosion or other defects.

(c) GIA 63W Integrated Avionics Units (2 ea.)

Inspect the GIA 63W #1 and #2 unit, rack, and connectors for corrosion or other defects.

(d) GEA 71 Engine/Airframe Units (2 ea.)

Inspect the GEA 71 unit, rack, and connectors for corrosion or other defects.

(e) GTX 33 Mode S Transponder(s) (if installed)

Inspect the GTX 33 unit, rack, and connectors for corrosion or other defects.

(f) GDL 69/69A - Datalink (if installed)

Inspect the GDL 69/69A unit, rack, and connectors for corrosion or other defects.

(g) PFD & MFD Cooling Fans

Inspect both PFD & MFD cooling fans for accumulation of dust or other damage. Remove excess dust as required. Close the AVIONICS FAN circuit breaker and verify that the PFD and MFD cooling fans are functional.

(h) Avionics Cooling Fan

Inspect the avionics cooling fan for accumulation of dust or other damage. Remove excess dust as required. Close the AVIONICS FAN circuit breaker and verify that the avionics cooling fan is functional.

(i) GRS 77 AHRS

Inspect the GRS 77 unit, rack, and connector for corrosion or other defects.

(j) GMU 44 Magnetometer

Remove the GMU 44 and inspect the mounting hardware and GMU 44 for corrosion or other damage.

(k) Perform electrical bond tests on all G1000 equipment. See 51-80-00.

(l) Perform visual checks on the shield terminations for any degradation.

H. Components

CAUTION: ENTER “MAINTENANCE MODE” WHEN ACTIVATING THE GARMIN G1000 SYSTEM ON THE GROUND TO CONDUCT TESTS, INSPECTIONS, ETC. SEE “MAINTENANCE MODE” ON PAGE 3425018.

(1) General

Maintenance of the G1000 system components is “On Condition” only. The following provides basic information regarding unit removal, installation, and troubleshooting. Basic guidance on loading and configuring software is provided under Post Installation Set-up, below. When removing and/or replacing any G1000 component, always ensure that aircraft power is off. Unplug any auxiliary power supplies.

Before removing any G1000 LRU, it is required that the technician verify the LRU software part numbers.

To check an LRU software part number and/or version:

(a) Start the G1000 system in configuration mode as described in “Configuration Mode Overview” on page 34250197.

(b) The System Status page shows a list of LRUs in the LRU window. Activate the cursor and highlight the LRU window.

(c) Use the FMS knob to scroll through the list in the window and select the desired LRU.

(d) The software part number and version is displayed in the DATA window. Compare this to the latest configuration shown in the appropriate Required Equipment List (REL), “Chart 30” on page 342501165.

NOTE: If a faulty LRU is not reporting its software version and part number, check aircraft maintenance logs for last software version loaded and verify against the latest configuration.

(2) GDU 1040 PFD and GDU 1040 MFD

See “Figure 11”.

CAUTION: THE GDU 1040S USE A LENS COATED WITH A SPECIAL ANTI-REFLECTIVE COATING THAT IS VERY SENSITIVE TO SKIN OILS, WAXES AND ABRASIVE CLEANERS. CLEANERS CONTAINING AMMONIA WILL HARM THE ANTI-REFLECTIVE COATING. IT IS VERY IMPORTANT TO CLEAN THE LENS USING A CLEAN, LINT-FREE CLOTH AND AN EYEGLASS LENS CLEANER THAT IS SPECIFIED AS SAFE FOR ANTI-REFLECTIVE COATINGS.

(a) Description

Three Garmin GDU 1040 units, 10.4 inch LCD displays with 1024x768 resolution, are installed in the instrument panel. Two GDU 1040 units are configured as PFD1 and PFD2; one GDU 1040 unit is configured as a MFD. The PFD displays are located on either side of the MFD, with the stand-by instruments located to the left of the Pilot’s PFD (PFD1). A GMA 1347 Audio Panel is located between the Pilot PFD and the MFD.



GDU 1040 PFD and MFD
Figure 11

Electrical power to PFD1 is from the Essential Bus #1 or the Emergency Bus, whereas the MFD and PFD2 receive power from Avionics Bus #2.

The GDU 1040 units communicate with each other and the GIA 63W units through a high-speed data bus (HSDB) Ethernet connection.

On the panel, an independent flight instrument display provides attitude, altitude, airspeed, heading, VSI and GPS indicators as a fully digital, backup instrument. This is mounted in a vertical configuration to the left of the pilot PFD.

(b) Troubleshooting

See “Chart 7” on page 34250140 and “Chart 8” on page 34250142.

(c) Removal

1) Using a 3/32nd hex tool, rotate all four ¼-turn fasteners counter-clockwise until they reach their stops.
2) Carefully remove the display from the panel.
3) While supporting the display, disconnect the connector.

(d) Installation

1) Visually inspect the connector and pins for signs of damage. Repair any damage. While supporting the display, connect the connector to the rear of the unit.
2) Carefully insert the display into the panel cutout, ensuring that all 4 ¼-turn fasteners align with the corresponding holes.
3) Seat the display in the panel cutout. Do not use excessive force while inserting the display.
4) Once seated, rotate all four ¼-turn fasteners clockwise to lock the display to the panel.

(e) Return to Service

1) Original Display Reinstalled.

If the removed display(s) are re-installed in their original positions, no software or configuration loading is required. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) Original PFD Displays Installed in Opposite Locations for Troubleshooting.

If the PFD1 and PFD2 are installed in opposite positions, no software or configuration loading is required.

3) New, Repaired, or Exchanged Display(s) Installed.

If a new, repaired, or exchanged GDU 1040 or GDU 1500 is installed, the correct software and configuration files must be loaded to the unit.

a) See “G1000 Software/Configuration Procedure” on page 342501107 and “MFD & PFD Software Load” on page 342501108.

b) If ChartView, Synthetic Vision, or TAWS were previously installed, these must be reactivated. See “Chartview, Synthetic Vision, and TAWS Unlock Procedures” on page 342501113. If any other options were previously installed, these must also be reactivated per “Optional Equipment Configuration” on page 342501109.

4) Test

Perform “PFD/MFD Test Procedure” on page 342501125.

CHART 7 (Sheet 1 of 2)
TROUBLESHOOTING - GDU 1040

GDU 1040 Troubleshooting / Common Problems	
Symptom	Recommended Action
Display is blank	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Cycle power. ✓ If GDU recovers, observe display for yellow text containing error information at the top of the screen. If message indicates software need to be re-loaded, then re-load software. Otherwise, replace the GDU.
Display resets	• Use a bright light to verify LCD is active. ✓ Adjust avionics dimmer control full clockwise. ✓ Manually turn up backlight on the PFD and load configuration files to the GDU. • Ensure slide lock is fully engaged with the locking tabs on the back of the unit. ✓ If slide lock is not fully engaged, remove connector and verify the locking tabs on the GDU are perpendicular to the connector. If necessary, straighten them before reseating connector.
Display flickers	• Ensure GDU is receiving power. If a circuit breaker is tripped, determine source of short before resetting breaker. • Ensure circuit breakers have not failed and power wire connections are secure. • Swap PFD1 and PFD2. ✓ If problem follows unit, replace the display. Please note the position it failed in (PFD1/2). ✓ If problem does not follow unit, troubleshoot aircraft wiring for fault.
SD card is stuck in GDU	• DO NOT insert a screwdriver of any length into the card slot. • DO NOT pry against the overlay. • DO NOT force the SD Card out. • Use a small screwdriver in the groove on the side of the exposed end of the card to help pull out the card. • Push the card in further to release the card locking mechanism. • Check SD Card for having more than one label. Two or more labels on the card will cause sticking. ✓ Remove all but one label. • Ensure the SD card is from Toshiba or SanDisk. Use of other SD Cards is not recommended. • If card was inserted with the label facing to the right, do not attempt to remove. Return the unit to Garmin for repair.
A button/knob/joystick does not appear to function	• Go to the GDU TEST page in configuration mode and verify button, knob, or joystick operates correctly by observing a change in color from red to green in the button/knob/joystick icon when the button/knob/joystick is pressed. If a button is stuck, the button icon will be green without pressing the button as soon as you turn to the GDU TEST page. ✓ If problem is verified, replace GDU.

CHART 7 (Sheet 2 of 2)
TROUBLESHOOTING - GDU 1040

GDU 1040 Troubleshooting / Common Problems (cont.)	
Symptom	Recommended Action
Terrain/Obstacle/Safetaxi does not display	• Ensure supplemental data cards are inserted correctly in the lower slots of all three GDU's. • Allow the system to verify the data on the cards for approximately five minutes after power-up. • If a database does not activate, reload the problem database onto the SD Card or replace the card.
Display will not track dimmer bus	• Reload GDU configuration files. • If display is a PFD, swap PFD1 and PFD2 to see if problem remains with display. ✓ Replace display if condition remains with the same unit. ✓ If condition remains in original position after swapping displays, check GDU dimmer input to verify voltage is present. • If display is the MFD, check dimmer input to verify voltage is present. ✓ Replace MFD if dimmer voltage is present.
Keyboard will not track dimmer bus	

CHART 8 (Sheet 1 of 6)
MESSAGE ADVISORIES - GDU 1040

GDU 1040 Messages / Software / Configuration Messages		
Failure Message	Cause	Solution
SW MISMATCH – GDU software version mismatch. Xtalk is off.	The system has found the PFDs and/or MFD software versions do not match.	• Load correct software version. See Software Loading procedure.
MANIFEST – PFD 1 software mismatch. Communication Halted.	The system has detected an incorrect software version loaded in the specified PFD.	
MANIFEST – PFD 2 software mismatch. Communication Halted.		
MANIFEST – MFD1 software mismatch. Communication Halted.	The system has detected an incorrect software version loaded in MFD.	
CNFG MODULE – PFD 1 configuration module is inoperative.	The PFD master configuration module has failed.	• Check master configuration module connector and wiring for damage inside the GDU connector backshell. ✓ Replace master configuration module wiring and pins. ✓ If problem persists, replace master configuration module. NOTE New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 System ID number will change to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number

CHART 8 (Sheet 2 of 6)
MESSAGE ADVISORIES - GDU 1040

GDU 1040 Messages / Software / Configuration Messages (cont.)		
Failure Message	Cause	Solution
MFD1 CONFIG – MFD1 configuration error. Config service req'd.	A configuration mismatch has occurred between the display and the Master Configuration Module.	• Reload the display configuration files from SD Loader Card. • Reload system configuration files by pressing the UPDT CFG softkey on the Configuration Upload Page in the PFD1 System Page Group to load configuration files into the configuration module. ✓ If message persists, check PFD1 config module wiring for faults and replace if necessary. ✓ If issue continues, replace PFD1 master configuration module. ✓ If problem persists, replace the display. NOTE New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 System ID number will change to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number
PFD 1 CONFIG – PFD 1 configuration error. Config service req'd.		
PFD 2 CONFIG – PFD 2 configuration error. Config service req'd.		

CHART 8 (Sheet 3 of 6)
MESSAGE ADVISORIES - GDU 1040

Database Message Advisories		
Failure Message	Cause	Solution
MFD1 DB ERR – MFD1 navigation database error exists.	The MFD or specified PFD has encountered an error in the navigation database.	• Reload navigation database into the display. • Contact Garmin Technical Support for assistance.
PFD 1 DB ERR – PFD 1 navigation database error exists.		
PFD 2 DB ERR – PFD 2 navigation database error exists.		
MFD1 DB ERR – MFD1 terrain database error exists.	The MFD has encountered an error in the terrain database.	• Confirm supplemental data card is inserted fully. • Reload the database or replace the supplemental datacard. • Contact Garmin Product Support for further assistance if needed.
PFD 1 DB ERR – PFD 1 terrain database error exists.	The specified PFD has encountered an error in the terrain database.	
PFD 2 DB ERR – PFD 2 terrain database error exists.		
MFD1 DB ERR – MFD1 obstacle database error exists.	The MFD has encountered an error in the obstacle database.	• Confirm supplemental data card is inserted fully. • Reload the database or replace the supplemental datacard. • Contact Garmin Product Support for further assistance if needed.
PFD 1 DB ERR – PFD 1 obstacle database error exists.	The specified PFD has encountered an error in the obstacle database.	
PFD 2 DB ERR – PFD 2 obstacle database error exists.		
MFD1 DB ERR – MFD1 airport terrain database error exists.	The MFD has encountered an error in the airport terrain database.	• Confirm supplemental data card is inserted fully. • Reload the database or replace the supplemental datacard. • Contact Garmin Product Support for further assistance if needed.
PFD 1 DB ERR – PFD 1 airport terrain database error exists.	The specified PFD has encountered an error in the airport terrain database.	
PFD 2 DB ERR – PFD 2 airport terrain database error exists.		
DB MISMATCH – Navigation database mismatch. Xtalk is off.	The system has found different navigation database versions or regions (i.e., Americas, International, Atlantic, etc.) installed in the PFDs and/or MFD. Crossfill is off.	• Check/Load current navigation database versions or regions in all displays. • Check database synchronization function complete.
DB MISMATCH – Standby Navigation database mismatch.	The system has found different navigation database versions or regions (i.e., Americas, International, Atlantic, etc.) installed in the PFDs and/or MFD.	• Check/Load current navigation database versions or regions in all displays. • Check database synchronization function complete.

CHART 8 (Sheet 4 of 6)
MESSAGE ADVISORIES - GDU 1040

Database Message Advisories (cont.)		
Failure Message	Cause	Solution
DB MISMATCH – Terrain database mismatch.	The system has found different terrain database versions or regions (i.e., Americas, International, Atlantic, etc.) installed in the PFDs and/or MFD.	• Check/Load current navigation database versions or regions in all displays. • Check database synchronization function complete.
DB MISMATCH – Obstacle database mismatch.	The system has found different obstacle database versions or regions (i.e., Americas, International, Atlantic, etc.) installed in the PFDs and/or MFD.	
DB MISMATCH – Airport Terrain database mismatch.	The system has found different airport terrain database versions or regions (i.e., Americas, International, Atlantic, etc.) installed in the PFDs and/or MFD.	

Cooling Message Advisories		
Failure Message	Cause	Solution
MFD1 COOLING – has poor cooling. Reducing power usage.	MFD1 has exceeded its operating temperature range.	• Check cooling fan and wiring for proper operation (if applicable). ✓ Replace cooling fan if unable to determine if operating correctly. ✓ If problem persists, replace the MFD. • If problem continues contact Garmin Aviation Product Support for assistance.
PFD 1 COOLING – has poor cooling. Reducing power usage.	PFD 1 has exceeded it's operating temperature range.	• Check cooling fan and wiring for proper operation (if applicable). ✓ Replace cooling fan if unable to determine if operating correctly. ✓ If problem persists, replace the PFD1. • If problem continues contact Garmin Aviation Product Support for assistance.
PFD 2 COOLING – has poor cooling. Reducing power usage.	PFD 2 has exceeded it's operating temperature range.	• Check cooling fan and wiring for proper operation (if applicable). ✓ Replace cooling fan if unable to determine if operating correctly. ✓ If problem persists, replace the PFD2. • If problem continues contact Garmin Aviation Product Support for assistance.

CHART 8 (Sheet 5 of 6)
MESSAGE ADVISORIES - GDU 1040

Key Message Advisories		
Failure Message	Cause	Solution
MFD1 “key” KEYSTK – key is stuck.	The SYSTEM has determined a key is stuck on MFD1.	• Go to the GDU TEST page in configuration mode and verify key is stuck (if key is stuck the corresponding indicator will be green). • Exercise suspected stuck key and reset GDU TEST page to see if indicator remains green without pressing the key. ✓ If problem persists replace the display.
PFD 1 “key” KEYSTK – key is stuck.	The system has determined a key is stuck on the PFD 1.	
PFD 2 “key” KEYSTK – key is stuck.	The system has determined a key is stuck on the PFD 2.	

Miscellaneous Message Advisories		
Failure Message	Cause	Solution
XTALK ERROR – A flight display cross talk error has occurred.	A communication error has occurred between the MFD and/or PFDs.	• Ensure a database error has not occurred (identified in the ALERTS window on the PFD). ✓ If a database error has occurred, correct error before proceeding. • Check display Ethernet interconnect wiring. • Replace PFD1 with a known good unit, to verify location of problem: ✓ If problem persists, reinstall original PFD1 and replace PFD2. ✓ If problem persists, reinstall PFD2 and replace MFD.
DATA LOST – Pilot stored data lost. Recheck settings.	Pilot stored data has been lost.	• If the CLR key was held during a power cycle, disregard message. • Cycle power to PFD1: ✓ Ensure CLR key is not stuck on the GDU TEST page. ✓ If problem persists, replace PFD1.

CHART 8 (Sheet 6 of 6)
MESSAGE ADVISORIES - GDU 1040

Miscellaneous Message Advisories		
Failure Message	Cause	Solution
MFD1 SERVICE – needs service. Return unit for repair.	The G1000 has determined MFD1 needs service.	• Ensure the MFD connector is fully seated and locked. • If the unit was started in a very dark environment the photocells may not have enough light to initially raise the CCFT level. Go to the GDU STATUS page in configuration mode, ensure CCFT CRNT 1 & 2 levels are above 50. ✓ If the CCFT levels are not above 50, apply light to the photocell and observe if the CCFT level rises. ✓ If the CCFT level rises, disregard the message. ✓ If the CCFT level does not rise, replace the MFD • Replace the MFD
PFD 1 SERVICE – needs service. Return unit for repair.	The G1000 has determined the specified PFD needs service.	• Ensure the PFD connector is fully seated and locked. • If the unit was started in a very dark environment the photocells may not have enough light to initially raise the CCFT level. Go to the GDU STATUS page in configuration mode, ensure CCFT CRNT 1 & 2 levels are above 50. ✓ If the CCFT levels are not above 50, apply light to the photocell and observe if the CCFT level rises. ✓ If the CCFT level rises, disregard the message. ✓ If the CCFT level does not rise, replace the PFD • Replace the PFD.
PFD 2 SERVICE – needs service. Return unit for repair.		
PFD 1 VOLTAGE – PFD 1 has low voltage. Reducing power usage.	The specified PFD supply voltage is low.	• Check input voltage to PFD. • If input voltage is ok, replace PFD..
PFD 2 VOLTAGE – PFD 2 has low voltage. Reducing power usage.		
MFD1 VOLTAGE – MDF1 has low voltage. Reducing power usage.	The MFD supply voltage is low.	• Check input voltage to MFD. • If input voltage is ok, replace MFD.

(3) GMA 1347 Audio Panel

See “Figure 12”.

(a) Description

The Garmin GMA 1347 Audio Panel integrates NAV/COM digital audio, intercom system and marker beacon controls. The GMA 1347 panel provides control of all cockpit intercom/mic systems as well as NAV/COM/ILS audio. The unit also provides display reversion mode control through a large red button. Power to the audio panel is normally received from the Essential Bus #1 when the Battery Master switch is turned on or by applying external power. An Emergency Bus also could supply power when the Emergency Power switch is Armed. The GMA 1347 unit interfaces with the existing marker beacon antenna as well as existing mic and phone jacks and oxygen mask mic.



GMA 1347 Audio Panel
Figure 12

(b) Troubleshooting

See “Chart 9” on page 34250149 and “Chart 10” on page 34250150.

(c) Removal

1) Using a 3/32nd hex tool, turn the hex nut counter-clockwise until the GMA 1347 is unlocked from its location.
2) Carefully remove the GMA 1347 from its rack.

(d) Installation

1) Visually inspect the connectors using a flashlight to ensure there are no bent or damaged pins. Repair any damage.
2) Gently insert the GMA 1347 into the rack until the locking tab engages the rack.
3) Begin to turn the hex nut clockwise. This draws the unit into the rack until seated. Do not overtighten the nut.

(e) Return to Service

1) Original GMA 1347 Reinstalled.

No software/configuration loading or testing is required if the removed GMA 1347 is reinstalled. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

Continue to the final return-to-service checks.

2) New, Repaired, or Exchanged GMA 1347 Installed.

If a new, repaired, or exchanged GMA 1347 is installed, the correct software and configuration files must be loaded to the unit. See “G1000 Software/Configuration Procedure” on page 342501107.

3) Test

Perform “GMA 1347 Test” on page 342501126.

CHART 9
TROUBLESHOOTING - GMA

Symptom/Failure Message	Recommended Action
Noise in Audio	Most often the cause of the noise is external to the GMA. Try the following to locate the source of the noise before replacing the GMA: • Try a different pair of headsets. Noise cancelling headsets may pick up and/or generate more noise than standard headsets from their own circuitry. • Check for noise with the engine turned off. ✓ If the noise is present only when the engine is running, check the generator and/or ignition system as possible sources of noise (see applicable airframe specific maintenance manual). • Check for noise as all electrical equipment is turned on and off (strobes, other radios, etc.). ✓ If the noise is identified from one electrical system or component refer to the applicable airframe specific maintenance manual. • Ensure the NAV/COM squelch is not open. • Ensure the ADF and DME audio is not active. • Ensure the marker beacon audio is not active. • Ensure the ICS squelch is not open. ✓ Master squelch level can be adjusted on the GMA CONFIGURATION page for higher noise environments. • Replace unit only after all possible external sources of noise are eliminated.
Buttons Do Not Work.	• Some buttons are disabled in the GMA CONFIGURATION page by default. This is to remove potential sources of audio noise for inputs that are not used. If in doubt as to which buttons should be disabled, reload GMA config files and other config files for optional equipment installed in the aircraft (i.e. ADF, HF, etc.) from the loader card.
COM Bleedover	• Verify on the GMA CONFIGURATION page that “Disable Split COM” has a green box. Due to the closeness of the COM antennas and high power of the COM transceivers, Split COM operation is not approved. If the box is black (indicating COM ½ button is active), highlight “Disable Split COM” with the cursor and press the ENT key to turn the box green which will deactivate Split COM mode.
Speaker Cuts Out	• Reduce volume level of the item that caused the speaker to cut out when turned up. A speaker protection circuit disables the speaker output if the volume is too high. If the volume is not sufficient, replace aircraft cabin speaker, reference the Airframe Maintenance Manual.
Mic Audio Heard in Speaker	• Reduce ICS Volume.

CHART 10
MESSAGE ADVISORIES - GMA

Failure Message	Cause	Solution
GMA1 SERVICE – GMA1 needs service. Return unit for repair.	The system has determined that the audio panel needs service.	• Replace GMA.
GMA1 FAIL – GMA1 is inoperative.	The system has detected a failure in the audio panel.	• Ensure GMA1 is receiving power. • Ensure both GIAs are receiving power. • Ensure all GDUs are receiving power. • Ensure the GMA/GIA RS-232 data lines are working properly. • Ensure the GIA/GDU Ethernet data lines are working properly. • Replace GMA.
MANIFEST – GMA1 software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the audio panel.	Load the correct unit software. See Software/Configuration (above) for the Software Loading procedure.
GMA1 CONFIG – GMA1 configuration error. Config service req'd.	The system has detected a audio panel configuration mismatch for the unit.	• Load GMA configuration files. See Software/Configuration (above). ✓ Replace GMA. ✓ If problem persists, replace master configuration module, check config module wiring for faults and replace if necessary. NOTE New Terrain/Obstacle cards, Navigation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 System ID number will change to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number.

(4) GIA 63W Integrated Avionics Unit

See “Figure 13”.



GIA 63W Integrated Avionics Unit
Figure 13

(a) Description

Two Garmin GIA 63W Integrated Avionics Units (IAUs) contain the VHF COM/NAV receivers, WAAS GPS receiver, Flight Director, and system integration microprocessors. The GIAs also serve as a communication interface to all other G1000 LRUs in the system. Each GIA 63W communicates directly with the on-side GDU 1040 display using a HSDB Ethernet connection. Both GIAs are located directly behind the PFD1 and PFD2, installed into their respective LRU racks. Power is provided to GIA1 by the Essential Bus #1 or the Emergency Bus. GIA2 is powered by Avionics Bus #2. Both GIA 63s interface to the following equipment:

- VOR/LOC/Glideslope Antenna System
- VHF COM #1 & #2 Antennas
- GA 36 GPS/WAAS Antennas
- GDU 1040A, #1 & #2
- GMA 1347
- GDC 74A #1 (& GDC 74A #2, if installed)
- GRS 77 #1 (& GRS 77 #2, if installed)
- GEA 71 (Left and Right)
- GTX 33

The GIA 63 #1 interfaces to the following additional equipment:

- S-TEC 55X Autopilot (if installed)
- Fuel Quantity Sensors (left and right)

The GIA 63 #2 interfaces to the following additional equipment:

- KR 87 ADF (if installed)
- KN 63 DME (if Installed)
- WX 500 Stormscope (if installed)
- GSR 56 (if installed)

NOTE: The GIA 63W is compatible only with the antennas listed in the GIA 63W Installation Manual, Garmin P/N 190-00303-05.

(b) Troubleshooting

See “Chart 11” on page 34250153 and “Chart 12” on page 34250155.

(c) Removal

1) Remove the MFD as described in “Removal” on page 34250138.
2) Unlock the GIA 63W handle by loosening the Phillips screw on the handle.
3) Pull the handle upward to unlock the GIA 63W. Gently remove the unit from the rack.

(d) Installation

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
2) Gently insert the GIA 63W into its rack. The handle should engage the dogleg track.
3) Press down on the GIA 63W hand to lock the unit into the rack.
4) Lock the handle to the GIA 63W body using the Phillips screw.
5) Reinstall the MFD as described in “Installation” on page 34250138.

(e) Return to Service

1) Original GIA 63W(s) Reinstalled

No software or configuration loading is required if the removed GIA is reinstalled in its original position (GIA1 and GIA2 in their original racks). This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) Original GIA 63W(s) Swapped for Troubleshooting

No software loading is required if the originally installed GIA units are reinstalled in opposite positions (GIA1 and GIA2 in opposite unit racks). However, configuration loading is required. See “G1000 Software/Configuration Procedure” on page 342501107.

3) New, Repaired, or Exchanged GIA 63W(s) Installed

If a new, repaired, or exchanged GIA 63 is installed, the correct software and configuration files must be loaded to the unit. See “G1000 Software/Configuration Procedure” on page 342501107.

4) Test

Perform “GIA 63W Test” on page 342501128.

CHART 11 (Sheet 1 of 2)
TROUBLESHOOTING - GIA 63W

GIA 63W Troubleshooting COM	
Symptom	Recommended Action
Weak COM transmit power	• Switch GIA1 and GIA2, to verify location of problem: ✓ If problem follows unit, replace GIA. ✓ If problem does not follow unit, check COM antenna and cabling for faults.
Weak COM receiver	• Switch GIA1 and GIA2, to verify location of problem: ✓ If problem follows unit, replace GIA. ✓ If problem does not follow unit, check COM antenna and cabling for faults.
No COM sidetone	• Switch GIA1 and GIA2, to verify location of problem: ✓ If problem follows unit, replace GIA. ✓ If problem persists, replace GMA1 with a known good unit.

GIA 63W Troubleshooting NAV	
Symptom	Recommended Action
Weak NAV receiver	• Set up a NAV/COM Ramp Test Set to radiate a test signal. • Switch GIA1 and GIA2, to verify location of problem: ✓ If problem follows unit, replace GIA. ✓ If problem does not follow unit, check NAV antenna, coupler, and cabling for faults.

GIA 63W Troubleshooting G/S	
Symptom	Recommended Action
Weak G/S receiver	• Set up a NAV/COM Ramp Test Set to radiate a test signal. • Switch GIA1 and GIA2, to verify location of problem: ✓ If problem follows unit, replace GIA. ✓ If problem does not follow unit, check NAV antenna, coupler, and cabling for faults.

CHART 11 (Sheet 2 of 2)
TROUBLESHOOTING - GIA 63W

GIA 63W Troubleshooting GPS	
Symptom	Recommended Action
Will Not Acquire Satellites	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Using the MFD AUX – GPS Status page, verify the signal strength bars are not erratic. If so, this indicates outside interference is affecting the GPS receivers. Find and remove the source of interference (i.e. cell phones, FBO datalink antennas, etc.). • Check date and time on Date/Time Setup Page. ✓ If date and time are incorrect, enter the correct date and time. • Switch GIA1 and GIA2, to verify location of problem: ✓ If problem follows unit, clear the GPS almanac by performing the following steps – o Using the PFD in config mode, go to the GIA RS-232/ARINC 429 Config Page. o At the top of the screen, select the GIA that can not acquire satellites (GIA1 or GIA2) and press the ENT key. o Press the “CLR NV” softkey at the bottom of the screen. o Select “OK” in the “Clear GIA nonvolatile memory?” pop-up window. o Next reload GIA Audio and Config files from a loader card, see G1000 Software/Configuration Procedure, for instructions. Be sure to reload the config files for any optional equipment installed on the aircraft that require the GIA config to be updated. • Cycle power on the system and allow it to restart in normal mode. Place the aircraft outside and allow 15-30 minutes for the GPS to acquire a position and download a new almanac. ✓ If clearing nonvolatile memory is unsuccessful and the GPS still can not acquire a position, replace the GIA. ✓ If problem does not follow unit, check GPS antenna and cabling.

CHART 12 (Sheet 1 of 6)
MESSAGE ADVISORIES - GIA 63W

GIA 63W Message Advisories COM Messages		
Failure Message	Cause	Solution
COM1 SERVICE – COM1 needs service. Return unit for repair.	The system has determined COM1 needs service.	• Replace GIA1 according to instructions in Removal/Installation.
COM2 SERVICE – COM2 needs service. Return unit for repair.	The system has determined COM2 needs service.	• Replace GIA2 according to instructions in Removal/Installation.
COM1 PTT – COM1 push-to-talk key is stuck.	The COM1 external push-to-talk (PTT) switch is stuck in the enabled (or “pressed”) state.	• Press the push-to-talk switch(s) again to cycle its operation. • Check push-to-talk switch(s) and wiring. • Check GIA1/GMA interconnect. • Switch GIA1 and GIA2, to identify whether the unit or connectors/wiring is at fault (Both GIAs must be configured when swapped; see Software/Configuration above): ✓ If problem follows the unit, replace GIA1. ✓ If problem persists replace defective GMA.
COM2 PTT – COM2 push-to-talk key is stuck.	The COM2 external push-to-talk (PTT) switch is stuck in the enabled (or “pressed”) state.	• Press the push-to-talk switch(s) again to cycle its operation. • Check push-to-talk switch(s) and wiring. • Check GIA2/GMA interconnect. • Switch GIA1 and GIA2, to identify whether the unit or connectors/wiring is at fault (Both GIAs must be configured when swapped, see Software/Configuration above): ✓ If problem follows the unit, replace GIA2. ✓ If problem persists replace defective GMA.

CHART 12 (Sheet 2 of 6)
MESSAGE ADVISORIES - GIA 63W

GIA 63W Message Advisories COM Messages		
Failure Message	Cause	Solution
COM1 RMT XFR – COM1 remote transfer key is stuck.	The COM1 external remote transfer switch is stuck in the enabled (or “pressed”) state.	Press the COM1 external remote transfer switch again to cycle its operation. Check COM1 external remote transfer switch and wiring. Switch GIA1 and GIA2, to identify whether the unit or connectors/wiring is at fault (Both GIAs must be configured when swapped, see Software/ Configuration above): ✓ If problem follows the unit, replace GIA1. ✓ If problem persists, continue to troubleshoot remote transfer switch & wiring.
COM2 RMT XFR – COM2 remote transfer key is stuck.	The COM2 external remote transfer switch is stuck in the enabled (or “pressed”) state.	Press the COM2 external remote transfer switch again to cycle its operation. Check COM2 external remote transfer switch and wiring. Switch GIA1 and GIA2, to identify whether the unit or connectors/wiring is at fault (Both GIAs must be configured when swapped, see Software/ Configuration above): ✓ If problem follows the unit, replace GIA2. ✓ If problem persists, continue to troubleshoot remote transfer switch & wiring.
COM1 TEMP – COM1 over temp. Reducing transmitter power.	The specified COM transceiver is reporting a high temperature condition and is reducing transmit power to prevent damage.	• Check fan, wiring and air tubing for proper operation (if applicable). • Replace cooling fan if unable to determine if operating correctly. • Replace GIA. • If problem persists contact Garmin Aviation Product Support for assistance.
COM2 TEMP – COM2 over temp. Reducing transmitter power.		

CHART 12 (Sheet 3 of 6)
MESSAGE ADVISORIES - GIA 63W

GIA Message Advisories NAV Messages		
Failure Message	Cause	Solution
NAV1 SERVICE – NAV1 needs service. Return unit for repair.	The system has detected a failure in NAV1 receiver.	• Replace GIA1.
NAV2 SERVICE – NAV2 needs service. Return unit for repair.	The system has detected a failure in NAV2 receiver.	• Replace GIA2.
NAV1 RMT XFR – NAV1 remote transfer key is stuck.	The NAV1 external remote transfer switch is stuck in the enabled (or “pressed”) state.	Press the NAV1 external remote transfer switch again to cycle its operation. Check NAV1 remote transfer switch and wiring. Switch GIA1 and GIA2, to identify whether the unit or connectors/wiring is at fault (Both GIAs must be configured when swapped, see Software/ Configuration above): ✓ If problem follows unit, replace GIA1. ✓ If problem persists, continue to troubleshoot remote transfer switch & wiring.
NAV2 RMT XFR – NAV2 remote transfer key is stuck.	The NAV2 external remote transfer switch is stuck in the enabled (or “pressed”) state.	Press the NAV2 external remote transfer switch again to cycle its operation. Check NAV2 remote transfer switch and wiring. Switch GIA1 and GIA2, to identify whether the unit or connectors/wiring is at fault (Both GIAs must be configured when swapped, see Software/ Configuration above): ✓ If problem follows unit, replace GIA1. ✓ If problem persists, continue to troubleshoot remote transfer switch & wiring.

CHART 12 (Sheet 4 of 6)
MESSAGE ADVISORIES - GIA 63W

GIA Message Advisories Glideslope Messages		
Failure Message	Cause	Solution
G/S1 SERVICE – G/S1 needs service. Return unit for repair.	The system has detected a failure in G/S1 receiver.	• Replace GIA1.
G/S2 SERVICE – G/S2 needs service. Return unit for repair.	The system has detected a failure in G/S2 receiver.	• Replace GIA2.
G/S1 FAIL – G/S1 is inoperative.	The system has detected a failure in G/S1 receiver.	• Switch GIA1 and GIA2 to verify location of problem: ✓ If problem follows the unit, replace GIA. ✓ If problem does not follow unit, check G/S1 antenna and cabling.
G/S2 FAIL – G/S2 is inoperative.	The system has detected a failure in G/S2 receiver.	• Switch GIA1 and GIA2 to verify location of problem: ✓ If problem follows the unit, replace GIA. ✓ If problem does not follow unit,. check G/S2 antenna and cabling.

GIA Message Advisories GPS Messages		
Failure Message	Cause	Solution
MANIFEST – GPS1 software mismatch. Communication halted.	The system has detected an incorrect software version loaded in GIA1.	• Load the correct GPS software. See Software/Configuration above): for the Software Loading procedure.
MANIFEST – GPS2 software mismatch. Communication halted.	The system has detected an incorrect software version loaded in GIA2.	
GPS1 SERVICE – GPS1 needs service. Return unit for repair.	The system has detected a failure in GPS1 receiver.	• Replace GIA.
GPS2 SERVICE – GPS2 needs service. Return unit for repair.	The system has detected a failure in GPS2 receiver.	
GPS1 FAIL – GPS1 is inoperative.	The system has detected a failure in GPS1 receiver.	• Switch GIA1 and GIA2, to verify location of problem: ✓ If problem follows the unit, replace GIA. ✓ If problem does not follow the unit, check GPS antenna and cabling.
GPS2 FAIL – GPS2 is inoperative.	The system has detected a failure in GPS2 receiver.	
NOTE Before troubleshooting, ensure that no cell phones or devices using cell phone technology are turned on, even in a monitoring state, in the cabin.		

CHART 12 (Sheet 5 of 6)
MESSAGE ADVISORIES - GIA 63W

GIA Message Advisories GPS Messages (cont.)		
Failure Message	Cause	Solution
LOI – GPS integrity lost. Crosscheck with other NAVS.	If the primary receiver is a WAAS sensor, the alert indicates that GPS position data has timed out.	• Verify the area the aircraft was traveling through did not have loss of GPS coverage. FAA NOTAMs may be issued for periods of outages, or the US Coast Guard website http://www.navcen.uscg.gov/ will have notices posted. • Using the MFD AUX – GPS Status page, verify the signal strength bars are not erratic. If so, this indicates outside interference is affecting the GPS receivers. Find and remove the source of interference (i.e. cell phones, FBO datalink antennas, etc.). • If GPS receivers can not aquire a position lock, troubleshoot per the “Will not acquire satellites” section.
GPS NAV LOST – Loss of GPS navigation. Insufficient satellites.	There is no GPS position fix available or the system is in dead reckoning mode.	
GPS NAV LOST – Loss of GPS navigation. Position error.	The G1000 has detected an internal position warning has occurred.	
GPS NAV LOST – Loss of GPS navigation. GPS fail.	The G1000 has detected a GPS engine failure.	

GIA Message Advisories GIA Cooling Messages		
Failure Message	Cause	Solution
GIA1 COOLING – GIA1 temperature too low.	GIA1 operating temperature is too low.	• Allow unit to warm up.
GIA2 COOLING – GIA2 temperature too low.	GIA2 operating temperature is too low.	• Allow unit to warm up.
GIA1 COOLING – GIA1 over temperature.	GIA1 has exceeded its operating temperature range.	• Check fan, wiring and air tubing for proper operation (if applicable). • Replace cooling fan if unable to determine if operating correctly. • Replace GIA1. • If problem persists contact Garmin Aviation Product Support for assistance.
GIA2 COOLING – GIA2 over temperature.	GIA2 has exceeded its operating temperature range.	• Check fan, wiring and air tubing for proper operation (if applicable). • Replace cooling fan if unable to determine if operating correctly. • Replace GIA2. • If problem persists contact Garmin Aviation Product Support for assistance.

CHART 12 (Sheet 6 of 6)
MESSAGE ADVISORIES - GIA 63W

GIA Message Advisories GIA Configuration Messages		
Failure Message	Cause	Solution
MANIFEST – GIA1 software mismatch. Communication Halted.	The system has detected an incorrect software version loaded in GIA1.	• Load the correct software. See Software/Configuration (above) for the Software Loading procedure.
MANIFEST – GIA2 software mismatch. Communication Halted.	The system has detected an incorrect software version loaded in GIA2.	
GIA1 CONFIG – GIA1 audio config error. Config service req'd.	The GIA's audio configuration files are incorrect or missing.	• Reload audio configuration files.
GIA2 CONFIG – GIA2 audio config error. Config service req'd.		
GIA1 CONFIG – GIA1 configuration error. Config service req'd.	The system has detected a GIA configuration mismatch. If GIAs are not properly configured after being swapped/replaced, this message appears.	• Load the configuration files for that GIA. See Software/Configuration (above) for GIA 63W Configuration Loading procedure. • If problem persists, replace master configuration module, check config module harness for faults and replace if necessary. NOTE New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 System ID number will change to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number.
GIA2 CONFIG – GIA2 configuration error. Config service req'd.		
HW MISMATCH – GIA hardware mismatch, GIA1 communication halted.	The G1000 has detected a non-WAAS GIA 63.	• Replace GIA with a WAAS unit.
HW MISMATCH – GIA hardware mismatch, GIA2 communication halted.		
GIA1 SERVICE – GIA1 needs service. Return unit for repair.	The G1000 has detected a failure in the specified GIA.	• Replace suspect GIA.
GIA2 SERVICE – GIA2 needs service. Return unit for repair.		

(5) GEA 71 Engine/Airframe Unit

See “Figure 14”.

(a) Description

The Garmin GEA 71 Engine/Airframe Unit (1 per engine) provides engine/airframe data to the G1000 system. Data received from transducers/ sensors is processed and sent to GIA 63s (via RS-485 digital interface), and subsequently to the GDU 1040 MFD. Engine parameters are normally displayed on the MFD. In the event of a MFD failure, all engine instruments can be displayed on the PFD’s. The GEAs are located behind PFD1 and MFD and are mounted in a vertical orientation. Power is received from the Essential buses or the Emergency bus. The GEA interfaces to the following:

- Manifold Pressure Sensor (MAP)
- Tachometer Sensor (RPM)
- Turbine Inlet Temperature (TIT) Sensor
- Fuel Flow Sensor
- Oil Pressure Sensor
- Oil Temperature Sensor
- Fuel Senders
- Flap Position potentiometer.
- Alternator & Battery Current Monitor Shunts
- Six Cylinder Head Temperature (CHT) Sensors
- Pitot Heat System
- Vacuum Transducer.
- Door Ajar switch



GEA 71 Engine/Airframe Unit
Figure 14

(b) Troubleshooting

See “Chart 13” on page 34250163.

(c) Removal

1) Remove the PFD/MFD as described in “Removal” on page 34250138.
2) Unlock the GEA 71 handle by unscrewing the Phillips screw.
3) Pull the handle upward to unlock the GEA 71.
4) Gently remove the GEA 71 from its rack.

(d) Installation

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
2) Gently insert the GEA 71 into the rack. The handle should engage the dogleg track.
3) Press down on the handle to lock the unit into place.
4) Lock the handle to the GEA 71 body using the Phillips screw.
5) Reinstall the PFD/MFD as described in “Installation” on page 34250138.

(e) Return to Service

1) Original GEA 71 Reinstalled

No software or configuration loading is required if the removed GEA 71 is reinstalled. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) New, Repaired, or Exchanged GEA 71 Installed

3) If a new, repaired, or exchanged GEA 71 is installed, the correct software and configuration files must be loaded to the unit. See “G1000 Software/Configuration Procedure” on page 342501107.

4) Test

Perform “GEA 71 Test” on page 342501130.

(6) GEA 71 Backshell Thermocouple

See “Figure 15”.

(a) Removal

1) Remove GEA 71 as described above.

2) Remove GEA connector backplate.

3) Remove connector J701 (5) from the backplate.

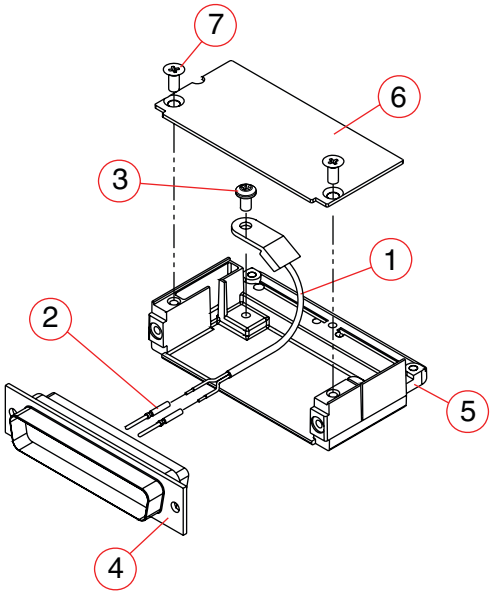
4) Remove cover (6) from the backshell.

5) Unscrew thermocouple from boss on backshell. Extract the thermocouple pins from the connector.

(b) Installation

1) Crimp pins (2) onto each of the thermocouple wires (1). Ensure that pre-stripped wire length is 1/8” prior to crimping.

2) Insert newly crimped pins and wires into the appropriate connector housing location (4) as specified in the electrical schematic.



GEA 71 Backshell Thermocouple
Figure 15

3) Place thermocouple body (1) onto the backshell boss (5). Place the thermocouple as shown in “Figure 15” so that the wires exit towards the bottom of the backshell.

4) Fasten thermocouple tightly to backshell using the provided screw (3).

5) Fasten cover (6) to backshell using the provided screws (7).

CHART 13
TROUBLESHOOTING / MESSAGE ADVISORIES - GEA

Failure Message	Cause	Solution
MANIFEST – GEA1 software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the specified GEA 71.	Load the correct software version. See GEA 71 Software Load Procedure.
MANIFEST – GEA2 software mismatch. Communication halted.		
GEA1 CONFIG – GEA1 configuration error. Config service req'd.	The system has detected a configuration mismatch in the specified GEA 71.	• Load GEA configuration files. See GEA 71 Configuration Procedure. • If problem persists, replace master configuration module, check config module harness for faults and replace if necessary. NOTE New Terrain/Obstacle cards, Navigation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 System ID number will change to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number.
GEA2 CONFIG – GEA2 configuration error. Config service req'd.		

(7) GTX 33 Mode S Transponder

See “Figure 16”.



GTX 33 Mode S Transponder
Figure 16

(a) Description

The Garmin GTX 33 provides Mode A, C, and S altitude and position reporting information from the G1000 system. The unit is mounted directly behind the MFD on the center avionics shelf. Power is received from Essential Bus #1. Similarly to the GEA 71, the GTX 33 sends data via RS-232 directly to a GIA 63W. Information is then sent to the PFD, where the pilot can control the transponder. The GTX 33 is connected to both GIA 63Ws for redundant communications. The GTX 33 interfaces with the transponder antenna.

(b) Troubleshooting

See “Chart 14” on page 34250165.

(c) Removal

1) Remove the MFD as described in “Removal” on page 34250138.
2) Unlock the GTX 33 handle by loosening the Phillips screw on the handle.
3) Pull the handle upward to unlock the GTX 33. Gently remove the unit from the rack.

(d) Installation

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
2) Gently insert the GTX 33 into its rack. The handle should engage the dogleg track.
3) Press down on the GTX 33 handle to lock the unit into the rack.
4) Lock the handle to the GTX 33 body using the Phillips screw.
5) Reinstall the MFD as described in “Installation” on page 34250138.

(e) Return to Service

1) Original GTX 33 is Reinstalled

No software or configuration loading is required if the removed GTX 33 is re-installed. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) New, Repaired, or Exchanged GTX 33 is Installed

If a new, repaired, or exchanged GTX 33 is installed, the correct software and configuration files must be loaded to the unit. See “G1000 Software/Configuration Procedure” on page 342501107, reload the software, and then enter the aircraft registration number per “GTX 33 Transponder” on page 342501130.

3) Test

Perform “Test” on page 342501130.

CHART 14
TROUBLESHOOTING / MESSAGE ADVISORIES - GTX 33

Failure Message	Cause	Solution
MANIFEST – GTX1 software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the specified GTX 33.	• Reload software. See Software Load / Configuration Procedure.
MANIFEST – GTX2 software mismatch. Communication halted.		
XPDR1 CONFIG – XPDR1 configuration error. Config service req'd.	The system has detected a configuration mismatch for the specified GTX 33.	• Perform a SET>ACTV configuration reset on the GTX Config page and verify the aircraft registration is present. • If error is still present, reload config files from a loader card. ✓ If problem persists, replace master configuration module, check config module harness for faults and replace if necessary. NOTE New Terrain/Obstacle cards, Navigation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 System ID number will change to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number.
XPDR2 CONFIG – XPDR2 configuration error. Config service req'd.		
XPDR1 SRVC – XPDR1 needs service. Return unit for repair.	The G1000 has detected a failure in the specified GTX 33.	• Replace GTX 33.
XPDR2 SRVC – XPDR2 needs service. Return unit for repair.		
XPDR1 FAIL – XPDR 1 is inoperative.	The specified GTX 33 is not responding.	• Check wiring between GIA's and GTX. • Replace GTX 33.
XPDR2 FAIL – XPDR 2 is inoperative.		

(8) GDC 74A Air Data Computer

See “Figure 17”.

(a) Description

The Garmin GDC 74A computer(s) compile information from the pitot/static system and the GTP 59 outside air temperature (OAT) probe(s) to digital air data computations to the G1000 system. The GDC 74A communicates with the GIA 63W, GDU 1040, and GRS 77 using ARINC 429 digital interface. The unit(s) are mounted behind PFD2. Power is provided to #1 GDC 74A unit (ADC1) from the Essential Bus #1 or the Emergency Bus. The #2 GDC 74A unit (ADC2) is powered by the Avionics bus #2.



GDC 74A Air Data Computer
Figure 17

NOTE: If any connections in the pitot/static system are opened for maintenance, the entire system must be rechecked per Pitot and Static Systems, Test, in 34-10-00.

(b) Troubleshooting

See “Chart 15” on page 34250168.

(c) Removal

1) Remove the PFD2 as described “Installation” on page 34250138.
2) Disconnect the pitot/static plumbing from the rear of the unit. Disconnect the single connector.
3) Loosen each thumbscrew on the hold-down clamp and remove the clamp.
4) Carefully remove the unit from its mount.

(d) Installation

1) Place the unit in the mounting tray.
2) Position the locking clamp and fasten using the thumbscrews.
3) Connect the pitot/static plumbing.
4) Inspect the connector and pins for damage. Repair any damage. Connect the connector to the unit.
5) Reinstall the PFD2 as described in “Installation” on page 34250138.

(e) Return to Service

NOTE: If any connections in the pitot / static system are opened for maintenance, the entire system must be rechecked per Pitot and Static System "Test" on page 34108.

1) Original GDC 74A is Reinstalled

No software or configuration loading is required if the removed GDC 74A is reinstalled. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) Original GDC 74A Installed in Opposite Locations for Troubleshooting

No software loading is required if the original GDC1 and GDC2 are installed in opposite locations.

3) New, Repaired, or Exchanged GDC 74A is Installed

If a new, repaired, or exchanged GDC 74A is installed, the correct software and configuration files must be loaded to the unit. See "G1000 Software/Configuration Procedure" on page 342501107.

4) New GDC 74A Configuration Module is Installed

The correct configuration files must be loaded if the GDC 74A configuration module has been replaced. See "G1000 Software/Configuration Procedure" on page 342501107.

5) Test

Perform "GDC 74A Tests" on page 342501132.

CHART 15
TROUBLESHOOTING / MESSAGE ADVISORIES - GDC 74A

Symptom	Recommended Action
Altitude is different than standby altimeter	• Perform a pitot/static check per 34-10 Pitot Static Systems in this manual. Allow the GDC to warm up for fifteen minutes before checking accuracy, per Garmin Service Advisory 0606. • Determine which instrument is outside limits and replace. The GDC may be field adjusted per Garmin Service Advisory 0720. Note: Both units may individually be in spec but show a difference in altitude. Do not return the GDC to Garmin for service if not outside limits.
GDC Config file does not load.	• Replace GDC config module. ✓ If problem persists, replace GDC config module wire harness.

Failure Message	Cause	Solution
MANIFEST – GDC1 software mismatch Communication halted.	The system has detected an incorrect software version loaded in the specified GDC 74A.	• Load correct software version. See Software Configuration Procedures.
MANIFEST – GDC2 software mismatch Communication halted.		
ADC1 ALT EC – ADC1 altitude error correction is unavailable.	The specified GDC 74A is reporting the absence of altitude error correction.	
ADC2 ALT EC – ADC2 altitude error correction is unavailable.		
ADC1 AS EC – ADC1 airspeed error correction is unavailable.	The specified GDC 74A is reporting the absence of airspeed error correction.	
ADC2 AS EC – ADC2 airspeed error correction is unavailable.		

(9) GTP 59 OAT Probe

See “Figure 18”.

(a) Description

The Garmin GTP 59 OAT Probe provides the GDC 74A with air temperature data. The OAT probe(s) are mounted to the underside of each wing, outboard of W.S. 164.72.

(b) Troubleshooting

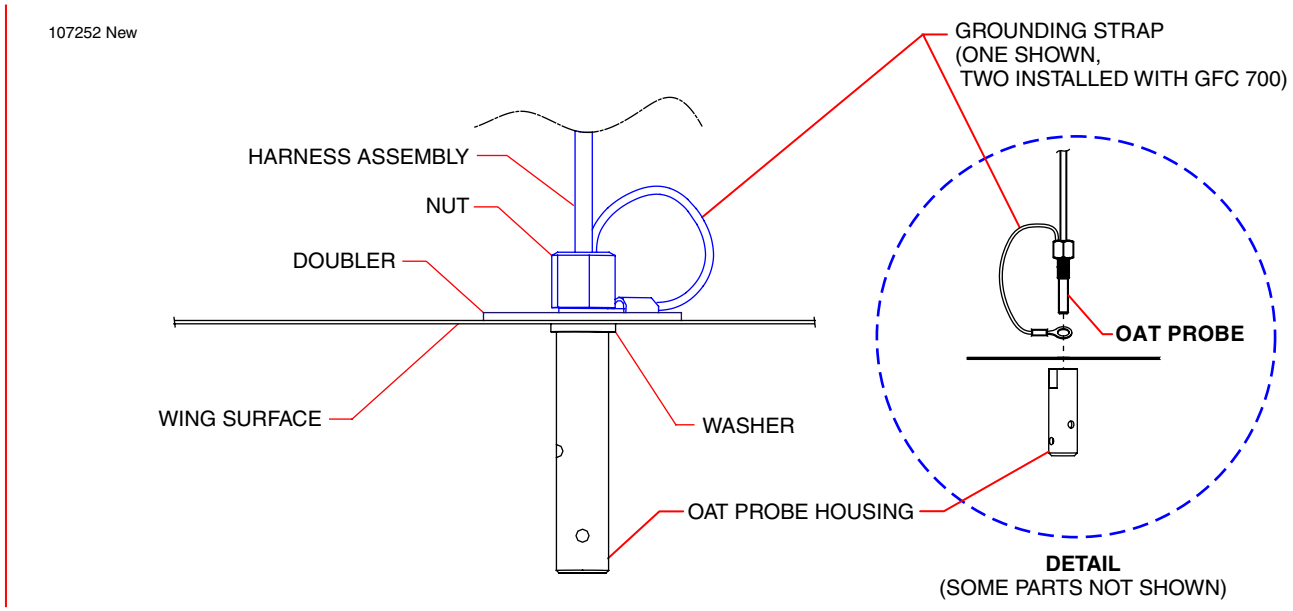
See “Chart 1” on page 34250112.

(c) Removal

1) Remove the access cover.
2) Disconnect the connector.
3) Use an open-end wrench to hold the probe in place on the inside of the wing. On the outside of the wing, loosen the GTP 59 Housing and remove the GTP 59.

(d) Installation

1) Place the GTP-59 grounding strap(s) O-ring connector(s) over the OAT probe end and insert the probe through the wing from the inside. Secure with washer and housing on the outside of the wing. Use an open-end wrench to hold the GTP-59 on the inside and torque the housing on the outside to 100 ± 20 in·lbs.
2) Reconnect the connector to the airplane wiring harness.
3) Reinstall access cover adjacent to the OAT probe.
4) Verify ambient temperature OAT display on each PFD.



GTP 59 OAT Probe
Figure 18

(10) GRS 77 Attitude and Heading Reference System

See “Figure 19”.



GRS 77 AHRS Unit
Figure 19

(a) Description

The Garmin GRS 77 AHRS unit(s) provide attitude and heading information to the G1000 system. The units, mounted below the aft passenger seats, contain advanced tilt sensors, accelerometers, and rate sensors. The unit interfaces with the GDC 74A and GMU 44 Magnetometer and utilizes GPS signals from the GIA 63Ws. Actual attitude and heading information is sent using ARINC 429 digital interface to both GDU 1040s and GIA 63Ws. Power is provided to #1 GRS 77 unit (AHRS 1) from the Essential Bus #1 or the Emergency Bus. The #2 GDC 74A (AHRS 2) is powered by the Avionics Bus #2. The GRS 77 interfaces with and provides power to the GMU 44 Magnetometer. The GRS 77 supplies attitude and heading information directly to the PFDs, MFD, and GIAs.

NOTE: GRS 77 Earth Magnetic Field Updates

The GRS 77 utilizes an Earth magnetic field model which is updated once every five years. The update is expected to be available from Garmin in each of the following years: 2010, 2015, and every five years thereafter, as long as the GRS 77 remains a Garmin-supported product. The G1000 system alerts the operator that the magnetic field database is out of date by issuing the message “AHRS SERVICE – AHRS Magnetic-field model needs update”. Garmin will distribute update instructions when updates are available.

(b) Troubleshooting

See “Chart 16” on page 34250172 and “Chart 17” on page 34250173.

(c) Removal

1) Remove aft passenger seats (See 25-10-00).
2) Disconnect the GRS 77 AHRS connector.
3) Remove the four Phillips thumbscrews with a screwdriver and set them aside.
4) Gently lift the GRS 77 from the mounting plate. (If the mounting plate is removed, the GRS 77 must be re-calibrated. See “Post-Installation Procedures” on page 342501119. See “LRU and System Test Procedures” on page 342501125)

(d) Installation

1) Level the airplane per 8-20-00.
2) Verify GRS 77 bracket assembly is level within 3.0 degrees.
3) Place the GRS 77 on the mounting plate, ensuring the general orientation is correct.
4) Align forward direction of the GRS 77 in heading to within 1.0 degree of the aircraft heading.
5) Fasten the unit to the plate using the Phillips thumbscrews. Recommended torque is 22-25 inch pounds.
6) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage. Connect the connector to the GRS 77, ensuring that each slidelock is secured on both sides.
7) Reinstall aft passenger seats (See 25-10-00).

(e) Return to Service

1) Original GRS 77 is Reinstalled

If the original GRS 77 was reinstalled, then no software loading is required. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

If the GRS rack was not removed or loosened, continue to Test, below.

If the GRS rack was removed or loosened, continue to the “GRS/GMU Calibration Procedures” on page 342501134.

2) Original GRS 77 Installed in Opposite Locations for Troubleshooting

If the original GRS1 and GRS2 are installed in opposite locations (GRS1 and GRS2 in opposite unit racks), no software loading is required.

If the GRS rack(s) was not removed or loosened, continue to Test, below.

If the GRS rack(s) was removed or loosened, continue to the “GRS/GMU Calibration Procedures” on page 342501134.

3) New, Repaired, or Exchanged GRS 77 is Installed

If a new, repaired, exchanged GRS 77 is installed, then software must be loaded and the unit must be calibrated. See “G1000 Software/Configuration Procedure” on page 342501107. Then, proceed as follows:

If the GRS rack(s) was not removed or loosened, continue to Test, below.

If the GRS rack(s) was removed or loosened, continue to the “GRS/GMU Calibration Procedures” on page 342501134.

4) New GRS 77 Configuration Module is Installed

If the GRS 77 configuration module was replaced, no software loading is required, but the unit must be recalibrated. Continue to the “GRS/GMU Calibration Procedures” on page 342501134.

5) Test

Perform “GRS/GMU Test” on page 342501142.

CHART 16
TROUBLESHOOTING - GRS 77 (AHRS) / GMU 44

Symptom	Recommended Action
AHRS does not complete initialization	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Ensure GPS has acquired at least four satellites, has a 3D navigation solution, and a DOP of less than 5.0. This is particularly important if this issue appears during ground operation only. • Calibrate the GRS 77. • Check GRS 77 configuration module wiring for damage. • Check GRS 77 connector for bent pins. ✓ If no damage can be found, replace GRS 77 configuration module. ✓ If problem persists, replace the GRS 77.
Attitude appears unstable	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Ensure the four GRS 77 mounting screws are tight. Finger tight is not sufficient, a screwdriver must be used to verify. • Ensure mounting rack and airframe shelf are secure and all hardware and brackets are present (CAUTION - do not loosen the mounting rack hardware to the airframe shelf or the aircraft will need to be re-leveled and the PITCH/ROLL OFFSET procedure performed). • Ensure GRS 77 connector is securely fastened and proper strain relief is provided. • Remove GRS 77 connector and verify there are no bent pins. • Replace the GRS 77. • Contact Garmin for further troubleshooting if required.

CHART 17 (Sheet 1 of 2)
MESSAGE ADVISORIES - GRS / GMU

Failure Message	Cause	Solution
MANIFEST – GRS1 software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the specified GRS 77.	• Load correct software version. See Software Configuration Procedures.
MANIFEST – GRS2 software mismatch. Communication halted.		
AHRS1 SRVC – AHRS1 magnetic-field model needs update.	The AHRS magnetic field model should be updated for the specified unit. Appears on ground only.	• Load updated AHRS magnetic field file. See GRS77 / Components for AHRS magnetic field update information.
AHRS2 SRVC – AHRS2 magnetic-field model needs update.		
GEO LIMITS – AHRS1 too far north/south, no magnetic compass.	No magnetic compass information available due to being too far north or south.	• Operate the aircraft only within the limits as specified in the POH.
GEO LIMITS – AHRS2 too far north/south, no magnetic compass.		
AHRS1 TAS – AHRS1 not receiving airspeed.	The specified GRS 77 is not receiving airspeed from the GDC 74A.	• Check GRS/GDC interconnect for faults. • Replace the GDC 74: • If problem persists, replace the GRS 77.
AHRS2 TAS – AHRS2 not receiving airspeed.		
AHRS1 GPS – AHRS1 not receiving backup GPS information.	The GRS 77 #1 is not receiving backup GPS information from either GIA 63.	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Check GPS status for GIA 1 and 2 on MFD - AUX GPS STATUS page. If one or both GPS receivers cannot acquire a position lock, see GPS troubleshooting section. • Troubleshoot GIA1/2 –GRS1 wiring for faults (ref. Failed Path troubleshooting section). • Replace the GRS 77 #1.
AHRS1 GPS – AHRS1 operating exclusively in no-GPS mode.	The GRS 77 #1 is operating in the absence of GPS.	
AHRS1 GPS – AHRS1 not receiving any GPS information.	The GRS 77 #1 is not receiving GPS data from the GPS receivers.	
AHRS1 GPS – AHRS1 using backup GPS source.	The GRS 77 #1 is using the backup GPS data path.	

CHART 17 (Sheet 2 of 2)
MESSAGE ADVISORIES - GRS / GMU

GRS Message Advisories (Cont.)		
Failure Message	Cause	Solution
AHRS2 GPS – AHRS2 not receiving backup GPS information.	The GRS 77 #2 is not receiving backup GPS information from either GIA 63.	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Check GPS status for GIA 1 and 2 on MFD - AUX GPS STATUS page. If one or both GPS receivers cannot acquire a position lock, see GPS troubleshooting section. • Troubleshoot GIA1/2 –GRS2 wiring for faults (ref. Failed Path troubleshooting section). • Replace the GRS 77 #2.
AHRS2 GPS – AHRS2 operating exclusively in no-GPS mode.	The GRS 77 #2 is operating in the absence of GPS.	
AHRS2 GPS – AHRS2 not receiving any GPS information.	The GRS 77 #2 is not receiving GPS data from the GPS receivers.	
AHRS2 GPS – AHRS2 using backup GPS source.	The GRS 77 #2 is using the backup GPS data path.	
AHRS MAG DB – AHRS magnetic model database version mismatch.	The G1000 has detected a magnetic model database version mismatch.	• Load AHRS magnetic model field in both GRS units.
MAG VAR WARN – Large magnetic variance. Verify all course angles.	Magnetic variance value from GMU 44 is not accurate.	• If flying near large radio towers or other sources of possible electromagnetic interference, the condition should correct itself as the aircraft leaves the area. • If problem persists, run magnetic interference check in LRU Test Procedures (below) to check for magnetic interference in the aircraft.
MANIFEST – GMU1 software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the specified GMU 44.	Load the correct software version. See G1000 Software/Configuration Procedure, for the Software Load Procedure.
MANIFEST – GMU2 software mismatch. Communication halted.		
HDG FAULT – AHRS1 magnetometer fault has occurred.	A fault has occurred in the specified magnetometer; heading will be flagged invalid.	• Check GMU 44/GRS 77 interconnect for faults. ✓ Replace GMU 44. ✓ If problem persists, replace GRS 77.
HDG FAULT – AHRS2 magnetometer fault has occurred.		

(11) GMU 44 Magnetometer

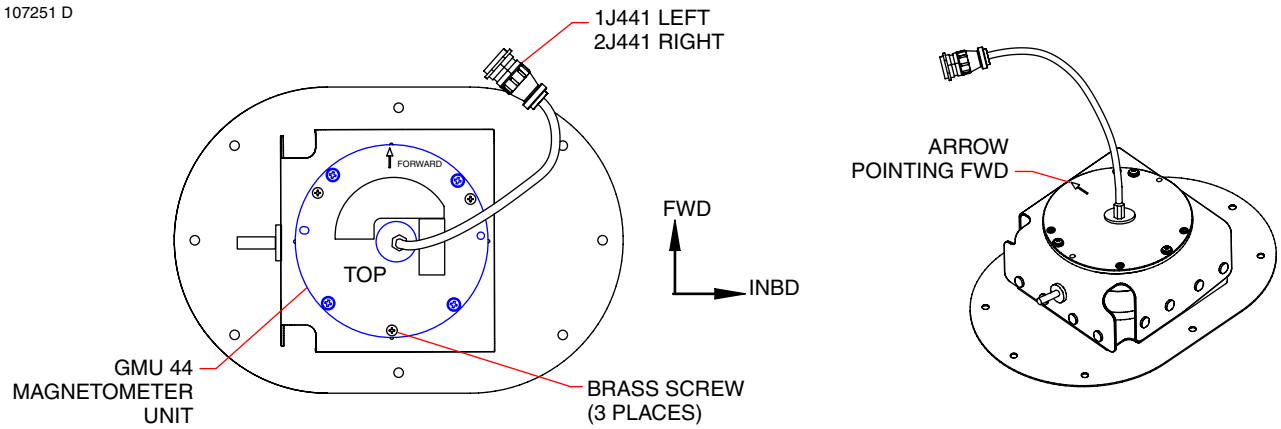
See “Figure 20”.

(a) Description

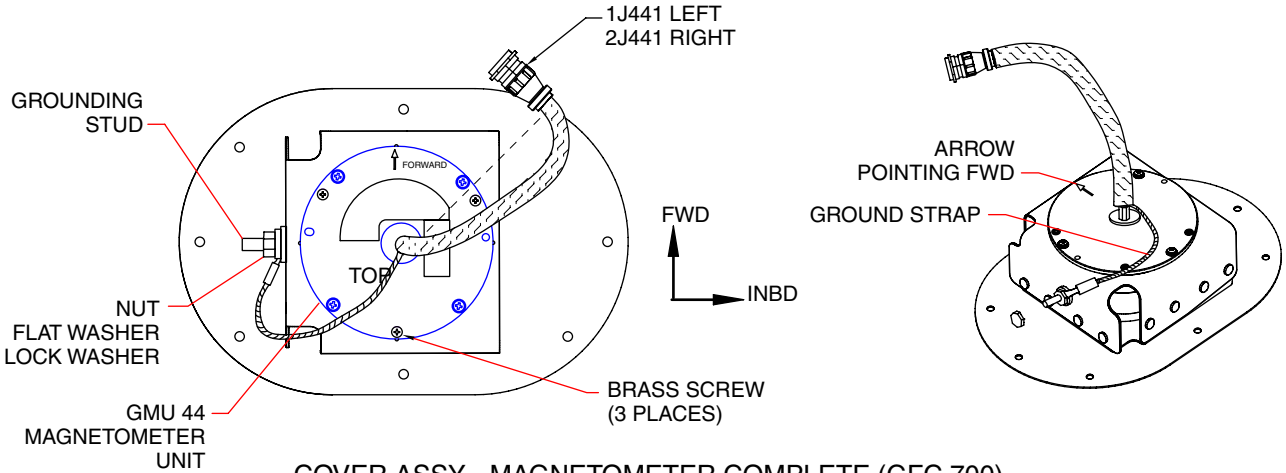
The GMU 44 provides horizontal and vertical magnetic field information to the GRS 77 AHRS. This allows heading to be calculated and provides assistance during AHRS alignment. The GMU 44 is mounted on an access cover under each wing between W.S. 183.98 and W.S. 208.33.

(b) Troubleshooting

See “Chart 16” on page 34250172 and ”Chart 17” on page 34250173.



COVER ASSY - MAGNETOMETER COMPLETE
(LEFT SHOWN, RIGHT OPPOSITE)



COVER ASSY - MAGNETOMETER COMPLETE (GFC 700)
(LEFT SHOWN, RIGHT OPPOSITE)

GMU 44 Magnetometer Installation
Figure 20

(c) Removal

1) Remove the access cover to expose the magnetometer.
2) Disconnect the GMU 44 harness from the wing harness.
3) In airplanes equipped with GFC 700 only, disconnect the grounding strap by removing the nut and washers.
4) Unscrew and remove the three brass screws securing the magnetometer to its mounting rack.
5) Carefully lift the GMU 44 from the rack.

(d) Installation

CAUTION: THE MAGNETOMETERS ARE SECURED TO THE MOUNTING BRACKET WITH BRASS SCREWS. ENSURE ONLY BRASS SCREWS ARE USED WHEN REINSTALLING.

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
2) Lower the GMU 44 into the rack and secure the plate with the three brass Phillips screws. Torque 12–15 inch pounds.
3) In airplanes equipped with GFC 700 only, connect the grounding strap and secure with washers and nut.
4) Connect GMU 44 harness to the wing harness.
5) Reinstall the access cover.
   NOTE: Ensure correct hardware is used when reinstalling access covers attached to magnetometers.

(e) Return to Service

1) Original GMU 44 is Reinstalled

   If the original GMU 44 was reinstalled, then no software loading is required. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) Original GMU 44 Installed in Opposite Locations for Troubleshooting

   If the original GMU1 and GMU2 are installed in opposite locations (GMU1 and GMU2 in opposite unit racks), no software loading is required. However, performing the “GRS 77/GMU 44 Magnetometer Calibration (Procedure B)” on page 342501137 is required.

3) New, Repaired, or Exchanged GMU 44 is Installed

   If a new, repaired, or exchanged GMU 44 is installed, then software must be loaded. See “G1000 Software/Configuration Procedure” on page 342501107. Then continue to the “GRS/GMU Calibration Procedures” on page 342501134.

4) Test

   Perform “GRS/GMU Test” on page 342501142.

(12) GDL 69A Data Link Unit

See “Figure 21”.

(a) Description

The GDL 69A provides XM Radio weather and music entertainment through means of a dedicated satellite data link. The GDL 69A is mounted in a mount rack on the upper left side of the fuselage aft of the cabin hatshelf and just forward of F.S. 220.164. Power to the GDL 69A is received from the Avionics Bus #1. The GDL 69A sends weather data through the HSDB bus to the MFD and PFD2, where the data link interface is controlled. Audio is sent directly to the GMA 1347 Audio Panel.

(b) Troubleshooting

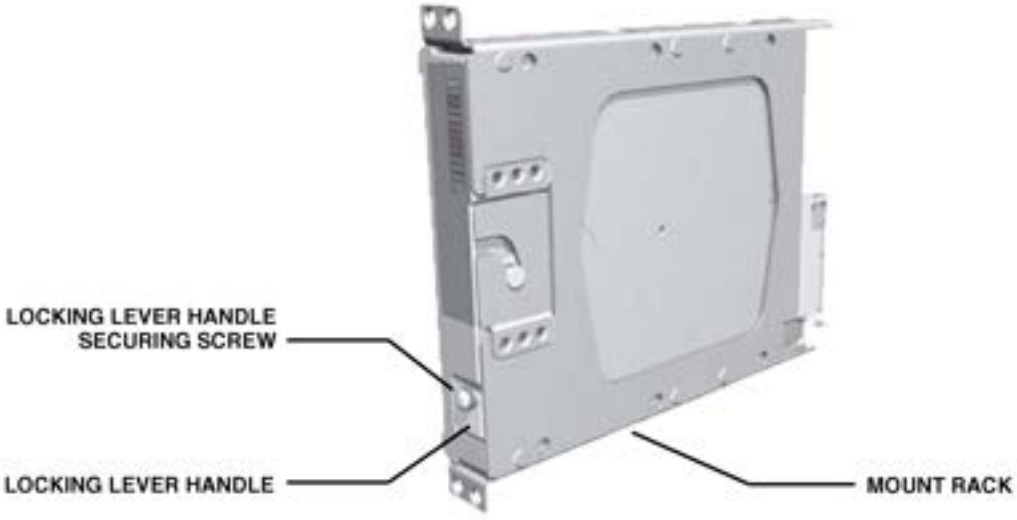
See “Chart 18” on page 34250179 and “Chart 19” on page 34250180.

(c) Removal

1) Remove the aft cabin hatshelf assembly.
2) Unlock the GDL 69A handle by unscrewing the Phillips screw.
3) Pull the handle upward to unlock the GDL 69A.
4) Gently remove the GDL 69A from its rack.

(d) Installation

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
2) Gently insert the GDL 69A into the rack. The handle should engage the dogleg track.
3) Press down on the handle to lock the unit into place.
4) Lock the handle to the GDL 69A body using the Phillips screw.
5) Reinstall the aft cabin hatshelf assembly.



GDL 69A Data Link Unit
Figure 21

(e) Return to Service

1) Original GDL 69A is Reinstalled

No software or configuration loading is required if the removed GDL 69A is reinstalled. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) New, Repaired, or Exchanged GDL 69A is Installed

If a new, repaired, or exchanged GDL 69A is installed, the correct software and configuration files must be loaded to the unit, see “G1000 Software/Configuration Procedure” on page 342501107. Then, the XM Satellite Radio subscription must be reactivated, see “Chart 18”.

3) Test

Perform the “GDL 69A Test (If installed)” on page 342501145.

CHART 18
TROUBLESHOOTING - GDL 69A

Symptom	Recommended Action
No XM audio is heard	• Ensure the following items are not preventing the audio panel from distributing XM audio (reference applicable G1000 Pilot's Guide): ✓ Verify the XM volume is not muted on the AUX – XM RADIO page on the MFD. ✓ Verify the COM squelch is not open. ✓ Verify the ICS squelch is not open. ✓ Verify the marker beacon tones are not being received. ✓ Verify the headphone (if equipped) volume is turned up. • Go to the AUX – SYSTEM STATUS page on the MFD and ensure unit is online. ✓ If a red X is present, verify the unit is receiving power at the rack connector. • Ensure there are no GDL alerts in the alert window. If there is an Alert for software or configuration error or mismatch, reload the file noted in the Alert. • Restart the PFD's and MFD in configuration mode and go to the GDL page.
No XM weather information is displayed	✓ Verify unit is active. ✓ Verify the Signal number is "2" or "3". If it is "0" or "1" check the GDL 69/69A antenna and cabling for faults. ✓Reseat the GDL 69/69A to verify the coax connector is fully seated. ✓ If unit is not active, contact XM Customer service at 1-800-985-9200 to have a refresh signal sent to your unit. You will need to provide them the Audio Radio ID (and Data Radio ID number for XM weather) numbers. Also verify with XM that the correct Weather package (Aviator Lite or Aviator) is on the account, and that no traffic service has been activated against that Radio ID. The unit must be on for approximately one hour after the request for the refresh has been sent to receive the signal. ✓Alternatively, you may also go to XM's website at http://www.siriusxm.com and enter the radio ID's to have a refresh signal sent. ✓ If there is still problems receiving weather products after performing the above step, call XM and have the account deactivated, and a new account activated to clear out any corrupt account information. ✓ Verify there is a good ground connection through the aircraft between the MFD and the GDL69/69A unit. Reference the Aircraft Maintenance Manual for instructions on how to check bonding and ground points. ✓ If problem persists, replace the GDL 69/69A.

CHART 19
MESSAGE ADVISORIES - GDL 69A

Failure Message	Cause	Solution
GDL 69 FAIL – GDL 69 has failed.	The G1000 has detected a failure in the GDL 69A.	• Replace GDL 69A. • Check GDL 69A antenna and cabling. • Check the GDL 69A and MFD interconnect.
GDL69 CONFIG – GDL 69 configuration error. Config service req'd.	The G1000 has detected a GDL 69A configuration mismatch.	• Reload configuration file. See Software Load / Configuration Procedure. • If problem persists, replace master configuration module, check config module harness for faults and replace if necessary. NOTE New Terrain/Obstacle cards, Navigation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 System ID number will change to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number
MANIFEST – GDL software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the GDL 69A.	• Load correct software version. See Software Load / Configuration Procedure.

(13) GRT 10 Wireless Transceiver Unit

See “Figure 22”.

(a) Description

The optional GRT 10/GRC 10 wireless remote system is for use by passengers in the aircraft to control the audio functions (audio volume and channel changes) of the Garmin GDL 69A Datalink Receiver. The system consists of two components:

1) The GRT 10 Wireless Transceiver mounted in a mount rack on the upper left side of the fuselage aft of the cabin hatshelf (just forward of F.S. 220.164) and connected to the GDL 69A serial port, and

2) The GRC 10 Wireless Remote with an LCD display.

(b) Removal

1) Remove the aft cabin hatshelf assembly.

2) Disconnect backshell assembly from unit.

3) Use a 3/16” hex drive tool to remove each of the four mounting screws.

(c) Installation

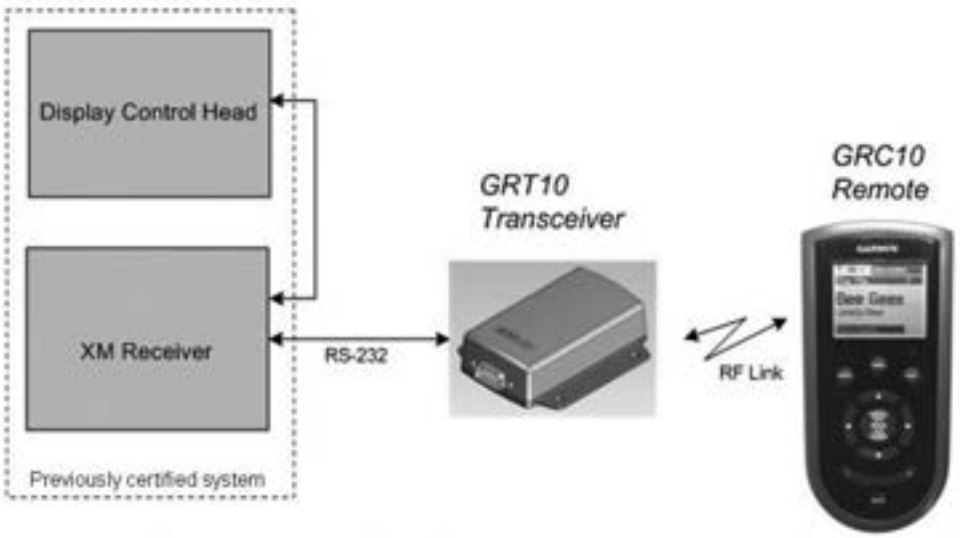
1) Inspect connector for damaged pins.

2) Hold unit flush with the mounting plate.

3) Use a 3/16” hex drive tool to tighten each of the four mounting screws.

4) Connect backshell assembly to unit.

5) Reinstall the aft cabin hatshelf assembly.



GRT-10
Figure 22

(14) GSR 56 Iridium Transceiver

See “Figure 23”.

(a) Description

The optional GSR 56 Iridium Transceiver provides airborne Iridium Satellite Telephone and SMS Messaging service. Iridium telephone and text meassaging are available to the flight crew through the MFD, audio panel, and headset. The Garmin GSR 56 is connected to the #2 GIA 63W and GMA 1347. Power is received from the Avionics Bus #1.



GSR 56 Iridium
Figure 23

(b) Troubleshooting

See “Chart 20”.

(c) Removal

1) Remove the aft cabin hatshelf assembly to gain access to the aft avionics shelf.
2) Loosen the GSR 56 unit by turning the ratchet mechanism located at the front of the unit counter-clockwise until it drops free of the locking pawl on the unit.
3) Gently remove the GSR 56 from its rack.

(d) Installation

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
2) Gently insert the GSR 56 into the rack and engage the connectors.
3) Lift the ratchet mechanism to allow the collar to engage the locking pawl on the unit.
4) Gently turn ratchet mechanism clockwise to secure the GSR 56 unit into place.
5) Reinstall the aft cabin hatshelf assembly.

CHART 20
MESSAGE ADVISORIES - GSR 56

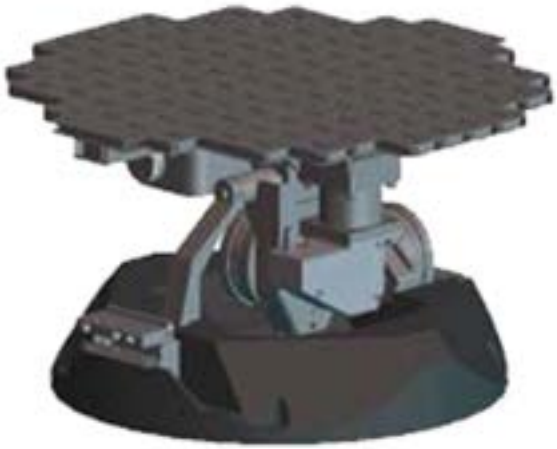
Failure Message	Cause	Solution
GSR1 FAIL – GSR1 has failed.	The system has detected a failure in the #1 GSR 56.	• Ensure GSR1 is receiving power. • Replace GSR.

(15) GWX 68 Weather Radar

See “Figure 24”.

(a) Description

The GWX 68 Airborne Weather Radar provides weather radar data output to the GDU 1040 MFD. The GWX 68 is mounted in the nose cone assembly, forward of the nose gear doors. Power to the GWX 68 is received from the Avionics Bus #1. Data received from the GWX 68 is routed to the MFD via high-speed data bus (Ethernet).



GWX 68 Weather Radar
Figure 24

(b) Troubleshooting

See “Chart 21” on page 34250185.

(c) Removal

NOTE: Take note of the number, type and location of any shims and washers during removal.

1) Disconnect backshell assembly (P400) from top of unit.
2) Use a 3/16” hex drive tool to remove each of the two mounting screws, washers and nuts connecting the unit to the upper beam assembly.
3) Use a #3 Phillips bit to remove two mounting screws which connect the upper beam to each side clip assembly. Remove the upper beam assembly.
4) Use a 3/16” hex drive tool to remove each of the two mounting screws connecting the unit to the lower bracket assembly.
5) Remove GWX 68 by rotating unit back over the bracket assembly.

(d) Installation

NOTE: Install shims and washers of same number and type in the same locations noted during removal.

1) Inspect connector for damaged pins.
2) Hold unit flush with the radar mount, ensuring that the connector is at the top.
3) Reattach and tighten each of the two mounting hex head screws to the lower bracket assembly.
4) Reattach and tighten each of the two mounting phillips head screws which connect the upper beam to each side clip assembly. Torque to 60 in·lbs.
5) Reattach and tighten each of the two mounting hex head screws, washers and nuts connecting the unit to the upper beam assembly. Torque to 60 in·lbs.
6) Connect backshell assembly (P400) to top of unit.

(e) Return to Service

1) Original GWX 68 Reinstalled

No software or configuration loading is required if the removed GWX 68 is reinstalled. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) New, Repaired, or Exchanged GWX 68 Installed

If a new ,repaired, or exchanged GWX 68 is installed, the correct software and configuration files must be loaded to the unit. See “G1000 Software/Configuration Procedure” on page 342501107.

3) Test

Perform “GWX 68 Weather Radar Test (If installed)” on page 342501143.

CHART 21
TROUBLESHOOTING / MESSAGE ADVISORIES - GWX 68

WARNING:

Before energizing the equipment, be sure microwave radiation safety precautions including both fuel and personnel safety considerations have been observed. These include clearing all personnel to an area beyond the maximum permissible exposure level (MPEL) boundary. The MPEL for the GWX 68 is 11 feet.

Failure Message	Cause	Solution
GWX CONFIG – GWX configuration error. Configuration service required.	The G1000 has detected a GWX 68 configuration mismatch.	• Load GWX configuration files, ✓ Replace GWX. ✓ If problem persists, replace master configuration module, check config module wiring for faults and replace if necessary. NOTE New Terrain/Obstacle cards, Navigation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 System ID number will change to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number
GWX FAIL – GWX is inoperative.	The G1000 has detected a failure in the GWX 68.	• Check Ethernet connection between the GWX and GDL69A for faults. • Replace the GWX 68
GWX SERVICE – Needs service. Return unit for repair.	The G1000 has detected a failure in the GWX 68.	• Replace the GWX 68.
MANIFEST – GWX software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the GWX 68.	• Load correct software version. See Software Configuration Procedures.
WX ALERT – Possible severe weather ahead.	The system has detected severe weather within ± 10 degrees of the aircraft heading at a range of 80 to 320 nm.	

(16) GTS 825 Traffic Advisory System

See “Figure 25”.

(a) Description

The optional GTS 825 Traffic Advisory System provides real-time traffic information to the PFD (and, indirectly, to the MFD). The Garmin GTS 825 connects with the #1 GDU 1040 and communicates with the MFD through an HSDB connection. Power is received from the Avionics Bus #1.

(b) Troubleshooting

See “Chart 22” on page 34250187.

(c) Removal

1) Remove the aft cabin hatshelf assembly to gain access to the aft avionics shelf.



GTS 825 TAS
Figure 25

2) Disconnect the GTS 825 connectors.

3) Loosen the GTS 825 unit by unscrewing the knurled retainers.

4) Move the retainer arms downward. Gently remove the GTS 825 from its rack.

(d) Installation

1) Insert the GTS 825 into the rack and lift each retainer arm to engage lip.

2) Gently screw retainers to secure the GTS 825 unit into place.

3) Visually inspect the connectors and pins for damage. Repair any damage. Connect the connectors to the unit.

4) Reinstall the aft cabin hatshelf assembly.

(e) Return to Service

1) Original GTS 8XX Reinstalled

No software or configuration loading is required if the original GTS unit is removed and reinstalled. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) New, Repaired, or Exchanged GTS 8XX Installed

If a new ,repaired, or exchanged GTS 8XX is installed, the correct software and configuration files must be loaded to the unit. See “G1000 Software/Configuration Procedure” on page 342501107.

3) Test

Perform “GTS 825 TAS Check (If installed)” on page 342501146.

CHART 22
MESSAGE ADVISORIES - GTS 825

Failure Message	Cause	Solution
MANIFEST – GTS software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the specified GTS 825.	Load the correct software version. See G1000 Software/Configuration Procedure, for the Software Load Procedure.
GTS CONFIG – GTS configuration error. Config service req'd.	The system has detected a configuration mismatch for the specified GTS 825.	

(17) Configuration Module

See “Figure 26”.

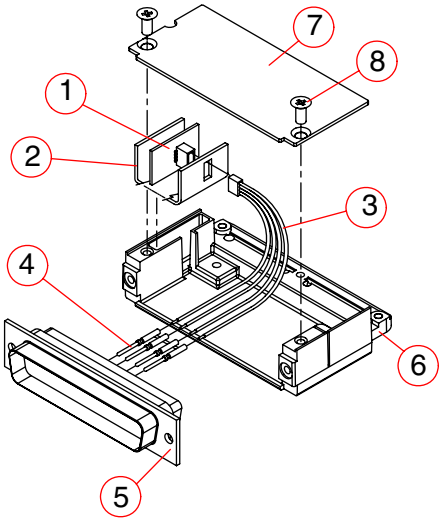
The configuration module is located in the backshell of the connector it is terminated to.

(a) Removal

1) Disconnect connector from LRU.
2) Remove 2 screws (8) from cover (7) and remove cover.
3) Unplug connector from configuration module (1).
4) Remove configuration module.

(b) Installation

1) Inspect connector for damaged pins (4).
2) Place configuration module (1) in position.
3) Insert connector into configuration module (1).
4) Assembly of the connector is the reverse of disassembly.



Configuration Module
Figure 26

(c) Return to Service

1) If a GRS 77 AHRS Configuration Module is replaced:

   All three GRS 77/GMU 44 calibration procedures must be performed. Proceed to “Post-Installation Procedures” on page 342501119.

2) If a GDC 74A Configuration Module is replaced:

   Configuration settings must be reloaded to the GDC 74A. Proceed to “Post-Installation Procedures” on page 342501119.

3) If the GEA 71 Configuration Module is replaced:

   Perform the “GEA 71 Test” on page 342501130.

4) If the Master Configuration Module is replaced:

   a) Start the G1000 system in configuration mode.
   b) Go to the Configuration Upload Page on the PFD.
   c) Press the UPDT CFG softkey.

5) If both the PFD and Master Configuration Module is replaced:

   The G1000 system (except GRS 77/GMU 44 and GDC 74A) must be reconfigured. Proceed to “Post-Installation Procedures” on page 342501119.

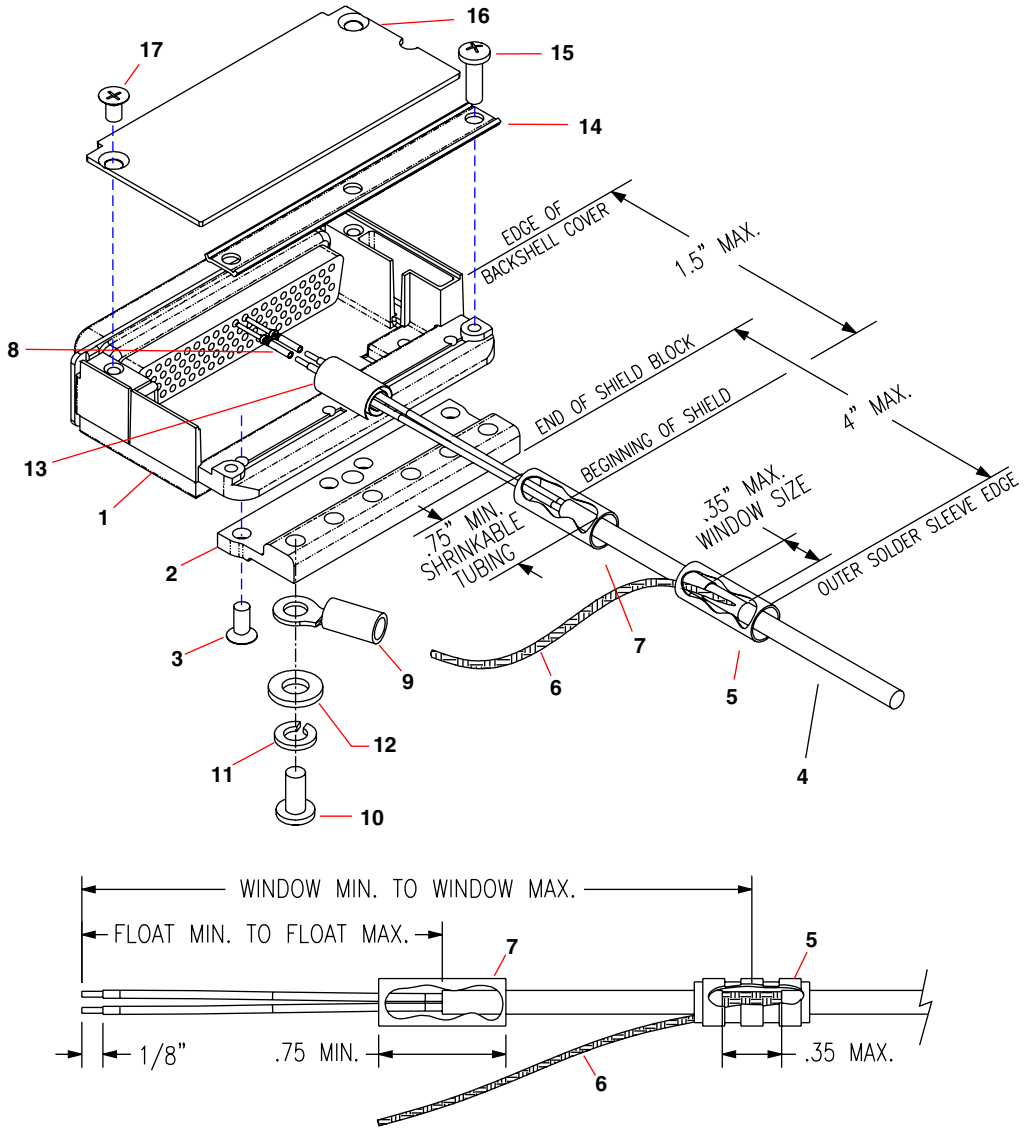
(18) Shield Block Installation

See “Figure 27” on page 34250189.

Most G1000 connectors employ a shield block grounding system to provide necessary ground reference to shielding and/or transducers.

(19) Backshell Connectors

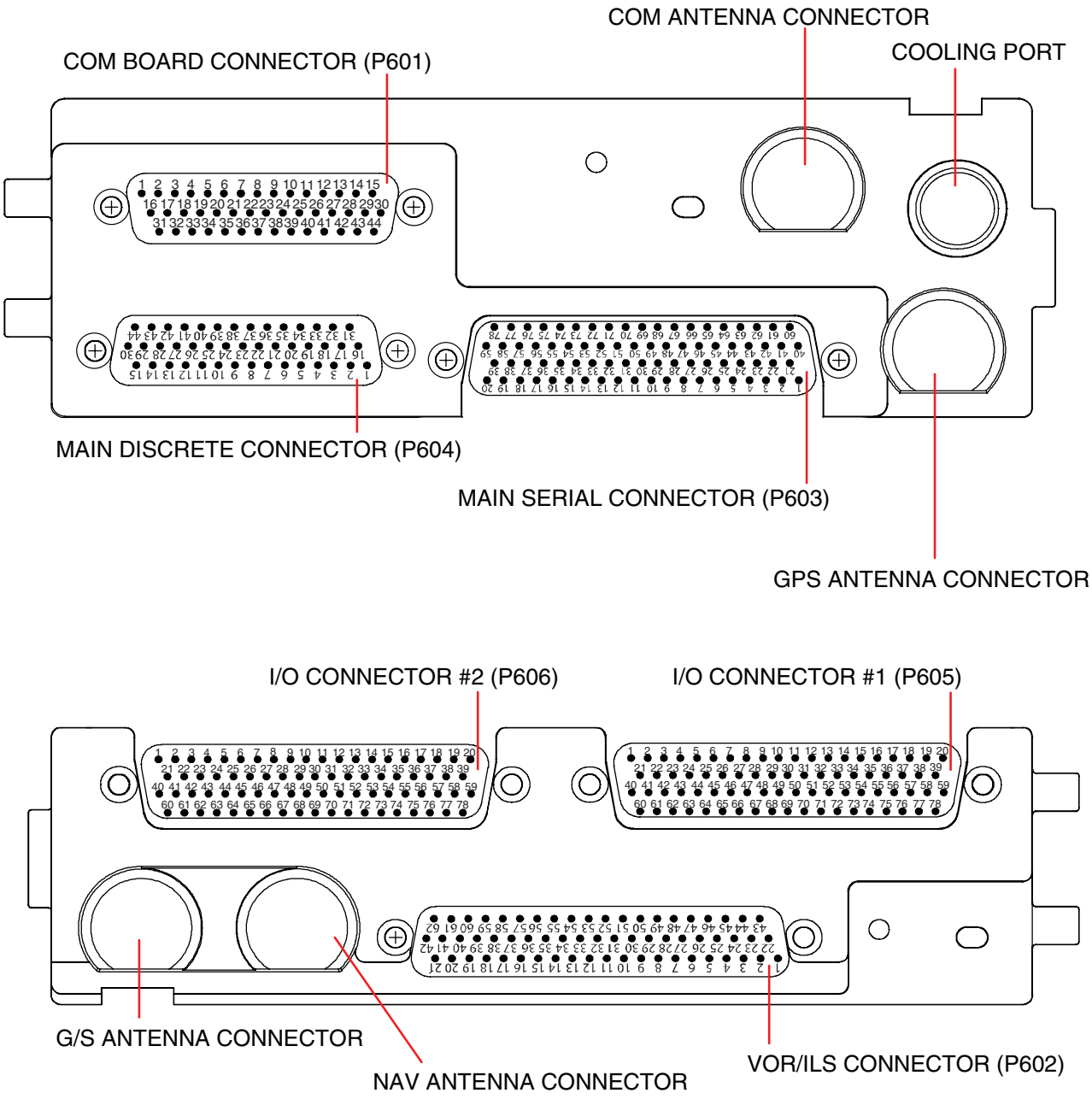
See “Figure 28” on page 34250190.



1. CAST HOUSING
2. SHIELD BLOCK(S)
3. SCREW
4. MULTIPLE CONDUCTOR SHIELDED CABLE (2–CONDUCTOR DEMONSTRATED HERE)
5. DRAIN WIRE SHIELD TERMINATION (METHOD OPTIONAL)
6. BRAID, FLAT
7. FLOATING SHIELD TERMINATION (METHOD OPTIONAL)
8. PINS
9. RING TERMINAL, #8, INSULATED
10. SCREW
11. SPLIT WASHER
12. FLAT WASHER
13. SILICON FUSION TAPE
14. STRAIN RELIEF
15. SCREW
16. LID
17. SCREW

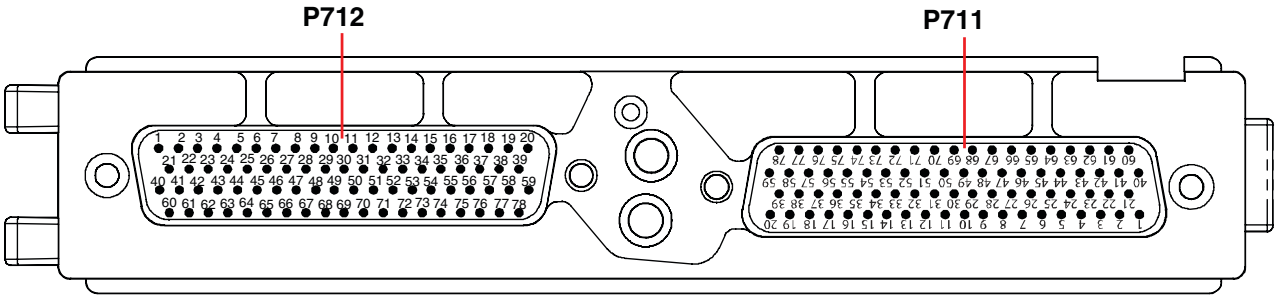
Shield Block Installation to Backshell
Figure 27

Backshell Connectors

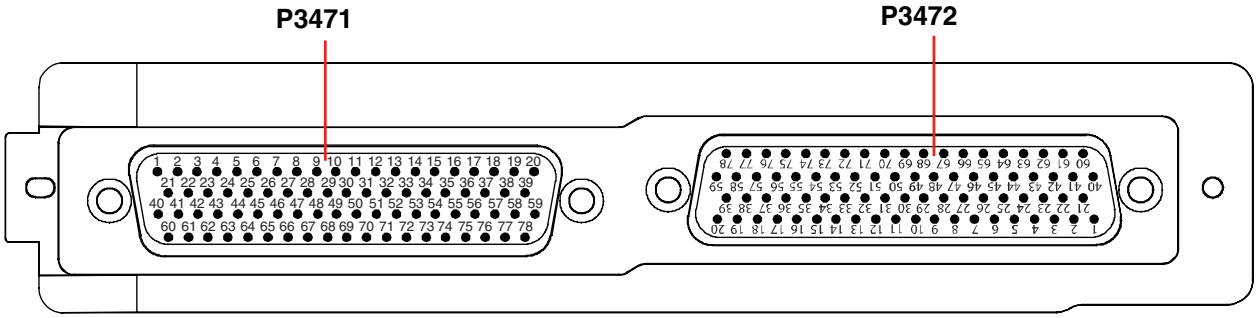


Backshell connectors as viewed with the LRU removed.

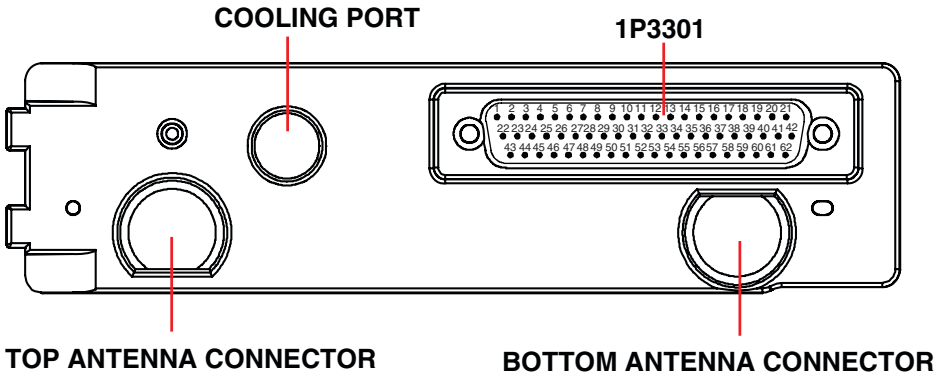
Backshell Connectors
Figure 28 (Sheet 1 of 4)



GEA 71 Backshell Connectors



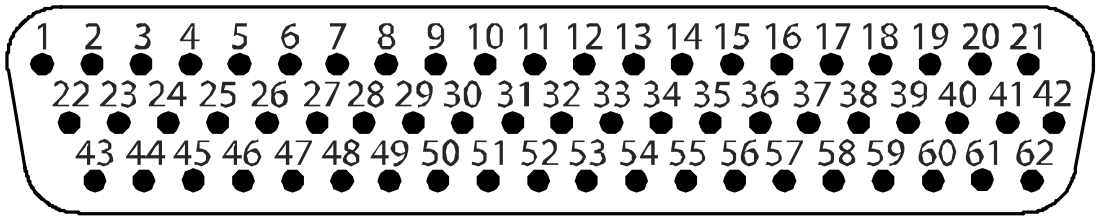
GMA 1347 Backshell Connectors



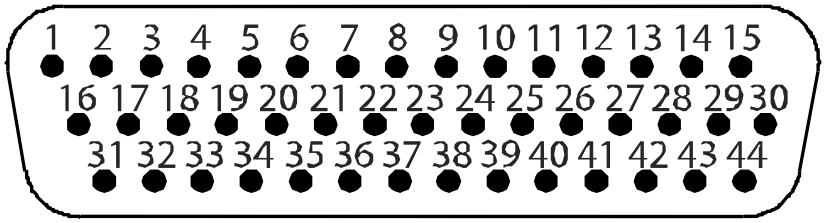
GTX 33 Backshell Connectors

Backshell connectors as viewed with the LRU removed.

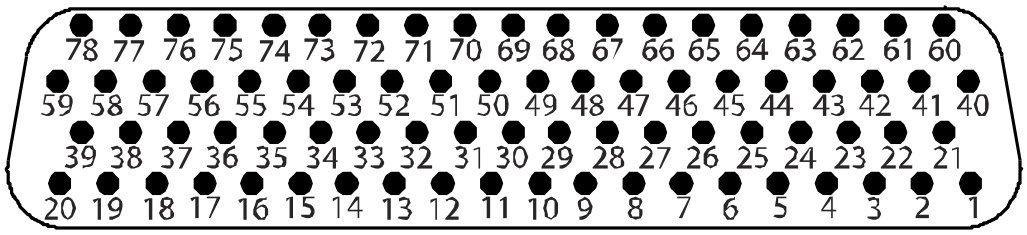
Backshell Connectors
Figure 28 (Sheet 2 of 4)



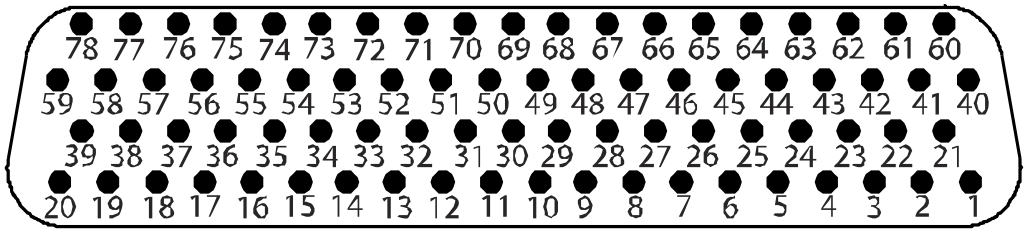
GDU 1040/1500 Backshell Connector (1P10001, 2P10001, and 3P10001)



GRS 77 Backshell Connector (1P771, 2P771)



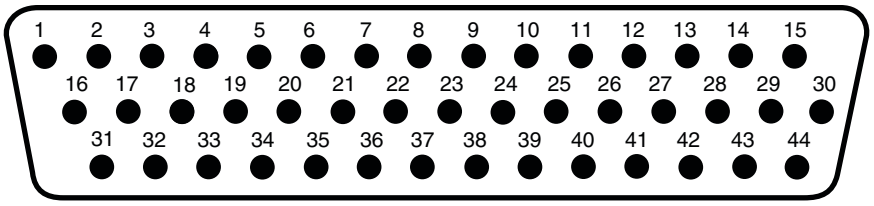
GDC 74A Backshell Connector (1P741, 2P741)



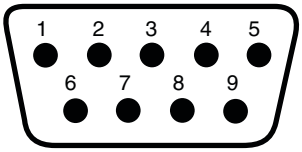
GDL 69A Backshell Connector (P691)

Backshell connectors as viewed with the LRU removed.

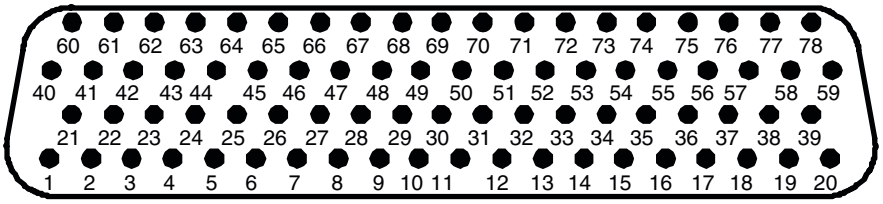
Backshell Connectors
Figure 28 (Sheet 3 of 4)



GWX 68 Backshell Connector (P400)



GRT 10 Backshell Connector (P771)



GTS 825 Backplate Connector (P8001, P8002)

Backshell connectors as viewed with the LRU removed.

Backshell Connectors
Figure 28 (Sheet 4 of 4)

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CHART 23
TROUBLESHOOTING / MESSAGE ADVISORIES - G1000 SOFTWARE CONFIGURATION

Problem	Solutions
MFD or PFD displays do not power up:	• Ensure power is present at display backshell connector. • Troubleshoot per the “Blank Display” GDU section.
Software file load fails: 	• Ensure that LRU is reporting data on System Status page (LRU is ‘ONLINE’). Check data path wiring as needed. • Retry software file load or try using a different card. Ensure that the MFD is not touched during the loading process. • Ensure that LRU part number is compatible with software version and Card Loader. • Replace LRU.
Configuration file load fails: 	• Ensure that LRU is reporting data on System Status page (LRU is ‘ONLINE’). Check data path wiring as needed. • Retry configuration file load or try using a different card. Ensure that the MFD is not touched during the loading process. • Ensure that LRU part number is compatible with Card Loader. • Replace LRU.
GIA1 and/or GIA2 to ‘LRU’ data path not working	• Ensure GIA1 and GIA2 are configured correctly. • Check wiring, connectors & pins as needed.
Software File Mismatch Alert appears in lower right corner of PFD when started in normal mode: 	• Ensure that proper software file part number and version were loaded to LRU. To attain the current sofware P/N and version for your aircraft contact: Piper Aircraft Inc. Customer Service. • Check and ensure that correct Card Loader was used during load process. Refer to the General Arrangement Drawing. • Reload software to LRU.

CHART 24
G1000 SYSTEM COMMUNICATION HIERARCHY

The following criteria must be satisfied to be able to perform the desired operation:	
Desired Operation	Criteria for Success
Load Software to MFD or PFD Displays	• G1000 Loader Card must be inserted in top slot for each display to be loaded. • CLR & ENT keys must be held during power up of display. • Power on only one display at a time during software loading.
Load AIRFRAME, SYSTEM, MFD, PFD1, PFD2 and MANIFEST configuration files to MFD and PFDs	• G1000 Loader Card must be inserted in top slot of PFD1. • PFD1 and MFD must be powered on. • PFD1 and MFD must have correct software.
Load Software/Configuration files to GIA 63Ws	• G1000 Loader Card must be inserted in top slot of PFD1. • G1000 system must be powered on. • PFD and MFD must have correct software. • PFD 1 and MFD must be successfully configured with AIRFRAME, SYSTEM, MANIFEST, MFD, PFD1 and PFD2 configuration files.
Load Software/Configuration files to: - GMA 1347 - GDC 74B - GEA 71 - GRS 77 (software only) - GMU 44 (software only) - GTX 33	• G1000 Loader Card must be inserted into PFD1 top slot. • G1000 must be powered on. • PFD1 and MFD must have correct software and configuration settings. • GIA 63Ws must have correct software. • GIA 63Ws must be successfully configured with GIA1 and GIA2 configuration files. • Data path from GIA1 to each LRU must be operational.

I. Configuration Mode Overview

CAUTION: THE CONFIGURATION MODE CONTAINS CERTAIN PAGES AND SETTINGS THAT ARE CRITICAL TO AIRCRAFT OPERATION AND SAFETY. THESE PAGES ARE PROTECTED AND CANNOT BE MODIFIED, UNLESS THE TECHNICIAN IS PROPERLY AUTHORIZED AND EQUIPPED. HOWEVER, MOST PROTECTED PAGES ARE VIEWABLE TO ALLOW SYSTEM AWARENESS FOR TROUBLESHOOTING.

The Configuration Mode exists to provide the avionics technician with a means of configuring, checking, and calibrating various G1000 sub-systems. Troubleshooting and diagnostics information can also be viewed in this mode.

(1) Start the system in Configuration Mode:

(a) Press and hold the ENT key on the co-pilot's PFD while applying power using the PFD2 circuit breaker.

(b) Release the ENT key after 'INITIALIZING SYSTEM' appears in the upper left corner of the PFD.

(c) Press the ENT key or the YES softkey to update system files if prompted.

(d) Repeat steps (a) thru (c) on the MFD using the MFD circuit breaker to apply power and using the far right soft key on the MFD.

(e) Repeat steps (a) thru (c) on the pilot's PFD using the PFD1 circuit breaker to apply power.

NOTE: For a complete description and breakdown of each Configuration Mode page, refer to the G1000 Line Maintenance & Configuration Manual, Garmin P/N 190-00303-04.

(2) SET>ACTV Configuration

Throughout the configuration mode pages, there are SET and ACTIVE columns for input/output settings and other parameters.

SET: Refers to a setting or group of settings that reside in PFD Internal Memory and/or the Master Configuration Module.

ACTIVE: Refers to an 'active' setting or parameter currently being used by the LRU. LRUs store the 'active' settings within internal memory.

Data can be manually copied from one column to the other (and consequently from PFD memory to the LRU memory and vice-versa) by using the following two softkeys, when available:

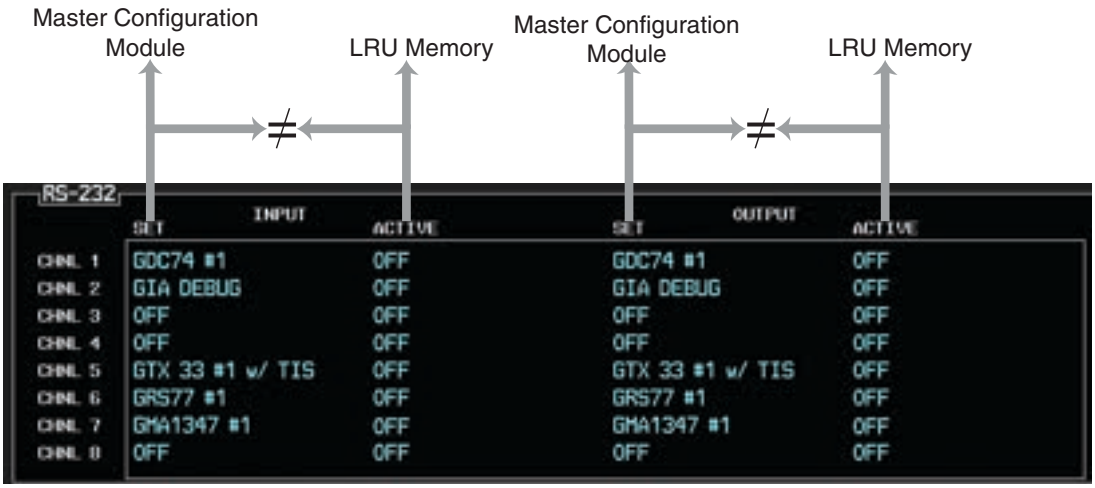
SET>ACTV (read 'Set to Active') softkey: Allows the installer to send the information in the SET column (data stored in the master config module) to the ACTV column (data used by LRU).

ACTV>SET (read 'Active to Set') softkey: Causes the LRUs current settings to be copied to the master configuration module as SET items.

CAUTION: THE ACTV>SET SOFTKEY MUST BE USED WITH CAUTION! IF AN IMPROPERLY CONFIGURED UNIT IS INSTALLED, THIS SOFTKEY CAUSES THE WRONG CONFIGURATION TO REPLACE THE CORRECT AIRCRAFT CONFIGURATION.

In the first example shown in "Figure 29" on page 34250198, the SET columns do not match the ACTIVE columns. The inequality between SET and ACTIVE indicates a configuration mismatch. By pressing the SET>ACTV softkey, this copies the SET column to the LRU unit's configuration memory. The settings then become the ACTIVE settings for the LRU being configured.

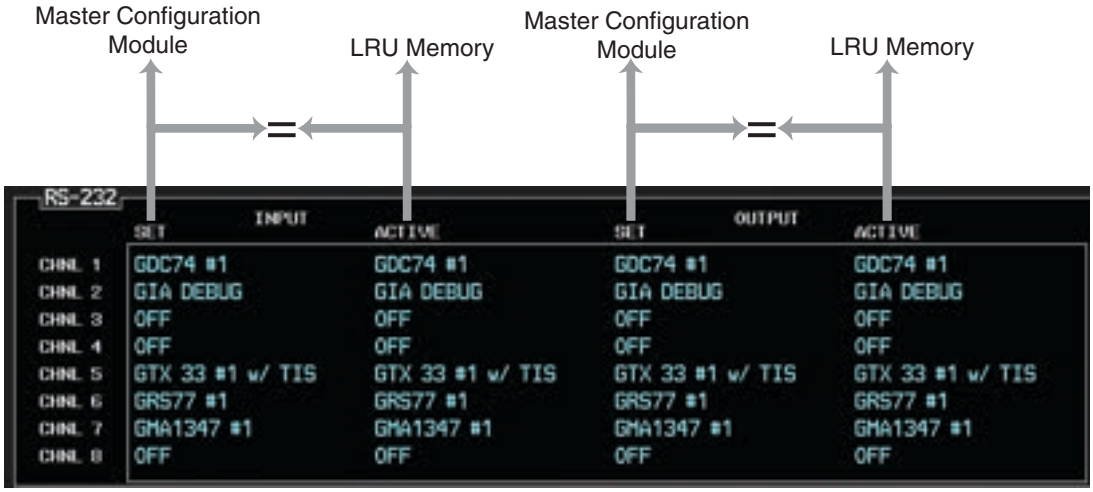
Configuration Mismatch



SET>ACTV Softkey



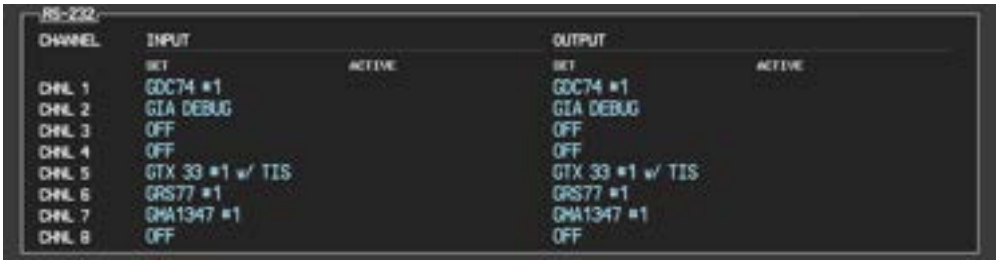
Configuration Correct



SET>ACTV Configuration
Figure 29

When troubleshooting the system, technicians can look for inequalities between SET and ACTIVE columns. Certain problems can be resolved simply by pressing the SET>ACTV softkey, which reloads settings to the specific LRU from the PFD. (Note that this can also be accomplished by reloading the configuration files for the LRU, using the software loader card.).

A blank active column, as shown in “Figure 30”, represents loss of communication between the display and the particular unit. See troubleshooting for more details.



Blank Active Columns
Figure 30

(3) Configuration Prompts

When configuration settings are changed, the technician receives on-screen prompts and/or confirmations such as those shown in “Figure 31”. Other prompts may be encountered during the configuration process.



Configuration Prompts
Figure 31

(4) Data Transmission Indicators

See “Figure 32”.

Several configuration screens utilize an indicator light system to show discrete (ON/OFF) data and/or hardware component status. Unless otherwise noted, the following applies to all such status indicators:

(a) Green Light: Expected data is successfully received and is ON. A green light could also indicate that the parameter/component is working correctly.

(b) Red Light: Expected data is not received. A red light could also indicate that a parameter/component is invalid.

(c) No Light (Black): Expected data is successfully received and is OFF, or no data is expected. A black light could also indicate that the parameter/component is not responding.

(5) Configuration Mode Navigation

Using the FMS knob, a user can navigate through different pages and page groups in the Configuration Mode. See “Chart 25” on page 342501101. For a complete description and breakdown of each page, refer to the G1000 Line Maintenance & Configuration manual.



Data Transmission Indicators
Figure 32

CHART 25
CONFIGURATION MODE NAVIGATION

System Page Group

1. System Status
2. Date/Time Setup
3. Main Lighting
4. Audio Alert Configuration
5. System Upload
6. Diagnostics Terminal
7. Maintenance Log
8. OEM Diagnostics
9. System Configuration
10. System Setup
11. Manifest Configuration

GDU Page Group

1. RS-232 / ARINC 429 Config
2. GDU Status
3. GDU Test
4. Diagnostics
5. Serial/Ethernet I/O
6. Alert Configuration
7. Airframe Configuration

GIA Page Group

1. RS-232 / ARINC 429 Config
2. CAN / RS-485 Config
3. GIA I/O Configuration
4. COM Setup
5. GIA Status

GEA Page Group

1. Engine Data
2. GEA Status
3. GEA Configuration

GTX Page Group

1. RS-232 / ARINC 429 Config
2. Transponder Configuration

GRS Page Group

1. AHRS / Air Data Input
2. GRS / GMU Calibration

GDC Page Group

1. GDC Configuration

GFC Page Group

1. GFC Configuration
2. GFC Status

GMA Page Group

1. GMA Configuration

GDL Page Group

1. GDL 69 Config

RMT Page Group

1. Remote Controller Status

GWX Page Group

1. GWX Configuration

CAL Page Group

1. Fuel Temperature Calibration
2. Flaps and Trim Calibration
3. HSCM Calibration

J. G1000 Software Information

NOTE: The following sections provide a detailed description of loading all G1000 software and configuration files, which may be excessive for individual LRU removal and replacement. If removing and replacing individual LRUs, refer to “Components” on page 34250137, for the necessary steps.

(1) G1000 Software Loader Card

CAUTION: IT IS CRITICAL THAT THE TECHNICIAN INSTALLS THE CORRECT SOFTWARE IMAGE PART NUMBER WHEN SERVICING THE G600 SYSTEM. APPROVED SOFTWARE DEFINITIONS FOR EACH CONFIGURATION ARE DEFINED IN THE REQUIRED EQUIPMENT LISTS (REL), “CHART 30” ON PAGE 342501165

All software and configuration files were certified by Garmin as part of the FAA-approved Type Design data. Approved software definitions for each configuration are defined in the Required Equipment List (REL), “Chart 30” on page 342501165.

G1000 software and configuration files are controlled via the approved software image part number listed in the REL. This software image is loaded into the G1000 using a software loader card. The installer shall create this software loader card by downloading the approved software image in accordance with the procedure provided immediately following.

CAUTION: BE CAUTIOUS WHEN USING SOFTWARE LOADER CARDS DURING MAINTENANCE. THE G600 SYSTEM IMMEDIATELY INITIALIZES THE CARD UPON POWER-UP. ON-SCREEN PROMPTS MUST BE GIVEN CAREFUL ATTENTION IN ORDER TO AVOID POTENTIAL LOSS OF DATA. ALWAYS READ THROUGH PROCEDURES GIVEN IN THIS SECTION, BEFORE ATTEMPTING TO USE THE SOFTWARE LOADER CARDS.

(2) Software Loader Card Creation

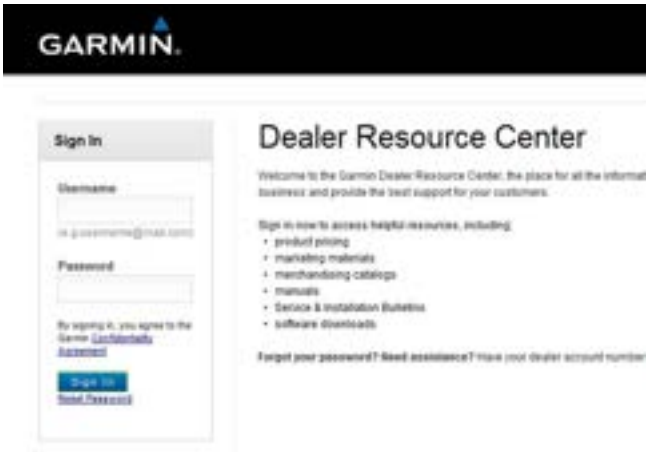
The following procedure requires a PC with an SD card reader. Piper recommends SanDisk® SDDR-99 and SDDR-93 card readers. Additionally, only SanDisk brand 16GB SD cards are recommended.

The software image is an executable self-extracting file which builds the correct file structure onto an SD card for use loading software to the G1000. To obtain the current file follow the procedures outlined below.

NOTE: In order to create a loader card, the installer completing these procedures must be an authorized Piper or Garmin service center to gain access to the necessary data via the Garmin website.

(a) Go to https://www.garmin.com and click on the ‘Dealer Resource Center’ link in the lower right hand portion of the home page. Enter username and password. See “Figure 33”.

(b) Select Support and then Software Downloads.



Dealer Login Screen
Figure 33

(c) Search for the appropriate image version (i.e., “1981.xx” from the appropriate “Chart 30” on page 342501165. in the “Search by Keyword” field.

(d) Save the file to a known location on the PC.

(e) Run the executable file that was downloaded. A window will appear to confirm software installation.

(f) Insert the SD card into the SD card reader and select the Next button in the pop-up window.

(g) In the pop-up window, select the drive that corresponds to the inserted SD card and then click the Next button.

NOTE: The selected drive will be completely erased. Click the Next button to acknowledge any warnings that appear. Another pop-up window will appear to monitor loader card creation progress. After the card is successfully completed, another pop-up window will appear.

(h) Select the Finish button.

(i) Stop the card reader using Windows stop device function.

(j) Eject the SD card from the card reader.

(k) Apply a label to the software loader card identifying the aircraft, G1000 SFTW and the appropriate software image part number as shown below:

NOTE: See appropriate “Chart 30” on page 342501165 for software image part number.

PA-34-220T G1000 SFWT 006-x1981-xx

(3) Handling a Software Download Card

The SD memory cards used for the Software Download Card are manufactured to the specifications of the SD Card Association for use in portable devices. As such they are reasonably robust devices that do not require special handling instructions. However, the memory cards are not unbreakable and reasonable precautions regarding their use and handling should be used including the following guidelines:

(a) Do no bend an SD memory card or subject it to violent impact.

(b) Do not leave SD cards in locations where they may be exposed to moisture or high temperatures.

(c) Avoid exposing the card to static electricity or electro-magnetic interference.

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K. Software Files

Software files are defined by part number and version number. Each G1000 LRU reports the software version it currently contains to the user in two places.

(1) Normal System Mode

The AUX – SYSTEM STATUS page lists each LRU and the reported software version.

(2) Configuration Mode

The SYSTEM STATUS page (SYSTEM page group) reports more detailed LRU information, including software version, part number, and LRU status. Software files are loaded to LRUs from the SYSTEM UPLOAD page in configuration mode.

(3) Configuration File Descriptions

(a) Configuration Files are divided into groups, and are only displayed at the System Upload page during the software/configuration loading process.

(b) Configuration files contain preset selections for input/output channels, aircraft-specific settings, and LRU-specific settings. The following list describes each of the configuration files:

AIRFRAME This file contains data such as airspeed parameters, engine/airframe sensor limitations, fuel tank parameters and alerting system settings that tailor a G1000 PFD or MFD to the PA34-220T.

AIRFRAME ALERTS PA34-220T-specific alerts are set when this file is loaded.

SYSTEM This file configures the G1000 high-speed data bus (HSDB) to expect two PFDs, MFD, two GIAs, and a GDL69.

MANIFEST This file loads a manifest checklist of all software part numbers and versions associated with an approved system configuration. The G1000 performs a software check between each LRU's reported version and the version contained on the manifest. If an inequality is detected for an LRU, this LRU is then excluded from the G1000 and a manifest alert is triggered to the operator.

MFD1 This file configures MFD serial/discrete communication and alert system settings.

PFD1/PFD2 These files configure PFD1/PFD2 serial/discrete communication and alert system settings.

GIA1/GIA2 These files configure GIA1/GIA2 serial/discrete communication settings.

GMA_PIL This file configures GMA 1347 audio and serial communication settings

GTX1/GTX2 These files configure GTX 1/GTX 2 transponder and serial communications settings.

GEA1/GEA2 These files configure GEA1/GEA2 engine/airframe parameters.

GDC_PIL/ GDC_COPIL This file configures GDC 74A air data values for the PA34-220T.

GDL_69 This file configures gains and cable loss for the PA34-220T installation.

GWX This file configures parameters as appropriate for the PA34-220T radar installation.

AUDIO This file configures audio alerts for the PA34-220T.

(4) Configuration File Storage

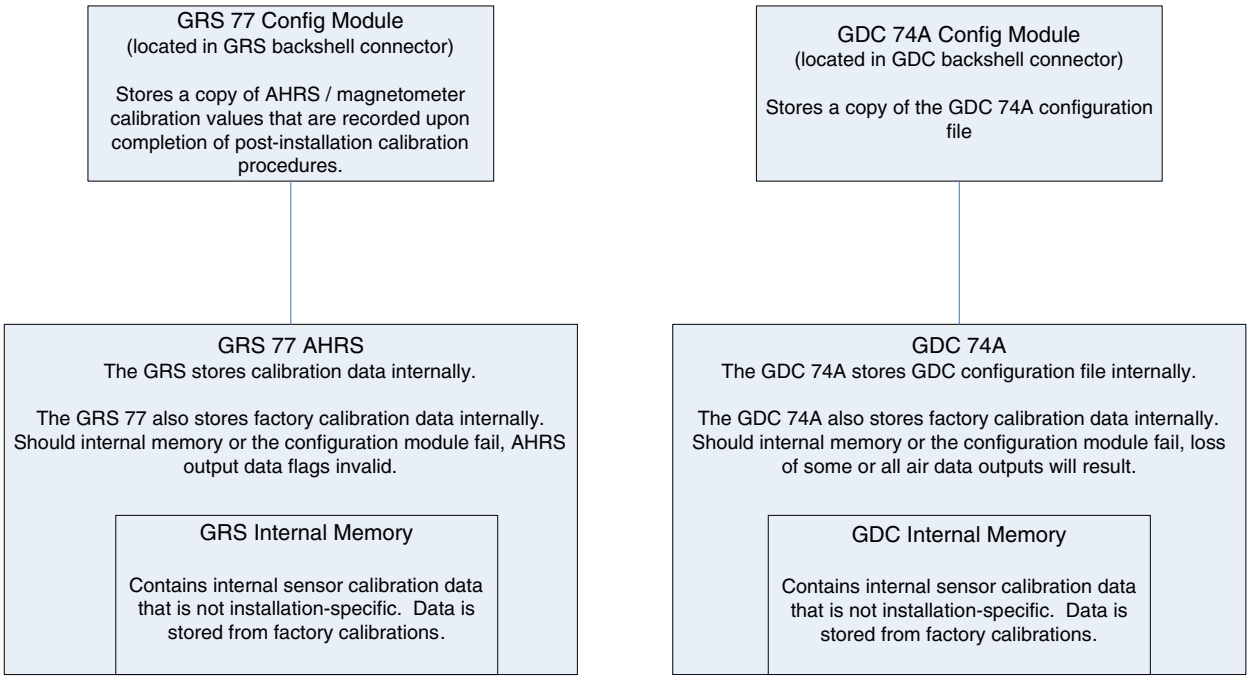
The G1000 system is designed to store all configuration settings in various places so that the configuration is retained in the aircraft during maintenance of units.

During system configuration, each file is sent directly to the applicable LRU where it is stored in local LRU memory (except GRS 77 & GDC 74A). Each file is also stored in the PFD internal memory. The applicable PFD also sends a copy of all configuration files to the 'Master Configuration module', located in the connector backshell (see "Configuration Mode Overview" on page 34250197). If the PFD is replaced, the configuration module retains all configuration files in the aircraft.

NOTE: The GRS 77 AHRS and GMU 44 Magnetometer do not have a configuration file. However, these LRUs do store calibration data acquired during the post installation checkout, which are characteristic to the specific installation. While performing maintenance on these units, re-calibration may be required. See "GRS/GMU Calibration Procedures" on page 342501134.

The GRS 77 and GDC 74A configuration modules, as shown in "Figure 34", function differently than the rest of the system. The GDC 74A's configuration file is loaded directly to GDC internal memory. A copy of the file is stored in the GDC configuration module.

The GRS 77 configuration module does not store any configuration settings. Instead, it stores calibration data recorded during installation calibration procedures.

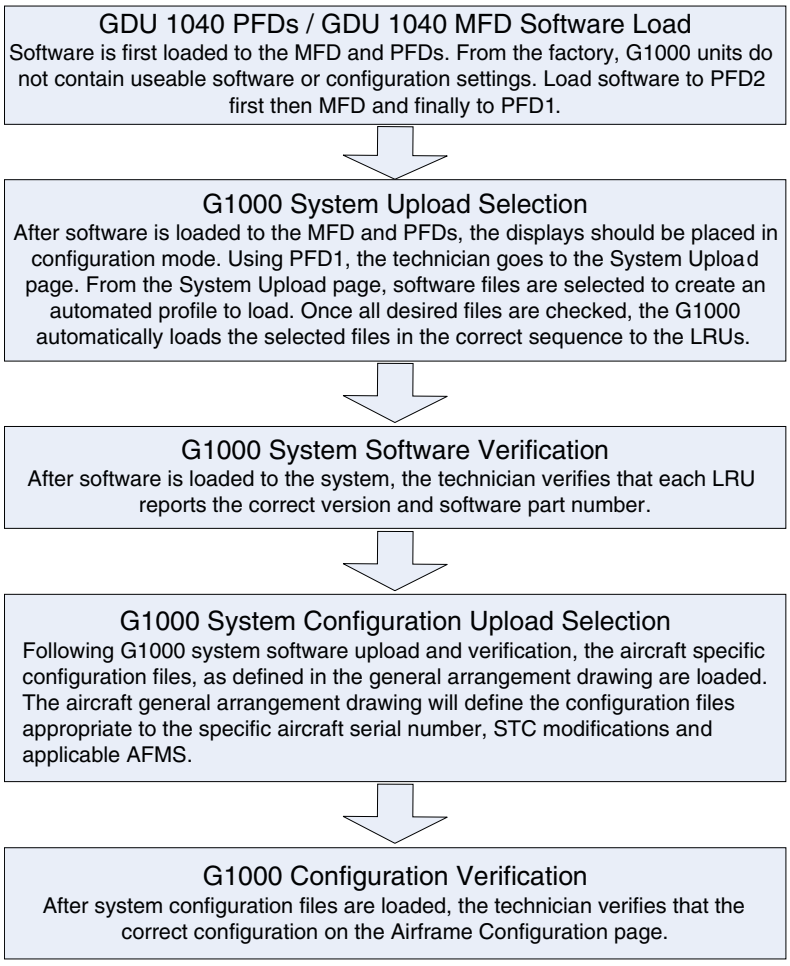


GRS/GDC Configuration Settings Storage
Figure 34

(5) G1000 Software/Configuration Procedure

This and the following sections summarize the procedures required to load software and configuration files to the G1000. It is intended to work as a central guide for technicians to use while performing maintenance on the aircraft. In sections of this manual where software is required to be reloaded, these sections will make reference back to this Section for instructions. The technician should use proper judgment regarding the context of maintenance required while following this section.

The diagram in “Figure 35” depicts an overview of the software/configuration sequence for the G1000 system. This applies mostly to a new G1000 system which has not previously been powered up and is for informative purposes only.



Software/Configuration Overview
Figure 35

L. Complete Software Load

(PIR-PPS55032, Rev. D./107867, Rev. C.)

(1) MFD & PFD Software Load

CAUTION: DO NOT INTERRUPT POWER TO THE AIRCRAFT AT ANY TIME DURING THESE PROCEDURES. INTERRUPTION OF POWER COULD RESULT IN DAMAGE TO THE EQUIPMENT.

(a) Apply power to the aircraft and all avionics.

(b) Pull the MFD and both PFD circuit breakers. Remove any SD cards that were previously inserted in either PFD or MFD.

(c) Insert G1000/PA-34 Loader Card into the PFD2 top card slot.

(d) While holding the ENT key on the PFD2, restore power to the PFD2 by pushing in the PFD2 circuit breaker.

(e) When the words “INITIALIZING SYSTEM” appear in the upper left corner of the PFD2, release the ENT key.

(f) Press the YES softkey at the following prompt (prompt may not appear):

“DO YOU WANT TO CLEAR USER SETTINGS?”
“YES WILL BE ASSUMED IN 10 SECONDS”

(g) Press the YES softkey at the following prompt:

“DO YOU WANT TO UPDATE SYSTEM FILES?”
“NO WILL BE ASSUMED IN 30 SECONDS”

“UPDATING SYSTEM FILES, PLEASE WAIT.”

(h) Press the YES softkey at the following prompt (prompt may not appear):

“DO YOU WANT TO UPDATE SPLASH SCREEN?”

(i) Press any softkey at the following prompt to confirm load completion:

“PRESS ANY KEY TO CONTINUE.”
“CONTINUING IN 10 SECONDS.”

After loading PFD2, go to step (j). For MFD, go to step (l). For PFD1, go to step (m).

(j) New software is loaded to the PFD2. When complete, the PFD2 starts in configuration mode.

(k) Remove the G1000/PA-34 Loader Card from PFD2 and insert it into the top card slot on MFD and repeat steps (d) thru (i). The rightmost soft key is used in place of the ENT key.

(l) Remove the G1000/PA-34 Loader Card from the MFD and insert it into the top card slot on PFD1 and repeat steps (d) thru (i). The ENT key is used in place of the softkey.

(m) When PFD1 update is complete, it starts in the configuration mode. DO NOT remove power.

CAUTION: FOR THE REST OF THE SOFTWARE/CONFIGURATION PROCEDURE, DO NOT OPERATE THE MFD OR PFD2 WHILE LOADING SOFTWARE OR CONFIGURATION FILES UNLESS SPECIFICALLY INSTRUCTED TO DO SO. A FAILED OR CANCELLED LOAD MAY RESULT.

(2) System Upload

NOTE: To Select/Deselect the cursor in fields, press the small FMS knob on the PFD. The large FMS knob changes Groups, and the small FMS knob changes Pages within a Group.

(a) Go to the SYSTEM UPLOAD page on PFD1 using the small FMS knob.

(b) Activate the cursor, select the Group field, and rotate the small FMS knob to generate a pick list. Rotate the small FMS knob to select Baseline.

(c) Press ENT.

(d) In the ITEM field, rotate the small FMS knob to generate a pick list and select PA-34-220T Seneca V.

(e) Press ENT.

(f) Press the LOAD softkey (The load takes approximately 1.5–2 hours to complete).

(g) Press ENT to accept the software load.

(h) Deselect cursor.

(3) Optional Equipment Configuration

Perform the following steps if any of the following options are installed:

KR 87 ADF	KN 63 DME
GWX 68	Air Conditioning
Dual GDC 74A/GTP 59	Dual GRS 77/GMU 44
GDL 69A	GSR 56
GTS 825 TAS (No GTX33 ES)	De-Ice
Oxygen	Hobbs Configuration (A, B, or C)
S-TEC 55x Autopilot	WX-500 Stormscope
Single GDC 74A/GTP 59	Single GRS 77/GMU 44
GTS 825 and GTX33 w/ ES	GTX33 w/ ES (No GTS 825)

NOTE: If Dual GDC option is not installed, the Single GDC option must be installed. If Dual GRS option is not installed, the Single GRS option must be installed.

(a) On PFD1, select the System Upload page and activate cursor.

(b) In the Group field, create a pick list by rotating the small FMS knob and select Options.

(c) Press ENT.

(d) In the ITEM field, create a pick list by rotating the small FMS knob and select the desired option, press ENT.

(e) Verify both Software and Configuration boxes are checked for Garmin options.

(f) Verify Configuration boxes checked for all other options.

(g) Press LOAD softkey.

(h) Verify the summary field lists the software and configuration are complete as required and PASS is displayed next to both appropriate boxes.

(i) Press ENT to accept.

(j) For other installed options, repeat steps (d) thru (i) by rotating the large FMS knob and highlighting the desired option in the ITEM Field.

(k) De-activate the cursor.

(l) Upon completion of all required baseline config, option config, and calibration loading procedures, remove the Loader card from top slot of PFD1.

(4) AFCS Installation (Optional)

The following options are available only with the GFC 700 autopilot. Perform the following steps if any of the following options are installed:

GFC 700 (without YD) Installation Option GFC 700 (with YD) Installation Option

Enhanced AFCS Support

NOTE: Enhanced AFCS Support must be loaded to any aircraft with the Enhanced AFCS option. This is required in addition to the Enhanced AFCS Unlock card to fully install the Enhanced AFCS option.

(a) On #1 PFD, rotate the FMS knob to select the SYSTEM UPLOAD page and activatecursor.

(b) In the GROUP field, create a pick list by rotating the small FMS knob and select GFC 700.

(c) Press ENT.

(d) In the ITEM field, create a pick list by rotating the small FMS knob and select the desired option, press ENT.

(e) Press the CHK ALL softkey.

(f) Press LOAD softkey.

(g) Verify the summary field lists the software and configuration are complete as required and PASS is displayed next to both appropriate boxes.

(h) Press ENT to accept.

(i) For other installed options repeat steps (d) thru (i) by rotating the large FMS knob and highlighting the desired option in the ITEM Field.

(j) De-activate the cursor.

(5) Calibration Option Load

NOTE: Only load configuration option once. Reloading of configuration option could result in a loss of system calibrations.

(a) Go to the System Upload page and activate cursor.

(b) In the Group field, create a pick list by rotating the small FMS knob and select Options.

(c) Press ENT.

(d) In the ITEM field, create a pick list by rotating the small FMS knob and select the desired calibration (Flaps Calibration Initialization or Fuel Calibration Initialization.

(e) Press LOAD softkey.

(f) Verify the Summary field lists the software and configuration are complete as required and PASS is displayed next to both appropriate boxes.

(g) Press ENT to select YES when prompted “Update Config Module?”.

(h) Press ENT to confirm completion.

(i) Upon completion of all required baseline config, option config, and calibration loading procedures, remove the Loader card from top slot of PFD1.

(6) Splash Screen Update

(a) Remove power from PFD1, PFD2, and MFD.

(b) Insert PA-34 Loader Card into the PFD2 top card slot.

(c) Apply power to PFD2.

(d) Press the NO softkey at the following prompt:

“DO YOU WANT TO UPDATE SYSTEM FILES?”

(e) Press the YES softkey at the following prompt:

“DO YOU WANT TO UPDATE SPLASH SCREEN?”

(f) At the prompt, press any key to continue.

(g) Remove PA-34 Loader Card from the PFD2 top card slot.

(h) Repeat steps (b) thru (g) for MFD, and then PFD1.

(i) Upon completion of splash screen update, remove the Loader card from top slot of PFD1.

(7) Software Load Confirmation

(a) Each PFD and MFD running in configuration mode.

(b) On PFD1, select the “Systems Status Page”. Activate the cursor and toggle to the LRU window.

(c) Highlight each of the listed items in the LRU window and check the LRU Part Number, software part number and software version against the current numbers and versions shown in the appropriate Required Equipment List (REL), ”Chart 30” on page 342501165.

NOTE: Not all of the software P/Ns and versions on the Systems Status page need to be verified, only those recorded in the appropriate REL.

(d) De-activate the cursor.

NOTE: If any software is not successfully loaded, do not continue with post installation procedures. Troubleshoot and resolve the issue before continuing.

(e) Re-start PFD1, PFD2 and MFD.

(f) On MFD splash screen (“Figure 41” on page 342501155), verify appropriate version number (“Piper PA-34-220T System xxxx.xx”) is displayed in the upper right corner of the MFD.

(g) Select softkey on MFD to proceed beyond splash screen.

(h) Verify that no manifest (software mismatch) alerts are shown on any of the displays.

(8) Configuration and Setup

(a) XM Satellite Radio

See “Chart 18” on page 34250179 if XM Satellite Radio is installed and requires activation.

(b) GRC 10 XM Pairing (If Installed)

NOTE: To pair the GRC 10 XM Handheld Remote Control to the GRT 10 Remote Transceiver you will need the GRT 10 Serial Number, and the GRC 10 will need to have two AA batteries installed.

1) Press and Release any key on the GRC 10 to power on the unit.
2) The GRC 10 will display: “GRT 10 not found”
3) Press the following buttons in sequential order:
   a) UP
   b) DOWN
   c) LEFT
   d) RIGHT
   e) MINUS (-)
   f) PLUS (+)
   g) PSET
4) Using the arrow buttons on the GRC 10, enter the GRT 10 Serial Number.
5) Verify that the correct Serial Number was entered.
6) Store the GRT 10 Serial Number by pressing the XM button.

(c) Aircraft Registration Number Entry

NOTE: The large FMS knob changes Groups and the small FMS knob changes Pages within a Group.

1) Start the PFD #1, MFD, and PFD #2 in Configuration Mode by holding the ENT key while applying power.
2) Select the GTX group by rotating the large FMS knob, and then select the TRANSPONDER CONFIGURATION page by rotating the small FMS knob on PFD #1.
3) Ensure that the ‘ADDRESS TYPE’ is ‘US TAIL’ under the ‘SET’ and ‘ACTIVE’ columns.
4) Ensure that the FLIGHT ID TYPE is SAME AS TAIL under the SET and ACTIVE columns.
5) Select the SYSTEM group by rotating the large FMS knob, and then select the AIRCRAFT CONFIGURATION page by rotating the small FMS knob.
6) Activate the cursor and highlight the AIRCRAFT REGISTRATION field. Use the small/large FMS knob to enter the aircraft registration number.
7) Once the correct registration number is entered, press the ENT key. Then press the SET GTX1 softkey to configure the transponder.
8) Press the ENT key to select OK at the dialog prompt.
9) Then press the ENT key again after configuration is completed.
10) Press the GTS softkey to configure the traffic system.
11) Press the ENT key to select OK at the dialog prompt.
12) Then press the ENT key again after configuration is completed.

(d) Flaps Calibration

NOTE: The large 'FMS' knob changes Groups and the small 'FMS' knob changes Pages within a Group.

1) Start the PFD1, MFD and PFD2 displays in the Configuration Mode by holding the ENT key while applying power.
2) On PFD #1, navigate to the CAL page group, then to the DAT CALIBRATION page.

NOTE: The FLAPS AND TRIM CALIBRATION page is not to be used to calibrate flaps for the G1000 PA-34 Seneca V.

NOTE: DAT CAL TBL 0, with the INPUT value of GEN_GAUGE_29 is the table that contains the flaps calibration data. All further references in this procedure are related to DAT CAL TBL 0. DO NOT modify any other tables in this page.

3) Ensure the flaps system is powered and functioning properly.
4) Set the flaps to the 0 degree detent position.
5) On PFD1, press the FMS knob to bring up the cursor and highlight the first point in the RAW PNTS column of the table.
6) Press the CAL softkey, and press the ENT key to acknowledge the pop-up box. The first RAW PNTS value will now be modified
7) Verify that the RAW VAL is unchanged from before the calibration (except for noise in the sensor sample updates), and that the CAL VAL now matches the desired flap position value (except for noise in the sensor sample updates).
8) Repeat steps 4) thru 7) for each of the 2nd, 3rd and 4th rows of the table, setting the flaps position to the 10 degree, 25 degree, and 40 degree detents (respectively).

(e) GSR 56 Activation Procedure (If Installed)

Contact FlightDataServices@Garmin.com PH: 866-739-5687 to activate Iridium services.

(f) Chartview, Synthetic Vision, and TAWS Unlock Procedures

WARNING: PRIOR TO UNLOCK OF ANY OF THE FOLLOWING OPTIONS, ENSURE THAT THE SOFTWARE IS PROPERLY LOADED AND THAT THE CONFIGURATION MODULE IS FUNCTIONAL.

1) CHARTVIEW Unlock (Optional)
   a) Remove power from both PFD's and MFD by opening the PFD and MFD circuit breakers.

   NOTE: The Chartview Enable card can only enable Charts on one system (one aircraft). A new Chartview Enable card must be used for each aircraft.

   b) Insert the Chart Unlock card, in the upper slot of the PFD1.
   c) While holding the ENT key on both PFD's and MFD, restore power by closing the circuit breakers.
   d) When the words "INITIALIZING SYSTEM" appear in the upper left corner of the MFD and PFD's release the ENT key.
   e) On the PFD1, go to the System Upload page with the small FMS knob.
   f) Activate the cursor and use the small FMS knob to create a pick list and select Configuration Files in the GROUP field. Press ENT.
   g) Use the small FMS knob to create a pick list and select Enable Chartview in the ITEM field. Press ENT.

h) Verify there is a check mark in the box in the configuration column for Airframe.

i) Press the LOAD softkey.

j) Monitor the status of the upload. When the upload is Complete and Pass is displayed, press ENT.

k) De-activate the cursor.

l) Power down the system and remove the Chartview Enable card from the PFD.

m) After the Chart function is unlocked, the card should be stored with the aircraft records.

2) Synthetic Vision Unlock (Optional)

a) Remove power from both PFD's and MFD by opening the PFD and MFD circuit breakers.

NOTE: The Synthetic Vision Enable card can only enable Synthetic Vision on one system (one aircraft). A new Synthetic Vision Enable card must be used for each aircraft.

b) Insert the Synthetic Vision Unlock card, in the upper slot of the PFD1.

c) While holding the ENT key on both PFD's and MFD, restore power by closing the PFD and MFD circuit breakers.

d) When the words "INITIALIZING SYSTEM" appear in the upper left corner of the PFD's and MFD, release the ENT key.

e) On the PFD1, go to the System Upload page with the small FMS knob.

f) Activate the cursor and use the small FMS knob to create a pick list and select Configuration Files in the GROUP field. Press ENT.

g) Use the small FMS knob to create a pick list and select Enable SVS DUAL PFD in the ITEM field. Press ENT.

h) Verify there is a check mark in the box in the configuration column for Airframe.

i) Press the LOAD softkey.

j) Monitor the status of the upload. When the upload is Complete and Pass is displayed, press ENT.

k) De-activate the cursor.

l) Power down the system and remove the Synthetic Vision Unlock card from the PFD.

m) After the Synthetic Vision function is unlocked, the card should be stored with the aircraft records.

3) TAWS Unlock (Optional)

a) Remove power from both PFD's and MFD by opening the PFD and MFD circuit breakers.

NOTE: The TAWS Enable card can only enable TAWS on one system (one aircraft). A new TAWS Enable card must be used for each aircraft.

b) Insert the TAWS Unlock card in the upper slot of the PFD1.

c) While holding the ENT key on the both PFD's and MFD, restore power by closing the PFD and MFD circuit breakers.

d) When the words "INITIALIZING SYSTEM" appear in the upper left corner of the PFD's and MFD release the ENT key.

e) On the PFD1, go to the System Upload page with the small FMS knob.

f) Activate the cursor and use the small FMS knob to create a pick list and select Configuration Files in the GROUP field. Press ENT.

g) Use the small FMS knob to create a pick list and select Enable TAWS in the ITEM field. Press ENT.

h) Verify there is a check mark in the box in the configuration column for Airframe.

i) Press the LOAD softkey.

j) Monitor the status of the upload. When the upload is Complete and Pass is displayed, press ENT.

k) De-activate the cursor.

l) Power down the system and remove the TAWS Unlock card from the PFD.

m) After the TAWS function is unlocked, the card should be stored with the aircraft records.

4) Enhanced AFCS Unlock (Optional)

The Enhanced AFCS Support option must be loaded in addition to using the Enhanced AFCS Enable card for the system to properly operate. See “AFCS Installation (Optional)” on page 342501110.

a) Remove power from the MFD and PFDs by opening the PFD and MFD circuit breakers.

NOTE: The Enhanced AFCS Enable card can only enable TAWS on one system (one aircraft). A new Enhanced AFCS Enable card must be used for each aircraft.

b) Insert the Enhanced AFCS Enable card, PS50207-22 in the upper slot of the #1 PFD.

c) While holding the ENT key on both PFDs and MFD, restore power by closing the PFD and MFD circuit breakers.

d) When the words “INITIALIZING SYSTEM” appear in the upper left corner of the PFDs and MFD release the ENT key.

e) On the #1 PFD, go to the SYSTEM UPLOAD page with the small FMS knob.

f) Activate the cursor and use the small FMS knob to create a pick list and select Configuration Files in the GROUP field. Press ENT.

g) Use the small FMS knob to create a pick list and select Enhanced AFCS in the ITEM field. Press ENT.

h) Verify there is a check mark in the box in the configuration column for Airframe. 8.

i) Press the LOAD softkey.

j) Monitor the status of the upload. When the upload is Complete and Pass is displayed, press ENT.

k) De-activate the cursor.

l) Power down the system and remove the Enhanced AFCS Enable card from the PFD.

m) After the Enhanced AFCS function is unlocked, the card should be stored with the aircraft records.

(g) Aviation Database Loading Procedures

1) GARMIN GDU 1040 Terrain Data Card Loading Procedure

NOTE: Garmin Field Service Engineering, resident at Piper, will be responsible for supplying updates for the GDU 1040 Terrain Data Base Card.

a) Remove power from the PFD's and MFD by pulling the PFD and MFD circuit breakers.

b) Identify each Terrain Data Base Card respective to the display in which it is to be installed (PFD1, MFD, PFD2).

c) Install a Terrain Database Card in the bottom slot of the PFD's and MFD.

d) This completes the installation of the Terrain Database Card.

2) JEPPS NAV Database Aircraft Loading Procedure

a) Remove power from the PFD's and MFD by pulling the PFD and MFD circuit breakers.

b) Install the NavData card in the top slot of the PFD1.

c) Restore power to the appropriate display (PFD1, PFD2, MFD) by closing the appropriate circuit breaker.

The following prompt is displayed in the upper left corner of the display.

"DO YOU WANT TO UPDATE THE AVIATION DATABASE?"

"PRESS CLR FOR NO AND ENT FOR YES "

"YOU HAVE 30 SECONDS BEFORE NO IS RETURNED"

d) Press the ENT key to confirm the database update. The following prompt is displayed:

"DO YOU WANT TO UPDATE THE AVIATION DATABASE?"

"PRESS CLR FOR NO AND ENT FOR YES"

"YOU HAVE 30 SECONDS BEFORE NO IS RETURNED"

"UPDATING AVIATION DATABASE"

"UPDATED 2 FILES SUCCESSFULLY!"

e) After the update completes, the PFD starts in normal mode. The PFD aviation database is now updated.

f) Remove power from the PFD1 circuit breaker and remove the NavData card from the top slot of the PFD1 and place it in the top slot of the PFD2. Repeat steps c) and d) for PFD2. After the update completes, PFD2 starts in normal mode. The PFD2 aviation database is now updated.

g) Remove power from the PFD2 circuit breaker and remove the NavData card from the top slot of the PFD2 and place it in the top slot of the MFD. Repeat steps c) and d) for the MFD. After the update completes, the MFD starts in normal mode. The MFD aviation database is now updated. Remove power from the MFD circuit breaker and remove the NavData card from the top slot of the MFD

h) After the JEPPS NAV Database loading procedure is completed, the card should be stored with the aircraft records.

3) JEPPS CHARTS Aircraft Loading Procedure
   a) Remove power from the MFD by pulling the MFD circuit breaker.
   b) Install Jepps Charts Card in the bottom slot of the MFD.
   c) Restore power to the MFD by closing the MFD circuit breaker.
   d) Press the ENT key to confirm the database update.
   e) After the update completes, the MFD starts in normal mode.
   f) The Jepps Charts are now loaded in the MFD and do not need to be loaded in the PFD.
   g) The Jepps Charts Card remains in the bottom slot of the MFD.

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M. Post-Installation Procedures

CAUTION: ENTER “MAINTENANCE MODE” WHEN ACTIVATING THE GARMIN G1000 SYSTEM ON THE GROUND TO CONDUCT TESTS, INSPECTIONS, ETC. SEE “MAINTENANCE MODE” ON PAGE 3425018.

NOTE: This entire procedure must be successfully accomplished in order for the G1000 system to be airworthy in your airplane.

NOTE: The following checks MUST be performed prior to electrical power-up with the associated wiring connected to the electronic equipment. Failure to perform these checks prior to powering up may result in damage to the equipment.

(1) Bus Power Check

(a) Do not apply external power to the aircraft.

(b) Verify that all circuit breakers are set.

(c) Turn ON Battery Master switch with Avionics switch and the Emergency switch in the “OFF” position.

1) Verify Avionics Bus 1, Avionics Bus 2, and Emergency Bus are not powered.
2) Pull Non-Ess Bus 1, Non-Ess Bus 2 and Lighting Bus circuit breakers.
3) Verify Non-Ess Bus 1, Non-Ess Bus 2 and Lighting Bus are not powered.
4) Reset Non-Ess Bus 1 circuit breaker.
5) Verify Non-Ess Bus 1 is powered. Verify Non-Ess Bus 2 and Lighting Bus are not powered.
6) Pull Non-Ess Bus 1 circuit breaker and reset Non-Ess Bus 2 circuit breaker.
7) Verify Non-Ess Bus 2 is powered. Verify Non-Ess Bus 1 and Lighting Bus are not powered.
8) Pull Non-Ess Bus 2 circuit breaker and reset Lighting Bus circuit breaker.
9) Verify Lighting Bus is powered. Verify Non-Ess Bus 1 and Non-Ess Bus 2 are not powered.
10) Pull Lighting Bus circuit breaker.
11) Turn ON Avionics Master switch.
12) Verify Non-Ess Bus 1, Non-Ess Bus 2 and Lighting Bus are not powered.
13) Verify Ess Bus 1, Ess Bus 2, Av Bus 1, and Av Bus 2 are powered.
14) Pull Ess Bus 1 circuit breaker.
15) Verify Ess Bus 1 is not powered. Verify Ess Bus 2, Av Bus 1, and Av Bus 2 are powered
16) Reset Ess Bus 1 circuit breaker. Pull Ess Bus 2 circuit breaker.
17) Verify Ess Bus 2 is not powered. Verify Ess Bus 1, Av Bus 1 and Av Bus 2 are powered
18) Pull AV Bus Tie circuit breaker.
19) Verify Ess Bus 2 and Av Bus 2 are not powered. Verify Ess Bus 1 and Av Bus 1 are powered.
20) Pull Av Bus 1 and Av Bus 2 circuit breakers. Reset Ess Bus 2 circuit breaker.
21) Verify Ess Bus 1 and Ess Bus 2 are powered. Verify Av Bus 1 and Av Bus 2 are not powered.

(d) Turn the Emergency switch to ARM, then turn OFF the Battery Master switch and Avionics switches.

1) Verify Ess Bus 1, Ess Bus 2, Non-Ess Bus 1, Non-Ess Bus 2, Av Bus 1, Av Bus 2 and Lighting Bus are not powered.

2) Verify Emergency Bus is powered.

(e) Turn OFF the Emergency switch, the Battery Master switch and Avionics Master Switch.

1) Verify the Battery Bus is powered.

2) Verify the Emergency Bus and Main Bus are not powered.

(2) Wiring Harness Checkout

The following is required only when a harness has been repaired, modified, or replaced, but is recommended when troubleshooting.

Prior to installing any LRUs, the wiring harness should be checked for proper connections to the aircraft systems and other avionics systems. Point to point continuity should be checked to expose any faults such as shorting to ground. Any faults or discrepancies found should be corrected before proceeding. Refer to wiring schematics in chapter 91-00-00 of this manual.

(3) Initial Display Testing

(a) Verify the following:

1) COM/NAV fields are valid in the top corners of the PFD's.

2) Altitude, airspeed, vertical speed, heading and OAT fields are valid on the PFD's.

3) In airplanes equipped with the S-TEC autopilot only, set FD/AP master switch to FD.

4) No BACKUP PATHs alerts on the PFD's. If an LRU is not communicating over its primary path, the BACKUP PATH alert will identify which LRU is having the problem.

5) Select Reversionary Mode on the GMA 1347 Audio Panel, and displays will display valid attitude, altitude, airspeed, vertical speed and engine instruments. De-activate the Reversionary Mode.

6) Verify flap position indication matches selected flap angles at each flap setting (0°, 10°, 25°, and 40°).

7) With Battery Master switch set to ON:

Verify Main bus voltage indication.

(b) BAGGAGE DOOR annunciation

1) Open door.

2) Verify "BAGGAGE DOOR" annunciation is an advisory (White).

3) Close door.

4) Verify "BAGGAGE DOOR" annunciation is cleared from display.

5) Defeat both squat switches.

6) Open baggage door.

7) Verify "BAGGAGE DOOR" annunciation is a warning (RED).

8) Close baggage door.

9) Verify "BAGGAGE DOOR" annunciation is cleared from display.

(c) HYDR PUMP ON and GEAR DOWN discrete I/O's

1) Suspend the aircraft on jacks.
2) Put #1 PFD in configuration mode by pressing the ENT key while applying power.
3) Using the FMS knob on the #1 PFD go to the GIA I/O Configuration page (GIA group, page 3).
4) Activate the cursor and using the large FMS knob scroll to channel "IN* 1A (Right), IN* 11A (Nose), and IN* 12A (Left)". Verify the DATA box next to CHANNEL "IN* 1A, IN* 11A, and IN* 12A" is green with the landing gear down and black with the landing gear up (this verifies the GEAR DOWN discrete I/O is functioning).
5) Verify that L GEAR DOWN, R GEAR DOWN, AND N GEAR DOWN indications (green circles) are displayed in the EIS strip on the MFD.
6) Activate the cursor and using the large FMS knob scroll to channel "IN 16".
7) Retract the landing gear and verify the DATA box next to CHANNEL "IN 16" is green during operation of the hydraulic pump and black when the hydraulic pump is off (this verifies the HYDR PUMP ON discrete I/O is functioning).
8) Verify that L GEAR, R GEAR, and N GEAR Transition Indication is displayed as White square with crosshatching in the EIS strip on the MFD while the gear is retracting.
9) Activate the cursor and using the large FMS knob scroll to channel "IN* 2A (Right), IN* 10A (Nose), and IN* 13A (Left)". Verify the DATA box next to CHANNEL "IN* 2A, IN* 10A, and IN* 13A" is green with the landing gear up and black with the landing gear down (this verifies the GEAR UP discrete I/O is functioning).
10) Verify that L GEAR, R GEAR, and N GEAR are displayed as WHITE hollow circles in the EIS strip on the MFD when the gear is up.
11) Select the gear down.

(d) Stall Annunciation

1) With flaps at 0 degrees, activate the outboard stall vane.
2) Verify "Stall" aural alert.
3) Move flaps to 25 degrees.
4) Activate the outboard stall vane and verify "Stall" aural alert.

(e) Pitot Heat Annunciations

1) With power removed from aircraft, remove connector J727.
2) Apply ground power to aircraft.
3) Verify "PITOT HT OFF" annunciation appears on display.
4) Turn the Pitot Heat switch to the ON position.
5) Verify "PITOT HT FAIL" annunciation appears on display.
6) Remove ground Power to aircraft and reconnect J727.

(f) Avionics Cooling Fan Fail Annunciation Test

1) Pull the “AVIONICS FAN” circuit breaker.
2) Verify “AVI FAN FAIL” advisory message annunciates after 5 seconds.
3) Close the “AVIONICS FAN” circuit breaker.
4) Verify “AVI FAN FAIL” advisory message is cleared from the PFD display.
5) Pull the “MFD FAN” circuit breaker.
6) Verify “MFD FAN FAIL” advisory message annunciates after 5 seconds.
7) Close the “MFD FAN” circuit breaker.
8) Verify “MFD FAN FAIL” advisory message is cleared from the PFD display.
9) Pull the “PFD 2 FAN” circuit breaker.
10) Verify “PFD2 FAN FAIL” advisory message annunciates after 5 seconds.
11) Close the “PFD 2 FAN” circuit breaker.
12) Verify “PFD2 FAN FAIL” advisory message is cleared from the PFD display.
13) Pull the “PFD 1 FAN” circuit breaker.
14) Verify “PFD1 FAN FAIL” advisory message annunciates after 5 seconds.
15) Close the “PFD 1 FAN” circuit breaker.
16) Verify “PFD1 FAN FAIL” message advisory is cleared from the PFD display.

(g) Hour Meter Operation Test

1) For aircraft with Hobbs Configuration B, defeat both squat switches. Hobbs shall run while aircraft is “in air”.
2) For aircraft with Hobbs Configuration C, Hobbs shall run whenever GIA 1 is powered and reports OK status.

(h) In airplanes equipped with GFC 700 only: with external power removed and Battery Master switch set to ON:

Verify Main bus voltage indication.

(i) With Emergency switch set to ARM, Battery Master switch set to OFF, and external power removed:

Verify Emergency bus voltage indication.

(4) G1000 Connection Check

Perform the following steps to verify the following:

NOTE: Do not have External Power on the aircraft while conducting the following tests.

(a) AHRS and ADC data path check

1) In the GIA page group, go to the GIA SERIAL CONFIGURATION page.
2) With GIA1 selected in the SELECT UNIT window, verify the RS232 Channel 1 (GDC1) and RS232 Channel 6 (GRS1) indicators are green.
3) With GIA1 selected in the SELECT UNIT window, verify the ARINC 429 IN5 (GDC1) and ARINC 429 IN6 (GRS1) indicators are green.
4) With GIA2 selected in the SELECT UNIT window, verify the RS232 Channel 1 (GDC2) and RS232 Channel 6 (GRS2) indicators are green.
5) With GIA2 selected in the SELECT UNIT window, verify the ARINC 429 IN5 (GDC2) and ARINC 429 IN6 (GRS2) indicators are green.
6) In the GDU page group, go to the SERIAL CONFIGURATION page.

7) With PFD1 selected in the SELECT UNIT window, verify the ARINC 429 IN1 (GRS1) and ARINC 429 IN2 (GDC1) indicators are green.
8) With PFD2 selected in the SELECT UNIT window, verify the ARINC 429 IN1 (GRS2) and ARINC 429 IN2 (GDC2) indicators are green.

(b) PFD and MFD Ethernet connection check

1) Re-start PFD2, PFD1 and MFD in normal mode.
2) On the circuit breaker panel (Av Bus 2), open the MFD circuit breaker:
3) Verify NAV1, COM1, NAV2 and COM2 remain valid on both PFD1 and PFD2.
4) Close the MFD circuit breaker and wait for MFD to initialize:
5) On the circuit breaker panel (Av Bus 2), open the PFD2 circuit breaker:
6) Press the DISPLAY BACK UP button.
7) Verify NAV2 and COM2 are each replaced by a red X and NAV1 and COM1 remain valid on both the MFD and PFD1.
8) Press the DISPLAY BACK UP button to return to normal mode.
9) Close the PFD2 circuit breaker and wait for PFD2 to initialize:
10) On the circuit breaker panel (Ess Bus 1), open the PFD1 circuit breaker:
11) Press the DISPLAY BACK UP button.
12) Verify NAV1 and COM1 are each replaced by a red X and NAV2 and COM2 remain valid on both the MFD and PFD2.
13) Press the DISPLAY BACK UP button to return to normal mode.
14) Close the PFD1 circuit breaker and wait for PFD1 to initialize:

(5) Trim Annunciator Check

For S-TEC equipped aircraft, see Autopilot Clutch Torque and Cable Tension Test under Automatic Flight Control System (AFCS) (If installed), above.

(6) Engine Data Check (GIA1 or GIA2 Inoperative)

NOTE: This procedure should be accomplished in conjunction with the engine run-up. Valid engine indications will be needed. Do the following to verify the engine data availability and validity, with GIA1 or GIA2 inoperative:

(a) Ensure the G1000 is in normal mode.
(b) With the engine running, note the indication values for MAP, TIT and oil pressure.
(c) On the circuit breaker panel (Ess Bus 1), open the INTEG AV1 circuit breaker.
(d) Verify the values for MAP, TIT and oil pressure remain unchanged.
(e) Close the INTEG AV1 circuit breaker and wait until the COM1 and NAV1 fields are valid.
(f) On the circuit breaker panel (Av Bus 2), open the INTEG AV2 circuit breaker.
(g) Verify the values for MAP, TIT and oil pressure remain unchanged.
(h) Close the INTEG AV2 circuit breaker and wait until the COM2 and NAV2 fields are valid.
(i) Put the G1000 in Reversionary mode.
(j) Repeat steps (b) thru (h), verifying engine gauge indications on PFD1, MFD and PFD2.
(k) The Engine Data Check is complete.

(7) G1000 Miscompare Checks (Optional)

This procedure will check the AHRS, airspeed and altitude miscompare monitors. Accomplish the following checks with the aircraft positioned where it can receive GPS signals and magnetic heading.

NOTE: Access to AHRS1 and AHRS2 will be required during this test.

(a) Ensure the G1000 is operating in normal mode.

(b) Connect a pitot/static test set to the aircraft ADC1 pitot and static ports (Pilot's side). (Do not connect the pitot/static tester to ADC2 ports at this time.)

(c) Set the baro-correction on PFD1 and PFD2 to 29.92.

(d) Slowly increase the pitot/static test set to simulate an ADC1 / ADC2 altitude miscompare of greater than 200 ft., verify ADC2 altitude does not increase and amber comparator window text "ALT MISCOMP" is displayed on PFD1 and PFD2.

(e) Reduce ADC1 altitude to ambient pressure.

(f) Repeat Step (e) with the pitot/static test set connected to the aircraft ADC2 pitot and static ports.

(g) Use the pitot/static test set to simulate an airspeed of 40 kts for ADC1 and 55 kts for ADC2. Verify the amber comparator window text "IAS MISCOMP" is displayed on PFD1 and PFD2.

(h) Use the pitot/static test set to simulate an airspeed of 55 kts for ADC1 and 40 kts for ADC2. Verify the amber comparator window text "IAS MISCOMP" is displayed on PFD1 and PFD2.

(i) Use the pitot/static test set to simulate an airspeed of 85 kts for ADC1 and 95 kts for ADC2. Verify the amber comparator window text "IAS MISCOMP" is displayed on PFD1 and PFD2.

(j) Use the pitot/static test set to simulate an airspeed of 95 kts for ADC1 and 85 kts for ADC2. Verify the amber comparator window text "IAS MISCOMP" is displayed on PFD1 and PFD2.

(k) Reduce ADC1 and ADC2 airspeeds to 0 kts.

(l) Slowly rotate AHRS1 along the lateral (pitch) axis to a pitch attitude of greater than 5 degrees, verify the following:

1) AHRS2 pitch attitude does not change.

2) An amber "PIT MISCOMP" is annunciated on PFD1 and PFD2.

(m) Replace AHRS1 to normal attitude and verify that attitude display on PFD1 displays current aircraft attitude.

(n) Slowly rotate AHRS1 along the longitudinal (roll) axis to a roll attitude of greater than 5 degrees, verify the following:

1) AHRS2 roll attitude does not change.

2) An amber "ROLL MISCOMP" is annunciated on PFD1 and PFD2.

(o) Replace AHRS1 to normal attitude and verify that attitude display on PFD1 displays current aircraft attitude.

(p) Repeat Steps (l) thru (o) for AHRS2.

(q) The G1000 Miscompare checks are complete.

N. LRU and System Test Procedures

CAUTION: ENTER “MAINTENANCE MODE” WHEN ACTIVATING THE GARMIN G1000 SYSTEM ON THE GROUND TO CONDUCT TESTS, INSPECTIONS, ETC. SEE “MAINTENANCE MODE” ON PAGE 3425018.

(1) PFD/MFD Test Procedure

(a) Allow displays to initialize for ~1 minute.

(b) Check that all COM/NAV display fields are valid in the top corners of both PFDs.

(c) Check that attitude, heading, altitude, airspeed, vertical speed and OAT fields are valid within 2 minutes of power up on both PFDs.

(d) Press the SENSOR softkey on each PFD and switch between ADC1 and ADC2. Verify that data from both GDC 74As is valid on both displays.

(e) Press the SENSOR softkey on each PFD and switch between AHRS1 and AHRS2. Verify that data from both GRS 77s is valid on both displays.

(f) Check that the engine instrument fields are valid on the MFD.

(g) Push the red DISPLAY BACKUP button on the pilot-side GMA 1347. Verify that the pilot-side PFD and MFD displays enter reversion mode. MFD should have valid altitude, airspeed, vertical speed, COM1, COM2, NAV1, NAV2 and engine instruments.

(h) De-activate pilot-side reversion mode by pushing the DISPLAY BACKUP button. Verify PFD1 and MFD return to normal display modes.

(i) Repeat Step (a) using the co-pilot-side PFD. Ensure that PFD2 and MFD enter reversion mode and MFD displays valid altitude, airspeed, vertical speed, COMM1, COMM2, NAV1, NAV2 and engine instruments.

(j) De-activate co-pilot’s side reversion mode by pushing the DISPLAY BACKUP button. Verify PFD2 and MFD return to normal display modes.

(k) If TAWS was activated, select the TAWS page (4th page in the MAP group) on the MFD.

1) Verify that the title at the top of the page reads “MAP – TAWS”. If TAWS has not been enabled, the title will read “MAP – TERRAIN PROXIMITY” or “MAP – TERRAIN”.

2) Press the MENU button and select “Test TAWS” from the pop-up menu.

3) After the TAWS test has completed, verify that “TAWS System Test Okay” is heard over the cockpit speaker.

(2) GMA 1347 Test

Except for marker beacon operation, an in-aircraft checkout may be performed in the aircraft with known good microphone, headset, and speaker.

(a) Check Pilot and Copilot Intercom System (ICS) to verify the pilot and copilot MIC and phone jack operation.

(b) Depress PA Key (for Passenger Address mode) on the Audio Panel, press the pilot MIC key and verify audio is heard over the cockpit speaker.

(c) Verify operation of COM1. Conduct VSWR check per “Chart 26” on page 342501127.

(d) Failsafe Operation Check – Audio Panel to COM #1

1) Turn the Audio Panel OFF by pulling the AUDIO MKR circuit breaker.

2) Verify COM1 transmits and receives normally over pilot headphones.

3) Verify COM1 audio is controlled by the PFD COM audio control.

4) Close the AUDIO MKR circuit breaker.

(e) Verify operation of COM2. Conduct VSWR check per “Chart 26” on page 342501127.

(f) Marker Beacon Test

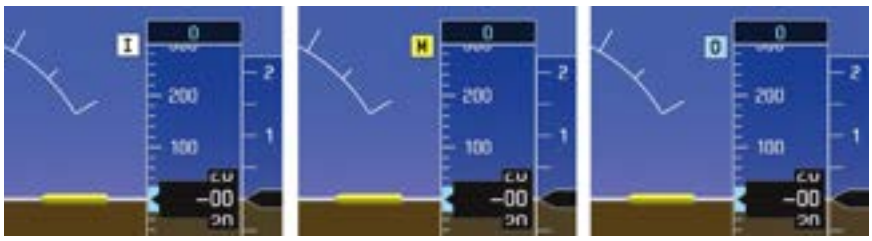
1) Verify that the outer, middle, and inner markers appear on PFD1 and PFD2 (See “Figure 36”).

2) Verify Marker Beacon Audio and tones.

3) Verify the MKR MUTE key mutes the Marker Beacon Audio.

4) Verify Marker HI/LO Sensitivity operation.

(g) Verify Passenger ICS is working at all stations.



Marker Beacon Symbology
Figure 36

CHART 26
VSWR TEST

VOLTAGE STANDING WAVE RATIO (VSWR)

The standing wave ratio is a measure of the amount of power transmitted to the antenna, compared to the amount reflected back to the transmitter. S.W.R.'s are a function of the transmission line, the antenna and their installation.

A. Required Equipment

Thruline Wattmeter – Model 43 Bird Electronics, Cleveland, Ohio.

B. S.W.R. Testing:

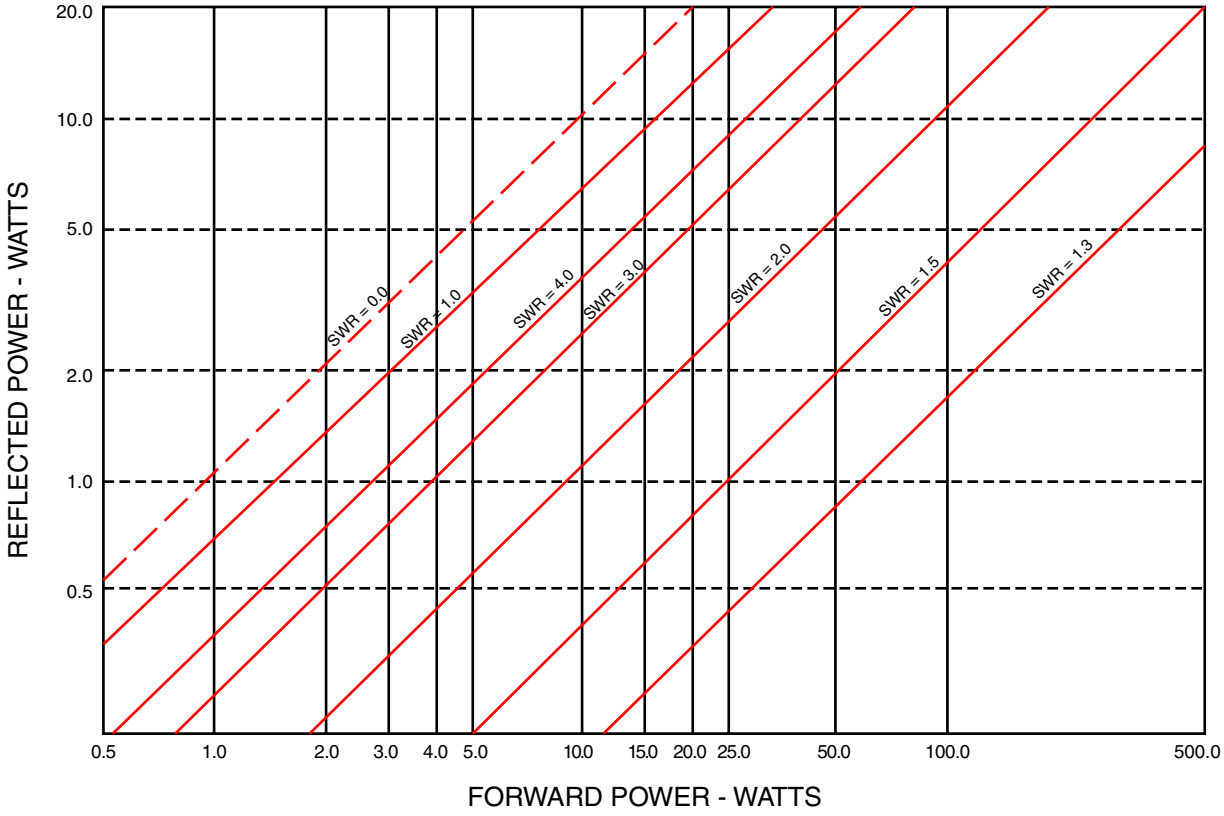
Test the antenna installation using the Thruline Wattmeter.

(1) Check VSWRs at 118.0, 123.0, 128.0, 132.0, and 135.95 MHZ by transmitting on COM 1 and then COM 2.

(2) Determine the VSWR value with the VSWR Conversion Chart below.

(3) VSWRs in excess of 2.5:1 are not acceptable.

VSWR CONVERSION CHART



(3) GIA 63W Test

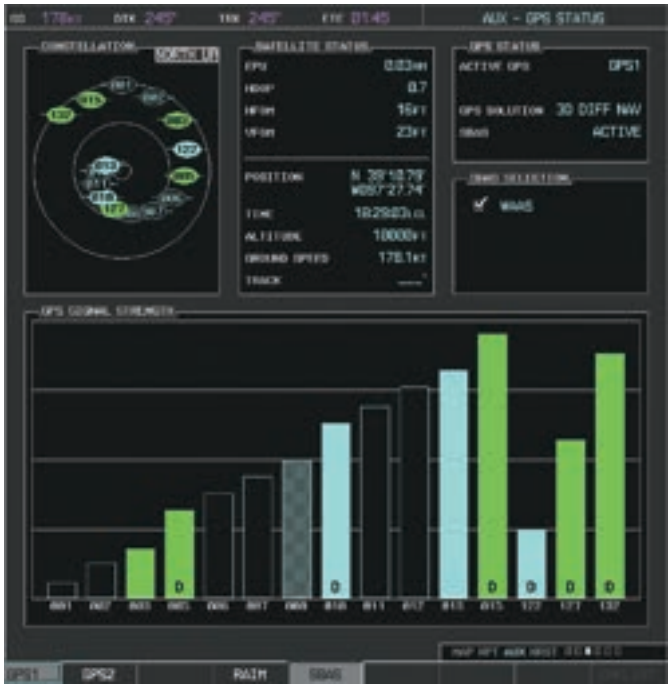
(a) GPS Signal Acquisition

See “Figure 37”.

The GIA 63W units should normally acquire a GPS navigation solution within 2 minutes of startup, provided the aircraft is outside (or indoors with a GPS repeater). Select the GPS STATUS page on the MFD (4th page in AUX group). Two softkeys on the bottom of the display allow the user to toggle between GPS1 and GPS2. Verify that both receivers show 3D DIFF NAV on the MFD.

Continue to the VHF COM Interference test.

NOTE: It may be necessary to temporarily disable or move away from GPS repeaters while testing, as repeaters may adversely affect GPS receiver performance.



GPS STATUS Page (MFD)
Figure 37

(b) VHF COM Interference Test

This test must be conducted outside. Use of a GPS repeater inside a hangar may result in a failed test.

This procedure assumes that the system is currently set to 25 kHz COM channel spacing. Once the signal acquisition test has been completed successfully, perform the following steps:

1) On the MFD, monitor GPS signal strength bars on the AUX – GPS STATUS page.
2) On the PFD, ensure that the CDI is set to GPS. If it is not, press the 'CDI' softkey until GPS ENR is displayed.
3) Verify that the GPS "INTEG" flag is out of view.
4) Select 121.150 MHz on the COM1 transceiver.
5) Transmit for a period of 35 seconds while monitoring GPS1 signal strength levels.
6) During the transmit period, verify that the GPS "INTEG" flag does not come into view on the PFD and verify that GPS1 does not lose a 3-D navigation solution on the MFD.
7) Repeat steps 5) and 6) and re-transmit while monitoring GPS2 signal levels on the MFD.
8) Repeat steps 4) thru 7) for each of the following frequencies:
   - 121.175 MHz
   - 121.200 MHz
   - 131.250 MHz
   - 131.275 MHz
   - 131.300 MHz
9) Repeat steps 4) thru 8) for the COM2 transceiver (GIA2).
10) On the MFD, select the AUX – SYSTEM SETUP page.
11) Under the COM CONFIG field, change the COM channel spacing from 25 kHz to 8.33 kHz.
12) Go back to the AUX – GPS STATUS page.
13) Select 121.185 MHz on the COM1 transceiver.
14) Transmit for a period of 35 seconds while monitoring GPS1 signal strength levels.
15) During the transmit period, verify that the GPS "INTEG" flag does not come into view on the PFD and verify that GPS1 does not lose a 3-D navigation solution on the MFD.
16) Repeat steps 14) and 15) and re-transmit while monitoring GPS2 signal levels on the MFD.
17) Repeat steps 14) thru 16) for each of the following frequencies:
   - 121.190 MHz
   - 130.285 MHz
   - 131.290 Mhz
18) Repeat steps 14) thru 17) for the COM2 transceiver (GIA2).
19) On the MFD, select the AUX – SYSTEM SETUP page and change the COM channel spacing back to 25 kHz.
20) Continue to the "VOR/LOC/GS Test" on page 342501130.

(c) VOR/LOC/GS Test

1) Perform NAV 1 and NAV 2 accuracy tests.
   a) Full scale.
   b) Half scale.
   c) OBS operation.
2) Check NAV 1 and NAV 2 audio and Ident.
3) Check NAV 1 and NAV 2 Sensitivity.
4) Perform LOC 1 and LOC 2 accuracy tests.
   a) Full scale.
   b) Half scale.
5) Check LOC 1 and LOC 2 Sensitivity.
6) Perform GS 1 and GS 2 accuracy tests.
   a) Full scale.
   b) Half scale.
7) Check GS 1 and GS 2 Sensitivity.

(4) GEA 71 Test

On the MFD (normal mode), check the indication for each of the sensor or monitor inputs with the aircraft engines off. In general, verify all engine and system instruments show valid static normal values and markings, with no red Xs or erratic indications. See “Figure 38” on page 342501131 (baseline configuration shown) for normal engine instrument markings.

(5) GTX 33 Transponder

(a) Configuration

“Aircraft Registration Number Entry” on page 342501112 configures the transponder:

(b) Test

Operation of the GTX 33 Mode-S transponder is accomplished using PFD1, PFD2 or the MFD. Refer to G1000 Cockpit Reference Guide, for basic operation.

The integrated transponder/altitude reporting system must be verified in accordance with Title 14 of the Code of Federal Regulations (CFR) §§ 91.411 and 91.413, every 24 calendar months, or any time the transponder is removed. This test requires the use of a Mode S ramp generator. Specific instructions for operating the ramp tester are contained in the applicable operator’s manual. Refer to Title 14 CFR Part 43.

(c) GTX 33 Extended Squitter Checkout (If installed)

1) Press the XPDR softkey on the PFD.
2) Verify that the ADS-B TX softkey is available and can be toggled (softkey 10).



NORMAL MODE



WHEN ENGINE PAGE DISPLAYED

Normal Engine Instruments
Figure 38

(6) GDC 74A Tests

NOTE: If any connections in the pitot / static system are opened for maintenance, the entire system must be rechecked per Pitot and Static Systems, "Test" on page 34108.

NOTE: Allow the unit to warm up for 15 minutes before performing the following tests.

Verification of the altimeter and airspeed must be performed using an air data test set (ADTS). The static port and altimeter must be verified in accordance with Title 14 of the Code of Federal Regulations (CFR) § 91.411 and Part 43 Appendix E. The PFD must be in Configuration mode and the MFD must be in Reversionary mode for performing the tests as outlined in Part 43 Appendix E.

(a) Part 43 Appendix E Testing

1) Preparation

CAUTION: CONFIGURATION MODE CONTAINS CERTAIN PAGES AND SETTINGS THAT ARE CRITICAL TO AIRCRAFT OPERATION AND SAFETY. THESE PAGES ARE PROTECTED AND CANNOT BE MODIFIED, UNLESS THE TECHNICIAN IS PROPERLY AUTHORIZED AND EQUIPPED. HOWEVER, MOST PROTECTED PAGES ARE VIEWABLE TO ALLOW SYSTEM AWARENESS FOR TROUBLESHOOTING.

a) Start the G1000 system in normal mode.

b) Remove power to PFD1.

c) Turn PFD1 on in Configuration mode by pressing and holding the ENT key on the PFD while applying power.

d) Release the ENT key after "INITIALIZING SYSTEM" appears in the upper left corner of the PFD.

NOTE: Configuration mode contains certain pages and settings that are critical to aircraft operation and safety. These pages are protected and cannot be modified, unless the technician is properly authorized and equipped. However, most protected pages are viewable to allow system awareness for troubleshooting.

2) Testing

a) Using the FMS knob on the PFD turn to the GRS page group. Use the B ALT field for all CFR Part 43 Appendix E tests for G1000 altitude.

b) Place the MFD in Reversionary mode by pressing the red "display backup" button on the GMA. The baro setting can then be read from the MFD for CFR Part 43 Appendix E tests.

NOTE: The baro setting on the MFD is controlled by PFD1. The baro setting will apply to both GDC1 and GDC2 regardless of the GDC selected on the MFD. The copilot's display can be ignored.

(b) After completing the tests specified by § 91.411 and Part 43 Appendix E, return both the MFD and the PFD to normal mode.

NOTE: The following tests are above and beyond the requirements set forth in Appendix E, and are required only when appendix E tests are required.

(c) Pitot/Static Airspeed Test

1) Command air data test set (ADTS) to simulate air speeds shown in the table below.

Calibrated Air Speed (Knots)	Allowed tolerance (± Knots)
50	5.0
80	3.5
100	2.0
120	2.0
150	2.0

2) Wait for ADTS to report that target values have been achieved.

3) Verify that computed air speeds shown on the PFD are within the tolerances specified in the OEM maintenance documentation.

(d) Static Port Vertical Speed (Rate of Climb) Test

1) Command ADTS to change the altitude at the rates shown in the table below.

2) Wait for ADTS to report that target rates have been achieved.

3) Verify that the Rate of Climb reported by the Vertical Speed field on the PFD is within the tolerances specified in the table below:

Vertical Speed (feet/minute)	Allowed tolerance, (± feet/minute)
2000	100
0	45
-2000	100

(7) OAT Probe Check

Ensure on-side sensors for PFD1 and PFD2. Ensure the outside air temperature (OAT) probes and a calibrated thermometer stabilize at ambient temperature. Verify that the OAT measurement shown on PFD1 and PFD2, in degrees Celsius, indicate within 2°C of the ambient temperature as measured by the calibrated thermometer.

(8) GRS/GMU Calibration Procedures

See “Chart 27” to determine when various calibrations are required.The following calibration procedures are provided for the GRS 77 and GMU 44:

- Pitch/Roll Offset:
- Magnetometer Calibration:
- Engine Run-Up Vibration Test:
- Magnetometer Interference Test:

When ready to perform the procedures, shut the PFDs and MFD off by pulling the PFD and MFD circuit breakers. Restart all displays in configuration mode. Follow the steps given for each procedure on-screen at the GRS/GMU CALIBRATION page. Note that the CALIBRATE command cannot be selected and activated until the installer acknowledges all required steps have been carried out by pressing the ENT key on each step.

CHART 27
G1000 - GRS 77 AHRS/GMU 44 MAGNETOMETER CALIBRATIONS

Required GRS/GMU Calibrations			
Condition	Calibrations Required		
	Procedure A: GRS 77 Pitch/Roll Offset	Procedure B: GRS/GMU Magnetic Calibration	Procedure C: Engine Run-up Vibration Test
Either GMU 44 was removed and reinstalled. (no change in serial number)	None Required. Continue to GRS/GMU Test section.		
GMU 44 was replaced with new unit. (New serial number)		X	
GRS 77 AHRS was removed and/or replaced. The mounting tray was NOT removed and the mounting tray bolts were NOT loosened.	None Required. Continue to GRS/GMU Test section.		
GRS 77 AHRS was removed and/or replaced. The mounting tray WAS removed and/or mounting tray bolts WERE loosened.	X	X	X
GRS 77 AHRS Configuration Module was replaced.	X	X	X

(9) GRS 77 Pitch/Roll Offset Calibration (Procedure A)

This procedure must be performed for both GRS 77 units installed in the aircraft. This first procedure must be carried out with the engine OFF.

(a) Level the aircraft to within ±0.25° of zero pitch and zero roll using a digital level. (Refer to 8-20-00.)

(b) Start the PFDs and MFD in Configuration mode by holding the ENT key on each display while closing the circuit breaker.

(c) Go to the GRS Page Group on both PFDs by rotating the large FMS knob.

(d) Select the GRS/GMU Calibration page by rotating the small FMS knob. This page is protected and requires a keystroke password to perform this test. Press the following softkeys in sequence on each PFD:

1) 9
2) 10
3) 11
4) 12 (far right softkey)

(e) Activate the cursor by pressing the small FMS knob and highlight the SELECT GRS UNIT window. Select GRS 77 #1 ON PFD1 AND GRS 77 #2 ON PFD2 (if dual sensors are installed).

(f) Highlight the SELECT PROCEDURE window and select **PITCH/ROLL OFFSET** on each PFD.

(g) Press the ENT key.

(h) Follow the checklist items displayed on each PFD and press the ENT key as each step is completed or confirmed.

(i) When the CALIBRATE field is blinking on both PFDs, press the ENT key on each PFD to begin the procedure.

(j) After several seconds, a new checklist appears in the lower half of the PFD. Press the ENT key as each step is confirmed. When the CONFIRM AIRCRAFT IS LEVEL field is blinking on both PFDs, press the ENT key to continue.

(k) The result of the pitch/roll offset compensation is displayed on each PFD. If successful, the PFDs report the successful calibration and return to normal operation.

(l) Press the ENT key on each PFD to conclude this procedure.

(m) Restart all displays in normal mode.

(10) Compass Rose Evaluation of Magnetic Disturbances for Magnetometer Calibration Procedure (Optional)

NOTE: The “GRS 77/GMU 44 Magnetometer Calibration (Procedure B)” on page 342501137 must be carried out at a site that is free of magnetic disturbances. If it is unsure whether the site is ‘clean’, the technician should verify that the site is ‘clean’ by performing the following procedure. The technician may skip the Compass Rose Evaluation Procedure if the site condition is acceptable.

NOTE: Typically, a compass rose is an acceptable location to perform the magnetometer calibration procedure. However, because not all compass roses are well maintained, even an existing compass rose should be regularly evaluated using the method described here to determine if it is free of magnetic disturbances. If evaluation of an existing compass rose indicates that magnetic disturbances are present, then an alternative location must be found to perform the Magnetometer Calibration procedure.

A G1000-equipped airplane that has completed the pitch/roll offset compensation procedure can be used to evaluate a candidate site for magnetic disturbances and determine whether it is a suitable location to perform the magnetometer calibration procedure. The magnetometer calibration procedure itself contains the logic to simultaneously survey the location for magnetic cleanliness while it is computing the magnetometer calibration parameters. In order to evaluate a candidate site, the Magnetometer Calibration procedure must be performed twice: once turning clockwise around the site, and once turning counter-clockwise. Both times, the procedure should be conducted as described in “GRS 77/GMU 44 Magnetometer Calibration (Procedure B)” on page 342501137 , with the exception of the direction of turns around the site.

NOTE: “GRS 77/GMU 44 Magnetometer Calibration (Procedure B)” on page 342501137 indicates that the Magnetometer Calibration procedure should be performed by making a series of clockwise turns around the site; however, the procedure can also be performed by making counterclockwise turns for the purpose of evaluating the site for magnetic disturbances.

If, upon completion of the Magnetometer Calibration procedure in each clockwise and counter-clockwise direction, the PFD displays the “CALIBRATION SUCCESSFUL / SITE IS CLEAN” message, then the candidate site is sufficiently free of magnetic disturbances and is acceptable for performing the Magnetometer Calibration procedure. It is important to obtain successful result in both the clockwise and counter-clockwise directions to ensure that the magnetometer sweeps over a large enough area at the candidate site.

If, upon completion of the Magnetometer Calibration procedure in either of the two directions, the PFD displays either the “MAG FIELD AT SITE NOT UNIFORM”, or “MAG FIELD AT SITE DIFFERS FROM IGRF MODEL” message, then the site contains magnetic disturbances that are too large.

(11) GRS 77/GMU 44 Magnetometer Calibration (Procedure B)

NOTE: If the GRS 77 was replaced with a new, repaired, or exchanged unit the "GRS 77 Pitch/Roll Offset Calibration (Procedure A)" on page 342501135 must first be successfully accomplished before performing Procedure B.

Calibration Procedure B must be carried out on a compass rose in order to guarantee measurements free of environmental magnetic disturbances. Attempting to carry out this maneuver on a typical ramp area may not yield a successful calibration. The accuracy of the AHRS cannot be guaranteed if this calibration is not performed on a magnetically clean compass rose. If the compass rose condition is not known, it is recommended that the technician follow the guidance in "Compass Rose Evaluation of Magnetic Disturbances for Magnetometer Calibration Procedure (Optional)" on page 342501136.

(a) Start the aircraft engine following the procedures referenced in the POH.

(b) After aircraft engine startup, taxi the aircraft to a properly calibrated compass rose.

CAUTION: CALIBRATION PROCEDURE B MUST BE CARRIED OUT ON A COMPASS ROSE IN ORDER TO GUARANTEE MEASUREMENTS FREE OF ENVIRONMENTAL MAGNETIC DISTURBANCES. ATTEMPTING TO CARRY OUT THIS MANEUVER ON A TYPICAL RAMP AREA MAY NOT YIELD A SUCCESSFUL CALIBRATION. THE ACCURACY OF THE AHRS CANNOT BE GUARANTEED IF THIS CALIBRATION IS NOT PERFORMED ON A MAGNETICALLY CLEAN COMPASS ROSE OR EQUIVALENT.

(c) At the compass rose, align the aircraft to a heading of magnetic north (±5°).

NOTE: This procedure provides instructions for calibrating both GRS 77 magnetometers simultaneously by putting both PFD1 and PFD2 in configuration mode and following the procedure below, using PFD1 to calibrate GRS 77 #1 and PFD2 to calibrate GRS 77 #2 (if installed).

(d) Start the PFDs and MFD in Configuration mode by holding the ENT key on each display while closing the circuit breaker.

(e) Go to the GRS Page Group on both PFDs by rotating the large FMS knob.

(f) Select the GRS/GMU Calibration page by rotating the small FMS knob. This page is protected and requires a keystroke password to perform this test. Press the following softkeys in sequence on each PFD:

1) 9
2) 10
3) 11
4) 12 (far right softkey)

(g) Activate the cursor by pressing the small FMS knob and highlight the SELECT GRS UNIT window. Select GRS 77 #1 ON PFD1 AND GRS 77 #2 (if installed) ON PFD2.

(h) Highlight the SELECT PROCEDURE window and select **MAGNETOMETER** on each PFD.

(i) Press the ENT key.

(j) Use the cursor to highlight the BEFORE CALIBRATION window on each PFD.

(k) Follow the checklist items displayed on each PFD and press the ENT key as each step is completed or confirmed.

(l) When the CALIBRATE field is blinking on both PFDs, press the ENT key on each PFD to begin the procedure.

(m) The PFD displays advise the operator when to turn the aircraft, when to stop, and when to turn again.

(n) Upon instruction to turn, taxi the aircraft in a right turn. After approximately 25° to 30° of turn from the last heading, the PFD displays advise the operator to stop the aircraft.

NOTE: Due to the difficulties in executing smooth, accurate turns, the PFDs may incorrectly interpret a station and instruct to “HOLD POSITION” prior to full completion of a 30° turn. If this scenario is encountered, it is best for the operator to ignore the “HOLD POSITION” command and instead use outside references to complete the approximate 30° turn. Instead of using the PFD instructions to turn as a real-time indication of when to turn, simply judge the 30° (±5°) turn increments of the aircraft by using the compass rose radials. Dwelling at these 30° increments for the time recommended by the PFDs should result in successful calibration.

(o) The PFDs guide the operator to dwell at multiple headings around a complete circle.

NOTE: Due to high winds or excessive airframe vibration, the operator may encounter a condition where a PFD restarts the 18-second countdown without full completion of the previous countdown. If this is encountered more than once for a given station, the operator should begin turning to the next station (approximately 30°). A minimum of 2 successful stations per quadrant (and for each GRS 77) is required, where a successful station is a full 18-second countdown followed by instruction to move. Ensure that at least 2 stations per quadrant are completed. Thus, it may sometimes be required to dwell at a station after a countdown restart. A maximum of 20 stations is allowed for the entire calibration procedure. If too many countdown restarts are encountered, the calibration will fail with the message, “TOO MANY STATIONS.”

(p) Repeat the turn-and-stop process until the PFDs advise that a successful calibration is complete. The GRS 77 AHRS UNITS then enter the normal operational mode. Press the ENT key on each PFD to conclude this procedure.

(12) Engine Run-Up Vibration Test (Procedure C)

NOTE: Calibration Procedure C is performed in order to guarantee that the AHRS mounting is sufficiently rigid and insensitive to vibration. This procedure must be performed for both GRS 77 units installed in the aircraft. Calibration Procedures A and B are not required prior to this procedure.

NOTE: This test must be performed after interior has been installed.

(a) Start the PFDs and MFD in Configuration mode by holding the ENT key on each display while closing the circuit breaker.

(b) Go to the GRS Page Group on both PFDs by rotating the large FMS knob.

(c) Select the GRS/GMU Calibration page by rotating the small FMS knob. This page is protected and requires a keystroke password to perform this test. Press the following softkeys in sequence on each PFD:

1) 9
2) 10
3) 11
4) 12 (far right softkey)

(d) Activate the cursor by pressing the small FMS knob and highlight the SELECT GRS UNIT window. Select GRS 77 #1 ON PFD1 AND GRS 77 #2 (if installed) ON PFD2.

(e) Highlight the SELECT PROCEDURE window on each PFD and select **ENGINE RUN-UP TEST**. Press ENT.

(f) Press the ENT key again on each PFD to select the first checklist item displayed.

(g) As each command is complied with and ENT is pressed, the associated box will turn green to confirm the selection has been complied with. When the CALIBRATE field is blinking on both PFDs, press the ENT key on each PFD to begin the procedure.

(h) The PFD displays instruct the operator to gradually increase power from idle to full throttle and back to idle over a period of 1–2 minutes.

(i) When the engine run-up is completed and the engine is back to an idle setting, press the ENT key on each PFD to indicate that the process is complete. When this is done, the TEST COMPLETE field stops blinking.

(j) Each PFD will state if the installation has passed or failed the vibration test. If the test fails, the specific measurements causing the failure are identified and associated numeric values are displayed on the PFD corresponding to each GRS 77 unit. Using the large FMS knob, scroll through the list and verify that each test passed.

NOTE: Should a failure occur, the technician may perform the Engine Run-up test up to three (3) times successively before corrective action must be taken. If the test does not pass after three attempts, then the installation should not be considered reliable until the source of the vibration problem is identified and remedied. In the event of repeated failure of the engine run-up test, record the values that are reported to be out of range for future reference.

The following are potential causes for failure of the engine run-up test:

1) Vibration motion of GRS 77 and/or GMU 44 caused by neighboring equipment and/or supports.
2) Mounting screws and other hardware for GRS 77 and/or GMU 44 not firmly attached.
3) GRS77 connector not firmly attached to unit.
4) Cabling leading to GRS 77 or GMU 44 not firmly secured to supporting structure.
5) An engine / propeller that is significantly out of balance.

(k) Press the ENT key on each PFD to conclude this procedure.

(13) Magnetometer Interference Test

A magnetometer interference test is available for troubleshooting and/or verifying a magnetically 'clean' installation of the GMU 44. This test exercises various devices on the aircraft that could potentially affect the magnetic field as measured by the GMU 44 (examples include navigation lights, control servos, landing gear motors, etc).

NOTE: This test is used to validate that no electronic device interferes with the operation of the GMU 44 magnetometer. It is highly recommended that this test be performed after installation or maintenance of electrical components on the aircraft and/or for troubleshooting the GMU 44.

NOTE: External power cart and other aircraft need to be as far away from the test aircraft as possible. Turn Battery Master ON, AV BUS Master ON, Autopilot ON (AP engaged to allow pitch servo movement.), and all circuit breakers in. Verify the Air Conditioning Condenser Blower Motor is OFF.

(a) Start the PFDs and MFD in the Configuration Mode by holding ENT on each display while closing the circuit breaker

(b) Go to the GRS Page Group on both PFDs by rotating the large FMS knob.

(c) Select the GRS/GMU Calibration page, as shown in "Figure 39" on page 342501141, by rotating the small FMS knob. This page is protected and requires a keystroke password to perform this test. Press the following softkeys in sequence on each PFD:

1) 9
2) 10
3) 11
4) 12 (far right softkey)

(d) Activate the cursor by pressing the small FMS knob and highlight the SELECT GRS UNIT window. Select GRS 77 #1 on PFD1 and GRS 77 #2 on PFD2. Highlight the SELECT PROCEDURE window and select MAGNETOMETER INTERFERENCE TEST on each PFD.

(e) Press ENT again to select the first checklist item displayed on each PFD. As each command is complied with and ENT is hit the associated box will turn green to confirm the selection has been complied with.

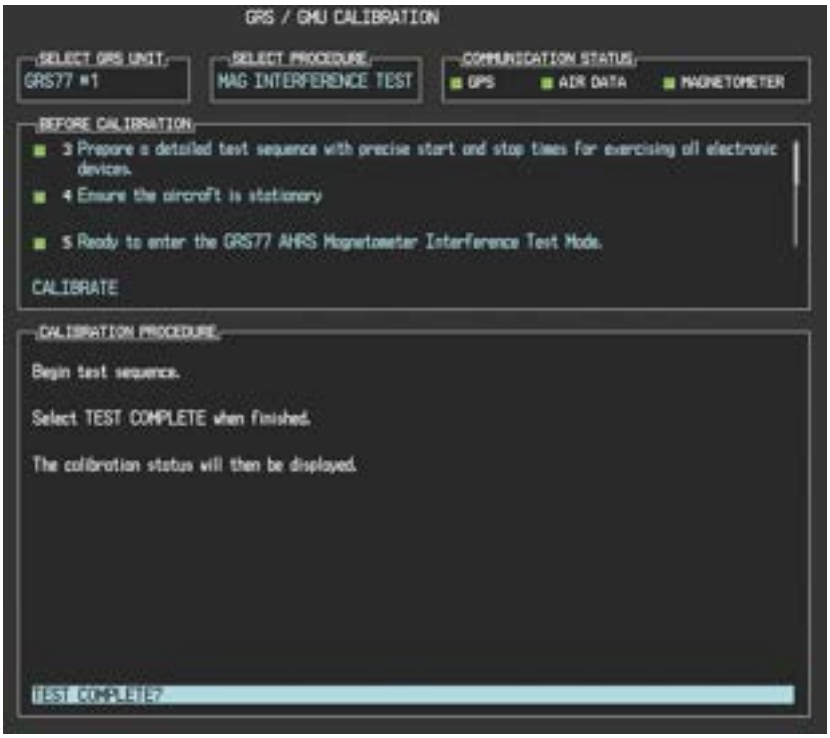
NOTE: Use a stopwatch to accurately time the tests. All actions are to be carried out in the order and at the precise elapsed time as specified in the prepared test sequence. MFD must be out of configuration mode prior to test being performed.

(f) When the CALIBRATE field is blinking on both PFDs, press the ENT keys to begin the procedure per "Chart 28" on page 342501141.

(g) When the test is complete press the ENT key on each PFD to indicate the process is complete and the TEST COMPLETE annunciation stops blinking at this time.

CHART 28
G1000 - MAGNETOMETER INTERFERENCE TEST

ELAPSED TIME Since start of test (Minutes:Seconds)	ACTION
0:00	Test Begins
0:10	Turn on strobe light
0:20	Turn off strobe light
0:30	Turn on fin strobe
0:40	Turn off fin strobe
0:50	Turn on NAV lights (if optional tail light installed)
1:00	Turn off NAV lights (if optional tail light installed)
1:10	Blower Motor on Low
1:20	Blower Motor off
1:30	Blower Motor on High
1:40	Air Conditioning Switch On
1:50	Blower Motor off & AC Switch off



Magnetometer Interference Test
Figure 39

(h) The PFD informs the operator if the installation has passed or failed the magnetometer interference test. If the test passes, no further action is required for this test. If the test fails, the installation should be considered unreliable until the source of magnetic interference is identified and remedied. When the magnetometer interference test fails, record the three magnetometer maximum deviation values and their corresponding timestamps. Any maximum deviation value greater than 2.5 milliGauss indicates a problem that must be resolved. Compare the corresponding timestamps with the prepared test sequence to identify which action produced the problem. Contact Garmin for assistance in resolving the problem.

NOTE: Two common reasons for a failed magnetometer interference test are:
1) New equipment is installed in close proximity to the GMU 44 magnetometer, and
2) An existing or new electronic device has become grounded through the aircraft structure instead of via the proper ground wire in a twisted shielded pair.

(i) Press ENT on each PFD to conclude the test.

(j) Press ENT on each PFD to conclude the test.

(14) GRS/GMU Test

The aircraft can now be taxied back and the engine can be shut down for final testing. Restart the displays in normal mode to conduct final system checks. When the PFDs power up in normal mode, the AHRS attitude and heading information displayed should become valid within one (1) minute of power-up, as shown in “Figure 40” (provided both GPS receivers have a valid position; if GPS is unavailable, AHRS initialization may take as long as 2 minutes).



Normal Mode AHRS Check
Figure 40

(15) GWX 68 Weather Radar Test (If installed)

NOTE: Before energizing the equipment, be sure microwave radiation safety precautions including both fuel and personnel safety considerations have been observed. These include clearing all personnel to an area beyond the maximum permissible exposure level (MPEL) boundary. The MPEL for the GWX 68 is 11 feet.

(a) Start the G1000 in the Normal mode.

(b) On the MFD, turn the large FMS knob to select the MAP Group on the MFD, and then turn the small FMS knob to select the WEATHER RADAR page.

(c) Select the MODE softkey, then select the STANDBY softkey to initiate the one minute warm-up period. Verify the radar enters the standby mode after the warm-up is complete.

(d) Press the MFD FMS knob to activate the cursor in the TILT field then turn the small FMS knob to select the desired antenna tilt angle. Press the MFD FMS knob to remove the cursor.

(e) Select the MODE softkey, then select the WEATHER softkey energize the radar array. Only after reading the CAUTION, and having followed the precautionary steps, select YES to continue. Press the BACK softkey to show weather radar scan and gain control softkeys.

(f) Select the GAIN softkey to activate the cursor in the ‘GAIN’ field. Turn the small FMS knob to adjust the gain for the desirable level then press ENTER on the MFD. Verify the gain setting is visible in the gain field as a movable horizontal bar in a flashing box with a line pointer reference depicting the calibrated position. Press the FMS knob to remove the cursor.

(g) Select the VERTICAL softkey, repeat item (f).

(h) Deselect the GAIN softkey. Verify ‘CALIBRATED’ is displayed in the ‘GAIN’ field.

(i) Select Horizon, Mode, and Ground softkeys sequentially. Verify scanning of and depiction of surrounding ground features. Verify Tilt, Bearing, Sector Scan, and Gain can be adjusted from the GROUND scan mode display via the FMS knob and RANGE knob, as appropriate.

(j) Select the MODE softkey, then select the STAB ON softkey to activate antenna stabilization. Verify that STAB ON is shown in the upper right corner of the weather radar display. Select the STAB OFF softkey to deactivate and verify that STAB OFF is shown in the upper right corner of the weather radar display.

(k) Select the OFF softkey at the conclusion of testing to turn off the weather radar.

(16) Stormscope (WX-500) Functional Check (If installed.)

See “Optional Equipment Configuration” on page 342501109, to configure Stormscope.

(a) Start the G1000 in the Normal mode.

(b) On the MFD, turn the large FMS knob to select the MAP group, then turn the small FMS knob to select the STORMSCOPE page.

(c) Apply power to the WX-500.

1) During initialization, the WX-500 stormscope runs a series of self-tests to ensure that all functions are operating properly. If no faults are detected, continue on to “Basic Test”.

2) If a fault is detected, consult the WX-500 installation manual for troubleshooting assistance.

(d) Basic Test

1) On the MFD, verify that it is possible to enter the following modes via softkeys: 360 degree, 120 degree, cell mode, strike mode.

2) Verify that the range can be changed from 25 nm to 100 nm via the range knob.

(e) Detailed Test

The following procedure requires access to and familiarity with the set-up and operation of the 3M WX-PA Portable Analyzer.

1) Following successful power up, verify the following modes:
   a) Access to both 3600 and 1200 weather view modes.
   b) All available ranges can be displayed.
   c) Access to the cell mode and strike mode.
   d) Strike counter is displayed in all weather modes and ranges.
2) Key COM1 and COM2 several times on different frequencies representing the lower, mid and upper portion of the VHF COM frequency band. Verify keying of COM1 or COM2 does not cause strike data to appear on the MFD.
3) Operate DME, XPDR1, XPDR2 and weather radar. Verify these systems do not cause strike data to appear on the MFD.
4) Connect the WX-PA cable to the WX-PA antenna.
5) Position the WX-PA antenna on the WX-500 antenna. Ensure the connection is tight. If necessary, use tape to secure the WX-PA antenna. Ensure the “FORWARD” arrows are aligned and the WX-PA antenna suction cups are positioned forward of center along the longitudinal axis.
6) Secure the WX-PA cable to the aircraft with the attached suction cup and route the cable to the cockpit.
7) Connect the remaining cable end to the WX-PA.
8) Power up the WX-PA and verify the WX-500 is in the weather mapping mode (i.e., 3600 weather view at the 200 NM range).
9) Set the WX-500 to STRIKE mode; 100 NM range (or next highest available range).
10) Select the Continuous Out mode displayed on the WX-PA menu and press MENU/ENTER.
11) Select the bottom mount antenna configuration on the WX-PA keyboard (“A” key).
12) Select a cardinal bearing and a range of 120 NM.
13) Use the F1 and F2 keys to adjust range and F3 and F4 keys to adjust heading.

    NOTE: The WX-500 will plot data at one-half the range selected on the WX-PA.

14) Press MENU/ENTER to start the test.
15) Observe the MFD to ensure the proper positioning of the strikes, based on range and azimuth settings on the WX-PA. Change the cardinal bearings and verify correct test strikes. Verify the strikes are within 10 degrees of the selected azimuth and plot at 60NM (1/2 120 NM).
16) Verify after 20 seconds of operation the strike counter reads 580 +/-40.
17) After testing for all ranges and bearings indicated, press 2ND then MENU/ENTER to return to the main menu.
18) Select Circular Pattern mode on the WX-PA mode menu and press MENU/ENTER.
19) Select bottom mount antenna configuration on the WX-PA keyboard (“A” key).

    NOTE: The MFD should be set at 100 NM (or next highest available range) on the 360 degree weather screen and in strike mode.

20) Use the F1 and F2 keys to select 120 NM range and press MENU/ENTER to start the test. Verify the WX-500 plots discharge points at approximately 60 NM.

21) Observe the MFD to ensure the proper positioning of the test strikes. Verify the strikes are within 10 degrees of the 30 degree azimuth increment and within 12 NM of 60 NM.

22) When complete, set WX-PA for 55 NM, set the MFD for 50 NM (or the next highest available range) and repeat test.

23) On the MFD, verify the sensor plots points just outside of 25 NM and the strikes are within 10 degrees of the 30 degree azimuth and within 5 NM of 27.5 NM.

24) Repeat above setting the MFD for 25 NM (or the next highest available range) and the WX-PA for 15 NM.

25) On the MFD, verify the positioning of test strikes are within 10 degrees of azimuth and within 2 NM of 7.5 NM.

26) Restart the G1000 in configuration mode by opening the PFD and MFD circuit breakers. While holding the ENT keys on the PFDs and MFD, restore power by closing the PFDs and MFD circuit breakers.

27) Select the OTHER page group on the MFD. The STORMSCOPE page is shown by default.

(17) GDL 69A Test (If installed)

(a) If an XM Satellite Radio subscription has already been activated for the GDL 69A, then power up the G1000 and go to the AUX – XM INFORMATION page and verify that the GDL 69A is working properly.

(b) GDL 69A Datalink Check

NOTE: This section verifies the correct installation of the GDL 69A.

1) On the MFD, select “AUX - XM RADIO”.

2) Run up and down the XM channels using the cabin controls. Select an active music channel when finished.

3) Increase and decrease the volume using the cabin controls, then set the volume at the mid-point.

4) If the GRT 10 is installed, repeat steps 2) and 3) using the GRC 10 remote control.

5) Verify XM audio operation at pilot, copilot and all passenger stations. Verify stereo operation.

6) Plug an external source into the AUX audio jack in the cabin and verify that it can be heard in all passenger stations. Verify that it is not audible via the pilot and copilot phone jacks.

7) Activate the Stall Warning horn and verify that the XM audio is muted.

(c) If the GDL 69A is replaced (new unit), the owner must re-activate the subscription, using the new GDL 69A’s Radio ID number(s). See “Chart 18” on page 34250179

(d) As a final operations check, make sure there are no MANIFEST errors shown on the PFD for the GDL 69A.

(18) GTS 825 TAS Check (If installed)

(a) Self-Test

1) Start the G1000 in the Normal mode and select the TRAFFIC MAP page on the MAP.

2) Verify that the STANDBY, OPERATE, TEST (in Standby mode) and ALT MODE softkeys are available on the bottom of the display. Verify that a TAS Status (not TAS FAIL) is displayed in the upper left corner of the traffic map. Verify that a yellow NO DATA message is not displayed in the center of the map. If TIS is displayed instead of TAS, refer to the "(3) Optional Equipment Configuration" on page 342501109 to install the GTS 825 option.

3) Press the OPERATE softkey and verify that OPERATING is displayed in the upper left corner of the traffic map.

4) Press the STANDBY softkey and verify that STANDBY is displayed in the upper left corner of the traffic map.

5) Set range to 6 NM or greater.

6) Press the TEST softkey and verigy that TEST is displayed in the upper left corner of the traffic map and that a traffic test pattern is displayed. Verigy that "TAS System Test Ok" annunciation is heard over the cockpit speaker at the conclusion of the test.

7) Open the TAS circuit breaker and verify that NO DATA is displayed at the center of the traffic map after several seconds.

8) Close the TAS circuit breaker and verify that NO DATA is removed from the traffic map.

(b) Operational Test

Perform the following test to verify GTS 825 operational and surveillance functions. Use a ramp tester such as a TIC TR220 or equivalent to perform the tests.

To select a scenario that will properly converge and intercept the GTS 825, the GTS 825 must be in ground test mode. To enable ground test mode, the aircraft must be on the ground and the GTS 825 must be in normal system mode and in standby.

Position the test set directional antenna with a clear line of sight to the GTS 825 antenna at 90 degrees. With the GTS 825 powered up and in Standby mode indicated on the CDTI, cycle the GTS 825 to 'Operate'.

1) Select the following:

a) Set the intruder type as ATCRBS.

b) Intruder Start Distance: 10 nm.

c) Intruder Start Altitude: 50,000 ft.

d) Vertical Speed: 0 fpm.

e) Velocity: 360 kts.

2) Initiate the intruder scenario and observe the following:

a) Traffic should be acquired at approximately 10 NM at 90 degree bearing and co-altitude. Observe intruder closes on own aircraft at a rate of 0.1 NM/sec.

b) The intruder should transition from Other Traffic (displayed as an open diamond with 00 displayed above), to proximate traffic (displayed as a filled white diamond with 00 displayed above), to a Traffic Advisory (TA) alarm.

c) The appropriate TA symbology (yellow filled circle with 00 displayed above, and an audio annunciation of "Traffic! 3 O'clock! At Altitude! 3 Miles!"), displayed when the intruder approaches within 3 NM.

(19) GSR 56 Checkout (If Installed)

This procedure verifies correct installation of the GSR 56, whether or not it has been activated. If the unit needs to be activated, see “GSR 56 Activation Procedure (If Installed)” on page 342501113.

(a) On the MFD navigate to AUX group, SATELLITE PHONE page using the FMS knobs.

(b) Press the DIAL softkey.

(c) Use the softkeys or FMS knobs to enter “1911135” and press ENT to confirm.

(d) Press ENT to call.

(e) If both parties can hear each other, then the installation has been verified.

1) If the unit has an active subscription, tell the Garmin representative that you are testing the GSR 56 installation and the unit should remain active.

2) If the unit has not been activated, tell the Garmin representative that you are testing the GSR 56 installation and that the unit can be deactivated.

a) If placed on hold when calling Garmin Flight Data Services, you may call anyone for a brief verification phone call. The cost for this phone call is charged to Garmin, so be brief.

b) Send an email flightdataservices@garmin.com after the brief phone call is complete so the GSR 56 unit can be turned off. Include the GSR 56 serial number in this email.

(20) ADF (Optional)

(a) Functional Check

This check verifies that the ADF / G1000 interface operates correctly.

1) Press the PFD softkey on PFD1 and PFD2. Toggle the BRG1 and BRG2 softkey until the ADF bearing is shown on PFD1 and PFD2.

2) Verify that the ADF window is not invalid (no red ‘X’).

3) Using the ADF control head select a known valid local ADF. Verify that the ADF bearing pointer moves towards and stabilizes on the correct bearing to that station.

4) Select ADF and SPKR on GMA audio panel. Using the ADF control head, select, ANT mode. Verify that the audio from the station tuned can be heard on the pilots and copilots headset and cockpit speaker.

5) Increment the ADF volume control from full low to full high. Verify the volume increases and decreases appropriately over pilots and copilots headset and cockpit speaker.

(b) Quadrantal Error Adjustment

1) Position aircraft on ramp in an area that is clear of surrounding buildings and other obstructions that may block or reflect radio signals.

2) Select ADF as the bearing source on the #1 PFD.

3) Select ADF mode on ADF receiver.

4) Tune the ADF to a known valid local ADF.

5) Orient aircraft such that the bearing indicator points to 0 degrees (aircraft is heading directly to station). Note the heading.

6) Using the heading indicator, rotate the aircraft 45 degrees to the left (counterclockwise). Note the indicated relative bearing and the amount of error.

7) Continue to turn the aircraft in 45 degree increments, noting the relative bearing error at each increment.

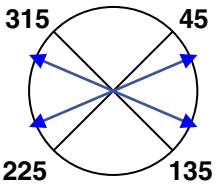
8) Verify that the relative bearing error at the cardinal points (90, 180, 270, 360) is within ±5 degrees.

NOTE: The following steps are not required if the quadrantal error is within tolerance.

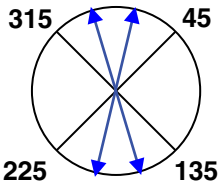
9) Average the errors at the quadrantal points (45, 135, 225, 315) to determine the amount of QE adjustment required.

NOTE: The QE adjustment potentiometer is located to the side of the antenna connector. The adjustment hole is protected by a threaded cap which must first be removed. If the antenna is inaccessible from the inside of the aircraft, the antenna will have to be unfastened and pulled away from the aircraft far enough to insert a jeweler's screwdriver into the adjustment hole. Take care of the O-ring seal beneath the adjustment screw cap. Following adjustment, the cap must be reinstalled to ensure that the electronics compartment is environmentally sealed.

10) Perform quadrantal error adjustment per the following figures. The QE adjustment potentiometer has a sensitivity of approximately 1 degree per turn.



Turn the QE adjustment pot clockwise



Turn the QE adjustment pot counterclockwise

11) Recheck the relative bearings and readjust the QE compensation pot as necessary to split the errors at the quadrantal points and obtain the lowest possible average error.

12) When QE adjustments are complete, reinstall the O-ring seal and the threaded cap in the adjustment hole to seal the antenna.

(21) DME (Optional) Functional Check

(a) Basic Test

This check verifies that the DME-to-G1000 NXi interface operates correctly, if check does not pass, conduct Detailed Test below. It requires appropriate DME Test Equipment.

1) Perform systems check and verify the DME is tuned by NAV 1 and NAV 2

2) Verify the distance is accurate.

3) Verify the DME audio is available on the GMA 1360 audio panel.

(b) Detailed Test

This check verifies that the DME-to-G1000 NXi interface operates correctly. It requires appropriate DME Test Equipment.

1) If DME is not displayed, press the PFD softkeys on PFD1 and PFD2, then press the DME softkey. Verify the DME window is displayed next to the PFD1 and PFD2 HSI.

2) On PFD1 and PFD2, press the BACK softkey then select DME softkeys. Verify the DME TUNING screen is displayed.

3) On PFD1 and PFD2, use the large FMS knob to select the NAV1, NAV2 and HOLD modes in the DME field. Verify that the NAV1, NAV2 and HOLD modes can be selected by turning the small FMS knob.

4) Set NAV1 and NAV2 frequencies to 108.00 and 117.00 respectively.

5) On PFD1 and PFD2, select the DME NAV1 mode. Verify that the DME window display is set to the NAV1 frequency of 108.00.

6) On PFD1 and PFD2, select the DME NAV2 mode by pressing the ENT softkey. Verify that the DME window display is set to the NAV2 frequency of 117.00.

7) On PFD1 and PFD2, select the DME HOLD mode by using the FMS knob. Verify that the last selected NAV frequency of 117.00 remains the same when the NAV2 frequency is changed. Deselect DME HOLD.

8) On the NAV Test Set, set up a DME test and note the nav frequency. Tune NAV1 to the test set frequency and set PFD1 and PFD2 DME MODE to NAV1. Set NAV2 to a frequency other than the test set frequency.

9) Verify that the DME distance on PFD1 and PFD2 match the test set.

10) Select, or verify selected, DME and SPKR buttons on GMA audio panel to select the DME audio and select speaker to ON. Verify that the DME audio can be heard over the cockpit speakers.

11) On PFD1 and PFD2, set the DME mode to NAV2 and verify that the DME distance is dashed out.

12) Tune NAV2 to the test set frequency. Set NAV1 to a frequency other than the test set frequency.

(22) ChartView (Optional) Check

ChartView, if installed, must be enabled, see “Chartview, Synthetic Vision, and TAWS Unlock Procedures” on page 342501113.

Ensure that a Chartview Jeppesen Nav Data U.S. database card is installed in the top card slot of the MFD.

NOTE: The required ChartView databases are subscription-based and are to be procured by the aircraft owner directly from Jeppesen.

(a) Select the Airport Information page (first page in the WPT group).

(b) Verify that the title at the top of the page reads “WPT- Airport Information”.

(c) Enter KVRB as the waypoint and verify that the CHRT softkey is available.

(d) Press the CHRT softkey and verify the electronic chart is viewable.

(23) Synthetic Vision (Optional) Check

(a) Ensure that matching terrain data base cards are installed in the bottom card slots in the PFD and MFD, and the aircraft has a GPS position.

(b) Press the PFD softkey on PFD1. Then press the SYN VIS softkey.

(c) Press the SYN TERR softkey.

NOTE: Synthetic Vision requires a valid GPS signal.

(d) Verify the traditional blue/brown attitude depiction is replaced with the Synthetic Vision rendering within 2–3 minutes.

(24) TAWS (Optional) Functional Check

(a) Ensure that matching terrain database cards are installed in the bottom slots of the PFD and MFD and that the aircraft has a GPS position.

(b) Select the TAWS-B page in the MAP Group.

(c) Verify that the title at the top of the page reads “MAP - “TAWS-B.”

NOTE: If TAWS has not been enabled, the title will read “MAP - TERRAIN PROXIMITY” or “MAP - TERRAIN.” See “Chartview, Synthetic Vision, and TAWS Unlock Procedures” on page 342501113.

(d) Press the MENU button and select “Test TAWS System” from the pop-up menu and press ENT key.

(e) After the TAWS test has completed, verify that “TAWS System Test Okay” is heard over the audio system.

(25) Terrain Proximity Check (if TAWS is not installed.)

(a) Ensure that matching terrain data base cards are installed in the bottom card slots in the PFD and MFD, and the aircraft has a GPS position.

(b) Select the TERRAIN PROXIMITY page (last page in the MAP group).

(c) Verify that the title at the top of the page reads “MAP - TERRAIN PROXIMITY”.

(d) Verify that the map is colored to show terrain and no failures are annunciated on the page.

(26) Automatic Flight Control System (AFCS) (S-TEC equipped aircraft only, If installed.)

See Garmin G1000 under S-TEC System 55 / 55X (Seneca V only.) - Post-Maintenance Operational Checkout in 22-10-00.

(27) NAV Source Selection Tests

NOTE: The CDI will not be depicted if the VOR antenna is not receiving a valid signal.

NOTE: For GPS, the word that is displayed with the GPS (i.e., GPS Term) depends on the phase of flight of the airplane. It is acceptable for GPS TERM or GPS ENR to be displayed.

(a) Start these tests with all avionics powered.

(b) Press the Direct-To-Key (--D->).

(c) Enter a Direct-To waypoint. Press ENT to accept waypoint.

(d) Enter an appropriate VOR frequency simulated with test equipment into the active NAV 1 frequency box on #1 PFD.

(e) Enter a different appropriate VOR frequency simulated with test equipment into the active NAV 2 frequency box on #1 PFD.

(f) On MFD, AUX Group, SYSTEM SETUP page, press the FMS knob to activate the cursor. Turn the large FMS knob to highlight CDI in the Synchronization window. Turn the small FMS knob to turn synchronization on. Press the FMS knob to deactivate the cursor.

(g) Verify that #1 PFD and #2 PFD synchronize (i.e. #1 PFD & #2 PFD match for the CDI).

(h) Press the CDI softkey on #1 PFD. Verify the NAV source on #1 PFD and #2 PFD changes from GPS TERM to VOR1.

(i) Press the CDI softkey on #1 PFD. Verify the NAV source on #1 PFD and #2 PFD changes from VOR1 TO VOR2.

(j) Press the CDI softkey on #1 PFD. Verify the NAV source on #1 PFD and #2 PFD changes from VOR2 TO GPS TERM.

(k) Repeat Step (f), but turn the synchronization off.

(l) By pressing the CDI softkey on #1 PFD, cycle the NAV source on #1 PFD from GPS TERM to VOR1 TO VOR2 TO GPS TERM. Verify that only the NAV source depicted on #1 PFD changes from GPS TERM to VOR1 TO VOR2 TO GPS TERM.

(m) By pressing the CDI softkey on #2 PFD, cycle the NAV source on #2 PFD from GPS TERM to VOR1 TO VOR2 TO GPS TERM. Verify that only the NAV source depicted on #2 PFD changes from GPS TERM to VOR1 TO VOR2 TO GPS TERM.

(28) Bearing Pointers and Information Windows Tests

NOTE: The bearing pointer will not be depicted if it is not receiving a valid signal.

(a) Start these tests with all avionics powered.

(b) Press the Direct-To-Key (--D->).

(c) Enter a Direct-To waypoint.

(d) Enter an appropriate VOR frequency simulated with test equipment into the NAV1 frequency box on #1 PFD.

(e) Enter a different appropriate VOR frequency simulated with test equipment into the NAV2 frequency box on #1 PFD.

(f) Enter a known local NDB frequency into the ADF receiver (if installed).

(g) Press the PFD softkey on #1 PFD to make the BRG1 and BRG2 softkey selections visible.

(h) Verify that the BRG1 softkey can be pressed to sequentially cycle through NAV1, GPS, and ADF (if installed).

(i) Verify that the BRG2 softkey can be pressed to sequentially cycle through NAV2, GPS, and ADF (if installed).

(j) Repeat Steps (g) thru (i) for PFD #2.

(29) BARO Selection Tests

(a) On MFD Aux Group, SYSTEM SETUP page, press the FMS knob to activate the cursor. Turn the large FMS knob to highlight BARO in the Synchronization window. Turn the small FMS knob to turn synchronization on. Press the FMS knob to deactivate the cursor.

(b) Rotate the BARO knob on the #1 PFD and verify that both PFD's BARO values change.

(c) Rotate the BARO knob on the #2 PFD and verify that both PFD's BARO values change.

(d) Set the current altimeter setting and verify that the both PFD's indicate within ±30' of current field elevation.

(e) On MFD Aux Group, SYSTEM SETUP page, press the FMS knob to activate the cursor. Turn the large FMS knob to highlight BARO in the Synchronization window. Turn the small FMS knob to turn synchronization off. Press the FMS knob to deactivate the cursor.

(f) Rotate the BARO knob on the #1 PFD and verify that only #1 PFD BARO values change and a delta indication of at least 0.01 in. Hg turns the indication yellow.

(g) Rotate the BARO knob on the #2 PFD and verify that only #2 PFD BARO values change and a delta indication of at least 0.01 in. Hg turns the indication yellow.

(h) On MFD Aux Group, SYSTEM SETUP page, press the FMS knob to activate the cursor. Turn the large FMS knob to highlight BARO in the Synchronization window. Turn the small FMS knob to turn synchronization on. Press the FMS knob to deactivate the cursor.

(30) Post-Interior Installation Checks

The following checks must be accomplished only after the interior has been installed.

(a) Audio Checks

1) Check Passenger ICS. See “GMA 1347 Test” on page 342501126.
2) If GDL 69A installed, perform “GDL 69A Datalink Check” on page 342501145.

(b) Lighting Tests

Increase and Decrease lights to ensure operation.

NOTE: A tarp must be used to cover the fuselage, and a flashlight is required to test photocell operation.

1) Courtesy Lights
   a) Right FWD Headliner
   b) Left AFT armrest switch
2) Reading Lights

   Passenger Switch panels (four)
3) Cockpit Dome Lights
4) Avionics dimmer switch controls the following:
   a) PFD #1 & #2
   b) MFD
   c) Audio Panel
5) Rotate the Avionics dimmer to full dim ensure the following switch to photocell:
   a) PFD #1 & #2
   b) MFD
6) Switch lights dimmer switch controls the following:
   a) Overhead Panel Switch Lights
   b) Instrument Panel Switch Lights
   c) AP/FD Switch (S-TEC equipped aircraft only - If installed)
   d) YD Switch (S-TEC equipped aircraft only - If installed)
   e) Air Cond Switch (If Installed)
   f) Pitot/Stall Warn Heat Switch
   g) Surface De-Ice Switch (If Installed)
   h) Prop Heat Switch (If Installed)
   i) Windshield Heat Switch (If Installed)
   j) Ice Light Switch (If Installed)
   k) Heat Fan Switch
   l) Defrost Switch
   m) ADF (If Installed)
   n) Autopilot Computer (S-TEC equipped aircraft only - If installed)
   o) Circuit Breaker Panel Lights
   p) Main Bus Circuit Breaker Lights
   q) Instrument panel lighted placards

7) Panel lights dimmer controls the following:
   a) Lights under glare shield
   b) Pedestal Light
8) With engine off and no external power applied to aircraft select Emergency Switch “ARM” then select Battery Master Switch and Avionics Switch “OFF”. The following items should remain lit:
   a) Aspen EBD Standby Instrument
   b) Master Caution/Warning Reset Switch Lights (if they are active)
   c) PFD #1
   d) Audio Panel
   e) All other flood lights and equipment lights extinguish.
   f) All Overhead Switch Panel and Instrument Panel switch lights extinguish.

(31) Emergency Bus Operation Verification

Verify all the following functions are operational when EMERGENCY switch is “ON” (in ARM position), Battery Master SWITCH is “OFF“, and Engine is “OFF”.

(a) PFD #1
(b) COM #1
(c) NAV #1
(d) AHRS #1
(e) Audio Panel #1
(f) Air Data Computer #1
(g) GEA 71 #1 and #2 (Engine Instruments)
(h) Standby Instrument
(i) Avionics Lights (i.e., Dimmer)

NOTE: MFD, PFD2, AHRS2, GDL 69A, Autopilot Controller (with S-TEC A/P only), Transponder #1, Air Data Computer 2, Radar, ADF, DME, GTS 825 and Stormscope are not available. Also note that NAV2, GPS2, COM2, Deice (optional), and O2 display invalid.

(32) Engine Run Checks

(a) Annunciation Verification

WARNING: UTILIZE ALL SAFETY PROCEDURES. NEVER WORK IN THE ENGINE AREA WHILE THE ENGINE IS RUNNING.

1) Remove fuses F102 and F202 in the engine compartments.
2) Start both engines.
3) With left engine running above 1100 RPM, turn on L ALTR switch. Verify L ALTR FAIL warning message. Reset L ALTR switch.
4) With right engine running above 1100 RPM, turn on R ALTR switch. Verify R ALTR FAIL warning message. Reset R ALTR switch.
5) Turn off engines.
6) Reinstall fuses F102 and F202.

(b) Various Systems Verification

1) Start PFD1 in config mode by holding ENT while applying power.

2) Using the FMS knob on the #1 PFD go to GIA I/O Configuration page on GIA group.

3) During left engine start, verify the data box next to DISCRETE IN 19A is green while left engine start switch is pressed.

4) During right engine start, verify the data box next to DISCRETE IN 20A is green while right engine start switch is pressed.

5) Verify the following with the engines ON:

a) Verify Hour Meter operation:

For aircraft with Hobbs Configuration A, run the engines. Hobbs shall run while engine is running.

b) Verify MAP, RPM, TIT, Fuel Flow, CHT, Oil Pressure, Oil Temp, Oxygen (if installed), and Vacuum (if installed) are operating in normal range.

c) Verify left alternator and right alternator current indications.

1] With the Battery Master Switch ON and left alternator and right alternator on line, turn the Battery Master and left alternator off.

2] Note right alternator current.

3] Turn the left alternator ON and right alternator off.

4] Note left alternator current.

5] Verify that current readings are equal (within +/- 3 amps).

O. Return to Service Procedure

CAUTION: ENTER "MAINTENANCE MODE" WHEN ACTIVATING THE GARMIN G1000 SYSTEM ON THE GROUND TO CONDUCT TESTS, INSPECTIONS, ETC. SEE MAINTENANCE MODE UNDER TROUBLESHOOTING, ABOVE.

(1) Software Verification

After reinstalling any G1000 LRU, use the checkoff list in "Chart 29" on page 342501164 to verify correct LRU software part numbers and versions against the current numbers and versions shown in the appropriate Required Equipment List (REL), "Chart 30" on page 342501165.

(a) Start the G1000 system in configuration mode.

(b) On MFD splash screen ("Figure 41") verify appropriate version "PA-34-220T System XXXX.XX" is displayed in the upper right corner of the MFD (where XXXX.XX is the system version number for the appropriate Required Equipment List (REL) in "Chart 30" on page 342501165).



MFD Splash Screen
Figure 41

(c) Select softkey on MFD to proceed beyond splash screen.

(d) Verify that no manifest (software mismatch) alerts are shown on any of the displays.

(e) Re-start the G1000 system in configuration mode.

(f) The PFD System Status Page (“Figure 42”) displays a list of LRUs in the LRU window.

1) Verify that the following are valid. (Green annunciations indicate that the system is healthy): MFD1, PFD1, PFD2, GIA1, GIA2, GDL 69 (optional), GTS (optional), and GWX (optional).

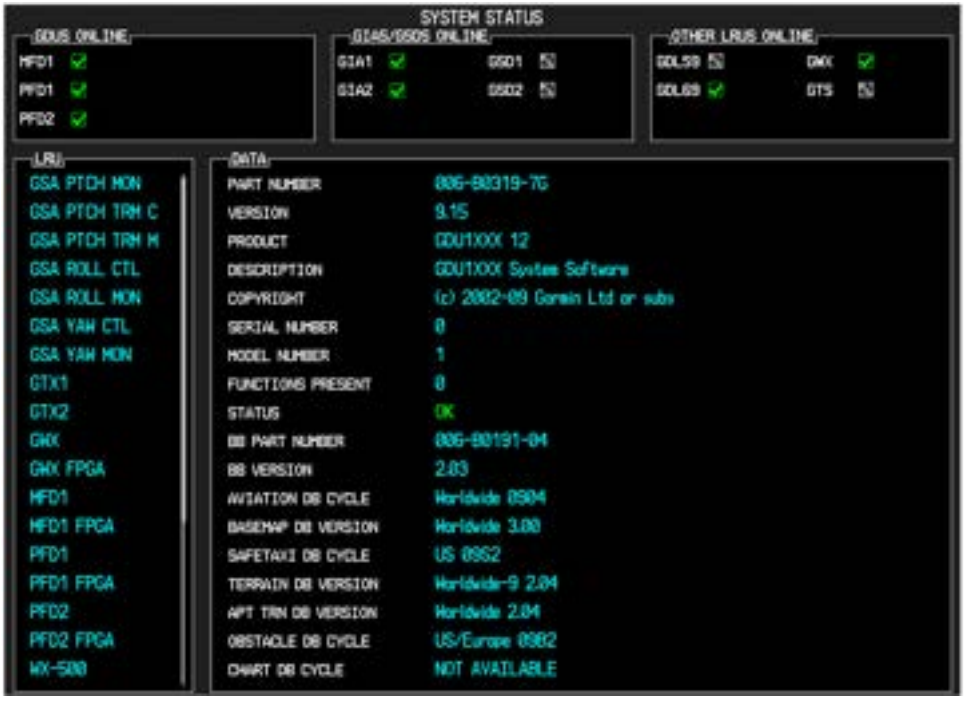
2) Rotate the small FMS knob to change to the “System Configuration Page” and verify that the following are valid. (Indicated by green annunciations): MFD1, PFD1, PFD2, GIA1, GIA2, GDL 69 (optional), GTS (optional), and GWX (optional).

3) Return to the “System Status Page.”

(g) Activate the cursor and highlight the LRU window.

(h) Use the FMS knob to scroll through the list, selecting each LRU in turn.

(i) The software part number and version is displayed in the DATA window. Compare this to the current numbers and versions shown in the appropriate Required Equipment List (REL).



System Status Page (Configuration Mode)
Figure 42

(2) GPS Failure Test

(a) Start these tests with all avionics powered.

(b) Valid GPS reception.

(c) Place a shroud over the GPS 1 antenna to prevent signal reception. Verify loss of GPS 1 signal on MFD AUX page 4. The following should remain valid on the PFD throughout the procedure.

- ✈ Attitude
- ✈ Heading
- ✈ Airspeed
- ✈ Altitude
- ✈ Vertical Speed
- ✈ OAT
- ✈ GPS CDI remains valid on PFD.

(d) Remove shroud from the GPS 1 antenna.

(e) Place a shroud over the GPS 2 antenna to prevent signal reception. Verify loss of GPS 2 signal on MFD AUX page 4. The following should remain valid on the PFD throughout the procedure.

- ✈ Attitude
- ✈ Heading
- ✈ Airspeed
- ✈ Altitude
- ✈ Vertical Speed
- ✈ OAT
- ✈ GPS CDI remains valid on PFD.

(f) Leave the shroud over the GPS 2 antenna and replace the shroud over the GPS 1 antenna. Verify loss of GPS 1 and 2 signals on MFD AUX page 4. The following should remain valid on the PFD throughout the procedure.

- ✈ Attitude
- ✈ Heading
- ✈ Airspeed
- ✈ Altitude
- ✈ Vertical Speed
- ✈ OAT
- ✈ Loss of GPS integrity annunciated on the PFD.

(3) GIA Failure Test

(a) Start these tests with all avionics powered.

(b) Valid GPS reception. Valid NAV signal.

(c) Remove power from GIA 1 by pulling the INTEG AV 1 and COM 1 circuit breakers.

(d) Verify the following system messages:

1) AHRS1 GPS – AHRS1 USING BACKUP GPS SOURCE
2) FAILED PATH
3) AHRS2 GPS – AHRS2 NOT RECEIVING BACKUP OF GPS INFORMATION (if installed)
4) ESP FAIL (If Enhanced AFCS installed)

(e) Verify the following flag invalid:

1) COM/NAV 1 field
2) NAV 1 CDI loses deviation bar.

(f) Verify BOTH ON GPS2 Sensor Reversion annunciation on the PFDs.

(g) Restore power to GIA1 by resetting the INTEG AV 1 and COM 1 circuit breakers.

(h) Verify that all invalid flags, system messages, and annunciations are removed. (#1 GPS will take time to reinitialize).

(i) Remove power from GIA 2 by pulling the INTEG AV 2 and COM 2 circuit breakers.

(j) Verify the following system messages:
1) AHRS2 GPS – AHRS2 USING BACKUP GPS SOURCE (if installed)
2) STRMSCP FAIL (if installed)
3) AHRS1 GPS – AHRS1 not receiving backup GPS information.
4) FAILED PATH
5) ESP FAIL (If Enhanced AFCS installed)

(k) Verify the following flag invalid:
1) COM/NAV 2 field
2) NAV 2 CDI loses deviation bar.
3) Stormscope (on MFD Stormscope Map page) via LIGHTNING FAILED yellow text (if installed)

(l) Verify BOTH ON GPS1 Sensor Reversion annunciation on the PFDs.

(m) Restore power to GIA2 by resetting the INTEG AV 2 and COM 2 circuit breakers.

(n) Verify that all invalid flags, system messages, and annunciations are removed. (#2 GPS will take time to reinitialize).

(o) Remove power from GIA 1 and GIA 2 by pulling the INTEG AV 1, INTEG AV 2, COM 1 and COM 2 circuit breakers.

(p) Verify the following system messages:
1) GPS NAV LOST
2) TRAFFIC FAIL
3) XPDR1 FAIL
4) GMA1 FAIL
5) TRN AUD FAIL (If TAWS is installed)

(q) Verify the following flags invalid:
1) COM/NAV 1 field
2) COM/NAV 2 field
3) XPDR field
4) Engine Instruments

(r) Verify the following:
1) All AHRS and ADC fields remain valid.
2) NAV 1 & NAV 2 CDI lose deviation bar.
3) “GEAR SYS” CAS warning

(4) Display Failure Test
(a) Start these tests with all avionics powered.
(b) Remove power from the MFD by pulling the MFD circuit breaker.
(c) Verify the following system messages on the PFDs:
1) XTALK ERROR system message.
2) GWX CONFIG FAIL system message (If Installed).
3) MFD1 VOLTAGE (This may or may not appear depending on power failure rate vs. MFD update rate).
4) FAILED PATH
5) ESP OFF (If Enhanced AFCS installed)

(d) Verify the following:
  1) PFDs switch to reversion mode.
  2) Attitude and Heading remain valid from AHRS.
  3) Airspeed, Altitude, Vertical Speed and OAT remain valid.
  4) Engine Instrumentation appears on PFDs.
  5) COM/NAV 2 fields flag invalid. (In installations with S-TEC A/P or no autopilot only.)

(e) Restore power to the MFD and deselect reversionary mode on GMA 1347 Audio Panel. The displays should return to Normal operation.

(f) Select reversionary mode on GMA 1347 Audio Panel.

(g) Remove power from the #1 PFD by pulling the #1 PFD circuit breaker.

(h) Verify the following system messages are displayed on #2 PFD and the MFD:
  1) XTALK ERROR
  2) FAILED PATH
  3) TRAFFIC FAIL (If GTS 825 installed)
  4) ESP FAIL (If Enhanced AFCS installed)

(i) Verify the following:
  1) MFD and PFD2 switch to reversion mode.
  2) Attitude and Heading remain valid from AHRS.
  3) Airspeed, Altitude, Vertical Speed and OAT remain valid.
  4) Reversionary sensor BOTH ON GPS2 is displayed.
  5) MFD retains engine instrumentation.
  6) COM/NAV 1 fields flag invalid.

(j) Restore power to the #1 PFD and deselect reversionary mode on GMA 1347 Audio Panel. The displays should return to normal operation.

(k) Select reversionary mode on GMA 1347 Audio Panel.

(l) Remove power from the #2 PFD by pulling the #2 PFD circuit breaker.

(m) Verify the following system messages are displayed on #1 PFD and the MFD:
  1) XTALK ERROR
  2) FAILED PATH
  3) STRMSCP FAIL (if installed)

(n) Verify the following:
  1) MFD and PFD1 switch to reversion mode.
  2) Attitude and Heading remain valid from AHRS.
  3) Airspeed, Altitude, Vertical Speed and OAT remain valid.
  4) MFD retains engine instrumentation.
  5) COM/NAV 2 fields flag invalid. (If GFC 700 installed)

(o) Restore power to the #2 PFD and deselect reversionary mode on GMA 1347 Audio Panel. The displays should return to normal operation.

(5) AHRS Failure Tests

(a) Single AHRS

NOTE: Allow for the AHRS to realign between tests.

1) Start these tests with all avionics powered.
2) Remove power from the #1 AHRS by pulling the #1 AHRS circuit breaker.
3) Verify Heading, Pitch, and Roll flag invalid on #1 PFD & #2 PFD.
4) Restore power to the #1 AHRS.
5) Verify normal operation and failure annunciations are removed.

(b) Dual AHRS (If Installed)

NOTE: Allow for the AHRS to realign between tests.

1) Start these tests with all avionics powered.
2) Remove power from the #1 AHRS by pulling the #1 AHRS circuit breaker.
3) Verify the following Comparator Annunciations are displayed on the PFDs:
   a) HDG NO COMP
   b) ROL NO COMP
   c) PIT NO COMP
4) Verify Heading, Pitch, and Roll flag invalid on #1 PFD.
5) Press the SENSOR key on #1 PFD.
6) Press the AHRS2 key on #1 PFD.
7) Verify the following:
   a) #2 AHRS parameters are displayed on #1 PFD without invalid flags.
   b) BOTH ON AHRS2 sensor reversion annunciation is displayed on the PFDs.
   c) White NO COMP Annunciations remain on the PFDs.
8) Restore power to the #1 AHRS.
9) Select AHRS1 key on #1 PFD.
10) Verify normal operation, failure and Comparator Annunciations are removed.
11) Repeat steps 2) thru 10) for the #2 AHRS and #2 PFD.

(c) Cross-Side AHRS Test (If Dual AHRS Installed)

12) Start this test with all avionics powered.
13) Press the Sensor key on the each PFD.
14) Press the AHRS2 Key on #1 PFD.
15) Press the AHRS1 Key on #2 PFD.
16) Verify the USING AHRS 2 annunciation on #1 PFD and USING AHRS 1 annunciation on #2 PFD.
17) Press the AHRS1 Key on #1 PFD.
18) Press the AHRS2 Key on #2 PFD.
19) Verify AHRS annunciations are removed.

(6) Air Data Computer (ADC) Failure Tests

(a) Single ADC

1) Start these tests with all avionics powered.
2) Remove power from the #1 ADC by pulling the #1 ADC circuit breaker.
3) Verify the following flag invalid #1 PFD & #2 PFD.
   a) Altitude
   b) Indicated Airspeed
   c) Vertical Speed
   d) Static Air Temperature (OAT) and ISA
4) Restore power to the #1 ADC.
5) Verify normal operation and invalid flags are removed.
6) Repeat steps 2) thru 5) for the #2 ADC and #2 PFD.

(b) Dual ADC (If Installed)

1) Start these tests with all avionics powered.
2) Remove power from the #1 ADC by pulling the #1 ADC circuit breaker.
3) Verify the following Comparator Annunciations are displayed on the PFDs:
   a) IAS NO COMP
   b) ALT NO COMP
4) Verify the following flag invalid #1 PFD. 4.
   a) Altitude
   b) Indicated Airspeed
   c) Vertical Speed
   d) Static Air Temperature (OAT) and ISA
5) Press the SENSOR key on #1 PFD.
6) Press the ADC2 key on #1 PFD.
7) Verify the following:
   a) #2 ADC parameters are displayed on #1 PFD without invalid flags.
   b) BOTH ON ADC2 sensor reversion annunciation is displayed on the PFDs.
   c) White NO COMP Annunciations remain on the PFDs.
8) Restore power to the #1 ADC.
9) Select ADC1 key on #1 PFD.
10) Verify normal operation, invalid flags and Comparator Annunciations are removed.
11) Repeat steps 2) thru 10) for the #2 ADC and #2 PFD.

(7) AHRS/ADC Backup Path Tests

(a) Single ADC/AHRS

1) Start these tests with all avionics powered.
2) Remove power from the #1 PFD by pulling the #1 PFD circuit breaker.
3) Press the SENSOR soft key and select ADC1 and AHRS1. This test verifies the secondary AHRS/ADC 1 paths. The following shall occur during this test:
   a) Attitude, Heading, Airspeed, Altitude, Vertical Speed and OAT remain valid.
   b) Engine Instrumentation remains valid.
   c) COM/NAV 1 fields flag invalid.
4) Restore power to the #1 PFD.
5) Remove power from the MFD and GIA 1 by pulling the MFD and INTEG AV 1 circuit breakers.
6) Press the SENSOR soft key and select ADC1 and AHRS1. This test verifies the primary AHRS/ADC 1 paths. The following shall occur during this test:
   a) Attitude, Heading, Airspeed, Altitude, Vertical Speed and OAT remain valid.
   b) COM/NAV1 and COM/NAV2 fields flag invalid.
7) All engine instruments flag invalid.
8) Restore power to the MFD and GIA 1.

(b) Dual ADC/AHRS (If Installed)

1) Start these tests with all avionics powered.
2) Remove power from the #1 PFD by pulling the #1 PFD circuit breaker.
3) Press the SENSOR softkey and select ADC 1 and AHRS 1. This test verifies the secondary AHRS/ADC 1 paths. The following shall occur during this test:
   a) Attitude, Heading, Airspeed, Altitude, Vertical Speed and OAT remain valid.
   b) Engine Instrumentation remains valid.
   c) COM/NAV 1 fields flag invalid.
4) Restore power to the #1 PFD.
5) Remove power from the #2 PFD and GIA 1 by pulling the PFD 2 and INTEG AV 1 circuit breakers.
6) Press the SENSOR softkey and select ADC 1 and AHRS 1. This test verifies the primary AHRS/ADC 1 paths. The following shall occur during this test:
   a) Attitude, Heading, Airspeed, Altitude, Vertical Speed and OAT remain valid.
   b) COM/NAV1 and COM/NAV2 fields flag invalid.
7) All engine instruments flag invalid.
8) Restore power to the #2 PFD and GIA 1.

(8) Flight Test

A flight test is recommended after the installation is complete to ensure satisfactory performance of the G1000.

(9) VHF Comm Tests

Verify the communications capability on both the high and low ends of the VHF COM spectrum.

(10) VOR/ILS Tests

Select a VOR channel within a 40 nautical mile range. Listen to the VOR audio and verify that no electrical interference such as magneto noise is present. Check the tone identifier filter operation. Fly inbound or outbound on a selected VOR radial and check for proper LEFT / RIGHT, TO / FROM flag indications on the CDI. Check the VOR accuracy.

(11) Maintenance Records

After conducting the function check flight, the airplane may be returned to service. Record the following information in the appropriate airplane maintenance logs:

(a) Part number of the loader card used to perform software loading or software updates.

(b) Any other applicable information related to the maintenance work performed on the airplane.

CHART 29
G1000 - RETURN TO SERVICE CHECK LIST

Group System	SW Ver. OK	System	SW Ver. OK	System	SW Ver. OK
Control & Display:					
PFD 1	_____	PFD 2 FPGA	_____		
PFD 1 FPGA	_____	MFD1	_____		
PFD 2	_____	MFD1 FPGA	_____		
CNS:					
GIA1	_____	GPS2	_____	GDL69	_____
GIA2	_____	GTX1	_____	GWX	_____
GMA1	_____	GIA1 AUDIO	_____	GWX FPGA	_____
GPS1	_____	GIA2 AUDIO	_____	GTS	_____
Sensors:					
GDC1	_____	GMU2	_____	GEA1	_____
GDC1 FPGA	_____	GMU2 FPGA	_____	GEA2	_____
GDC2	_____	GRS1	_____		
GDC2 FPGA	_____	GRS1 FPGA	_____		
GMU1	_____	GRS2	_____		
GMU1 FPGA	_____	GRS2 FPGA	_____		

CHART 30
REQUIRED EQUIPMENT LISTS (REL)

G1000 VER. 1578.02 (Sheet 1 of 2)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
COM1	006-B0081-xx	11.00	
COM2	006-B0081-xx	11.00	
GDC1	006-B0261-19	3.09	
GDC1 FPGA	006-C0055-00	01.05	
GDC2	006-B0261-19	3.09	(Verify if installed)
GDC2 FPGA	006-C0055-00	01.05	(Verify if installed)
GDL69	006-B0317-22	4.02.00	(Verify if installed)
GEA1	006-B0193-05	2.07	
GEA2	006-B0193-05	2.07	
GIA1	006-B0544-52	7.08	
GIA1 AUDIO	006-D4432-00	2.00	
GIA2	006-B0544-52	7.08	
GIA2 AUDIO	006-D4432-00	2.00	
GMA1	006-B0203-44	4.04	
GMU1	006-B0224-01	2.05	
GMU1 FPGA	006-C0048-00	2.00	
GMU2	006-B0224-01	2.05	(Verify if installed)
GMU2 FPGA	006-C0048-00	2.00	(Verify if installed)
GPS1	006-B0339-20	5.0	
GPS2	006-B0339-20	5.0	
GRS1	006-B0223-23	3.03	
GRS1 FPGA	006-C0049-00	02.00	
GRS2	006-B0223-23	3.03	(Verify if installed)
GRS2 FPGA	006-C0049-00	02.00	(Verify if installed)
GS1	006-B0082-xx	6.02	
GS2	006-B0082-xx	6.02	
GTS	006-B1244-03	2.00	(Verify if installed)
GTS AUDIO	006-D0726-50	2.00	(Verify if installed)
GTS FPGA	006-C0081-20	2.0	(Verify if installed)
GTS MAGNET V	006-D0159-02	2010.00	(Verify if installed)

CHART 30
REQUIRED EQUIPMENT LISTS (REL)

G1000 VER. 1578.02 (Sheet 2 of 2)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version
GTX1	006-B0172-xx	7.02
GWX	006-B0266-11	2.12
GWX FPGA	006-B0042-06	3.01
MFD1	006-B0319-B1	13.01
MFD1 FPGA	006-C0036-04	1.04
NAV1	006-B0082-12	6.02
NAV1 BB	006-B0082-BC	2.01
NAV1 FPGA	006-C0124-00	1.0
NAV2	006-B0082-12	6.02
NAV2 BB	006-B0082-BC	2.01
NAV2 FPGA	006-C0124-00	1.0
PFD1	006-B0319-B1	13.01
PFD1 FPGA	006-C0036-04	1.04
PFD2	006-B0319-B1	13.01
PFD2 FPGA	006-C0036-04	1.04
LOADER IMAGE	006-B1578-02	1578.02 (1-2)

(1) See “Figure 41” on page 342501155.

(2) PIR-107399-002, Rev. B. Factory installed in S/N’s 3449459, 3449467 and up.

CHART 30
REQUIRED EQUIPMENT LISTS (REL)

G1000 VER. 1578.03 (Sheet 1 of 2)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
COM1	006-B0081-xx	11.00	
COM2	006-B0081-xx	11.00	
GDC1	006-B0261-19	3.09	
GDC1 FPGA	006-C0055-00	01.05	
GDC2	006-B0261-19	3.09	(Verify if installed)
GDC2 FPGA	006-C0055-00	01.05	(Verify if installed)
GDL69	006-B0317-22	4.02.00	(Verify if installed)
GEA1	006-B0193-05	2.07	
GEA2	006-B0193-05	2.07	
GIA1	006-B0544-52	7.08	
GIA1 AUDIO	006-D4432-00	2.00	
GIA2	006-B0544-52	7.08	
GIA2 AUDIO	006-D4432-00	2.00	
GMA1	006-B0203-44	4.04	
GMU1	006-B0224-01	2.05	
GMU1 FPGA	006-C0048-00	2.00	
GMU2	006-B0224-01	2.05	(Verify if installed)
GMU2 FPGA	006-C0048-00	2.00	(Verify if installed)
GPS1	006-B0339-20	5.0	
GPS2	006-B0339-20	5.0	
GRS1	006-B0223-23	3.03	
GRS1 FPGA	006-C0049-00	02.00	
GRS2	006-B0223-23	3.03	(Verify if installed)
GRS2 FPGA	006-C0049-00	02.00	(Verify if installed)
GS1	006-B0082-xx	6.02	
GS2	006-B0082-xx	6.02	
GTS	006-B1244-03	2.00	(Verify if installed)
GTS AUDIO	006-D0726-50	2.00	(Verify if installed)
GTS FPGA	006-C0081-20	2.0	(Verify if installed)
GTS MAGNET V	006-D0159-02	2010.00	(Verify if installed)

CHART 30
REQUIRED EQUIPMENT LISTS (REL)

G1000 VER. 1578.03 (Sheet 2 of 2)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version
GTX1	006-B0172-xx	7.02
GWX	006-B0266-11	2.12
GWX FPGA	006-B0042-06	3.01
MFD1	006-B0319-B3	13.03
MFD1 FPGA	006-C0036-04	1.04
NAV1	006-B0082-12	6.02
NAV1 BB	006-B0082-BC	2.01
NAV1 FPGA	006-C0124-00	1.0
NAV2	006-B0082-12	6.02
NAV2 BB	006-B0082-BC	2.01
NAV2 FPGA	006-C0124-00	1.0
PFD1	006-B0319-B3	13.03
PFD1 FPGA	006-C0036-04	1.04
PFD2	006-B0319-B3	13.03
PFD2 FPGA	006-C0036-04	1.04
LOADER IMAGE	006-B1578-03	1578.03 (1-2)

(1) See “Figure 41” on page 342501155.
(2) PIR-107399-002, Rev. C. Factory installed in S/N’s 3449479 and up.

CHART 30
REQUIRED EQUIPMENT LISTS (REL)

G1000 VER. 1578.05 (Sheet 1 of 2)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
COM1	006-B0081-XX	11.10	
COM2	006-B0081-XX	11.10	
GDC1	006-B0261-19	3.09	
GDC1 FPGA	006-C0055-00	01.05	
GDC2	006-B0261-19	3.09	Optional, Verify If Installed
GDC2 FPGA	006-C0055-00	01.05	Optional, Verify If Installed
GDL69	006-B0317-22	4.02.00	Optional, Verify If Installed
GEA1	006-B0193-05	2.07	
GEA2	006-B0193-05	2.07	
GIA1	006-B0544-4N	7.14	
GIA1 AUDIO	006-D4432-00	2.00	
GIA2	006-B0544-4N	7.14	
GIA2 AUDIO	006-D4432 -00	2.00	
GMA1	006-B0203-44	4.04	
GMU1	006-B0224-01	2.05	
GMU1 FPGA	006-C0048-00	2.00	
GMU2	006-B0224-01	2.05	Optional, Verify If Installed
GMU2 FPGA	006-C0048-00	2.00	Optional, Verify If Installed
GPS1	006-B0339-20	5.0	
GPS2	006-B0339-20	5.0	
GRS1	006-B0223-23	3.03	
GRS1 FPGA	006-C0049-00	02.00	
GRS1 MV DB	006-D0159-02	2010.00	
GRS2	006-B0223-23	3.03	Optional, Verify If Installed
GRS2 FPGA	006-C0049-00	02.00	Optional, Verify If Installed
GRS2 MV DB	006-D0159-02	2010.00	Optional, Verify If Installed
GS1	006-B0082-XX	6.02	
GS2	006-B0082-XX	6.02	

CHART 30
REQUIRED EQUIPMENT LISTS (REL)

G1000 VER. 1578.05 (Sheet 2 of 2)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
GTS	006-B1244-07	2.22	Optional, Verify If Installed
GTS AUDIO	006-D0726-50	2.00	Optional, Verify If Installed
GTS FPGA	006-C0081-20	2.0	Optional, Verify If Installed
GTS MAGNET V	006-D0159-02	2010.00	Optional, Verify If Installed
GTX1	006-B0172-XX	7.04	
GWX	006-B0266-11	2.12	
GWX FPGA	006-C0042-06	3.01	
MFD1	006-B0319-B9	13.08	
MFD1 FPGA	006-C0036-04	1.04	
NAV1	006-B0082-12	6.02	
NAV1 BB	006-B0082-BC	2.01	
NAV1 FPGA	006-C0124-00	1.0	
NAV2	006-B0082-12	6.02	
NAV2 BB	006-B0082-BC	2.01	
NAV2 FPGA	006-C0124-00	1.0	
PFD1	006-B0319-B9	13.08	
PFD1 FPGA	006-C0036-04	1.04	
PFD2	006-B0319-B9	13.08	
PFD2 FPGA	006-C0036-04	1.04	
PA-34 IMAGE	006-B1578-05	1578.05 (1-2)	

(1) See “Figure 41” on page 342501155.
(2) PIR-107399-002, Rev. H.

CHART 30
REQUIRED EQUIPMENT LISTS (REL)

G1000 VER. 1981.00 (Sheet 1 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
COM1	006-B0081-xx	11.00	
COM2	006-B0081-xx	11.00	
GDC1	006-B0261-19	3.09	
GDC1 FPGA	006-C0055-00	01.05	
GDC2	006-B0261-19	3.09	(Verify if installed)
GDC2 FPGA	006-C0055-00	01.05	(Verify if installed)
GDL69	006-B0317-22	4.02.00	(Verify if installed)
GEA1	006-B0193-05	2.07	
GEA2	006-B0193-05	2.07	
GIA1	006-B0544-4H	7.13	
GIA1 AUDIO	006-D4432-00	2.00	
GIA2	006-B0544-4H	7.13	
GIA2 AUDIO	006-D4432-00	2.00	
GMA1	006-B0203-44	4.04	
GMU1	006-B0224-01	2.05	
GMU1 FPGA	006-C0048-00	2.00	
GMU2	006-B0224-01	2.05	(Verify if installed)
GMU2 FPGA	006-C0048-00	2.00	(Verify if installed)
GPS1	006-B0339-20	5.0	
GPS2	006-B0339-20	5.0	
GRS1	006-B0223-23	3.03	
GRS1 FPGA	006-C0049-00	02.00	
GRS2	006-B0223-23	3.03	(Verify if installed)
GRS2 FPGA	006-C0049-00	02.00	(Verify if installed)
GS1	006-B0082-xx	6.02	
GS2	006-B0082-xx	6.02	
GSA PTCH CTL	006-B0398-37	3.30	(Verify if installed)
GFC CERT P C	006-D4801-00	2.00	(Verify if installed)
GSA PTCH MON	006-B0398-37	3.30	(Verify if installed)
GFC CERT P M	006-D4801-00	2.00	(Verify if installed)

CHART 30
REQUIRED EQUIPMENT LISTS (REL)

G1000 VER. 1981.00 (Sheet 2 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
GSA PTCH TRIM C	006-B0398-37	3.30	(Verify if installed)
GFC CERT PT C	006-D4801-00	2.00	(Verify if installed)
GSA PTCH TRIM M	006-B0398-37	3.30	(Verify if installed)
GFC CERT PT M	006-D4801-00	2.00	(Verify if installed)
GSA ROLL CTL	006-B0398-37	3.30	(Verify if installed)
GFC CERT R C	006-D4801-00	2.00	(Verify if installed)
GSA ROLL MON	006-B0398-37	3.30	(Verify if installed)
GFC CERT R M	006-D4801-00	2.00	(Verify if installed)
GSA YAW CTL	006-B0398-37	3.30	(Verify if installed)
GFC CERT Y C	006-D4801-00	2.00	(Verify if installed)
GSA YAW MON	006-B0398-37	3.30	(Verify if installed)
GFC CERT Y M	006-D4801-00	2.00	(Verify if installed)
GTS	006-B1244-03	2.00	(Verify if installed)
GTS AUDIO	006-D0726-50	2.00	(Verify if installed)
GTS FPGA	006-C0081-20	2.0	(Verify if installed)
GTS MAGNET V	006-D0159-02	2010.00	(Verify if installed)
GTX1	006-B0172-xx	7.02	
GWX	006-B0266-11	2.12	
GWX FPGA	006-B0042-06	3.01	
MFD1	006-B0319-B7	13.06	
MFD1 FPGA	006-C0036-04	1.04	
NAV1	006-B0082-12	6.02	
NAV1 BB	006-B0082-BC	2.01	
NAV1 FPGA	006-C0124-00	1.0	
NAV2	006-B0082-12	6.02	
NAV2 BB	006-B0082-BC	2.01	
NAV2 FPGA	006-C0124-00	1.0	
PFD1	006-B0319-B7	13.06	
PFD1 FPGA	006-C0036-04	1.04	

CHART 30
REQUIRED EQUIPMENT LISTS (REL)

G1000 VER. 1981.00 (Sheet 3 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version
PFD2	006-B0319-B7	13.06
PFD2 FPGA	006-C0036-04	1.04
LOADER IMAGE	006-B1981-00	1981.00 (1-2)

(1) See “Figure 41” on page 342501155.
(2) PIR-107399-003, Rev. E.

CHART 30
REQUIRED EQUIPMENT LISTS (REL)

G1000 VER. 1981.01 (Sheet 1 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
COM1	006-B0081-xx	11.00	
COM2	006-B0081-xx	11.00	
GDC1	006-B0261-19	3.09	
GDC1 FPGA	006-C0055-00	01.05	
GDC2	006-B0261-19	3.09	(Verify if installed)
GDC2 FPGA	006-C0055-00	01.05	(Verify if installed)
GDL69	006-B0317-22	4.02.00	(Verify if installed)
GEA1	006-B0193-05	2.07	
GEA2	006-B0193-05	2.07	
GIA1	006-B0544-4H	7.13	
GIA1 AUDIO	006-D4432-00	2.00	
GIA2	006-B0544-4H	7.13	
GIA2 AUDIO	006-D4432-00	2.00	
GMA1	006-B0203-44	4.04	
GMU1	006-B0224-01	2.05	
GMU1 FPGA	006-C0048-00	2.00	
GMU2	006-B0224-01	2.05	(Verify if installed)
GMU2 FPGA	006-C0048-00	2.00	(Verify if installed)
GPS1	006-B0339-20	5.0	
GPS2	006-B0339-20	5.0	
GRS1	006-B0223-23	3.03	
GRS1 FPGA	006-C0049-00	02.00	
GRS2	006-B0223-23	3.03	(Verify if installed)
GRS2 FPGA	006-C0049-00	02.00	(Verify if installed)
GS1	006-B0082-xx	6.02	
GS2	006-B0082-xx	6.02	
GSA PTCH CTL	006-B0398-37	3.30	(Verify if installed)
GFC CERT P C	006-D4801-00	2.00	(Verify if installed)
GSA PTCH MON	006-B0398-37	3.30	(Verify if installed)
GFC CERT P M	006-D4801-00	2.00	(Verify if installed)

CHART 30
REQUIRED EQUIPMENT LISTS (REL)

G1000 VER. 1981.01 (Sheet 2 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
GSA PTCH TRIM C	006-B0398-37	3.30	(Verify if installed)
GFC CERT PT C	006-D4801-00	2.00	(Verify if installed)
GSA PTCH TRIM M	006-B0398-37	3.30	(Verify if installed)
GFC CERT PT M	006-D4801-00	2.00	(Verify if installed)
GSA ROLL CTL	006-B0398-37	3.30	(Verify if installed)
GFC CERT R C	006-D4801-00	2.00	(Verify if installed)
GSA ROLL MON	006-B0398-37	3.30	(Verify if installed)
GFC CERT R M	006-D4801-00	2.00	(Verify if installed)
GSA YAW CTL	006-B0398-37	3.30	(Verify if installed)
GFC CERT Y C	006-D4801-00	2.00	(Verify if installed)
GSA YAW MON	006-B0398-37	3.30	(Verify if installed)
GFC CERT Y M	006-D4801-00	2.00	(Verify if installed)
GTS	006-B1244-03	2.00	(Verify if installed)
GTS AUDIO	006-D0726-50	2.00	(Verify if installed)
GTS FPGA	006-C0081-20	2.0	(Verify if installed)
GTS MAGNET V	006-D0159-02	2010.00	(Verify if installed)
GTX1	006-B0172-xx	7.02	
GWX	006-B0266-11	2.12	
GWX FPGA	006-B0042-06	3.01	
MFD1	006-B0319-B7	13.06	
MFD1 FPGA	006-C0036-04	1.04	
MFD1 BOOT	006-B0191-04	2.03	
NAV1	006-B0082-12	6.02	
NAV1 BB	006-B0082-BC	2.01	
NAV1 FPGA	006-C0124-00	1.0	
NAV2	006-B0082-12	6.02	
NAV2 BB	006-B0082-BC	2.01	
NAV2 FPGA	006-C0124-00	1.0	

CHART 30
REQUIRED EQUIPMENT LISTS (REL)

G1000 VER. 1981.01 (Sheet 3 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version
PFD1	006-B0319-B7	13.06
PFD1 FPGA	006-C0036-04	1.04
PFD1 BOOT	006-B0191-04	2.03
PFD2	006-B0319-B7	13.06
PFD2 FPGA	006-C0036-04	1.04
PFD2 BOOT	006-B0191-04	2.03
LOADER IMAGE	006-B1981-01	1981.01 (1-2)

(1) See “Figure 41” on page 342501155.
(2) PIR-107399-003, Rev. F.

CHART 30
REQUIRED EQUIPMENT LISTS (REL)

G1000 VER. 1981.02 (Sheet 1 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
COM1	006-B0081-xx	11.00	
COM2	006-B0081-xx	11.00	
GDC1	006-B0261-19	3.09	
GDC1 FPGA	006-C0055-00	01.05	
GDC2	006-B0261-19	3.09	(Verify if installed)
GDC2 FPGA	006-C0055-00	01.05	(Verify if installed)
GDL69	006-B0317-22	4.02.00	(Verify if installed)
GEA1	006-B0193-05	2.07	
GEA2	006-B0193-05	2.07	
GIA1	006-B0544-4H	7.13	
GIA1 AUDIO	006-D4432-00	2.00	
GIA2	006-B0544-4H	7.13	
GIA2 AUDIO	006-D4432-00	2.00	
GMA1	006-B0203-44	4.04	
GMU1	006-B0224-01	2.05	
GMU1 FPGA	006-C0048-00	2.00	
GMU2	006-B0224-01	2.05	(Verify if installed)
GMU2 FPGA	006-C0048-00	2.00	(Verify if installed)
GPS1	006-B0339-20	5.0	
GPS2	006-B0339-20	5.0	
GRS1	006-B0223-23	3.03	
GRS1 FPGA	006-C0049-00	02.00	
GRS2	006-B0223-23	3.03	(Verify if installed)
GRS2 FPGA	006-C0049-00	02.00	(Verify if installed)
GS1	006-B0082-xx	6.02	
GS2	006-B0082-xx	6.02	
GSA PTCH CTL	006-B0398-37	3.30	(Verify if installed)
GFC CERT P C	006-D4801-00	2.00	(Verify if installed)
GSA PTCH MON	006-B0398-37	3.30	(Verify if installed)
GFC CERT P M	006-D4801-00	2.00	(Verify if installed)

CHART 30
REQUIRED EQUIPMENT LISTS (REL)

G1000 VER. 1981.02 (Sheet 2 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
GSA PTCH TRIM C	006-B0398-37	3.30	(Verify if installed)
GFC CERT PT C	006-D4801-00	2.00	(Verify if installed)
GSA PTCH TRIM M	006-B0398-37	3.30	(Verify if installed)
GFC CERT PT M	006-D4801-00	2.00	(Verify if installed)
GSA ROLL CTL	006-B0398-37	3.30	(Verify if installed)
GFC CERT R C	006-D4801-00	2.00	(Verify if installed)
GSA ROLL MON	006-B0398-37	3.30	(Verify if installed)
GFC CERT R M	006-D4801-00	2.00	(Verify if installed)
GSA YAW CTL	006-B0398-37	3.30	(Verify if installed)
GFC CERT Y C	006-D4801-00	2.00	(Verify if installed)
GSA YAW MON	006-B0398-37	3.30	(Verify if installed)
GFC CERT Y M	006-D4801-00	2.00	(Verify if installed)
GTS	006-B1244-03	2.00	(Verify if installed)
GTS AUDIO	006-D0726-50	2.00	(Verify if installed)
GTS FPGA	006-C0081-20	2.0	(Verify if installed)
GTS MAGNET V	006-D0159-02	2010.00	(Verify if installed)
GTX1	006-B0172-xx	7.02	
GWX	006-B0266-11	2.12	
GWX FPGA	006-B0042-06	3.01	
MFD1	006-B0319-B7	13.06	
MFD1 FPGA	006-C0036-04	1.04	
MFD1 BOOT	006-B0191-04	2.03	
NAV1	006-B0082-12	6.02	
NAV1 BB	006-B0082-BC	2.01	
NAV1 FPGA	006-C0124-00	1.0	
NAV2	006-B0082-12	6.02	
NAV2 BB	006-B0082-BC	2.01	
NAV2 FPGA	006-C0124-00	1.0	

CHART 30
REQUIRED EQUIPMENT LISTS (REL)

G1000 VER. 1981.02 (Sheet 3 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version
PFD1	006-B0319-B7	13.06
PFD1 FPGA	006-C0036-04	1.04
PFD1 BOOT	006-B0191-04	2.03
PFD2	006-B0319-B7	13.06
PFD2 FPGA	006-C0036-04	1.04
PFD2 BOOT	006-B0191-04	2.03
LOADER IMAGE	006-B1981-02	1981.02 (1-2)

(1) See “Figure 41” on page 342501155.
(2) PIR-107399-003, Rev. G.

CHART 30
REQUIRED EQUIPMENT LISTS (REL)

G1000 VER. 1981.03 (Sheet 1 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
COM1	006-B0081-xx	11.10	
COM2	006-B0081-xx	11.10	
GDC1	006-B0261-19	3.09	
GDC1 FPGA	006-C0055-00	01.05	
GDC2	006-B0261-19	3.09	(Verify if installed)
GDC2 FPGA	006-C0055-00	01.05	(Verify if installed)
GDL69	006-B0317-22	4.02.00	(Verify if installed)
GEA1	006-B0193-05	2.07	
GEA2	006-B0193-05	2.07	
GIA1	006-B0544-4N	7.14	
GIA1 AUDIO	006-D4432-00	2.00	
GIA2	006-B0544-4N	7.14	
GIA2 AUDIO	006-D4432-00	2.00	
GMA1	006-B0203-44	4.04	
GMU1	006-B0224-01	2.05	
GMU1 FPGA	006-C0048-00	2.00	
GMU2	006-B0224-01	2.05	(Verify if installed)
GMU2 FPGA	006-C0048-00	2.00	(Verify if installed)
GPS1	006-B0339-20	5.0	
GPS2	006-B0339-20	5.0	
GRS1	006-B0223-23	3.03	
GRS1 FPGA	006-C0049-00	02.00	
GRS2	006-B0223-23	3.03	(Verify if installed)
GRS2 FPGA	006-C0049-00	02.00	(Verify if installed)
GS1	006-B0082-xx	6.02	
GS2	006-B0082-xx	6.02	
GSA PTCH CTL	006-B0398-37	3.30	(Verify if installed)
GFC CERT P C	006-D4801-00	2.00	(Verify if installed)
GSA PTCH MON	006-B0398-37	3.30	(Verify if installed)
GFC CERT P M	006-D4801-00	2.00	(Verify if installed)

CHART 30
REQUIRED EQUIPMENT LISTS (REL)

G1000 VER. 1981.03 (Sheet 2 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
GSA PTCH TRIM C	006-B0398-37	3.30	(Verify if installed)
GFC CERT PT C	006-D4801-00	2.00	(Verify if installed)
GSA PTCH TRIM M	006-B0398-37	3.30	(Verify if installed)
GFC CERT PT M	006-D4801-00	2.00	(Verify if installed)
GSA ROLL CTL	006-B0398-37	3.30	(Verify if installed)
GFC CERT R C	006-D4801-00	2.00	(Verify if installed)
GSA ROLL MON	006-B0398-37	3.30	(Verify if installed)
GFC CERT R M	006-D4801-00	2.00	(Verify if installed)
GSA YAW CTL	006-B0398-37	3.30	(Verify if installed)
GFC CERT Y C	006-D4801-00	2.00	(Verify if installed)
GSA YAW MON	006-B0398-37	3.30	(Verify if installed)
GFC CERT Y M	006-D4801-00	2.00	(Verify if installed)
GTS	006-B1244-07	2.22	(Verify if installed)
GTS AUDIO	006-D0726-50	2.00	(Verify if installed)
GTS FPGA	006-C0081-20	2.0	(Verify if installed)
GTS MAGNET V	006-D0159-02	2010.00	(Verify if installed)
GTX1	006-B0172-xx	7.04	
GWX	006-B0266-11	2.12	
GWX FPGA	006-B0042-06	3.01	
MFD1	006-B0319-B9	13.08	
MFD1 FPGA	006-C0036-04	1.04	
MFD1 BOOT	006-B0191-04	2.03	
NAV1	006-B0082-12	6.02	
NAV1 BB	006-B0082-BC	2.01	
NAV1 FPGA	006-C0124-00	1.0	
NAV2	006-B0082-12	6.02	
NAV2 BB	006-B0082-BC	2.01	
NAV2 FPGA	006-C0124-00	1.0	

CHART 30
REQUIRED EQUIPMENT LISTS (REL)

G1000 VER. 1981.03 (Sheet 3 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version
PFD1	006-B0319-B9	13.08
PFD1 FPGA	006-C0036-04	1.04
PFD1 BOOT	006-B0191-04	2.03
PFD2	006-B0319-B9	13.08
PFD2 FPGA	006-C0036-04	1.04
PFD2 BOOT	006-B0191-04	2.03
LOADER IMAGE	006-B1981-03	1981.03 (1-2)

(1) See “Figure 41” on page 342501155.
(2) PIR-107399-003, Rev. H.

INTEGRATED AVIONICS SYSTEM - GARMIN G1000 NXi

Integrated Avionics System - Garmin G1000 NXi

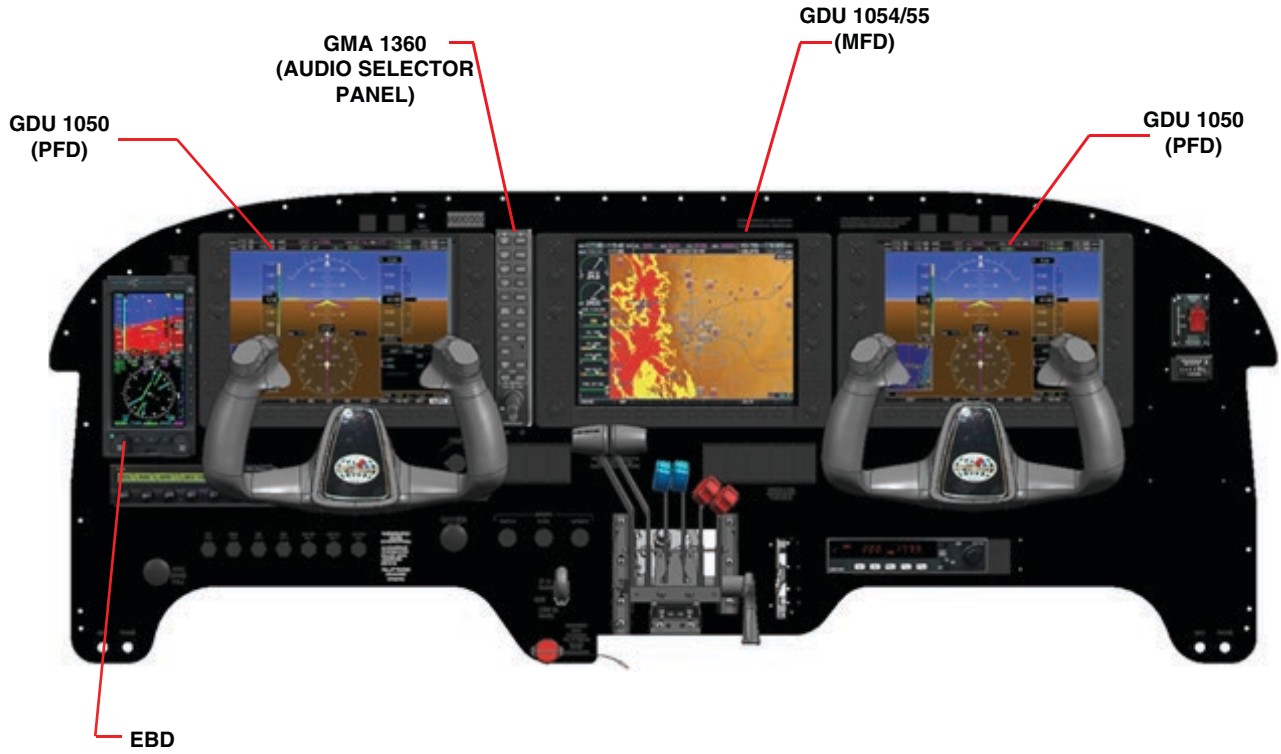
S/N's 3449509 and up.

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

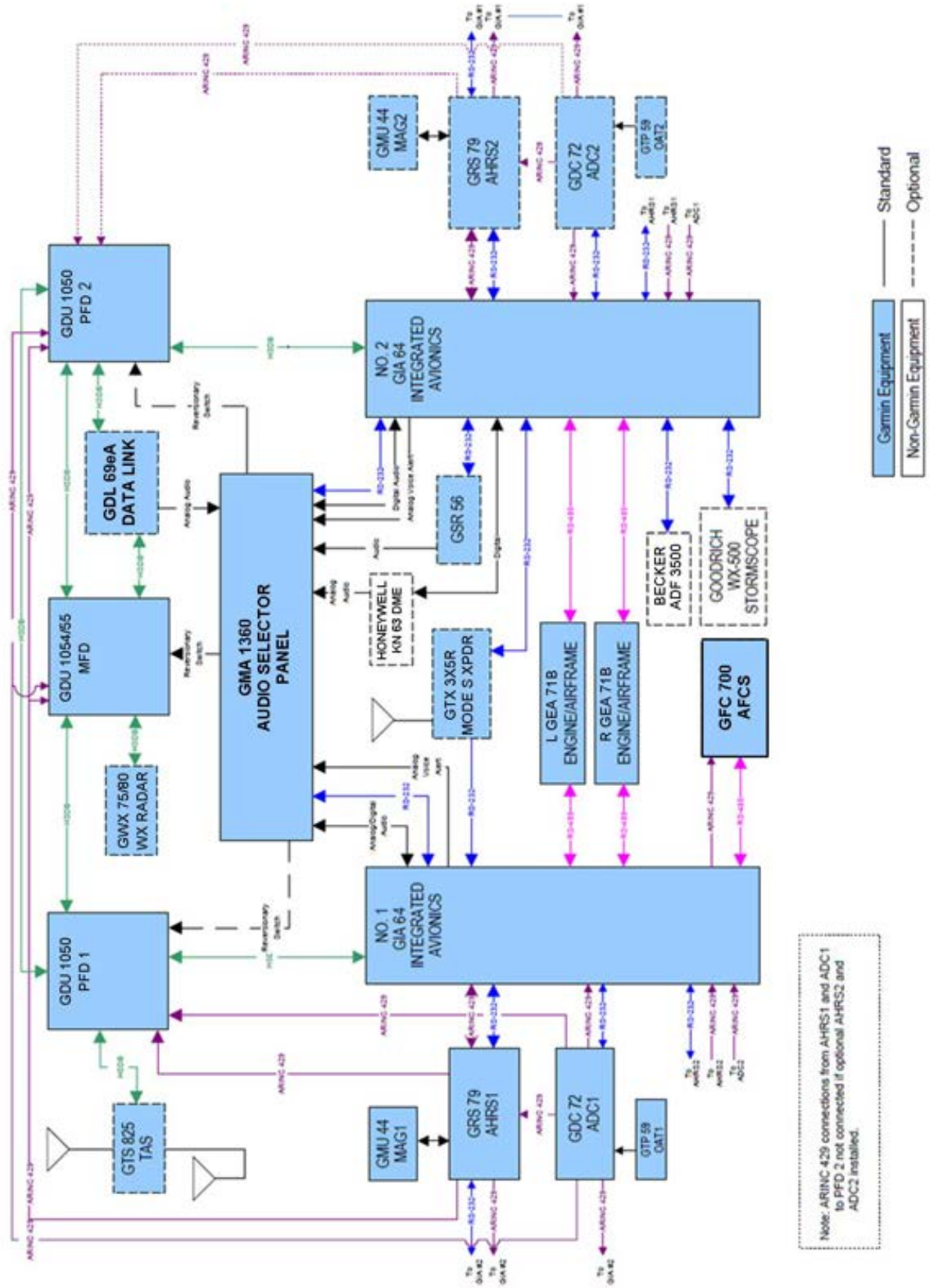
A. Description

The Garmin G1000 NXi Integrated Avionics System provides a fully integrated avionics suite that interfaces with a Garmin GFC 700 Autopilot.

It consists of three (3) displays, two (2) GDU 1050s dedicated as primary flight displays (PFD1/PFD2) and one as a multi-function display (MFD). The PFD displays are located on either side of the MFD, with the stand-by instruments located to the left of the Pilot's PFD (PFD1). Functions provided by the Integrated Avionics System include display of attitude, heading, navigation, traffic (optional), air data, engine and airframe status, and a moving map display with position derived by GPS. In addition to display functions, GPS navigation, VHF/Com, VOR/ILS and transponder functions are provided by the G1000 NXi system. See "Figure 1".



Garmin G1000 NXi Displays
Figure 1



Electrical/Avionics Component Block Diagram
Figure 2

See the latest revision of the Garmin G1000 NXi Pilot's Guide for control and operation. See Vendor Publications under Supplementary Publications in the Introduction.

See "Figure 2" on page 3425022.

(1) Flight Instrumentation

The GRS 79 AHRS(s), GDC 72 Air Data Computer(s), GMU 44B Magnetometer(s), and GTP 59 Temperature Probe(s) are responsible for providing the G1000 NXi system with aircraft attitude, heading, altitude, airspeed, vertical speed, and outside air temperature information, all displayed on the PFD (data is displayed on the MFD in reversionary mode only).

Primary data outputs from the GRS and GDC are sent directly to the PFD1 via ARINC 429. Secondary data paths connect the GRS and GDC to the MFD and PFD2. Additional communications paths connect the GRS and GDC to both GIA 64 units.

The GRS 79 receives GPS data from both GIAs, airspeed data from the GDC, and magnetic heading from the GMU. Using these three external sources, combined with internal sensor data, the GRS accurately calculates aircraft attitude and heading.

(2) Engine Indicator System

The GEA 71B (L/R) provides engine/airframe data to the G1000 NXi system. The unit interfaces to transducers. Analog data is received from the transducers and is converted to a digital signal by the GEA 71B. Digital information is then sent through the primary RS-485 serial path to the #1 GIA 64. From the GIA, data is sent through the HSDB connection to the PFD, then on to the MFD for display. A backup data path from the GEA to the #2 GIA 64, then on to the MFD, exists in the event the primary path fails.

(3) Communications/Navigation Systems

The GIA 64 IAUs contain VHF COM, VHF NAV, and GPS receivers. COM and NAV audio is sent via digital audio to the GMA 1360 Audio Panel. The GMA 1360 Audio Panel controls the display reversionary mode.

GPS information is sent to the GRS 79 AHRS and to both displays for processing.

The GTX 3X5 Mode A, C, S Transponder communicates with both GIAs in a single transponder installation. Transponder data is sent from the GIAs to the PFD where control and operation occurs.

The #1 GIA outputs analog HSI signals to the GFC 700, along with ARINC 429 roll steering, and altitude data, and digital selected altitude data.

(4) Components Details

See "Components" on page 34250237, for detailed descriptions of the individual G1000 NXi system components/Line Replaceable Units (LRU's).

B. Power Distribution

Distribution of power to the G1000 NXi system occurs on nine aircraft electrical busses. See "Figure 3" on page 3425024 and "Figure 4" on page 3425025.

(1) Main Bus

The Main bus current from the battery is controlled by the Master switch on the overhead panel. The battery feed is protected by a 250 amp current limiter (fuse). Alternator power to the main bus is introduced by Alternator switches on the overhead panel. The two Alternator feeds are protected by two 90 amp circuit breakers.

(2) Essential Buses

The two Essential busses are fed through the Main bus and are protected by 60-amp circuit breakers.

(3) Non-Essential Buses

The two Non-Essential busses are fed through the Main bus and are protected by 35-amp circuit breakers.

(4) Avionics Buses

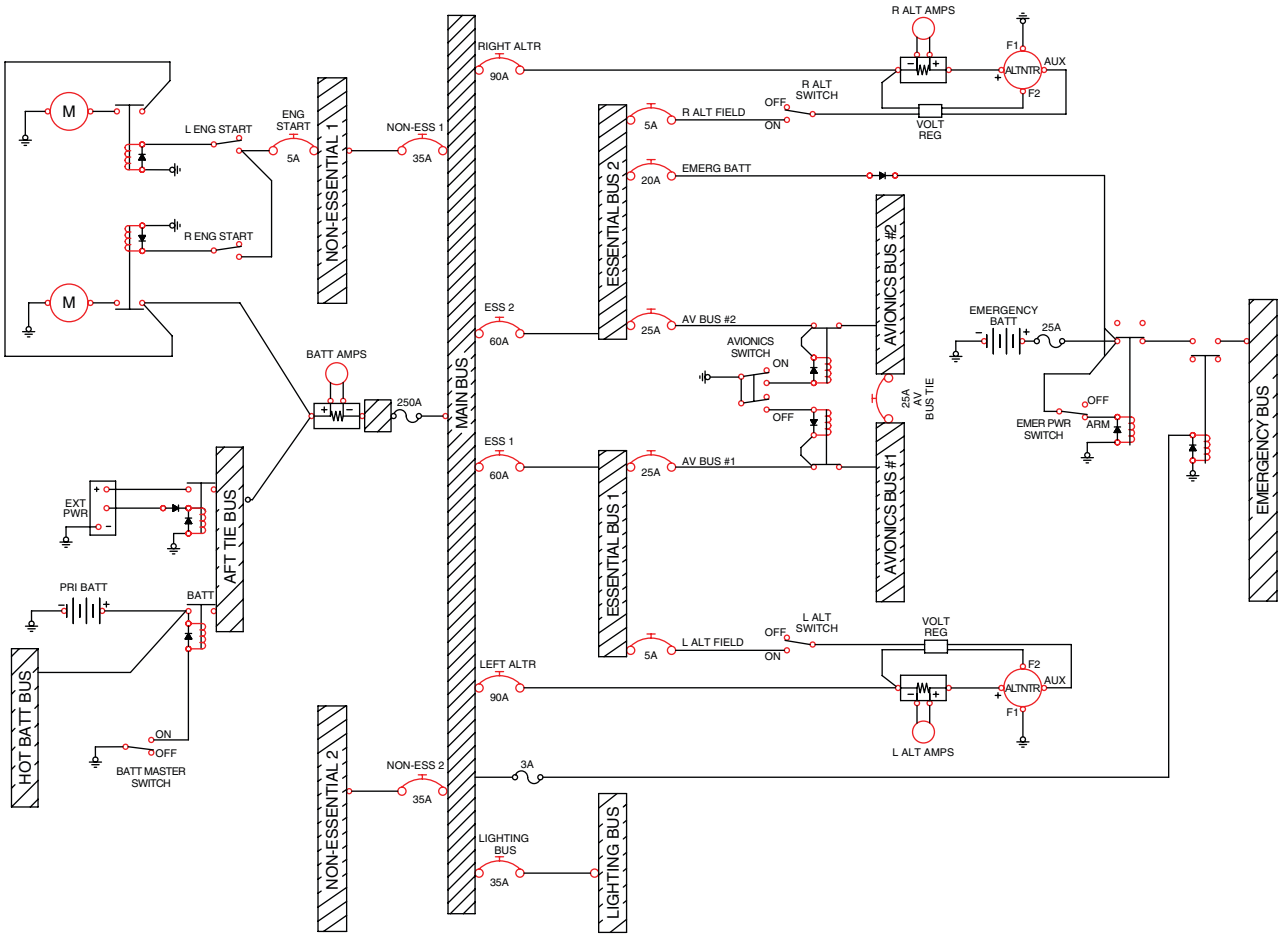
The two avionics busses are fed through independent essential busses which are protected by 25-amp circuit breakers. The busses are interconnected by a 25-amp bus tie and controlled by the avionics master switch. When the AVIONICS switch on the overhead switch panel is pressed, both avionics contactors (normally closed) relax allowing current to flow to both avionics busses. Should the need arise, either avionics bus can be isolated by pulling the avionics bus BUS TIE circuit breaker and the appropriate avionics circuit breaker.

(5) Lighting Bus

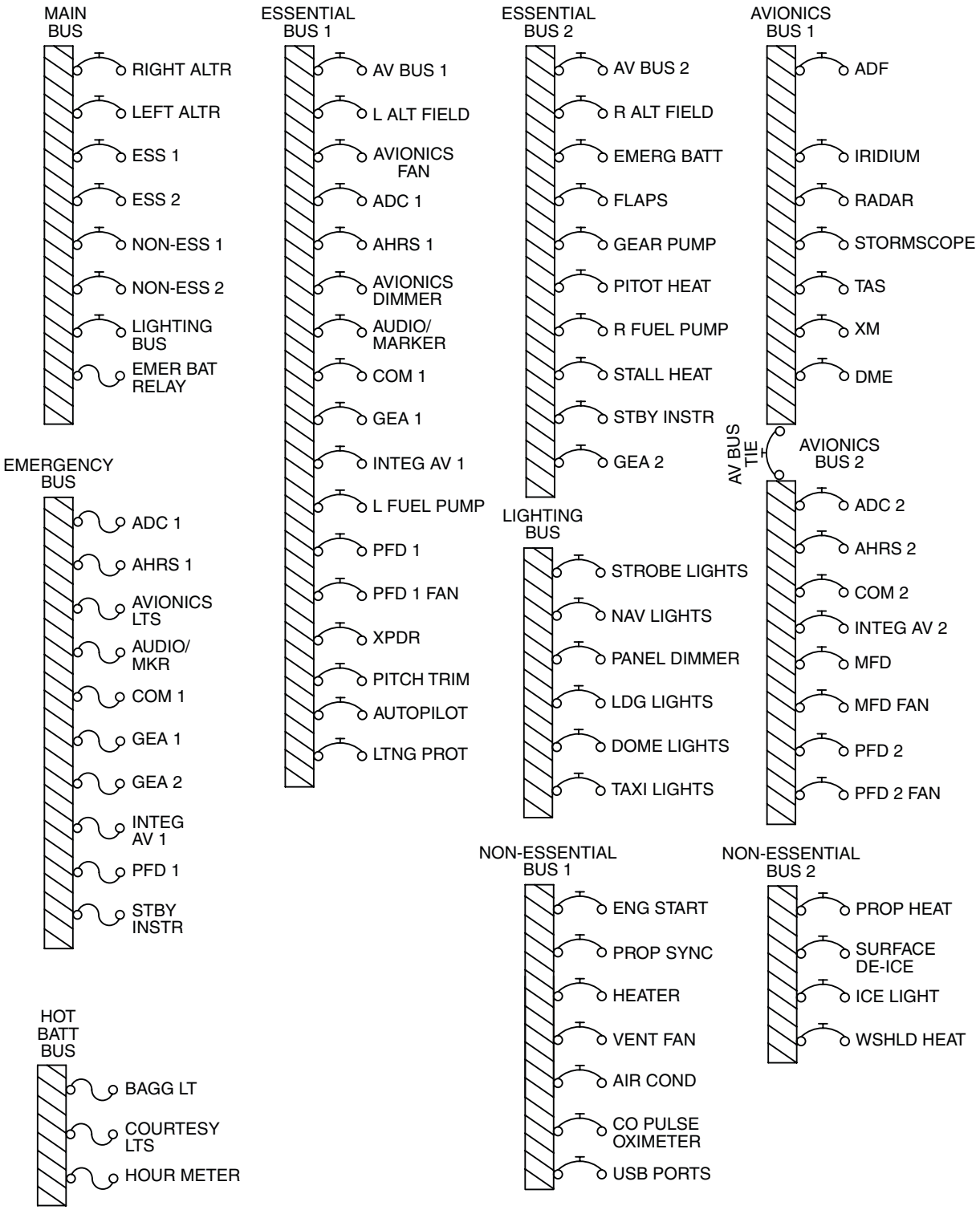
The Lighting bus is fed through the Main bus and is protected by a 35-amp circuit breaker.

(6) Emergency Bus

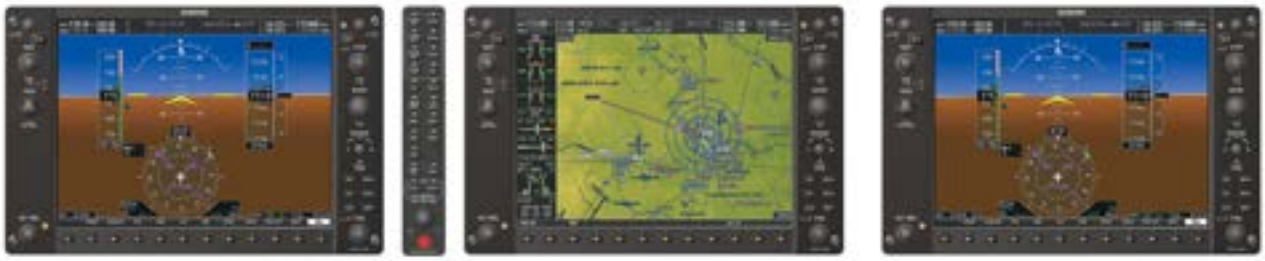
The Emergency bus exists as a backup power bus to the G1000 NXi system. When the EMER switch is turned to the ARM position, power must be removed from the Main bus by turning the battery master switch to the “OFF” position. At the same time, the relay connecting the Emergency bus to the emergency battery is closed, providing emergency power to the connected equipment.



Bus Feeds
Figure 3



Component Power Sources
Figure 4



Normal Mode
Figure 5



Reversionary Mode #1 PFD Fail
Figure 6



Reversionary Mode MFD Fail
Figure 7

C. G1000 NXi System Controls and Operation

(1) System Controls

The G1000 NXi system controls are located on the PFD/MFD bezels and above the respective PFD/MFD displays. Indicators above the PFD, MFD, NAV, and COM keys are illuminated when their respective control mode(s) are selected. The unit is in MFD control mode by default on system power-up. NAV/COM radio tuning can be accomplished in either PFD or MFD control mode. The appropriate frequency box on the selected display is outlined by a light blue selection box, which flashes for a few seconds to indicate Control Unit activity. Selection of a different display control or radio tuning mode results in cancelation of the previous radio tuning mode.

(2) System Operation

The displays are connected together via multiple data busses, thus allowing for high-speed communication. Each GIA 64 is connected to the on-side PFD. This section discusses normal and reversionary modes of operation as well as the various AHRS modes of the G1000 NXi system.

(a) Normal Operation, see “Figure 5” on page 3425026.

1) PFD

In normal mode, the PFDs present graphical flight instrumentation (attitude, heading, airspeed, altitude and vertical speed), thereby replacing the traditional flight instrument cluster. The PFDs also offer control for COM and NAV frequency selection.

2) MFD

In normal mode, the right portion of the MFD displays a full-color moving map with navigation information, while the left portion of the MFD is dedicated to the Engine Indication System (EIS). The MFD also offers control for COM and NAV frequency selection.

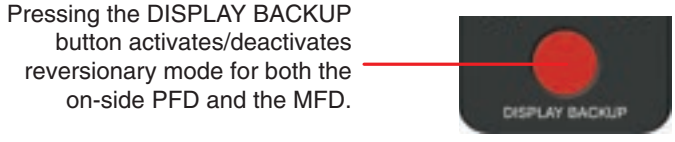
(b) Reversionary Mode, see “Figure 6” on page 3425016 and “Figure 7” on page 3425026.

NOTE: The G1000 NXi system alerts the pilot when backup paths are utilized by the LRUs.

Reversionary mode is a mode of operation in which all important flight information is presented identically on at least one of the remaining displays. Transition to reversionary mode should be straight-forward for the pilot, for flight parameters are presented in the same format as in normal mode. In the event of a PFD failure, the G1000 NXi system automatically switches to reversionary (backup) mode. In reversionary mode, all important flight information is presented on the remaining display(s) in the same format as in normal operating mode.

- PFD1 failure – MFD enters reversionary mode; PFD2 remains in normal mode.
- MFD failure – PFD1 automatically switches to reversionary mode; PFD2 remains in normal mode.
- PFD2 failure – PFD1 and the MFD remain in normal mode.

Each display may be put into reversionary mode by manually pressing the DISPLAY BACKUP button at the bottom of the audio panel (see “Figure 8”). Pressing this button again deactivates reversionary mode.



Display Backup Button
Figure 8

D. Maintenance Mode

CAUTION: ENTER “MAINTENANCE MODE” WHEN ACTIVATING THE GARMIN G1000 NXI SYSTEM ON THE GROUND TO CONDUCT TESTS, INSPECTIONS, ETC.

The “Hobbs” Meter is activated by the GIA 64 Integrated Avionics Units and will run when one of the following is true, depending on how it was configured:

(1) the engine is running (Option A); or

(2) the airplane is inflight (Option B); or

(3) the GIA 64 No. 1 is ON (Option C).

To prevent needless accumulation of “hours” (i.e., to disable the “Hobbs” Meter) on the ground a Maintenance Mode has been implemented in the Garmin G1000 NXi software.

Enter “Maintenance Mode” before exiting the splash screen at system startup by pressing SoftKeys 1, 2, 3, 1 in that order. The “MAINT MODE ON” advisory will then be displayed while the system remains in maintenance mode.

E. Troubleshooting

Troubleshoot the G1000 NXi system by first identifying, then isolating the specific failure to the responsible LRU. There are several indications that the G1000 NXi presents to the pilot or technician, showing overall system condition. A course of action should be determined based on the information presented on the display.

The 5th AUX group page on the MFD (“Figure 9” on page 3425029) provides LRU health status by means of a green check or a red ‘X’. Also, LRU software versions are shown along with other database versions and dates. If a red ‘X’ is shown for an LRU, see the appropriate LRU troubleshooting chart for guidance.

Typically, the G1000 NXi Alerting System provides alerts/annunciations in conjunction with the information presented at the 5th AUX page.

CAUTION: RETURN TO SERVICE PARAGRAPHS UNDER INDIVIDUAL ”COMPONENTS” ON PAGE 34250237 PROVIDE DETAILED INSTRUCTIONS ON EQUIPMENT CONFIGURATION AND RETURN TO SERVICE TESTING REQUIRED ANYTIME A G1000 NXI COMPONENT OR LRU IS REMOVED, SWAPPED, OR REPLACED, THE TECHNICIAN MUST FOLLOW THESE PROCEDURES TO ENSURE PROPER OPERATION OF THE SYSTEM.

Troubleshooting information is provided as follows:

(1) Line Replaceable Unit (LRU) Failures
See “Chart 1” on page 34250212.

(2) Engine / Airframe Sensor Failures
See “Chart 2” on page 34250217.

(3) Data Path Failures
See “Chart 3” on page 34250221.

(4) GDU 1050
See “Chart 7” on page 34250240 and “Chart 8” on page 34250242.

(5) GMA 1360
See “Chart 9” on page 34250249 and “Chart 10” on page 34250250.

(6) GIA 64
See “Chart 11” on page 34250253 and “Chart 12” on page 34250255.

(7) GEA 71B
See “Chart 13” on page 34250263.
(8) GTX 3X5R
See “Chart 14” on page 34250265.
(9) GDC 72
See “Chart 15” on page 34250268.
(10) GRS 79
See “Chart 16” on page 34250272 and “Chart 17” on page 34250273.
(11) GMU 44B
See “Chart 16” on page 34250272 and “Chart 17” on page 34250273.
(12) GDL 69/69A
See “Chart 18” on page 34250279 and “Chart 19” on page 34250280.
(13) GSR 56
See “Chart 21” on page 34250283.
(14) GWX 75/80
See “Chart 22” on page 34250286.
(15) GTS 825
See “Chart 23” on page 34250288.
(16) Software Configuration
See “Chart 24” on page 34250295.



System Status Page (AUX Group) (Typical)
Figure 9

(17) Software Status Verification

(a) Enter the configuration mode on the PFDs and the MFD by applying power while holding ENT on each display.

(b) Select the “Systems Status Page”.

1) Verify that the following are valid. (Green annunciations indicate that the system is healthy): MFD1, PFD1, PFD2, GIA1, GIA2, GDL 69 (optional), GTS (optional), and GWX (optional).

2) Rotate the small FMS knob to change to the “System Configuration Page” and verify that the following are valid. (Indicated by green annunciations): MFD1, PFD1, PFD2, GIA1, GIA2, GDL 69 (optional), GTS (optional), and GWX (optional).

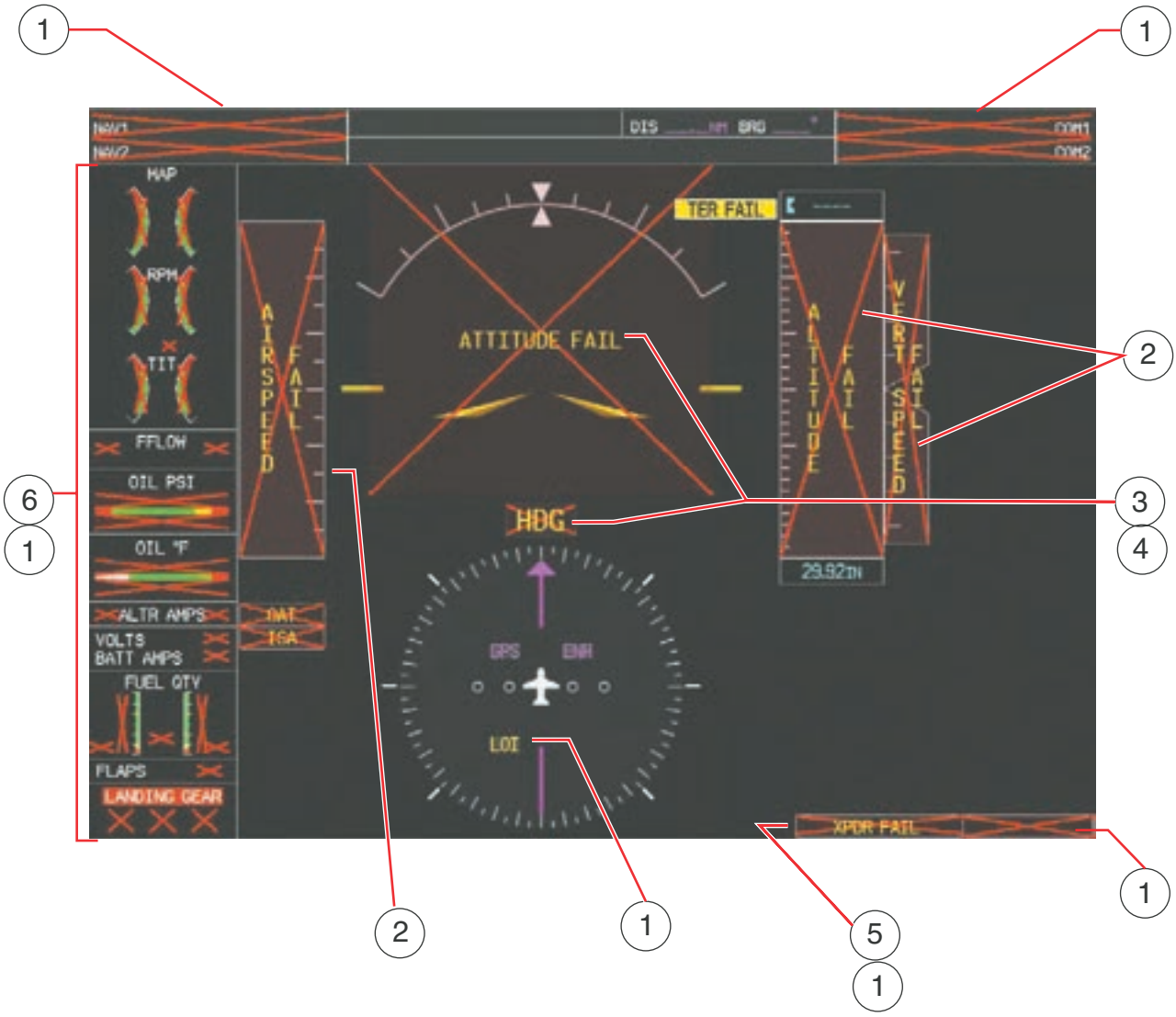
(18) System Configuration Error Removal

NOTE: If any configuration or calibration is purposely changed on an aircraft, it may result in a System Configuration Error. The process below details the procedure to remove this error.

(a) Start the PFD and MFD in configuration mode by applying power to each while holding the ENT key.

(b) Navigate to the Configuration Manager page of the System group using the small FMS knob.

(c) Press the CNFM CFG softkey. Press ENT to select OK on popup prompt.



1. GIA 64 Integrated Avionics Units
2. GDC 72 Air Data Computer
3. GRS 79 AHRS
4. GMU 44 Magnetometer
5. GTX 335R/345R Transponder
6. GEA 71B Engine Airframe Units

System Fail (Typical)
Figure 10

CHART 1 (Sheet 1 of 5)
LRU TROUBLESHOOTING

System Failure Troubleshooting

The following table provides basic troubleshooting guidance for LRU failures.

Invalid Data Field	Associated LRU(s)	Solution
Low Speed Awareness Band permanently displayed 	GDU	• With aircraft weight on wheels and the G1000 in Configuration Mode, check that the **RH GEAR ON GROUND** discrete in indication on GIA I/O CONFIGURATION page is illuminated. • If indication is illuminated Green, switch GIA1 and GIA2, to verify location of problem: ✓ If problem is resolved, replace defective unit. ✓ If indication is not illuminated, check for wiring faults between the GIAs and aircraft squat switch system.
Low Speed Awareness Band permanently inhibited 	GDU	• With aircraft weight off wheels and the G1000 in Configuration Mode, check that the **RH GEAR ON GROUND** discrete in indication on GIA I/O CONFIGURATION page is not illuminated. • If indication is not illuminated Green, switch GIA1 and GIA2, to verify location of problem: ✓ If problem is resolved, replace unit. ✓ If indication is illuminated, check for wiring faults between the GIAs and aircraft squat switch system.
NAV1 & COM1 	GIA1	• Check configuration settings for GIA1 and PFD1. • Swap GIA1 and GIA2, to verify location of problem: ✓ If problem follows GIA1, replace GIA1. • Check Ethernet interconnect from GIA1 to PFD1. ✓ If problem persists, replace PFD.
NAV2 & COM2 	GIA2	• Check configuration settings for GIA2 and PFD2. • Swap GIA1 and GIA2, to verify location of problem: ✓ If problem follows GIA2, replace GIA2. • Check Ethernet interconnect from GIA2 to PFD2. ✓ If problem persists, replace PFD.

CHART 1 (Sheet 2 of 5)
LRU TROUBLESHOOTING

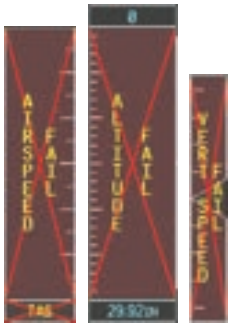
Invalid Data Field	Associated LRU(s)	Solution
GPS DR 	GIA1 or GIA2	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Check GPS1 and GPS2 signal strength on AUX page 4. Refer to GIA GPS section if GPS can not acquire a position lock for troubleshooting. • Check corresponding GPS antenna and cable for faults. • Check Ethernet interconnect between the PFD1 to GIA1 or PFD2 to GIA2 for faults. • Swap GIA1 and GIA2, to verify location of problem: ✓ If problem follows the GIA, replace the GIA. ✓ If problem persists, replace the PFD1 or PFD2.
XPDR FAIL 	GTX 3X5R	• Check GTX 3X5R configuration settings for GIA1 and GIA2. • Check GTX 3X5R/GIA1 interconnect. • Check GTX 3X5R/GIA2 interconnect. • Perform a SET>ACTV configuration reset on the GTX Config page for each GTX, and verify the aircraft registration is present. • Replace GTX 3X5R. ✓ If problem persists, replace GIA.
TAS FAIL AIRSPEED FAIL ALTITUDE FAIL VERT SPEED FAIL 	GDC 72	• Inspect GDC 72 pitot/static plumbing integrity. • Inspect pitot/static ports and associated equipment for faults. • For TAS failure, also check GTP 59 OAT probe. • Check GDC 72 configuration settings for the PFDs, MFD, GIA1, and GIA2. ✓ If PFDs, MFD, and GIA configuration settings are correct, replace the GDC configuration module. ✓ If problem persists, replace the GDC 72.
OAT 	GTP 59	• Check OAT probe wiring and connectors for faults. • Check GDC config module wiring for damage, replace if any is found. • Replace GDC config module • Replace GDC 72 with a known good unit: ✓ If problem persists, replace the GTP 59

CHART 1 (Sheet 3 of 5)
LRU TROUBLESHOOTING

Invalid Data Field	Associated LRU(s)	Solution
HDG FAIL 	GRS 79 & GMU 44	• Ensure metal objects (tool boxes, power carts, etc.) are not interfering with the magnetometer and aircraft is not in hangar, near other buildings, parked over metal drainage culverts or on hard surfaces that may contain steel reinforcements. • Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Cycle power after moving aircraft away from metal objects to determine if metal objects were the source of the interference. Allow up to five minutes for the heading to reinitialize. • Perform a Magnetometer Interference Test to check for interference from onboard electrical system components (e.g. NAV lights). Pay particular attention to any new electrical devices that have been installed since the aircraft was new. Correct any discrepancies that do not allow this test to pass before continuing. • Ensure GRS 79 and GMU 44 connectors are secure. • Check the wiring and any inline connectors between the GRS and GMU for faults. • Recalibrate the GMU 44. • Load configuration files to the PFD1, PFD2, GIA1, and GIA2. ✓ If problem persists replace the GMU 44. ✓ If problem persists, replace the GRS 79.
	GRS 79 & GMU 44	• If this message persists longer than five minutes, perform AHRS calibration procedures as described later in this chapter.

CHART 1 (Sheet 4 of 5)
LRU TROUBLESHOOTING

Invalid Data Field	Associated LRU(s)	Solution
ATTITUDE FAIL 	GRS 79	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Cycle GRS 79 power to restart initialization. • Ensure GRS 79 connecter is secure and proper wire harness strain relief is provided. • Ensure the GRS 79 is fastened down tightly in it's mounting rack and that the mounting rack is not loose (CAUTION - do not loosen the mounting rack hardware to the airframe shelf or the aircraft will need to be re-leveled and the PITCH/ROLL OFFSET procedure performed). • Ensure GPS has acquired at least four satellites, has a 3D navigation solution, and a DOP of less than 5.0. This is particularly important for an ATTITUDE FAIL that appears during ground operation only. • Perform an Engine Run-Up Test to check if engine vibration is causing the GRS 79 to go offline. • Load configuration files to the PFD, MFD, GIA1, and GIA2. • Calibrate the GRS 79 (Pitch/Roll Offset and Magnetometer Calibration). • Replace GRS 79. ✓ If problem persists replace GRS 79 configuration module. ✓ Contact Garmin Aviation Product Support if condition continues after replacing the GRS 79 and config module for additional assistance.

CHART 1 (Sheet 5 of 5)
LRU TROUBLESHOOTING

A DME or ADF failure is represented by the following red X's. The PFD displays function as a control head for the remote-mounted Becker ADF 3500 and KN 63 DME.

Invalid Field	Sensor	Solutions
 	Becker ADF	• Ensure that GIA 64 #2 is properly functioning. • Reload the ADF option configurations. • Check for proper operation of the ADF receiver. Ensure that the receiver is receiving power. • Check ADF – GIA2 interconnect.
 	Honeywell DME	• Ensure that GIA 64 #2 is properly functioning. • Reload the DME option configurations. • Check for proper operation of the DME receiver. Ensure that the receiver is receiving power. • Check DME – GIA2 interconnect.

CHART 2 (Sheet 1 of 3)
TROUBLESHOOTING ENGINE / AIRFRAME SENSOR FAILURES

Invalid Data Field	Associated LRU(s)	Solution
Engine/Airframe Sensors (All Invalid) 	GEA 71B & GIA 64	• If software was loaded to a new GDU display, be sure that the user settings for the replaced display were cleared. Clear user settings by pressing the CLR key on the replaced display while applying power to it. Acknowledge the on-screen prompt by pressing the ENT key or the right-most softkey. • Check for GEA related Alert messages on the PFD. Correct any Alerts concerning software or configuration errors by reloading software or configuration as noted. • Verify GEA internal power supply, configuration, and calibration status in configuration mode. ✓ The internal power supply, configuration and calibration boxes should be green. If they are red, replace the GEA 71B. • Verify internal, external, and reference voltages listed in the Main Analog and I/O A Analog boxes are not dashed out (does not include Aircraft Power 1 and 2). ✓ If any voltages are dashed out, replace the GEA. • Ensure the GEA is online (green checkmark on the AUX – SYSTEM STATUS page). ✓ If GEA is not online verify unit is receiving power at the rack connector. ✓ Check the GIA/GEA interconnects for faults. ✓ Reload configuration files to both GIA's and the GEA. • If problem persists, replace the GEA 71B.

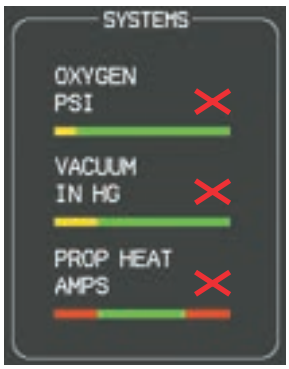
CHART 2 (Sheet 2 of 3)
TROUBLESHOOTING ENGINE / AIRFRAME SENSOR FAILURES

Engine/Airframe Instrument Failures

The following table provides guidance for troubleshooting individual engine/airframe sensor failures. Be sure to also follow previous guidance given for the GEA 71B. The technician should troubleshoot to isolate the fault by checking sensor-to-GEA wiring, replacing the suspect sensor, and finally by replacing the GEA 71B. Replace one part at a time. Check for correct operation of the sensors and GEA 71B after any part has been replaced. Refer to G1000 NXi in 91-00-00 Wiring Diagrams.

Invalid Field	Sensor	Possible Solutions (for applicable engine/system)
	Manifold Pressure	• Check manifold pressure transducer wiring. • Replace manifold pressure transducer wiring. • Replace GEA 71B.
	Propeller RPM	• Check engine tachometer sensor wiring. • Replace engine tachometer sensor. • Replace GEA 71B.
	Turbine Inlet Temperature	• Check TIT probe wiring. • Replace TIT probe sensor. • Replace GEA 71B.
	Fuel Flow	• Check fuel flow transmitter wiring. • Check fuel flow transmitter - GEA wiring. • Check power input to fuel flow transmitter. • Replace fuel flow transmitter. • Replace GEA 71B.
	Oil Pressure	• Check oil pressure sensor – GEA wiring. • Replace oil pressure sensor. • Replace GEA 71B.
	Oil Temperature	• Check oil temperature sensor – GEA wiring. • Replace oil temperature sensor. • Replace GEA 71B.
	Fuel Quantity	• Check Fuel Senders - #1 GIA 64 wiring. • Check #1 GIA 64 - #1PFD wiring. • Check MFD - #1 PFD connections. • Replace Fuel Senders left or right • Replace #1 GIA 64 • Replace #1 PFD • For more details see 28-00-00.

CHART 2 (Sheet 3 of 3)
TROUBLESHOOTING ENGINE / AIRFRAME SENSOR FAILURES

Invalid Field	Sensor	Possible Solutions (for applicable engine/system)
	Amp/Volts	• Check Alt amp sensor - GEA 71B wiring. • Replace Alt amp sensor. • Replace GEA 71B. • Check main bus - GEA 71B wiring. • Check emergency bus - GEA 71B wiring. • Replace GEA 71B
	Flaps	• Check Potentiometer - GEA 71B wiring. • Replace Potentiometer. • Replace GEA 71B.
	Oxygen	• Check oxygen pressure transducer wiring. • Replace oxygen pressure transducer sensor. • Replace GEA 71B.
	Vacuum (w/Deice - Opt)	• Check vacuum sensor - GEA 71B wiring. • Replace vacuum sensor. • Replace GEA 71B.
	Prop Heat (w/Deice - Opt)	

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(19) Backup Communications Path Checks

The G1000 NXi system architecture is designed with redundant communication ports for several LRUs so that critical information can continue to be displayed in the event of an equipment or wiring failure. Of most importance is flight attitude, heading, and air data information. The GRS 79 and GDC 72 each have four separate ARINC 429 data lines which are all capable of sending data to the displays. The GEA 71B, GTX 3X5R, and GMA 1360 each have two redundant serial communication paths for the same purpose. See “Figure 2” on page 3425022 for a basic G1000 NXi block diagram depicting this architecture. When troubleshooting, refer to the G1000 NXi Wiring Diagrams listed in Chapter 91.

(20) Failed Path Messages

The following message indicates there is a data path connected to the GDU (PFD1, PFD2, or MFD) or the GIA 64 (1 or 2) that has failed.

FAILED PATH – A data path has failed. Check configuration mode.

The FAILED PATH message is triggered by a timeout of any one digital channel. The channels that are checked are listed on these pages in config mode:

(a) GDU RS-232 / ARINC 429 CONFIG (PFD1/2, and MFD).

(b) GIA RS-232 / ARINC 429 CONFIG (GIA1 and GIA2).

(c) GIA RS-485 CONFIGURATION (GIA1 and GIA2).

NOTE: Once the FAILED PATH message has been triggered, it will remain on the list of messages until the next power cycle. This latching was implemented so that for intermittent failures, the message would remain at the end of the flight (to alert maintenance crew). Also, this keeps the crew from having to acknowledge message repeatedly in the case of intermittent failures.

The box next to each channel indicates the current status of the channel per the below:

Red = data path is known to be failed.

Black = data path status is unknown.

Green = data path is known to be good.

The applicable data paths can be verified by viewing the configuration mode pages listed in “Chart 3” through “Chart 6” on page 34250233. See “Chart 26” on page 342502101 to navigate to a particular configuration page.

CHART 3
TROUBLESHOOTING - DATA PATH FAILURES

Failure Message	Cause	Remedy
FAILED PATH – a data path has failed.	A communications data path status is not being received by the G1000 NXi.	Determine which data path has failed - check configuration mode. Check wiring continuity for the failed path. Swap or replace the affected LRU.

CHART 4 (Sheet 1 of 3)
FAILED PATH MESSAGES, CONFIG PAGE (PFD1/2, AND MFD)

GDU RS-232 / ARINC 429 CONFIG Page
PFD 1 ARINC 429

Channel	LRU	Indicator	Status
IN 1	GRS 79 #1	GREEN	PFD1/GRS 79 #1 data path is functioning correctly.
		RED	PFD1/GRS 79 #1 data path is not functioning correctly. • Verify GRS 79 #1 status is "OK" using the System Status page on the MFD. If it is not, correct condition before proceeding, reference GRS troubleshooting sections. o Swap GRS1 and GRS2 (no reconfiguration required) to confirm if the problem is in the original GRS1. o Replace original GRS1 if box turns green after swapping units. • Load PFD1 configuration file. • Swap PFD1 and PFD2 to confirm if the problem is in the original PFD1. o Replace original PFD1 if box turns green after swapping displays. • Swap GRS1 and GRS2 (no reconfiguration required) to confirm if the problem is in the original GRS1. o Replace original GRS1 if box turns green after swapping units. Check the PFD1/GRS 79 #1 interconnect wiring for faults.
		BLACK	PFD1/GRS 79 #1 data path functionally is unknown. Reload PFD1 configuration file.
IN 2	GDC 72 #1	GREEN	PFD1/GDC 72 #1 data path is functioning correctly
		RED	PFD1/GDC 72 #1 data path is not functioning correctly. • Verify GDC 72 #1 status is "OK" using the System Status page on the MFD. If it is not, correct condition before proceeding, reference GDC troubleshooting sections. • Load PFD1 and GDC 72 #1 configuration files. • Swap PFD1 and PFD2 to confirm if the problem is in the original PFD1. o Replace original PFD1 if box turns green after swapping displays. • Check the PFD1/GDC 72 #1 interconnect wiring for faults. Replace GDC 72 #1 if problem remains.
		BLACK	PFD1/GDC 72 #1 data path functionality is unknown. Reload PFD1 configuration file.

CHART 4 (Sheet 2 of 3)
FAILED PATH MESSAGES, CONFIG PAGE (PFD1/2, AND MFD)

PFD 2 ARINC 429

Channel	LRU	Indicator	Status
IN 1	GRS 79 #2	GREEN	PFD2/GRS 79 #2 data path is functioning correctly.
		RED	PFD2/GRS 79 #2 data path is not functioning correctly. • Verify GRS 79 #2 status is “OK” using the System Status page on the MFD. If it is not, correct condition before proceeding, reference GRS troubleshooting sections. o Swap GRS1 and GRS2 (no reconfiguration required) to confirm if the problem is in the original GRS1. o Replace original GRS1 if box turns green after swapping units. • Load PFD2 configuration file. • Swap PFD2 and PFD1 to confirm if the problem is in the original PFD2. o Replace original PFD2 if box turns green after swapping displays. • Swap GRS1 and GRS2 (no reconfiguration required) to confirm if the problem is in the original GRS1. o Replace original GRS1 if box turns green after swapping units. • Check the PFD2/GRS 79 #2 interconnect wiring for faults.
		BLACK	PFD2/GRS 79 #2 data path functionality is unknown. Reload PFD2 configuration file.
IN 2	GDC 72 #2	GREEN	PFD2/GDC 72 #2 data path is functioning correctly.
		RED	PFD2/GDC 72 #2 data path is not functioning correctly. • Verify GDC 72 #2 status is “OK” using the System Status page on the MFD. If it is not, correct condition before proceeding, reference GDC troubleshooting sections. • Load PFD2 and GDC 72 #2 configuration files. • Swap PFD2 and PFD1 to confirm if the problem is in the original PFD2. o Replace original PFD2 if box turns green after swapping displays. • Check the PFD2/GDC 72 #2 interconnect wiring for faults. Replace GDC 72 #2 if problem remains.
		BLACK	PFD2/GDC 72 #2 data path functionality is unknown. Reload PFD 2 configuration file.

CHART 4 (Sheet 3 of 3)
FAILED PATH MESSAGES, CONFIG PAGE (PFD1/2, AND MFD)

MFD ARINC 429

Channel	LRU	Indicator	Status
IN 1	GRS 79 #1	GREEN	MFD/GRS 79 #1 data path is functioning correctly.
		RED	MFD/GRS 79 #1 data path is not functioning correctly. • Verify GRS 79 #1 status is “OK” using the System Status page on the MFD. If it is not, correct condition before proceeding, reference GRS troubleshooting sections. o Swap GRS1 and GRS2 (no reconfiguration required) to confirm if the problem is in the original GRS1. o Replace original GRS1 if box turns green after swapping units. • Load MFD configuration file. • Swap MFD and PFD2 to confirm if the problem is in the original MFD. o Replace original MFD if box turns green after swapping displays. • Swap GRS1 and GRS2 (no reconfiguration required) to confirm if the problem is in the original GRS1. o Replace original GRS1 if box turns green after swapping units. Check the MFD/GRS 79 #1 interconnect wiring for faults.
		BLACK	MFD/GRS 79 #1 data path functionally is unknown. Reload MFD configuration file.
IN 2	GDC 72 #1	GREEN	MFD/GDC 72 #1 data path is functioning correctly
		RED	MFD/GDC 72 #1 data path is not functioning correctly. • Verify GDC 72 #1 status is “OK” using the System Status page on the PFD. If it is not, correct condition before proceeding, reference GDC troubleshooting sections. • Load MFD and GDC 72 #1 configuration files. • Swap MFD and PFD2 to confirm if the problem is in the original MFD. o Replace original MFD if box turns green after swapping displays. • Check the MFD/GDC 72 #1 interconnect wiring for faults. Replace GDC 72 #1 if problem remains.
		BLACK	MFD/GDC 72 #1 data path functionality is unknown. Reload MFD configuration file.

CHART 5 (Sheet 1 of 8)
FAILED PATH MESSAGES, GIA RS-232 / ARINC 429 CONFIG PAGE (GIA1 AND GIA2)

GIA1 RS-232

Channel	LRU	Indicator	Status
CHNL 1	GDC 72 #1	GREEN	GIA1/GDC 72 #1 data path is functioning correctly.
		RED	GIA1/GDC 72 #1 data path is not functioning correctly. • Load GIA1 and GDC 72 #1 configuration files. • Swap GIA1 and GIA2, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA1. o Replace original GIA1 if box turns green after swapping units. • Check the GIA1/GDC 72 #1 interconnect wiring for faults. Replace GDC 72 #1 if problem remains.
		BLACK	GIA1/GDC 72 #1 data path functionality is unknown. Reload GIA1 configuration files.
CHNL 4	GRS 79 #2 (output only)	BLACK	GIA1/GRS 79 #2 output data path is not monitored. A black box is normal.
CHNL 5	GTX3X5	GREEN	GIA1/GTX3X5 data path is functioning correctly.
		RED	GIA1/GTX3X5 data path is not functioning correctly. • Load GIA1 and GTX3X5 configuration files. • Swap GIA1 and GIA2, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA1. o Replace original GIA1 if box turns green after swapping units. • Check the GIA1/GTX3X5 interconnect wiring for faults. Replace GTX3X5 if problem remains.
		BLACK	GIA2/GTX3X5 data path functionality is unknown Reload GIA1 configuration files.

CHART 5 (Sheet 2 of 8)
FAILED PATH MESSAGES, GIA RS-232 / ARINC 429 CONFIG PAGE (GIA1 AND GIA2)

GIA1 RS-232

Channel	LRU	Indicator	Status
CHNL 6	GRS 79 #1	GREEN	GIA1/GRS 79 #1 data path is functioning correctly.
		RED	GIA1/GRS 79 #1 data path is not functioning correctly. • Load GIA1 configuration files. • Swap GIA1 and GIA2, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA1. o Replace original GIA1 if box turns green after swapping units. • Swap GRS1 and GRS2 (no reconfiguration required) to confirm if the problem is in the original GRS1. o Replace original GRS1 if box turns green after swapping units. Check the GIA1/GRS 79 #1 interconnect wiring for faults.
		BLACK	GIA1/GRS 79 #1 data path functionality is unknown. Reload GIA1 configuration files.
CHNL 7	GMA 1347	GREEN	GIA1/GMA 1360 data path is functioning correctly.
		RED	GIA1/GMA 1360 data path is not functioning correctly. • Load GIA1 and GMA 1360 configuration files. • Swap GIA1 and GIA2, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA1. o Replace original GIA1 if box turns green after swapping units. Check the GIA1/GMA 1360 interconnect wiring for faults.
		BLACK	GIA1/GMA 1360 data path functionality is unknown. Reload GIA1 configuration files.

CHART 5 (Sheet 3 of 8)
FAILED PATH MESSAGES, GIA RS-232 / ARINC 429 CONFIG PAGE (GIA1 AND GIA2)

GIA1 ARINC 429

Channel	LRU	Indicator	Status
IN 5	GDC 72 #1	GREEN	GIA1/GDC 72 #1 data path is functioning correctly.
		RED	GIA1/GDC 72 #1 data path is not functioning correctly. • Check PFD Alert Window for GIA or GDC configuration or software error messages. Correct any errors before proceeding. • Verify GDC 72 status is OK using the AUX – SYSTEM STATUS page on the MFD. If it has a Red-X, troubleshoot why the GDC is offline before proceeding. • Swap GIA1 and GIA2, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA1. – Replace original GIA1 if box turns green after swapping units. • Check the GIA1/GDC 72 #1 interconnect wiring and unit connector pins for faults. • Replace GDC 72 #1.
		BLACK	GIA1/GDC 72 #1 data path functionality is unknown. Reload GIA1 configuration files.
IN 6	GRS 79 #1	GREEN	GIA1/GRS 79 #1 data path is functioning correctly.
		RED	GIA1/GRS 79 #1 data path is not functioning correctly. • Check PFD Alert Window for GIA or GRS configuration or software error messages. Correct any errors before proceeding. • Verify GRS 79 status is OK using the AUX - SYSTEM STATUS PAGE on the MFD. If it has a Red-X, troubleshoot why the GRS is offline before proceeding. • Ensure GRS 79 connecter is secure and proper wire harness strain relief is provided. • Swap GIA1 and GIA2, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA1. – Replace original GIA1 if box turns green after swapping units. • Check the GIA1/GRS 79 #1 interconnect wiring and unit connector pins for faults. Replace GRS 79 #1.
		BLACK	GIA1/GRS 79 #1 data path functionality is unknown. Reload GIA1 configuration files.

CHART 5 (Sheet 4 of 8)
FAILED PATH MESSAGES, GIA RS-232 / ARINC 429 CONFIG PAGE (GIA1 AND GIA2)

RESERVED

CHART 5 (Sheet 5 of 8)
FAILED PATH MESSAGES, GIA RS-232 / ARINC 429 CONFIG PAGE (GIA1 AND GIA2)

GIA2 RS-232

Channel	LRU	Indicator	Status
CHNL 1	GDC 72 #2	GREEN	GIA2/GDC 72 #2 data path is functioning correctly.
		RED	GIA2/GDC 72 #2 data path is not functioning correctly. • Load GIA2 and GDC 72 #2 configuration files. • Swap GIA2 and GIA1, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA1. o Replace original GIA2 if box turns green after swapping units. • Check the GIA1/GDC 72 #2 interconnect wiring for faults. Replace GDC 72 #2 if problem remains.
		BLACK	GIA2/GDC 72 #2 data path functionality is unknown. Reload GIA2 configuration files.
CHNL 2	GRS 79 #1 (output only)	BLACK	GIA2/GRS 79 #1 output data path is not monitored. A black box is normal.
CHNL 5	GTX 3X5	GREEN	GIA2/GTX 3X5 #2 data path is functioning correctly.
		RED	GIA2/GTX 3X5 #2 data path is not functioning correctly. • Load GIA2 and GTX 3X5 #2 configuration files. • Swap GIA2 and GIA1, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA1. o Replace original GIA2 if box turns green after swapping units. • Check the GIA2/GTX 3X5 #2 interconnect wiring for faults. Replace GTX 3X5 #2 if problem remains.
		BLACK	GIA2/GTX 3X5 #2 data path functionality is unknown Reload GIA2 configuration files.

CHART 5 (Sheet 6 of 8)
FAILED PATH MESSAGES, GIA RS-232 / ARINC 429 CONFIG PAGE (GIA1 AND GIA2)

GIA2 RS-232

Channel	LRU	Indicator	Status
CHNL 6	GRS 79 #2	GREEN	GIA2/GRS 79 #2 data path is functioning correctly.
		RED	GIA2/GRS 79 #2 data path is not functioning correctly. • Load GIA2 configuration files. • Swap GIA2 and GIA1, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA2. o Replace original GIA2 if box turns green after swapping units. • Swap GRS2 and GRS1 (no reconfiguration required) to confirm if the problem is in the original GRS2. o Replace original GRS2 if box turns green after swapping units. Check the GIA2/GRS 79 #2 interconnect wiring for faults.
		BLACK	GIA2/GRS 79 #2 data path functionality is unknown. Reload GIA2 configuration files.
CHNL 7	GMA 1347	GREEN	GIA2/GMA 1360 data path is functioning correctly.
		RED	GIA2/GMA 1360 data path is not functioning correctly. • Load GIA2 and GMA 1360 configuration files. • Swap GIA2 and GIA1, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA2. o Replace original GIA2 if box turns green after swapping units. Check the GIA2/GMA 1360 interconnect wiring for faults.
		BLACK	GIA2/GMA 1360 data path functionality is unknown. Reload GIA2 configuration files.

CHART 5 (Sheet 7 of 8)
FAILED PATH MESSAGES, GIA RS-232 / ARINC 429 CONFIG PAGE (GIA1 AND GIA2)

RESERVED

CHART 5 (Sheet 8 of 8)
FAILED PATH MESSAGES, GIA RS-232 / ARINC 429 CONFIG PAGE (GIA1 AND GIA2)

GIA2 ARINC 429

Channel	LRU	Indicator	Status
IN 5	GDC 72 #2	GREEN	GIA2/GDC 72 #2 data path is functioning correctly.
		RED	GIA2/GDC 72 #2 data path is not functioning correctly. • Check PFD Alert Window for GIA or GDC configuration or software error messages. Correct any errors before proceeding. • Verify GDC 72 status is OK using the AUX – SYSTEM STATUS page on the MFD. If it has a Red-X, troubleshoot why the GDC is offline before proceeding. • Swap GIA2 and GIA1, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA2. – Replace original GIA2 if box turns green after swapping units. • Check the GIA2/GDC 72 #2 interconnect wiring and unit connector pins for faults. • Replace GDC 72 #2.
		BLACK	GIA2/GDC 72 #2 data path functionality is unknown. Reload GIA1 configuration files.
IN 6	GRS 79 #2	GREEN	GIA2/GRS 79 #2 data path is functioning correctly.
		RED	GIA2/GRS 79 #2 data path is not functioning correctly. • Check PFD Alert Window for GIA or GRS configuration or software error messages. Correct any errors before proceeding. • Verify GRS 79 status is OK using the AUX - SYSTEM STATUS PAGE on the MFD. If it has a Red-X, troubleshoot why the GRS is offline before proceeding. • Ensure GRS 79 connecter is secure and proper wire harness strain relief is provided. • Swap GIA2 and GIA1, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA2. – Replace original GIA2 if box turns green after swapping units. • Check the GIA2/GRS 79 #2 interconnect wiring and unit connector pins for faults. Replace GRS 79 #2.
		BLACK	GIA2/GRS 79 #2 data path functionality is unknown. Reload GIA2 configuration files.

CHART 6 (Sheet 1 of 2)
FAILED PATH MESSAGES, GIA RS-485 CONFIG PAGE (GIA1 AND GIA2)

GIA RS-485 CONFIGURATION Page

GIA1 RS-485

Channel	LRU	Indicator	Status
CHNL 1	GEA1	GREEN	GIA1/GEA1 data path is functioning correctly.
		RED	GIA1/GEA1 data path is not functioning correctly. • Check PFD1 Alert Window for GIA or GEA configuration or software error messages. Correct any errors before proceeding. • Verify GEA1 status is OK using the AUX - SYSTEM STATUS PAGE on the MFD. If it has a Red-X, troubleshoot why the GEA is offline before proceeding. • Swap GIA1 and GIA2, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA1. • Replace original GIA1 if box turns green after swapping units. • Swap GEA1 and GEA2, reconfigure both GEA's to their new locations to confirm if the problem is in the original GEA1. • Replace original GEA1 if box turns green after swapping units. • Check the GIA1/GEA1 interconnect wiring and connector for faults.
		BLACK	GIA1/GEA1 data path functionality is unknown. Reload GIA1 configuration files.
CHNL 2	GEA2	GREEN	GIA1/GEA2 data path is functioning correctly.
		RED	GIA1/GEA2 data path is not functioning correctly. • Check PFD1 Alert Window for GIA or GEA configuration or software error messages. Correct any errors before proceeding. • Verify GEA2 status is OK using the AUX - SYSTEM STATUS PAGE on the MFD. If it has a Red-X, troubleshoot why the GEA is offline before proceeding. • Swap GIA1 and GIA2, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA1. • Replace original GIA1 if box turns green after swapping units. • Swap GEA1 and GEA2, reconfigure both GEA's to their new locations to confirm if the problem is in the original GEA2. • Replace original GEA2 if box turns green after swapping units. • Check the GIA1/GEA2 interconnect wiring and connector for faults.
		BLACK	GIA1/GEA2 data path functionality is unknown. Reload GIA1 configuration files.

CHART 6 (Sheet 2 of 2)
FAILED PATH MESSAGES, GIA RS-485 CONFIG PAGE (GIA1 AND GIA2)

GIA2 RS-485

Channel	LRU	Indicator	Status
CHNL 1	GEA1	GREEN	GIA2/GEA1 data path is functioning correctly.
		RED	GIA2/GEA1 data path is not functioning correctly. • Check PFD1 Alert Window for GIA or GEA configuration or software error messages. Correct any errors before proceeding. • Verify GEA1 status is OK using the AUX - SYSTEM STATUS PAGE on the MFD. If it has a Red-X, troubleshoot why the GEA is offline before proceeding. • Swap GIA1 and GIA2, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA2. • Replace original GIA2 if box turns green after swapping units. • Swap GEA1 and GEA2, reconfigure both GEA's to their new locations to confirm if the problem is in the original GEA1. • Replace original GEA1 if box turns green after swapping units. • Check the GIA2/GEA1 interconnect wiring and connector for faults.
		BLACK	GIA2/GEA1 data path functionality is unknown. Reload GIA2 configuration files.
CHNL 2	GEA2	GREEN	GIA2/GEA2 data path is functioning correctly.
		RED	GIA2/GEA2 data path is not functioning correctly. • Check PFD1 Alert Window for GIA or GEA configuration or software error messages. Correct any errors before proceeding. • Verify GEA2 status is OK using the AUX - SYSTEM STATUS PAGE on the MFD. If it has a Red-X, troubleshoot why the GEA is offline before proceeding. • Swap GIA1 and GIA2, reconfigure both GIA's to their new locations to confirm if the problem is in the original GIA2. • Replace original GIA2 if box turns green after swapping units. • Swap GEA1 and GEA2, reconfigure both GEA's to their new locations to confirm if the problem is in the original GEA2. • Replace original GEA2 if box turns green after swapping units. • Check the GIA2/GEA2 interconnect wiring and connector for faults.
		BLACK	GIA2/GEA2 data path functionality is unknown. Reload GIA2 configuration files.

F. Recommended Tools

The following tools (or equivalent) are recommended to perform various maintenance tasks on the G1000 NXi system.

(1) Calibrated voltmeter capable of measuring 0-32 Volts DC.
(2) #2 Phillips Screwdriver.
(3) 3/32nd inch Hex Tool.
(4) Calibrated digital Level with 0.25 degrees of accuracy capability.
(5) Calibrated VHF NAV/COM/ILS ramp tester.
(6) Calibrated transponder ramp tester including Mode S capability for Mode S transponder equipped aircraft.
(7) Air Data Test Set (ADTS) capable of simulating altitude up to the aircraft's service ceiling.
(8) Headset/Microphone.
(9) Outdoor line-of-site to GPS satellite signals or GPS indoor repeater.
(10) Ground Power Unit (Capable of supplying 28 Vdc).
(11) Calibrated Flight Control Cable Tension Meter or equivalent.
(12) Servo Mount Slip Clutch Adjustment Tool – Garmin P/N T10-00110-01 and a 2 Amp, 24 V, DC Power Supply.
(13) Calibrated digital thermometer suitable for measuring ambient temperature.
(14) Laptop with RS232 emulation software.
(15) M22885/108T8234 extraction tool.
(16) An 0.060" 6-Spline wrench.
(17) WX-PA portable analyzer kit (If Stormscope is installed).
(18) Calibrated torque wrench capable of measuring 0–70 in·lbs.
(19) Standard sockets & wrenches (3/8", 9/16", and 13/16").

G. Inspections

(1) Annual / 100 Hour Inspection

Each 12 months or 100 hours time-in-service, whichever comes first: perform the “Avionics Cooling Fan Fail Annunciation Test” on page 342502121.

(2) 500 Hour Inspection

Perform the following test of the PFD1, MFD, PFD2 and AVIONICS cooling fans:

(a) Turn OFF all electrical components and avionics systems.

(b) Verify the master switch is ON.

(c) Pull the PFD1 FAN, MFD FAN, PFD2 FAN and AVIONICS FAN circuit breakers.

(d) One at a time, push IN each of the four FAN circuit breakers. Verify airflow from each fan by motor sound.

(e) After checking each fan, turn OFF master switch and push IN all four FAN circuit breakers.

(3) 1000 Hour Inspection

Each 1000 hours time-in-service visually inspect each unit listed below for corrosion, damage, or other defects. Replace any damaged parts as required. Inspection may require the temporary removal of a unit or units to gain access to connectors.

(a) GDU 1050 PFD’s & MFD

Remove the PFD & MFD. Inspect the mounting surface and connector for corrosion, heavy oxidation, or other damage.

(b) GMA 1360 Audio Panel

Inspect the GMA 1360 unit, rack, and connectors for corrosion or other defects.

(c) GIA 64 Integrated Avionics Units (2 ea.)

Inspect the GIA 64 #1 and #2 unit, rack, and connectors for corrosion or other defects.

(d) GEA 71B Engine/Airframe Units (2 ea.)

Inspect the GEA 71B unit, rack, and connectors for corrosion or other defects.

(e) GTX 3X5R Mode A, C, S Transponder(s) (if installed)

Inspect the GTX 3X5R unit, rack, and connectors for corrosion or other defects.

(f) GDL 69/69A - Datalink (if installed)

Inspect the GDL 69/69A unit, rack, and connectors for corrosion or other defects.

(g) PFD & MFD Cooling Fans

Inspect both PFD & MFD cooling fans for accumulation of dust or other damage. Remove excess dust as required. Close the AVIONICS FAN circuit breaker and verify that the PFD and MFD cooling fans are functional.

(h) Avionics Cooling Fan

Inspect the avionics cooling fan for accumulation of dust or other damage. Remove excess dust as required. Close the AVIONICS FAN circuit breaker and verify that the avionics cooling fan is functional.

(i) GRS 79 AHRS

Inspect the GRS 79 unit, rack, and connector for corrosion or other defects.

(j) GMU 44B Magnetometer

Remove the GMU 44B and inspect the mounting hardware and GMU 44B for corrosion or other damage.

(k) Perform electrical bond tests on all G1000 NXi equipment. See 51-80-00.

(l) Perform visual checks on the shield terminations for any degradation.

H. Components

CAUTION: ENTER “MAINTENANCE MODE” WHEN ACTIVATING THE GARMIN G1000 NXI SYSTEM ON THE GROUND TO CONDUCT TESTS, INSPECTIONS, ETC. SEE “MAINTENANCE MODE” ON PAGE 3425028.

(1) General

Maintenance of the G1000 NXi system components is “On Condition” only. The following provides basic information regarding unit removal, installation, and troubleshooting. Basic guidance on loading and configuring software is provided under Post Installation Set-up, below. When removing and/or replacing any G1000 NXi component, always ensure that aircraft power is off. Unplug any auxiliary power supplies.

Before removing any G1000 NXi LRU, it is required that the technician verify the LRU software part numbers.

To check an LRU software part number and/or version:

(a) Start the G1000 NXi system in configuration mode as described in “Configuration Mode Overview” on page 34250297.

(b) The System Status page shows a list of LRUs in the LRU window. Activate the cursor and highlight the LRU window.

(c) Use the FMS knob to scroll through the list in the window and select the desired LRU.

(d) The software part number and version is displayed in the DATA window. Compare this to the latest configuration shown in the appropriate Required Equipment List (REL), “Chart 31” on page 342502165.

NOTE: If a faulty LRU is not reporting its software version and part number, check aircraft maintenance logs for last software version loaded and verify against the latest configuration.

(2) GDU 1050 PFD and MFD

See “Figure 11” on page 34250238.

CAUTION: THE GDU 1050’S USE A LENS COATED WITH A SPECIAL ANTI-REFLECTIVE COATING THAT IS VERY SENSITIVE TO SKIN OILS, WAXES AND ABRASIVE CLEANERS. CLEANERS CONTAINING AMMONIA WILL HARM THE ANTI-REFLECTIVE COATING. IT IS VERY IMPORTANT TO CLEAN THE LENS USING A CLEAN, LINT-FREE CLOTH AND AN EYEGLASS LENS CLEANER THAT IS SPECIFIED AS SAFE FOR ANTI-REFLECTIVE COATINGS.

(a) Description

Two Garmin GDU 1050 units, 10.4 inch LCD displays with 1024x768 resolution, are installed in the instrument panel as PFD1 and PFD2; one Garmin GDU 1054 (with GFC 700 w/o Yaw damper) or GDU 1055 (with GFC 700 with Yaw Damper) as the MFD. The PFD displays are located on either side of the MFD, with the stand-by instruments located to the left of the Pilot's PFD (PFD1).

Electrical power to PFD1 is from the Essential Bus #1 or the Emergency Bus, whereas the MFD and PFD2 receive power from Avionics Bus #2.

The GDU units communicate with each other and the GIA 64 units through a high-speed data bus (HSDB) Ethernet connection.

On the panel, an independent flight instrument display provides attitude, altitude, airspeed, heading, VSI and GPS indicators as a fully digital, backup instrument. This is mounted in a vertical configuration to the left of the pilot PFD.

(b) Troubleshooting

See "Chart 7" on page 34250240 and "Chart 8" on page 34250242.



GDU 1050 - PFD and GDU 1054/55 MFD
Figure 11

(c) Removal

1) Using a 3/32nd hex tool, rotate all four ¼-turn fasteners counter-clockwise until they reach their stops.
2) Carefully remove the display from the panel.
3) While supporting the display, disconnect the connector.

(d) Installation

1) Visually inspect the connector and pins for signs of damage. Repair any damage. While supporting the display, connect the connector to the rear of the unit.
2) Carefully insert the display into the panel cutout, ensuring that all 4 ¼-turn fasteners align with the corresponding holes.
3) Seat the display in the panel cutout. Do not use excessive force while inserting the display.
4) Once seated, rotate all four ¼-turn fasteners clockwise to lock the display to the panel.

(e) Return to Service

1) Original Display Reinstalled.

   If the removed display(s) are re-installed in their original positions, no software or configuration loading is required. Continue to “PFD/MFD Test Procedure” on page 342502125.

   NOTE: This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) Original PFD Displays Installed in Opposite Locations for Troubleshooting.

   If the PFD1 and PFD2 are installed in opposite positions, no software or configuration loading is required. Continue to “PFD/MFD Test Procedure” on page 342502125.

3) New, Repaired, or Exchanged Display(s) Installed.

   If a new, repaired, or exchanged GDU 1050 is installed, the correct software and configuration files must be loaded to the unit.

   a) See “G1000 NXi Software/Configuration Procedure” on page 342502107 and “MFD & PFD Software Load” on page 342502108.
   b) If ChartView, Synthetic Vision, or TAWS were previously installed, these must be reactivated. See “Options Unlock Procedures” on page 342502114. If any other options were previously installed, these must also be reactivated per “Optional Equipment Installation” on page 342502109.
   c) Perform “PFD/MFD Test Procedure” on page 342502125.

CHART 7 (Sheet 1 of 2)
TROUBLESHOOTING - GDU 1050

GDU 1050 Troubleshooting / Common Problems	
Symptom	Recommended Action
Display is blank	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Cycle power. ✓ If GDU recovers, observe display for yellow text containing error information at the top of the screen. If message indicates software need to be re-loaded, then re-load software. Otherwise, replace the GDU.
Display resets	• Use a bright light to verify LCD is active. ✓ Adjust avionics dimmer control full clockwise. ✓ Manually turn up backlight on the PFD and load configuration files to the GDU. • Ensure slide lock is fully engaged with the locking tabs on the back of the unit. ✓ If slide lock is not fully engaged, remove connector and verify the locking tabs on the GDU are perpendicular to the connector. If necessary, straighten them before reseating connector.
Display flickers	• Ensure GDU is receiving power. If a circuit breaker is tripped, determine source of short before resetting breaker. • Ensure circuit breakers have not failed and power wire connections are secure. • Swap PFD1 and PFD2. ✓ If problem follows unit, replace the display. Please note the position it failed in (PFD1/2). ✓ If problem does not follow unit, troubleshoot aircraft wiring for fault.
SD card is stuck in GDU	• DO NOT insert a screwdriver of any length into the card slot. • DO NOT pry against the overlay. • DO NOT force the SD Card out. • Use a small screwdriver in the groove on the side of the exposed end of the card to help pull out the card. • Push the card in further to release the card locking mechanism. • Check SD Card for having more than one label. Two or more labels on the card will cause sticking. ✓ Remove all but one label. • Ensure the SD card is from Toshiba or SanDisk. Use of other SD Cards is not recommended. • If card was inserted with the label facing to the right, do not attempt to remove. Return the unit to Garmin for repair.
A button/knob/joystick does not appear to function	• Go to the GDU TEST page in configuration mode and verify button, knob, or joystick operates correctly by observing a change in color from red to green in the button/knob/joystick icon when the button/knob/joystick is pressed. If a button is stuck, the button icon will be green without pressing the button as soon as you turn to the GDU TEST page. ✓ If problem is verified, replace GDU.

CHART 7 (Sheet 2 of 2)
TROUBLESHOOTING - GDU 1050

GDU 1050 Troubleshooting / Common Problems (cont.)	
Symptom	Recommended Action
Terrain/Obstacle/Safetaxi does not display	• Ensure supplemental data cards are inserted correctly in the lower slots of all three GDU's. • Allow the system to verify the data on the cards for approximately five minutes after power-up. • If a database does not activate, reload the problem database onto the SD Card or replace the card.
Display will not track dimmer bus	• Reload GDU configuration files. • If display is a PFD, swap PFD1 and PFD2 to see if problem remains with display. ✓ Replace display if condition remains with the same unit. ✓ If condition remains in original position after swapping displays, check GDU dimmer input to verify voltage is present. • If display is the MFD, check dimmer input to verify voltage is present. ✓ Replace MFD if dimmer voltage is present.
Keyboard will not track dimmer bus	

CHART 8 (Sheet 1 of 6)
MESSAGE ADVISORIES - GDU 1050

GDU 1050 Messages / Software / Configuration Messages		
Failure Message	Cause	Solution
SW MISMATCH – GDU software version mismatch. Xtalk is off.	The system has found the PFDs and/or MFD software versions do not match.	• Load correct software version. See Software Loading procedure.
MANIFEST – PFD 1 software mismatch. Communication Halted.	The system has detected an incorrect software version loaded in the specified PFD.	
MANIFEST – PFD 2 software mismatch. Communication Halted.		
MANIFEST – MFD1 software mismatch. Communication Halted.	The system has detected an incorrect software version loaded in MFD.	
CNFG MODULE – PFD 1 configuration module is inoperative.	The PFD master configuration module has failed.	• Check master configuration module connector and wiring for damage inside the GDU connector backshell. ✓ Replace master configuration module wiring and pins. ✓ If problem persists, replace master configuration module. NOTE New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 NXi System ID number will change to a new number when installing a newmaster config module. The old Terrainand other cards will no longer work asthey will remain locked to the old System ID number.

CHART 8 (Sheet 2 of 6)
MESSAGE ADVISORIES - GDU 1050

GDU 1050 Messages / Software / Configuration Messages (cont.)		
Failure Message	Cause	Solution
MFD1 CONFIG – MFD1 configuration error. Config service req'd.	A configuration mismatch has occurred between the display and the Master Configuration Module.	• Reload the display configuration files from SD Loader Card. • Reload system configuration files by pressing the UPDT CFG softkey on the Configuration Upload Page in the PFD1 System Page Group to load configuration files into the configuration module. ✓ If message persists, check PFD1 config module wiring for faults and replace if necessary. ✓ If issue continues, replace PFD1 master configuration module. ✓ If problem persists, replace the display. NOTE New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 NXi System ID number will change to a new number when installing a newmaster config module. The old Terrainand other cards will no longer work asthey will remain locked to the old System ID number.
PFD 1 CONFIG – PFD 1 configuration error. Config service req'd.		
PFD 2 CONFIG – PFD 2 configuration error. Config service req'd.		

CHART 8 (Sheet 3 of 6)
MESSAGE ADVISORIES - GDU 1050

Database Message Advisories		
Failure Message	Cause	Solution
MFD1 DB ERR – MFD1 navigation database error exists.	The MFD or specified PFD has encountered an error in the navigation database.	• Reload navigation database into the display. • Contact Garmin Technical Support for assistance.
PFD 1 DB ERR – PFD 1 navigation database error exists.		
PFD 2 DB ERR – PFD 2 navigation database error exists.		
MFD1 DB ERR – MFD1 terrain database error exists.	The MFD has encountered an error in the terrain database.	• Confirm supplemental data card is inserted fully. • Reload the database or replace the supplemental datacard. • Contact Garmin Product Support for further assistance if needed.
PFD 1 DB ERR – PFD 1 terrain database error exists.	The specified PFD has encountered an error in the terrain database.	
PFD 2 DB ERR – PFD 2 terrain database error exists.		
MFD1 DB ERR – MFD1 obstacle database error exists.	The MFD has encountered an error in the obstacle database.	• Confirm supplemental data card is inserted fully. • Reload the database or replace the supplemental datacard. • Contact Garmin Product Support for further assistance if needed.
PFD 1 DB ERR – PFD 1 obstacle database error exists.	The specified PFD has encountered an error in the obstacle database.	
PFD 2 DB ERR – PFD 2 obstacle database error exists.		
MFD1 DB ERR – MFD1 airport terrain database error exists.	The MFD has encountered an error in the airport terrain database.	• Confirm supplemental data card is inserted fully. • Reload the database or replace the supplemental datacard. • Contact Garmin Product Support for further assistance if needed.
PFD 1 DB ERR – PFD 1 airport terrain database error exists.	The specified PFD has encountered an error in the airport terrain database.	
PFD 2 DB ERR – PFD 2 airport terrain database error exists.		
DB MISMATCH – Navigation database mismatch. Xtalk is off.	The system has found different navigation database versions or regions (i.e., Americas, International, Atlantic, etc.) installed in the PFDs and/or MFD. Crossfill is off.	• Check/Load current navigation database versions or regions in all displays. • Check database synchronization function complete.
DB MISMATCH – Standby Navigation database mismatch.	The system has found different navigation database versions or regions (i.e., Americas, International, Atlantic, etc.) installed in the PFDs and/or MFD.	• Check/Load current navigation database versions or regions in all displays. • Check database synchronization function complete.

CHART 8 (Sheet 4 of 6)
MESSAGE ADVISORIES - GDU 1050

Database Message Advisories (cont.)		
Failure Message	Cause	Solution
DB MISMATCH – Terrain database mismatch.	The system has found different terrain database versions or regions (i.e., Americas, International, Atlantic, etc.) installed in the PFDs and/or MFD.	• Check/Load current navigation database versions or regions in all displays. • Check database synchronization function complete.
DB MISMATCH – Obstacle database mismatch.	The system has found different obstacle database versions or regions (i.e., Americas, International, Atlantic, etc.) installed in the PFDs and/or MFD.	
DB MISMATCH – Airport Terrain database mismatch.	The system has found different airport terrain database versions or regions (i.e., Americas, International, Atlantic, etc.) installed in the PFDs and/or MFD.	

Cooling Message Advisories		
Failure Message	Cause	Solution
MFD1 COOLING – has poor cooling. Reducing power usage.	MFD1 has exceeded its operating temperature range.	• Check cooling fan and wiring for proper operation (if applicable). ✓ Replace cooling fan if unable to determine if operating correctly. ✓ If problem persists, replace the MFD. • If problem continues contact Garmin Aviation Product Support for assistance.
PFD 1 COOLING – has poor cooling. Reducing power usage.	PFD 1 has exceeded it's operating temperature range.	• Check cooling fan and wiring for proper operation (if applicable). ✓ Replace cooling fan if unable to determine if operating correctly. ✓ If problem persists, replace the PFD1. • If problem continues contact Garmin Aviation Product Support for assistance.
PFD 2 COOLING – has poor cooling. Reducing power usage.	PFD 2 has exceeded it's operating temperature range.	• Check cooling fan and wiring for proper operation (if applicable). ✓ Replace cooling fan if unable to determine if operating correctly. ✓ If problem persists, replace the PFD2. • If problem continues contact Garmin Aviation Product Support for assistance.

CHART 8 (Sheet 5 of 6)
MESSAGE ADVISORIES - GDU 1050

Key Message Advisories		
Failure Message	Cause	Solution
MFD1 “key” KEYSTK – key is stuck.	The SYSTEM has determined a key is stuck on MFD1.	• Go to the GDU TEST page in configuration mode and verify key is stuck (if key is stuck the corresponding indicator will be green). • Exercise suspected stuck key and reset GDU TEST page to see if indicator remains green without pressing the key. ✓ If problem persists replace the display.
PFD 1 “key” KEYSTK – key is stuck.	The system has determined a key is stuck on the PFD 1.	
PFD 2 “key” KEYSTK – key is stuck.	The system has determined a key is stuck on the PFD 2.	

Miscellaneous Message Advisories		
Failure Message	Cause	Solution
XTALK ERROR – A flight display cross talk error has occurred.	A communication error has occurred between the MFD and/or PFDs.	• Ensure a database error has not occurred (identified in the ALERTS window on the PFD). ✓ If a database error has occurred, correct error before proceeding. • Check display Ethernet interconnect wiring. • Replace PFD1 with a known good unit, to verify location of problem: ✓ If problem persists, reinstall original PFD1 and replace PFD2. ✓ If problem persists, reinstall PFD2 and replace MFD.
DATA LOST – Pilot stored data lost. Recheck settings.	Pilot stored data has been lost.	• If the CLR key was held during a power cycle, disregard message. • Cycle power to PFD1: ✓ Ensure CLR key is not stuck on the GDU TEST page. ✓ If problem persists, replace PFD1.

CHART 8 (Sheet 6 of 6)
MESSAGE ADVISORIES - GDU 1050

Miscellaneous Message Advisories		
Failure Message	Cause	Solution
MFD1 SERVICE – needs service. Return unit for repair.	The G1000 NXi has determined MFD1 needs service.	• Ensure the MFD connector is fully seated and locked. • If the unit was started in a very dark environment the photocells may not have enough light to initially raise the CCFT level. Go to the GDU STATUS page in configuration mode, ensure CCFT CRNT 1 & 2 levels are above 50. ✓ If the CCFT levels are not above 50, apply light to the photocell and observe if the CCFT level rises. ✓ If the CCFT level rises, disregard the message. ✓ If the CCFT level does not rise, replace the MFD • Replace the MFD
PFD 1 SERVICE – needs service. Return unit for repair.	The G1000 NXi has determined the specified PFD needs service.	• Ensure the PFD connector is fully seated and locked. • If the unit was started in a very dark environment the photocells may not have enough light to initially raise the CCFT level. Go to the GDU STATUS page in configuration mode, ensure CCFT CRNT 1 & 2 levels are above 50. ✓ If the CCFT levels are not above 50, apply light to the photocell and observe if the CCFT level rises. ✓ If the CCFT level rises, disregard the message. ✓ If the CCFT level does not rise, replace the PFD • Replace the PFD.
PFD 2 SERVICE – needs service. Return unit for repair.		
PFD 1 VOLTAGE – PFD 1 has low voltage. Reducing power usage.	The specified PFD supply voltage is low.	• Check input voltage to PFD. • If input voltage is ok, replace PFD..
PFD 2 VOLTAGE – PFD 2 has low voltage. Reducing power usage.		
MFD1 VOLTAGE – MDF1 has low voltage. Reducing power usage.	The MFD supply voltage is low.	• Check input voltage to MFD. • If input voltage is ok, replace MFD.

(3) GMA 1360 Audio Panel

See “Figure 12”.



GMA 1360 Audio Panel
Figure 12

(a) Description

The Garmin GMA 1360 Audio Panel integrates NAV/COM digital audio, intercom system and marker beacon controls. The GMA 1360 panel provides control of all cockpit intercom/mic systems as well as NAV/COM/ILS audio. The unit also provides display reversion mode control through a large red button. Power to the audio panel is normally received from the Essential Bus #1 when the Battery Master switch is turned on or by applying external power. An Emergency Bus also could supply power when the Emergency Power switch is Armed. The GMA 1360 unit interfaces with the existing marker beacon antenna as well as existing mic and phone jacks and oxygen mask mic.

(b) Troubleshooting

See “Chart 9” on page 34250249 and “Chart 10” on page 34250250.

(c) Removal

1) Using a 3/32nd hex tool, turn the hex nut counter-clockwise until the GMA 1360 is unlocked from its location.
2) Carefully remove the GMA 1360 from its rack.

(d) Installation

1) Visually inspect the connectors using a flashlight to ensure there are no bent or damaged pins. Repair any damage.
2) Gently insert the GMA 1360 into the rack until the locking tab engages the rack.
3) Begin to turn the hex nut clockwise. This draws the unit into the rack until seated. Do not overtighten the nut.

(e) Return to Service

1) Original GMA 1360 Reinstalled.

No software/configuration loading or testing is required if the removed GMA 1360 is reinstalled. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

Continue to the final return-to-service checks.

2) New, Repaired, or Exchanged GMA 1360 Installed.

If a new, repaired, or exchanged GMA 1360 is installed, the correct software and configuration files must be loaded to the unit. See “G1000 NXi Software/Configuration Procedure” on page 342502107.

3) Test

Perform “GMA 1360 Test” on page 342502126.

CHART 9
TROUBLESHOOTING - GMA

Symptom/Failure Message	Recommended Action
Noise in Audio	Most often the cause of the noise is external to the GMA. Try the following to locate the source of the noise before replacing the GMA: • Try a different pair of headsets. Noise cancelling headsets may pick up and/or generate more noise than standard headsets from their own circuitry. • Check for noise with the engine turned off. ✓ If the noise is present only when the engine is running, check the generator and/or ignition system as possible sources of noise (see applicable airframe specific maintenance manual). • Check for noise as all electrical equipment is turned on and off (strobes, other radios, etc.). ✓ If the noise is identified from one electrical system or component refer to the applicable airframe specific maintenance manual. • Ensure the NAV/COM squelch is not open. • Ensure the ADF and DME audio is not active. • Ensure the marker beacon audio is not active. • Ensure the ICS squelch is not open. ✓ Master squelch level can be adjusted on the GMA CONFIGURATION page for higher noise environments. • Replace unit only after all possible external sources of noise are eliminated.
Buttons Do Not Work.	• Some buttons are disabled in the GMA CONFIGURATION page by default. This is to remove potential sources of audio noise for inputs that are not used. If in doubt as to which buttons should be disabled, reload GMA config files and other config files for optional equipment installed in the aircraft (i.e. ADF, HF, etc.) from the loader card.
COM Bleedover	• Verify on the GMA CONFIGURATION page that “Disable Split COM” has a green box. Due to the closeness of the COM antennas and high power of the COM transceivers, Split COM operation is not approved. If the box is black (indicating COM ½ button is active), highlight “Disable Split COM” with the cursor and press the ENT key to turn the box green which will deactivate Split COM mode.
Speaker Cuts Out	• Reduce volume level of the item that caused the speaker to cut out when turned up. A speaker protection circuit disables the speaker output if the volume is too high. If the volume is not sufficient, replace aircraft cabin speaker, reference the Airframe Maintenance Manual.
Mic Audio Heard in Speaker	• Reduce ICS Volume.

CHART 10
MESSAGE ADVISORIES - GMA

Failure Message	Cause	Solution
GMA1 SERVICE – GMA1 needs service. Return unit for repair.	The system has determined that the audio panel needs service.	• Replace GMA.
GMA1 FAIL – GMA1 is inoperative.	The system has detected a failure in the audio panel.	• Ensure GMA1 is receiving power. • Ensure both GIAs are receiving power. • Ensure all GDUs are receiving power. • Ensure the GMA/GIA RS-232 data lines are working properly. • Ensure the GIA/GDU Ethernet data lines are working properly. • Replace GMA.
MANIFEST – GMA1 software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the audio panel.	Load the correct unit software. See Software/Configuration (above) for the Software Loading procedure.
GMA1 CONFIG – GMA1 configuration error. Config service req'd.	The system has detected a audio panel configuration mismatch for the unit.	• Load GMA configuration files. See Software/Configuration (above). ✓ Replace GMA. ✓ If problem persists, replace master configuration module, check config module wiring for faults and replace if necessary. NOTE New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 NXi System ID number will change to a new number when installing a newmaster config module. The old Terrainand other cards will no longer work asthey will remain locked to the old System ID number.

(4) GIA 64 Integrated Avionics Unit

See “Figure 13”.



GIA 64 Integrated Avionics Unit
Figure 13

(a) Description

Two Garmin GIA 64 Integrated Avionics Units (IAUs) contain the VHF COM/NAV receivers, WAAS GPS receiver, Flight Director, and system integration microprocessors. The GIAs also serve as a communication interface to all other G1000 NXi LRUs in the system. Each GIA 64 communicates directly with the on-side GDU 1050 display using a HSDB Ethernet connection. Both GIAs are located directly behind the PFD1 and PFD2, installed into their respective LRU racks. Power is provided to GIA1 by the Essential Bus #1 or the Emergency Bus. GIA2 is powered by Avionics Bus #2. Both GIA 64s interface to the following equipment:

- VOR/LOC/Glideslope Antenna System
- VHF COM #1 & #2 Antennas
- GA 36 GPS/WAAS Antennas
- GDU 1050, #1 & #2
- GMA 1360
- GDC 72 #1 (& GDC 72 #2, if installed)
- GRS 79 #1 (& GRS 79 #2, if installed)
- GEA 71B (Left and Right)
- GTX 3X5

The GIA 64 #1 interfaces to the following additional equipment:

- GFC 700 Autopilot (if installed)
- Fuel Quantity Sensors (left and right)

The GIA 64 #2 interfaces to the following additional equipment:

- Becker ADF 3500
- KN 63 DME (if Installed)
- WX 500 Stormscope (if installed)
- GSR 56 (if installed)

NOTE: The GIA 64 is compatible only with the antennas listed in the GIA 64 Installation Manual, Garmin P/N 190-01912-00.

(b) Troubleshooting

See “Chart 11” on page 34250253 and “Chart 12” on page 34250255.

(c) Removal

1) Remove the MFD as described in “Removal” on page 34250239.
2) Unlock the GIA 64 handle by loosening the Phillips screw on the handle.
3) Pull the handle upward to unlock the GIA 64. Gently remove the unit from the rack.

(d) Installation

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
2) Gently insert the GIA 64 into its rack. The handle should engage the dogleg track.
3) Press down on the GIA 64 hand to lock the unit into the rack.
4) Lock the handle to the GIA 64 body using the Phillips screw.
5) Reinstall the MFD as described in “Installation” on page 34250239.

(e) Return to Service

1) Original GIA 64(s) Reinstalled

No software or configuration loading is required if the removed GIA is reinstalled in its original position (GIA1 and GIA2 in their original racks). This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) Original GIA 64(s) Swapped for Troubleshooting

No software loading is required if the originally installed GIA units are reinstalled in opposite positions (GIA1 and GIA2 in opposite unit racks). However, configuration loading is required. See “G1000 NXi Software/Configuration Procedure” on page 342502107.

3) New, Repaired, or Exchanged GIA 64(s) Installed

If a new, repaired, or exchanged GIA 64 is installed, the correct software and configuration files must be loaded to the unit. See “G1000 NXi Software/Configuration Procedure” on page 342502107

4) Test

Perform “GIA 64 Test” on page 342502128.

Chart 11 (Sheet 1 of 2)
TROUBLESHOOTING - GIA 64

GIA 64 Troubleshooting COM	
Symptom	Recommended Action
Weak COM transmit power	• Switch GIA1 and GIA2, to verify location of problem: ✓ If problem follows unit, replace GIA. ✓ If problem does not follow unit, check COM antenna and cabling for faults.
Weak COM receiver	• Switch GIA1 and GIA2, to verify location of problem: ✓ If problem follows unit, replace GIA. ✓ If problem does not follow unit, check COM antenna and cabling for faults.
No COM sidetone	• Switch GIA1 and GIA2, to verify location of problem: ✓ If problem follows unit, replace GIA. ✓ If problem persists, replace GMA1 with a known good unit.

GIA 64 Troubleshooting NAV	
Symptom	Recommended Action
Weak NAV receiver	• Set up a NAV/COM Ramp Test Set to radiate a test signal. • Switch GIA1 and GIA2, to verify location of problem: ✓ If problem follows unit, replace GIA. ✓ If problem does not follow unit, check NAV antenna, coupler, and cabling for faults.

GIA 64 Troubleshooting G/S	
Symptom	Recommended Action
Weak G/S receiver	• Set up a NAV/COM Ramp Test Set to radiate a test signal. • Switch GIA1 and GIA2, to verify location of problem: ✓ If problem follows unit, replace GIA. ✓ If problem does not follow unit, check NAV antenna, coupler, and cabling for faults.

CHART 11 (Sheet 2 of 2)
TROUBLESHOOTING - GIA 64

GIA 64 Troubleshooting GPS	
Symptom	Recommended Action
Will Not Acquire Satellites	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Using the MFD AUX – GPS Status page, verify the signal strength bars are not erratic. If so, this indicates outside interference is affecting the GPS receivers. Find and remove the source of interference (i.e. cell phones, FBO datalink antennas, etc.). • Check date and time on Date/Time Setup Page. ✓ If date and time are incorrect, enter the correct date and time. • Switch GIA1 and GIA2, to verify location of problem: ✓ If problem follows unit, clear the GPS almanac by performing the following steps – o Using the PFD in config mode, go to the GIA RS-232/ARINC 429 Config Page. o At the top of the screen, select the GIA that can not acquire satellites (GIA1 or GIA2) and press the ENT key. o Press the “CLR NV” softkey at the bottom of the screen. o Select “OK” in the “Clear GIA nonvolatile memory?” pop-up window. o Next reload GIA Audio and Config files from a loader card, see G1000 NXi Software/Configuration Procedure, for instructions. Be sure to reload the config files for any optional equipment installed on the aircraft that require the GIA config to be updated. • Cycle power on the system and allow it to restart in normal mode. Place the aircraft outside and allow 15-30 minutes for the GPS to acquire a position and download a new almanac. ✓ If clearing nonvolatile memory is unsuccessful and the GPS still can not acquire a position, replace the GIA. ✓ If problem does not follow unit, check GPS antenna and cabling.

Chart 12 (Sheet 1 of 6)
MESSAGE ADVISORIES - GIA 64

GIA 64 Message Advisories COM Messages		
Failure Message	Cause	Solution
COM1 SERVICE – COM1 needs service. Return unit for repair.	The system has determined COM1 needs service.	• Replace GIA1 according to instructions in Removal/Installation.
COM2 SERVICE – COM2 needs service. Return unit for repair.	The system has determined COM2 needs service.	• Replace GIA2 according to instructions in Removal/Installation.
COM1 PTT – COM1 push-to-talk key is stuck.	The COM1 external push-to-talk (PTT) switch is stuck in the enabled (or “pressed”) state.	• Press the push-to-talk switch(s) again to cycle its operation. • Check push-to-talk switch(s) and wiring. • Check GIA1/GMA interconnect. • Switch GIA1 and GIA2, to identify whether the unit or connectors/wiring is at fault (Both GIAs must be configured when swapped; see Software/Configuration above): ✓ If problem follows the unit, replace GIA1. ✓ If problem persists replace defective GMA.
COM2 PTT – COM2 push-to-talk key is stuck.	The COM2 external push-to-talk (PTT) switch is stuck in the enabled (or “pressed”) state.	• Press the push-to-talk switch(s) again to cycle its operation. • Check push-to-talk switch(s) and wiring. • Check GIA2/GMA interconnect. • Switch GIA1 and GIA2, to identify whether the unit or connectors/wiring is at fault (Both GIAs must be configured when swapped, see Software/Configuration above): ✓ If problem follows the unit, replace GIA2. ✓ If problem persists replace defective GMA.

CHART 12 (Sheet 2 of 6)
MESSAGE ADVISORIES - GIA 64

GIA 64 Message Advisories COM Messages		
Failure Message	Cause	Solution
COM1 RMT XFR – COM1 remote transfer key is stuck.	The COM1 external remote transfer switch is stuck in the enabled (or “pressed”) state.	Press the COM1 external remote transfer switch again to cycle its operation. Check COM1 external remote transfer switch and wiring. Switch GIA1 and GIA2, to identify whether the unit or connectors/wiring is at fault (Both GIAs must be configured when swapped, see Software/ Configuration above): ✓ If problem follows the unit, replace GIA1. ✓ If problem persists, continue to troubleshoot remote transfer switch & wiring.
COM2 RMT XFR – COM2 remote transfer key is stuck.	The COM2 external remote transfer switch is stuck in the enabled (or “pressed”) state.	Press the COM2 external remote transfer switch again to cycle its operation. Check COM2 external remote transfer switch and wiring. Switch GIA1 and GIA2, to identify whether the unit or connectors/wiring is at fault (Both GIAs must be configured when swapped, see Software/ Configuration above): ✓ If problem follows the unit, replace GIA2. ✓ If problem persists, continue to troubleshoot remote transfer switch & wiring.
COM1 TEMP – COM1 over temp. Reducing transmitter power.	The specified COM transceiver is reporting a high temperature condition and is reducing transmit power to prevent damage.	• Check fan, wiring and air tubing for proper operation (if applicable). • Replace cooling fan if unable to determine if operating correctly. • Replace GIA. • If problem persists contact Garmin Aviation Product Support for assistance.
COM2 TEMP – COM2 over temp. Reducing transmitter power.		

CHART 12 (Sheet 3 of 6)
MESSAGE ADVISORIES - GIA 64

GIA Message Advisories NAV Messages		
Failure Message	Cause	Solution
NAV1 SERVICE – NAV1 needs service. Return unit for repair.	The system has detected a failure in NAV1 receiver.	• Replace GIA1.
NAV2 SERVICE – NAV2 needs service. Return unit for repair.	The system has detected a failure in NAV2 receiver.	• Replace GIA2.
NAV1 RMT XFR – NAV1 remote transfer key is stuck.	The NAV1 external remote transfer switch is stuck in the enabled (or “pressed”) state.	Press the NAV1 external remote transfer switch again to cycle its operation. Check NAV1 remote transfer switch and wiring. Switch GIA1 and GIA2, to identify whether the unit or connectors/wiring is at fault (Both GIAs must be configured when swapped, see Software/ Configuration above): ✓ If problem follows unit, replace GIA1. ✓ If problem persists, continue to troubleshoot remote transfer switch & wiring.
NAV2 RMT XFR – NAV2 remote transfer key is stuck.	The NAV2 external remote transfer switch is stuck in the enabled (or “pressed”) state.	Press the NAV2 external remote transfer switch again to cycle its operation. Check NAV2 remote transfer switch and wiring. Switch GIA1 and GIA2, to identify whether the unit or connectors/wiring is at fault (Both GIAs must be configured when swapped, see Software/ Configuration above): ✓ If problem follows unit, replace GIA1. ✓ If problem persists, continue to troubleshoot remote transfer switch & wiring.

CHART 12 (Sheet 4 of 6)
MESSAGE ADVISORIES - GIA 64

GIA Message Advisories Glideslope Messages		
Failure Message	Cause	Solution
G/S1 SERVICE – G/S1 needs service. Return unit for repair.	The system has detected a failure in G/S1 receiver.	• Replace GIA1.
G/S2 SERVICE – G/S2 needs service. Return unit for repair.	The system has detected a failure in G/S2 receiver.	• Replace GIA2.
G/S1 FAIL – G/S1 is inoperative.	The system has detected a failure in G/S1 receiver.	• Switch GIA1 and GIA2 to verify location of problem: ✓ If problem follows the unit, replace GIA. ✓ If problem does not follow unit, check G/S1 antenna and cabling.
G/S2 FAIL – G/S2 is inoperative.	The system has detected a failure in G/S2 receiver.	• Switch GIA1 and GIA2 to verify location of problem: ✓ If problem follows the unit, replace GIA. ✓ If problem does not follow unit,. check G/S2 antenna and cabling.

GIA Message Advisories GPS Messages		
Failure Message	Cause	Solution
MANIFEST – GPS1 software mismatch. Communication halted.	The system has detected an incorrect software version loaded in GIA1.	• Load the correct GPS software. See Software/Configuration above): for the Software Loading procedure.
MANIFEST – GPS2 software mismatch. Communication halted.	The system has detected an incorrect software version loaded in GIA2.	
GPS1 SERVICE – GPS1 needs service. Return unit for repair.	The system has detected a failure in GPS1 receiver.	• Replace GIA.
GPS2 SERVICE – GPS2 needs service. Return unit for repair.	The system has detected a failure in GPS2 receiver.	
GPS1 FAIL – GPS1 is inoperative.	The system has detected a failure in GPS1 receiver.	• Switch GIA1 and GIA2, to verify location of problem: ✓ If problem follows the unit, replace GIA. ✓ If problem does not follow the unit, check GPS antenna and cabling.
GPS2 FAIL – GPS2 is inoperative.	The system has detected a failure in GPS2 receiver.	
NOTE Before troubleshooting, ensure that no cell phones or devices using cell phone technology are turned on, even in a monitoring state, in the cabin.		

CHART 12 (Sheet 5 of 6)
MESSAGE ADVISORIES - GIA 64

GIA Message Advisories GPS Messages (cont.)		
Failure Message	Cause	Solution
LOI – GPS integrity lost. Crosscheck with other NAVS.	If the primary receiver is a WAAS sensor, the alert indicates that GPS position data has timed out.	• Verify the area the aircraft was traveling through did not have loss of GPS coverage. FAA NOTAMs may be issued for periods of outages, or the US Coast Guard website http://www.navcen.uscg.gov/ will have notices posted. • Using the MFD AUX – GPS Status page, verify the signal strength bars are not erratic. If so, this indicates outside interference is affecting the GPS receivers. Find and remove the source of interference (i.e. cell phones, FBO datalink antennas, etc.). • If GPS receivers can not aquire a position lock, troubleshoot per the "Will not acquire satellites" section.
GPS NAV LOST – Loss of GPS navigation. Insufficient satellites.	There is no GPS position fix available or the system is in dead reckoning mode.	
GPS NAV LOST – Loss of GPS navigation. Position error.	The G1000 NXi has detected an internal position warning has occurred.	
GPS NAV LOST – Loss of GPS navigation. GPS fail.	The G1000 Nxi has detected a GPS engine failure.	

GIA Message Advisories GIA Cooling Messages		
Failure Message	Cause	Solution
GIA1 COOLING – GIA1 temperature too low.	GIA1 operating temperature is too low.	• Allow unit to warm up.
GIA2 COOLING – GIA2 temperature too low.	GIA2 operating temperature is too low.	• Allow unit to warm up.
GIA1 COOLING – GIA1 over temperature.	GIA1 has exceeded its operating temperature range.	• Check fan, wiring and air tubing for proper operation (if applicable). • Replace cooling fan if unable to determine if operating correctly. • Replace GIA1. • If problem persists contact Garmin Aviation Product Support for assistance.
GIA2 COOLING – GIA2 over temperature.	GIA2 has exceeded its operating temperature range.	• Check fan, wiring and air tubing for proper operation (if applicable). • Replace cooling fan if unable to determine if operating correctly. • Replace GIA2. • If problem persists contact Garmin Aviation Product Support for assistance.

CHART 12 (Sheet 6 of 6)
MESSAGE ADVISORIES - GIA 64

GIA Message Advisories GIA Configuration Messages		
Failure Message	Cause	Solution
MANIFEST – GIA1 software mismatch. Communication Halted.	The system has detected an incorrect software version loaded in GIA1.	• Load the correct software. See Software/Configuration (above) for the Software Loading procedure.
MANIFEST – GIA2 software mismatch. Communication Halted.	The system has detected an incorrect software version loaded in GIA2.	
GIA1 CONFIG – GIA1 audio config error. Config service req'd.	The GIA's audio configuration files are incorrect or missing.	• Reload audio configuration files.
GIA2 CONFIG – GIA2 audio config error. Config service req'd.		
GIA1 CONFIG – GIA1 configuration error. Config service req'd.	The system has detected a GIA configuration mismatch. If GIAs are not properly configured after being swapped/replaced, this message appears.	• Load the configuration files for that GIA. See Software/Configuration (above) for GIA 63W Configuration Loading procedure. • If problem persists, replace master configuration module, check config module harness for faults and replace if necessary. NOTE New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 NXi System ID number will change to a new number when installing a newmaster config module. The old Terrainand other cards will no longer work asthey will remain locked to the old System ID number.
GIA2 CONFIG – GIA2 configuration error. Config service req'd.		
HW MISMATCH – GIA hardware mismatch, GIA1 communication halted.	The G1000 has detected a non-WAAS GIA 63.	• Replace GIA with a WAAS unit.
HW MISMATCH – GIA hardware mismatch, GIA2 communication halted.		
GIA1 SERVICE – GIA1 needs service. Return unit for repair.	The G1000 has detected a failure in the specified GIA.	• Replace suspect GIA.
GIA2 SERVICE – GIA2 needs service. Return unit for repair.		

(5) GEA 71B Engine/Airframe Unit

See “Figure 14”.



GEA 71B Engine/Airframe Unit
Figure 14

(a) Description

The Garmin GEA 71B Engine/Airframe Unit (1 per engine) provides engine/airframe data to the G1000 NXi system. Data received from transducers/sensors is processed and sent to GIA 64s (via RS-485 digital interface), and subsequently to the GDU 1050 MFD. Engine parameters are normally displayed on the MFD. In the event of a MFD failure, all engine instruments can be displayed on the PFD’s. The GEAs are located behind PFD1 and MFD and are mounted in a vertical orientation. Power is received from the Essential buses or the Emergency bus. The GEA interfaces to the following:

- Manifold Pressure Sensor (MAP)
- Tachometer Sensor (RPM)
- Turbine Inlet Temperature (TIT) Sensor
- Fuel Flow Sensor
- Oil Pressure Sensor
- Oil Temperature Sensor
- Fuel Senders
- Flap Position potentiometer.
- Alternator & Battery Current Monitor Shunts
- Six Cylinder Head Temperature (CHT) Sensors
- Pitot Heat System
- Vacuum Transducer.
- Door Ajar switch

(b) Troubleshooting

See “Chart 13” on page 34250263.

(c) Removal

1) Remove the PFD/MFD as described in “Removal” on page 34250239.
2) Unlock the GEA 71B handle by unscrewing the Phillips screw.
3) Pull the handle upward to unlock the GEA 71B.
4) Gently remove the GEA 71B from its rack.

(d) Installation

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
2) Gently insert the GEA 71B into the rack. The handle should engage the dogleg track.
3) Press down on the handle to lock the unit into place.
4) Lock the handle to the GEA 71B body using the Phillips screw.
5) Reinstall the PFD/MFD as described in “Installation” on page 34250239.

(e) Return to Service

1) Original GEA 71B Reinstalled

No software or configuration loading is required if the removed GEA 71B is reinstalled. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) New, Repaired, or Exchanged GEA 71B Installed

3) If a new, repaired, or exchanged GEA 71B is installed, the correct software and configuration files must be loaded to the unit. See “G1000 NXi Software/Configuration Procedure” on page 342502107.

4) Test

Perform “GEA 71B Test” on page 342502130.

(6) GEA 71B Backshell Thermocouple

See “Figure 15”.

(a) Removal

1) Remove GEA 71B as described above.

2) Remove GEA connector backplate.

3) Remove connector J701 (5) from the backplate.

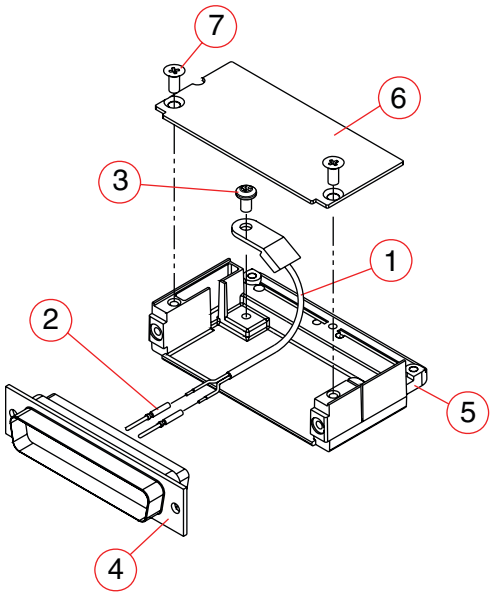
4) Remove cover (6) from the backshell.

5) Unscrew thermocouple from boss on backshell. Extract the thermocouple pins from the connector.

(b) Installation

1) Crimp pins (2) onto each of the thermocouple wires (1). Ensure that pre-stripped wire length is 1/8” prior to crimping.

2) Insert newly crimped pins and wires into the appropriate connector housing location (4) as specified in the electrical schematic.



GEA 71B Backshell Thermocouple
Figure 15

3) Place thermocouple body (1) onto the backshell boss (5). Place the thermocouple as shown in “Figure 15” so that the wires exit towards the bottom of the backshell.

4) Fasten thermocouple tightly to backshell using the provided screw (3).

5) Fasten cover (6) to backshell using the provided screws (7).

CHART 13
TROUBLESHOOTING / MESSAGE ADVISORIES - GEA

Failure Message	Cause	Solution
MANIFEST – GEA1 software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the specified GEA 71B.	Load the correct software version. See GEA 71B Software Load Procedure.
MANIFEST – GEA2 software mismatch. Communication halted.		
GEA1 CONFIG – GEA1 configuration error. Config service req'd.	The system has detected a configuration mismatch in the specified GEA 71B.	• Load GEA configuration files. See GEA 71B Configuration Procedure. • If problem persists, replace master configuration module, check config module harness for faults and replace if necessary. NOTE New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 NXi System ID number will change to a new number when installing a newmaster config module. The old Terrainand other cards will no longer work asthey will remain locked to the old System ID number.
GEA2 CONFIG – GEA2 configuration error. Config service req'd.		

(7) GTX 3X5R Mode A, C, S Transponder

See “Figure 16”.



GTX 3X5R Mode A, C, S Transponder
Figure 16

(a) Description

The Garmin GTX 3X5R provides Mode A, C, and S altitude and position reporting information from the G1000 NXi system. The unit is mounted directly behind the MFD on the center avionics shelf. Power is received from Essential Bus #1. Similarly to the GEA 71B, the GTX 3X5R sends data via RS-232 directly to a GIA 64. Information is then sent to the PFD, where the pilot can control the transponder. The GTX 3X5R is connected to both GIA 64s for redundant communications. The GTX 3X5R interfaces with the transponder antenna. In addition the GTX 3X5R (optional) provides all the same features plus ADS-B in as well as well as ADS-B in (UAT, TIS-B, & FIS-B).

(b) Troubleshooting

See “Chart 14” on page 34250265.

(c) Removal

1) Remove the MFD as described in “Removal” on page 34250239.
2) Unlock the GTX 3X5R handle by loosening the Phillips screw on the handle.
3) Pull the handle upward to unlock the GTX 3X5R. Gently remove the unit from the rack.

(d) Installation

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
2) Gently insert the GTX 3X5R into its rack. The handle should engage the dogleg track.
3) Press down on the GTX 3X5R handle to lock the unit into the rack.
4) Lock the handle to the GTX 3X5R body using the Phillips screw.
5) Reinstall the MFD as described in “Installation” on page 34250239.

(e) Return to Service

1) Original GTX 3X5R is Reinstalled

No software or configuration loading is required if the removed GTX 3X5R is re-installed. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) New, Repaired, or Exchanged GTX 3X5R is Installed

If a new, repaired, or exchanged GTX 3X5R is installed, the correct software and configuration files must be loaded to the unit. See “G1000 NXi Software/Configuration Procedure” on page 342502107, reload the software, and then enter the aircraft registration number per “Flight Instrumentation” on page 3425023.

3) Test

Perform “Test” on page 342502130.

CHART 14
TROUBLESHOOTING / MESSAGE ADVISORIES - GTX 3X5R

Failure Message	Cause	Solution
MANIFEST – GTX1 software mismatch. Communication halted. MANIFEST – GTX2 software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the specified GTX 3X5R.	• Reload software. See Software Load / Configuration Procedure.
XPDR1 CONFIG – XPDR1 configuration error. Config service req'd. XPDR2 CONFIG – XPDR2 configuration error. Config service req'd.	The system has detected a configuration mismatch for the specified GTX 3X5R.	• Perform a SET>ACTV configuration reset on the GTX Config page and verify the aircraft registration is present. • If error is still present, reload config files from a loader card. ✓ If problem persists, replace master configuration module, check config module harness for faults and replace if necessary. NOTE New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 NXi System ID number will change to a new number when installing a newmaster config module. The old Terrainand other cards will no longer work asthey will remain locked to the old System ID number.
XPDR1 SRVC – XPDR1 needs service. Return unit for repair. XPDR2 SRVC – XPDR2 needs service. Return unit for repair.	The G1000 has detected a failure in the specified GTX 3X5R.	• Replace GTX 3X5R.
XPDR1 FAIL – XPDR 1 is inoperative. XPDR2 FAIL – XPDR 2 is inoperative.	The specified GTX 3X5R is not responding.	• Check wiring between GIA's and GTX. • Replace GTX 3X5R.

(8) GDC 72 Air Data Computer

See “Figure 17”.

(a) Description

The Garmin GDC 72 computer(s) compile information from the pitot/static system and the GTP 59 outside air temperature (OAT) probe(s) to digital air data computations to the G1000 NXi system. The GDC 72 communicates with the GIA 64, GDU 1050, and GRS 79 using ARINC 429 digital interface. The unit(s) are mounted behind PFD2. Power is provided to #1 GDC 72 unit (ADC1) from the Essential Bus #1 or the Emergency Bus. The #2 GDC 72 unit (ADC2) is powered by the Avionics bus #2.



GDC 72 Air Data Computer
Figure 17

NOTE: If any connections in the pitot/static system are opened for maintenance, the entire system must be rechecked per Pitot and Static Systems, “Test” on page 34108.

(b) Troubleshooting

See “Chart 15” on page 34250268.

(c) Removal

1) Remove the PFD2 as described in “Removal” on page 34250239.
2) Disconnect the pitot/static plumbing from the rear of the unit. Disconnect the single connector.
3) Loosen each thumbscrew on the hold-down clamp and remove the clamp.
4) Carefully remove the unit from its mount.

(d) Installation

1) Place the unit in the mounting tray.
2) Position the locking clamp and fasten using the thumbscrews.
3) Connect the pitot/static plumbing.
4) Inspect the connector and pins for damage. Repair any damage. Connect the connector to the unit.
5) Reinstall the PFD2 as described in “Installation” on page 34250239.

(e) Return to Service

NOTE: If any connections in the pitot / static system are opened for maintenance, the entire system must be rechecked per Pitot and Static System “Test” on page 34108.

1) Original GDC 72 is Reinstalled

No software or configuration loading is required if the removed GDC 72 is reinstalled. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) Original GDC 72 Installed in Opposite Locations for Troubleshooting

No software loading is required if the original GDC1 and GDC2 are installed in opposite locations.

3) New, Repaired, or Exchanged GDC 72 is Installed

If a new, repaired, or exchanged GDC 72 is installed, the correct software and configuration files must be loaded to the unit. See “G1000 NXi Software/Configuration Procedure” on page 342502107.

4) New GDC 72 Configuration Module is Installed

The correct configuration files must be loaded if the GDC 72 configuration module has been replaced. See “G1000 NXi Software/Configuration Procedure” on page 342502107.

5) Test

Perform “GDC 72 Tests” on page 342502132.

CHART 15
TROUBLESHOOTING / MESSAGE ADVISORIES - GDC 72

Symptom	Recommended Action
Altitude is different than standby altimeter	• Perform a pitot/static check per 34-10 Pitot Static Systems in this manual. Allow the GDC to warm up for fifteen minutes before checking accuracy, per Garmin Service Advisory 0606. • Determine which instrument is outside limits and replace. The GDC may be field adjusted per Garmin Service Advisory 0720. Note: Both units may individually be in spec but show a difference in altitude. Do not return the GDC to Garmin for service if not outside limits.
GDC Config file does not load.	• Replace GDC config module. ✓ If problem persists, replace GDC config module wire harness.

Failure Message	Cause	Solution
MANIFEST – GDC1 software mismatch Communication halted.	The system has detected an incorrect software version loaded in the specified GDC 72.	• Load correct software version. See Software Configuration Procedures.
MANIFEST – GDC2 software mismatch Communication halted.		
ADC1 ALT EC – ADC1 altitude error correction is unavailable.	The specified GDC 72 is reporting the absence of altitude error correction.	
ADC2 ALT EC – ADC2 altitude error correction is unavailable.		
ADC1 AS EC – ADC1 airspeed error correction is unavailable.	The specified GDC 72 is reporting the absence of airspeed error correction.	
ADC2 AS EC – ADC2 airspeed error correction is unavailable.		

(9) GTP 59 OAT Probe

See “Figure 18”.

(a) Description

The Garmin GTP 59 OAT Probe provides the GDC 72 with air temperature data. The OAT probe(s) are mounted to the underside of each wing, outboard of W.S. 164.72.

(b) Troubleshooting

See “Chart 1” on page 34250212.

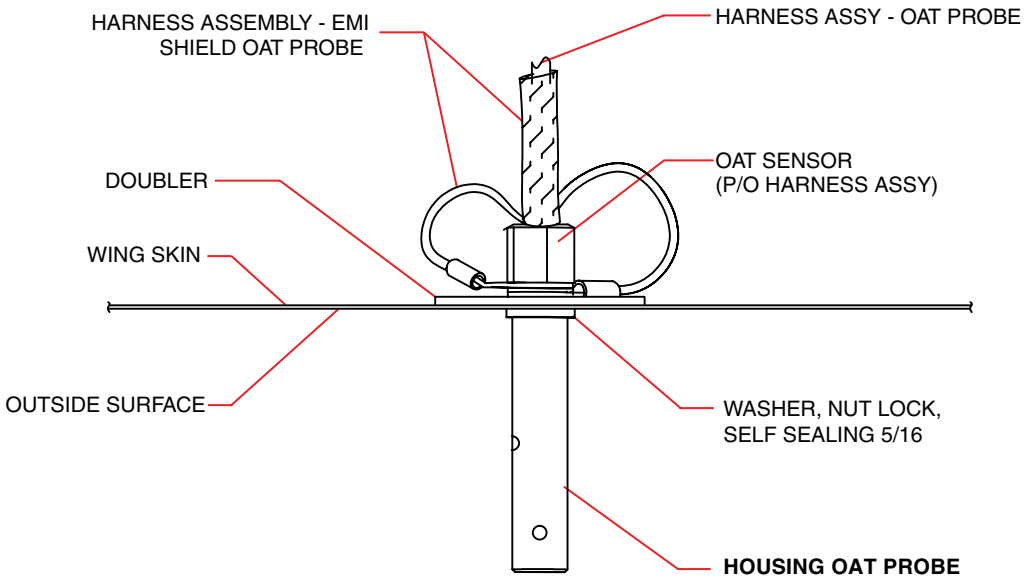
(c) Removal

1) Remove the access cover.
2) Disconnect the connector.
3) Use an open-end wrench to hold the probe in place on the inside of the wing. On the outside of the wing, loosen the GTP 59 Housing and remove the GTP 59.

(d) Installation

1) Place the GTP-59 grounding strap(s) O-ring connector(s) over the OAT probe end and insert the probe through the wing from the inside. Secure with washer and housing on the outside of the wing. Use an open-end wrench to hold the GTP-59 on the inside and torque the housing on the outside to 100 ± 20 in·lbs.
2) Reconnect the connector to the airplane wiring harness.
3) Reinstall access cover adjacent to the OAT probe.
4) Verify ambient temperature OAT display on each PFD.

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GTP 59 OAT Probe
Figure 18

(10) GRS 79 Attitude and Heading Reference System

See “Figure 19”.



GRS 79 AHRS Unit
Figure 19

(a) Description

The Garmin GRS 79 AHRS unit(s) provide attitude and heading information to the G1000 NXi system. The units, mounted below the aft passenger seats, contain advanced tilt sensors, accelerometers, and rate sensors. The unit interfaces with the GDC 72 and GMU 44B Magnetometer and utilizes GPS signals from the GIA 64s. Actual attitude and heading information is sent using ARINC 429 digital interface to both GDU 1050s and GIA 64s. Power is provided to #1 GRS 79 unit (AHRS 1) from the Essential Bus #1 or the Emergency Bus. The #2 GDC 72 (AHRS 2) is powered by the Avionics Bus #2. The GRS 79 interfaces with and provides power to the GMU 44B Magnetometer. The GRS 79 supplies attitude and heading information directly to the PFDs, MFD, and GIAs.

NOTE: GRS 79 Earth Magnetic Field Updates

The GRS 79 utilizes an Earth magnetic field model which is updated once every five years. The update is expected to be available from Garmin in each of the following years: 2010, 2015, and every five years thereafter, as long as the GRS 79 remains a Garmin-supported product. The G1000 NXi system alerts the operator that the magnetic field database is out of date by issuing the message “AHRS SERVICE – AHRS Magnetic-field model needs update”. Garmin will distribute update instructions when updates are available.

(b) Troubleshooting

See “Chart 16” on page 34250272 and “Chart 17” on page 34250273.

(c) Removal

1) Remove aft passenger seats (See 25-10-00).
2) Disconnect the GRS 79 AHRS connector.
3) Remove the four Phillips thumbscrews with a screwdriver and set them aside.
4) Gently lift the GRS 79 from the mounting plate. (If the mounting plate is removed, the GRS 79 must be re-calibrated. See”Post-Installation Procedures” on page 342502119 and “LRU and System Test Procedures” on page 342502125.)

(d) Installation

1) Level the airplane per 8-20-00.
2) Verify GRS 79 bracket assembly is level within 3.0 degrees.
3) Place the GRS 79 on the mounting plate, ensuring the general orientation is correct.
4) Align forward direction of the GRS 79 in heading to within 1.0 degree of the aircraft heading.
5) Fasten the unit to the plate using the Phillips thumbscrews. Recommended torque is 22-25 inch pounds.
6) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage. Connect the connector to the GRS 79, ensuring that each slidelock is secured on both sides.
7) Reinstall aft passenger seats (See 25-10-00).

(e) Return to Service

1) Original GRS 79 is Reinstalled

If the original GRS 79 was reinstalled, then no software loading is required. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

If the GRS rack was not removed or loosened, continue to Test, below.

If the GRS rack was removed or loosened, continue to the “GRS/GMU Calibration Procedures” on page 342502134.

2) Original GRS 79 Installed in Opposite Locations for Troubleshooting

If the original GRS1 and GRS2 are installed in opposite locations (GRS1 and GRS2 in opposite unit racks), no software loading is required.

If the GRS rack(s) was not removed or loosened, continue to Test, below.

If the GRS rack(s) was removed or loosened, continue to the “GRS/GMU Calibration Procedures” on page 342502134.

3) New, Repaired, or Exchanged GRS 79 is Installed

If a new, repaired, exchanged GRS 79 is installed, then software must be loaded and the unit must be calibrated. See “G1000 NXi Software/Configuration Procedure” on page 342502107. Then, proceed as follows:

If the GRS rack(s) was not removed or loosened, continue to Test, below.

If the GRS rack(s) was removed or loosened, continue to the “GRS/GMU Calibration Procedures” on page 342502134.

4) New GRS 79 Configuration Module is Installed

If the GRS 79 configuration module was replaced, no software loading is required, but the unit must be recalibrated. Continue to the “GRS/GMU Calibration Procedures” on page 342502134.

5) Test

Perform “GRS/GMU Test” on page 342502142.

CHART 16
TROUBLESHOOTING - GRS 79 (AHRS) / GMU 44B

Symptom	Recommended Action
AHRS does not complete initialization	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Ensure GPS has acquired at least four satellites, has a 3D navigation solution, and a DOP of less than 5.0. This is particularly important if this issue appears during ground operation only. • Calibrate the GRS 79. • Check GRS 79 configuration module wiring for damage. • Check GRS 79 connector for bent pins. ✓ If no damage can be found, replace GRS 79 configuration module. ✓ If problem persists, replace the GRS 79.
Attitude appears unstable	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Ensure the four GRS 79 mounting screws are tight. Finger tight is not sufficient, a screwdriver must be used to verify. • Ensure mounting rack and airframe shelf are secure and all hardware and brackets are present (CAUTION - do not loosen the mounting rack hardware to the airframe shelf or the aircraft will need to be re-leveled and the PITCH/ROLL OFFSET procedure performed). • Ensure GRS 79 connector is securely fastened and proper strain relief is provided. • Remove GRS 79 connector and verify there are no bent pins. • Replace the GRS 79. • Contact Garmin for further troubleshooting if required.

CHART 17 (Sheet 1 of 2)
MESSAGE ADVISORIES - GRS / GMU

Failure Message	Cause	Solution
MANIFEST – GRS1 software mismatch. Communication halted. MANIFEST – GRS2 software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the specified GRS 79.	• Load correct software version. See Software Configuration Procedures.
AHRS1 SRVC – AHRS1 magnetic-field model needs update. AHRS2 SRVC – AHRS2 magnetic-field model needs update.	The AHRS magnetic field model should be updated for the specified unit. Appears on ground only.	• Load updated AHRS magnetic field file. See GRS 79 / Components for AHRS magnetic field update information.
GEO LIMITS – AHRS1 too far north/south, no magnetic compass. GEO LIMITS – AHRS2 too far north/south, no magnetic compass.	No magnetic compass information available due to being too far north or south.	• Operate the aircraft only within the limits as specified in the POH.
AHRS1 TAS – AHRS1 not receiving airspeed. AHRS2 TAS – AHRS2 not receiving airspeed.	The specified GRS 79 is not receiving airspeed from the GDC 72.	• Check GRS/GDC interconnect for faults. • Replace the GDC 72: • If problem persists, replace the GRS 79.
AHRS1 GPS – AHRS1 not receiving backup GPS information.	The GRS 79 #1 is not receiving backup GPS information from either GIA 64.	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Check GPS status for GIA 1 and 2 on MFD - AUX GPS STATUS page. If one or both GPS receivers cannot acquire a position lock, see GPS troubleshooting section. • Troubleshoot GIA1/2 –GRS1 wiring for faults (ref. Failed Path troubleshooting section). • Replace the GRS 79 #1.
AHRS1 GPS – AHRS1 operating exclusively in no-GPS mode.	The GRS 79 #1 is operating in the absence of GPS.	
AHRS1 GPS – AHRS1 not receiving any GPS information.	The GRS 79 #1 is not receiving GPS data from the GPS receivers.	
AHRS1 GPS – AHRS1 using backup GPS source.	The GRS 79 #1 is using the backup GPS data path.	

CHART 17 (Sheet 2 of 2)
MESSAGE ADVISORIES - GRS / GMU

GRS Message Advisories (Cont.)		
Failure Message	Cause	Solution
AHRS2 GPS – AHRS2 not receiving backup GPS information.	The GRS 79 #2 is not receiving backup GPS information from either GIA 64.	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Check GPS status for GIA 1 and 2 on MFD - AUX GPS STATUS page. If one or both GPS receivers cannot acquire a position lock, see GPS troubleshooting section. • Troubleshoot GIA1/2 –GRS2 wiring for faults (ref. Failed Path troubleshooting section). • Replace the GRS 79 #2.
AHRS2 GPS – AHRS2 operating exclusively in no-GPS mode.	The GRS 79 #2 is operating in the absence of GPS.	
AHRS2 GPS – AHRS2 not receiving any GPS information.	The GRS 79 #2 is not receiving GPS data from the GPS receivers.	
AHRS2 GPS – AHRS2 using backup GPS source.	The GRS 79 #2 is using the backup GPS data path.	
AHRS MAG DB – AHRS magnetic model database version mismatch.	The G1000 NXi has detected a magnetic model database version mismatch.	• Load AHRS magnetic model field in both GRS units.
MAG VAR WARN – Large magnetic variance. Verify all course angles.	Magnetic variance value from GMU 44B is not accurate.	• If flying near large radio towers or other sources of possible electromagnetic interference, the condition should correct itself as the aircraft leaves the area. • If problem persists, run magnetic interference check in LRU Test Procedures (below) to check for magnetic interference in the aircraft.
MANIFEST – GMU1 software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the specified GMU 44B.	Load the correct software version. See G1000 NXi Software/Configuration Procedure, for the Software Load Procedure.
MANIFEST – GMU2 software mismatch. Communication halted.		
HDG FAULT – AHRS1 magnetometer fault has occurred.	A fault has occurred in the specified magnetometer; heading will be flagged invalid.	• Check GMU 44B/GRS 79 interconnect for faults. ✓ Replace GMU 44B. ✓ If problem persists, replace GRS 79.
HDG FAULT – AHRS2 magnetometer fault has occurred.		

(11) GMU 44B Magnetometer

See “Figure 20”.

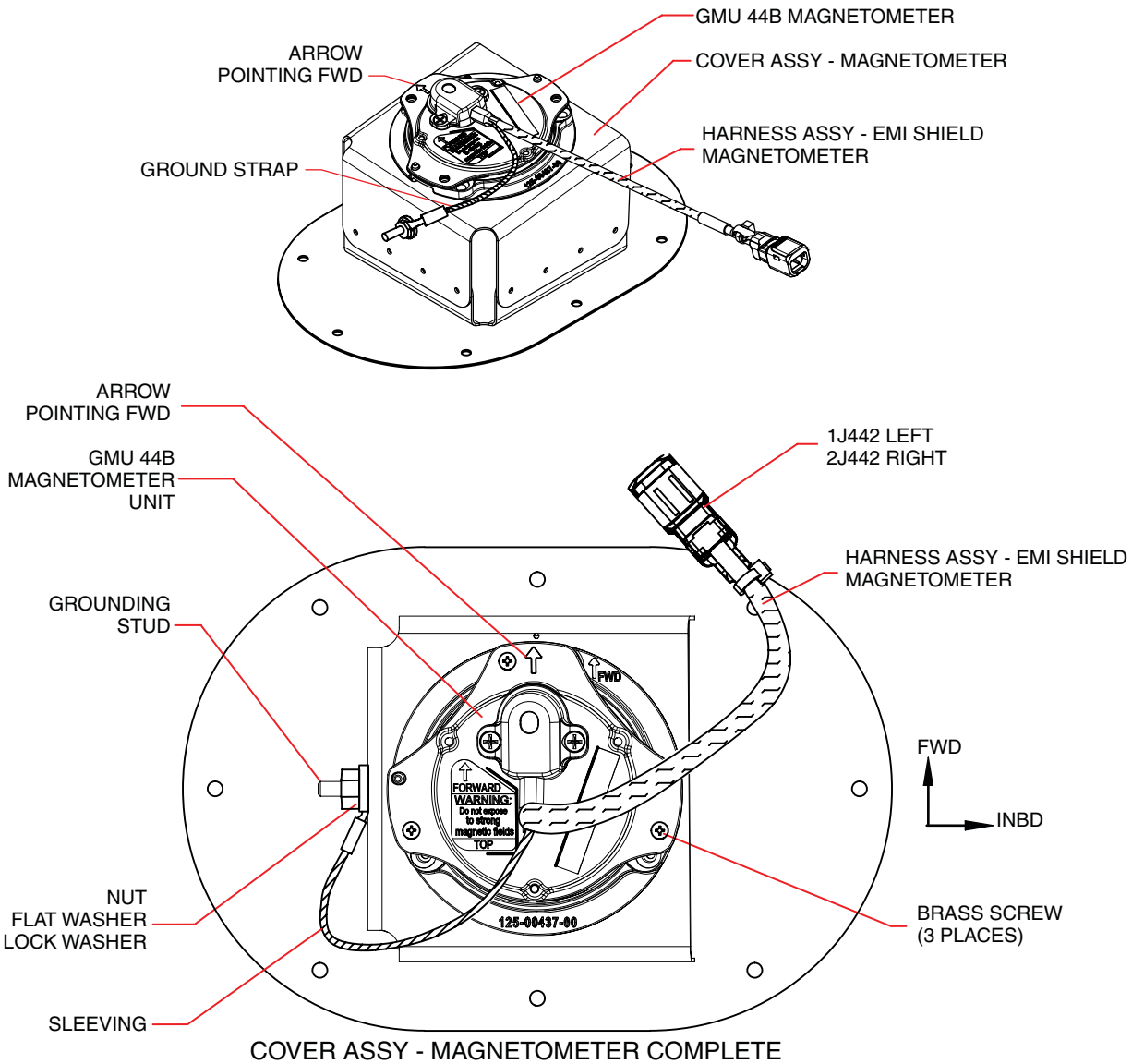
(a) Description

The GMU 44B provides horizontal and vertical magnetic field information to the GRS 79 AHRS. This allows heading to be calculated and provides assistance during AHRS alignment. The GMU 44B is mounted on an access cover under each wing between W.S. 183.98 and W.S. 208.33.

(b) Troubleshooting

See “Chart 16” on page 34250272 and ”Chart 17” on page 34250273.

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COVER ASSY - MAGNETOMETER COMPLETE
VIEW LOOKING DOWN
(LEFT SHOWN, RIGHT OPPOSITE)

GMU 44B Magnetometer Installation
Figure 20

(c) Removal

1) Remove the access cover to expose the magnetometer.
2) Disconnect the GMU 44B harness from the wing harness.
3) In airplanes equipped with GFC 700 only, disconnect the grounding strap by removing the nut and washers.
4) Unscrew and remove the three brass screws securing the magnetometer to its mounting rack.
5) Carefully lift the GMU 44B from the rack.

(d) Installation

CAUTION: THE MAGNETOMETERS ARE SECURED TO THE MOUNTING BRACKET WITH BRASS SCREWS. ENSURE ONLY BRASS SCREWS ARE USED WHEN REINSTALLING.

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
2) Lower the GMU 44B into the rack and secure the plate with the three brass Phillips screws. Torque 12–15 inch pounds.
3) In airplanes equipped with GFC 700 only, connect the grounding strap and secure with washers and nut.
4) Connect GMU 44B harness to the wing harness.
5) Reinstall the access cover.
   NOTE: Ensure correct hardware is used when reinstalling access covers attached to magnetometers.

(e) Return to Service

1) Original GMU 44B is Reinstalled

   If the original GMU 44B was reinstalled, then no software loading is required. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) Original GMU 44B Installed in Opposite Locations for Troubleshooting

   If the original GMU1 and GMU2 are installed in opposite locations (GMU1 and GMU2 in opposite unit racks), no software loading is required. However, performing the "GRS 79/GMU 44B Magnetometer Calibration (Procedure B)" on page 342502137 is required.

3) New, Repaired, or Exchanged GMU 44B is Installed

   If a new, repaired, or exchanged GMU 44B is installed, then software must be loaded. See "G1000 NXi Software/Configuration Procedure" on page 342502107. Then continue to the "GRS/GMU Calibration Procedures" on page 342502134.

4) Test

   Perform "GRS/GMU Test" on page 342502142.

(12) GDL 69eA Data Link Unit

See “Figure 21”.

(a) Description

The GDL 69eA provides XM Radio weather and music entertainment through means of a dedicated satellite data link. The GDL 69eA is mounted in a mount rack on the upper left side of the fuselage aft of the cabin hatshelf and just forward of F.S. 220.164. Power to the GDL 69eA is received from the Avionics Bus #1. The GDL 69eA sends weather data through the HSDB bus to the MFD and PFD2, where the data link interface is controlled. Audio is sent directly to the GMA 1360 Audio Panel.

(b) Troubleshooting

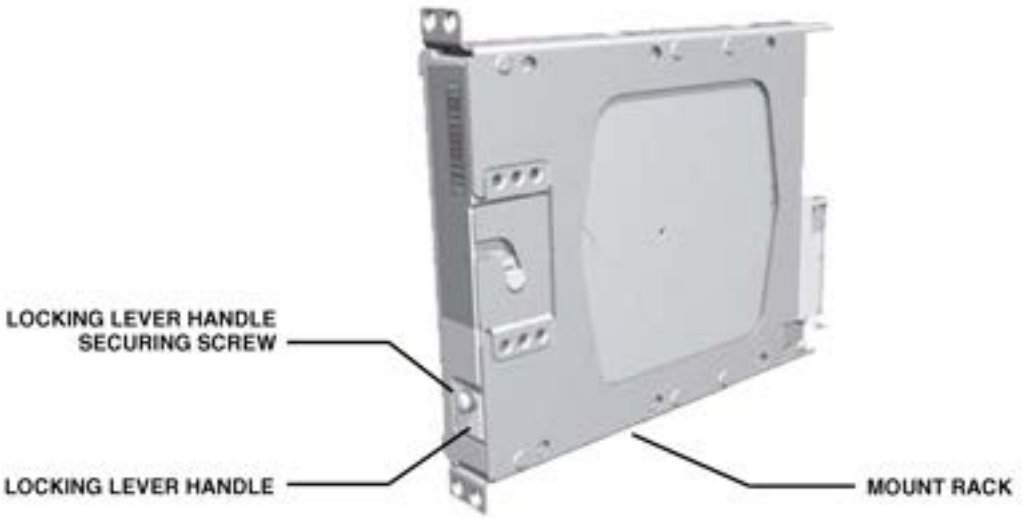
See “Chart 18” on page 34250279 and “Chart 19” on page 34250280.

(c) Removal

1) Remove the aft cabin hatshelf assembly.
2) Unlock the GDL 69eA handle by unscrewing the Phillips screw.
3) Pull the handle upward to unlock the GDL 69eA.
4) Gently remove the GDL 69eA from its rack.

(d) Installation

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
2) Gently insert the GDL 69eA into the rack. The handle should engage the dogleg track.
3) Press down on the handle to lock the unit into place.
4) Lock the handle to the GDL 69eA body using the Phillips screw.
5) Reinstall the aft cabin hatshelf assembly.



GDL 69eA Data Link Unit
Figure 21

(e) Return to Service

1) Original GDL 69eA is Reinstalled

No software or configuration loading is required if the removed GDL 69eA is reinstalled. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) New, Repaired, or Exchanged GDL 69eA is Installed

If a new, repaired, or exchanged GDL 69eA is installed, the correct software and configuration files must be loaded to the unit, see “G1000 NXi Software/Configuration Procedure” on page 342502107. Then, the XM Satellite Radio subscription must be reactivated, see “Chart 18” on page 34250279.

3) Test

Perform the “GDL 69eA Test (If installed)” on page 342502144.

CHART 18
TROUBLESHOOTING - GDL 69eA

Symptom	Recommended Action
No XM audio is heard No XM weather information is displayed	• Ensure the following items are not preventing the audio panel from distributing XM audio (reference applicable G1000 NXi Pilot's Guide): ✓ Verify the XM volume is not muted on the AUX – XM RADIO page on the MFD. ✓ Verify the COM squelch is not open. ✓ Verify the ICS squelch is not open. ✓ Verify the marker beacon tones are not being received. ✓ Verify the headphone (if equipped) volume is turned up. • Go to the AUX – SYSTEM STATUS page on the MFD and ensure unit is online. ✓ If a red X is present, verify the unit is receiving power at the rack connector. • Ensure there are no GDL alerts in the alert window. If there is an Alert for software or configuration error or mismatch, reload the file noted in the Alert. • Restart the PFD's and MFD in configuration mode and go to the GDL page. ✓ Verify unit is active. ✓ Verify the Signal number is "2" or "3". If it is "0" or "1" check the GDL 69eA antenna and cabling for faults. ✓Reseat the GDL 69eA to verify the coax connector is fully seated. ✓ If unit is not active, contact XM Customer service at 1-800-985-9200 to have a refresh signal sent to your unit. You will need to provide them the Audio Radio ID (and Data Radio ID number for XM weather) numbers. Also verify with XM that the correct Weather package (Aviator Lite or Aviator) is on the account, and that no traffic service has been activated against that Radio ID. The unit must be on for approximately one hour after the request for the refresh has been sent to receive the signal. ✓Alternatively, you may also go to XM's website at http://www.siriusxm.com and enter the radio ID's to have a refresh signal sent. ✓ If there is still problems receiving weather products after performing the above step, call XM and have the account deactivated, and a new account activated to clear out any corrupt account information. ✓ Verify there is a good ground connection through the aircraft between the MFD and the GDL69eA unit. Reference the Aircraft Maintenance Manual for instructions on how to check bonding and ground points. ✓ If problem persists, replace the GDL 69eA.

CHART 19
MESSAGE ADVISORIES - GDL 69eA

Failure Message	Cause	Solutions
GDL 69eA FAIL – GDL 69eA has failed.	The G1000 NXi has detected a failure in the GDL 69eA.	• Replace GDL 69eA. • Check GDL 69eA antenna and cabling. • Check the GDL 69eA and MFD interconnect.
GDL 69eA CONFIG – GDL 69eA configuration error. Config service req'd.	The G1000 has detected a GDL 69eA configuration mismatch.	• Reload configuration file. See Software Load / Configuration Procedure. • If problem persists, replace master configuration module, check config module harness for faults and replace if necessary. NOTE New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The change G1000 NXi System ID number will to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number.
MANIFEST – GDL software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the GDL 69eA.	• Load correct software version. See Software Load / Configuration Procedure.

(13) FS 510 Bluetooth and Wi-Fi Connectivity (Optional)

See”Figure 22”.

(a) Description

The FlightStream 510 is an SD card-sized unit which is inserted into the lower SD card slot on the face of the MFD. It provides a secure (password protected) Bluetooth and Wi-Fi interface to the G1000 NXi to provide wireless database updating via the Garmin Pilot app, situational data available to mobile apps, GPS and attitude information, ADS-B (Traffic and FIS-B), Flight Plans, PVT, AHRS, TIS-B, FIS-B, Data logging, and SXM Weather.

(b) Troubleshooting

See “Chart 20” on page 34250282.

(c) Removal

1) Press and release unit.
2) Remove unit from card slot.

(d) Installation

3) Place unit in card slot.
4) Press unit into card slot until you feel unit engage slot.



FS 510 (Optional)
Figure 22

CHART 20
TROUBLESHOOTING - FS 510 (Optional)

Problem	Action
Flight Stream Not Found	Check Flight Stream wiring and power. Check that Flight Stream is not obstructed by RF blocking material. Check that the control/display is in RF range of the Flight Stream. Check that pairing mode is enabled.
Flight Stream connection lost.	Check that the control/display is in RF range of the Flight Stream. Check that Flight Stream is not obstructed by RF blocking material. Remove the Bluetooth pairing from the portable electronic device and the Flight Stream, and then attempt to pair the device again.

(14) GSR 56 Iridium Transceiver

See “Figure 23”.

(a) Description

The optional GSR 56 Iridium Transceiver provides airborne Iridium Satellite Telephone and SMS Messaging service. Iridium telephone and text meassaging are available to the flight crew through the MFD, audio panel, and headset. The Garmin GSR 56 is connected to the #2 GIA 64 and GMA 1360. Power is received from the Avionics Bus #1.



GSR 56 Iridium
Figure 23

(b) Troubleshooting

See “Chart 21” on page 34250283.

(c) Removal

1) Remove the aft cabin hatshelf assembly to gain access to the aft avionics shelf.
2) Loosen the GSR 56 unit by turning the ratchet mechanism located at the front of the unit counter-clockwise until it drops free of the locking pawl on the unit.
3) Gently remove the GSR 56 from its rack.

(d) Installation

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
2) Gently insert the GSR 56 into the rack and engage the connectors.
3) Lift the ratchet mechanism to allow the collar to engage the locking pawl on the unit.
4) Gently turn ratchet mechanism clockwise to secure the GSR 56 unit into place.
5) Reinstall the aft cabin hatshelf assembly.

(e) Datapath Monitoring Disablement

GSR 56 Datapath Monitoring can be disabled as follows:

1) Start the PFD and MFD in configuration mode by applying power to each while holding the ENT key.
2) Navigate to the System Data Path Configuration page of the System group using the small FMS knob.
3) Press the RS-232 softkey.
4) Activate the cursor by pressing in the FMS knob.
5) Rotate the large FMS knob to scroll down to GIA2 CHNL 3.
6) Rotate the small knob to generate a pick list and select OFF. Press ENT key.
7) Press the FMS knob to deactivate the cursor.
8) Navigate to the Configuration Manager page of the System group using the small FMS knob.
9) Press the CNFM CFG softkey. Press ENT to select OK on popup prompt.

CHART 21
MESSAGE ADVISORIES - GSR 56

Failure Message	Cause	Solution
GSR1 FAIL – GSR1 has failed.	The system has detected a failure in the #1 GSR 56.	• Ensure GSR1 is receiving power. • Replace GSR.

(15) GWX 75 Weather Radar

GWX 80 Weather Radar (Optional).

See “Figure 24”.

(a) Description

The GWX 75 Airborne Weather Radar (GWX 80 optional) provides weather radar data output to the GDU 1050 MFD. The GWX 75/80 is mounted in the nose cone assembly, forward of the nose gear doors. Power to the GWX 75/80 is received from the Avionics Bus #1. Data received from the GWX 75/80 is routed to the MFD via high-speed data bus (Ethernet).

(b) Troubleshooting

See “Chart 22” on page 34250286.

(c) Removal

NOTE: Take note of the number, type and location of any shims and washers during removal.

1) Disconnect backshell assembly (P400) from top of unit.
2) Use a 3/16” hex drive tool to remove each of the two mounting screws, washers and nuts connecting the unit to the upper beam assembly.
3) Use a #3 Phillips bit to remove two mounting screws which connect the upper beam to each side clip assembly. Remove the upper beam assembly.
4) Use a 3/16” hex drive tool to remove each of the two mounting screws connecting the unit to the lower bracket assembly.
5) Remove GWX 75/80 by rotating unit back over the bracket assembly.

(d) Installation

NOTE: Install shims and washers of same number and type in the same locations noted during removal.

1) Inspect connector for damaged pins.
2) Hold unit flush with the radar mount, ensuring that the connector is at the top.
3) Reattach and tighten each of the two mounting hex head screws to the lower bracket assembly.
4) Reattach and tighten each of the two mounting phillips head screws which connect the upper beam to each side clip assembly. Torque to 60 in·lbs.
5) Reattach and tighten each of the two mounting hex head screws, washers and nuts connecting the unit to the upper beam assembly. Torque to 60 in·lbs.
6) Connect backshell assembly (P400) to top of unit.

(e) Return to Service

1) Original GWX 75/80 Reinstalled

No software or configuration loading is required if the removed GWX 75/80 is reinstalled. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

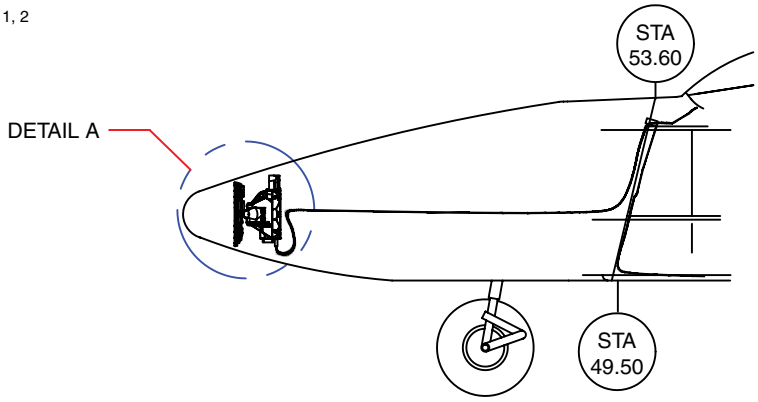
2) New, Repaired, or Exchanged GWX 75/80 Installed

If a new ,repaired, or exchanged GWX 75/80 is installed, the correct software and configuration files must be loaded to the unit. See “G1000 NXi Software/Configuration Procedure” on page 342502107.

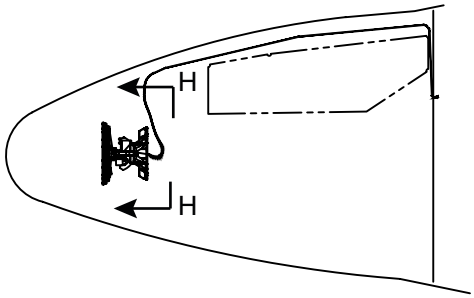
3) Test

Perform “GWX 75/80 Weather Radar Test (If installed)” on page 342502143.

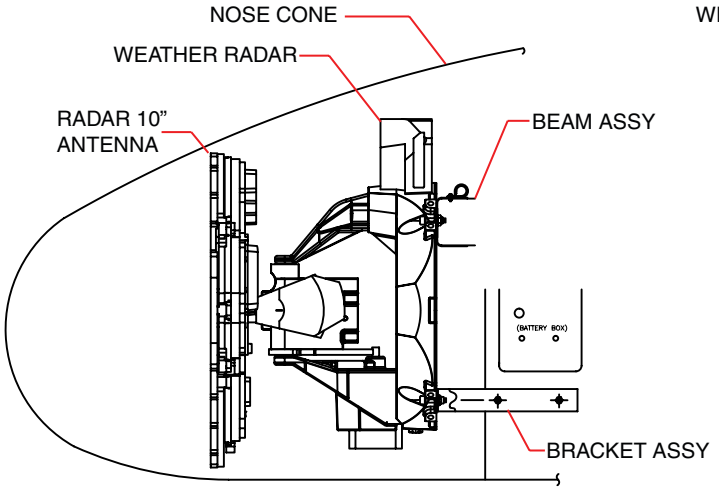
34F34A011 New 1, 2



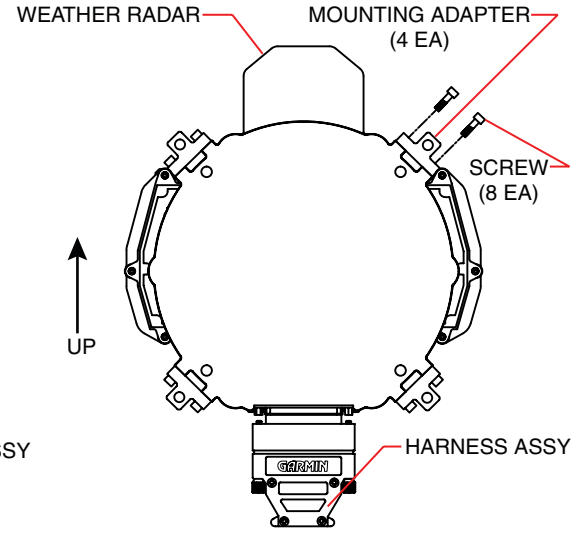
LH SIDE VIEW



TOP VIEW



DETAIL A
RADAR SIDE VIEW



VIEW H -H
VIEW LOOKING FORWARD

GWX 75 RADAR

GWX 75 Weather Radar (GWX 80 Weather Radar Optional)
Figure 24

CHART 22
TROUBLESHOOTING / MESSAGE ADVISORIES - GWX 75/80

WARNING:

Before energizing the equipment, be sure microwave radiation safety precautions including both fuel and personnel safety considerations have been observed. These include clearing all personnel to an area beyond the maximum permissible exposure level (MPEL) boundary. The MPEL for the GWX 75/80 is 11 feet.

Failure Message	Cause	Solution
GWX CONFIG – GWX configuration error. Configuration service required.	The G1000 has detected a GWX 75/80 configuration mismatch.	• Load GWX configuration files, ✓ Replace GWX. ✓ If problem persists, replace master configuration module, check config module wiring for faults and replace if necessary. NOTE New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 NXi System ID number will change to a new number when installing a newmaster config module. The old Terrainand other cards will no longer work asthey will remain locked to the old System ID number.
GWX FAIL – GWX is inoperative.	The G1000 NXi has detected a failure in the GWX 75/80.	• Check Ethernet connection between the GWX and GDL69eA for faults. • Replace the GWX 75/80
GWX SERVICE – Needs service. Return unit for repair.	The G1000 NXi has detected a failure in the GWX 75/80.	• Replace the GWX 75/80.
MANIFEST – GWX software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the GWX 75/80.	• Load correct software version. See Software Configuration Procedures.
WX ALERT – Possible severe weather ahead.	The system has detected severe weather within ± 10 degrees of the aircraft heading at a range of 80 to 320 nm.	

(16) GTS 825 Traffic Advisory System

See “Figure 25”.

(a) Description

The optional GTS 825 Traffic Advisory System provides real-time traffic information to the PFD (and, indirectly, to the MFD). The Garmin GTS 825 connects with the #1 GDU 1050 and communicates with the MFD through an HSDB connection. Power is received from the Avionics Bus #1.

(b) Troubleshooting

See “Chart 23” on page 34250288.

(c) Removal

1) Remove the aft cabin hatshelf assembly to gain access to the aft avionics shelf.



GTS 825 TAS
Figure 25

2) Disconnect the GTS 825 connectors.

3) Loosen the GTS 825 unit by unscrewing the knurled retainers.

4) Move the retainer arms downward. Gently remove the GTS 825 from its rack.

(d) Installation

1) Insert the GTS 825 into the rack and lift each retainer arm to engage lip.

2) Gently screw retainers to secure the GTS 825 unit into place.

3) Visually inspect the connectors and pins for damage. Repair any damage. Connect the connectors to the unit.

4) Reinstall the aft cabin hatshelf assembly.

(e) Return to Service

1) Original GTS 825 Reinstalled

No software or configuration loading is required if the original GTS unit is removed and reinstalled. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) New, Repaired, or Exchanged GTS 825 Installed

If a new ,repaired, or exchanged GTS 825 is installed, the correct software and configuration files must be loaded to the unit. See“G1000 NXi Software/Configuration Procedure” on page 342502107.

3) Test

Perform “GTS 825 TAS Tests (If installed)” on page 342502145.

CHART 23
MESSAGE ADVISORIES - GTS 825

Failure Message	Cause	Solution
MANIFEST – GTS software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the specified GTS 825.	Load the correct software version. See G1000 NXi Software/Configuration Procedure,for the Software Load Procedure.
GTS CONFIG – GTS configuration error. Config service req'd.	The system has detected a configuration mismatch for the specified GTS 825.	

(17) Configuration Module

See “Figure 26”.

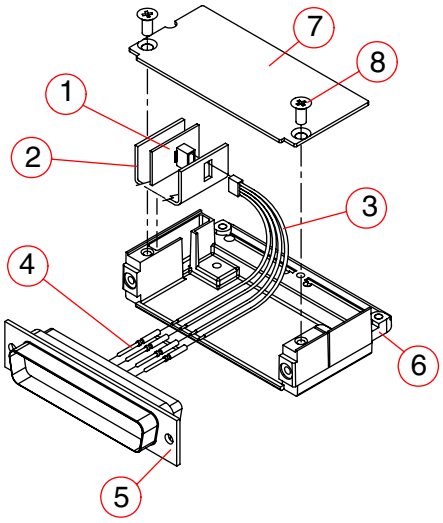
The configuration module is located in the backshell of the connector it is terminated to.

(a) Removal

1) Disconnect connector from LRU.
2) Remove 2 screws (8) from cover (7) and remove cover.
3) Unplug connector from configuration module (1).
4) Remove configuration module.

(b) Installation

1) Inspect connector for damaged pins (4).
2) Place configuration module (1) in position.
3) Insert connector into configuration module (1).
4) Assembly of the connector is the reverse of disassembly.



Configuration Module
Figure 26

(c) Return to Service

1) If a GRS 79 AHRS Configuration Module is replaced:

All three GRS 79/GMU 44B calibration procedures must be performed. Proceed to “Post-Installation Procedures” on page 342502119.

2) If a GDC 72 Configuration Module is replaced:

Configuration settings must be reloaded to the GDC 72. Proceed to “Post-Installation Procedures” on page 342502119.

3) If the GEA 71B Configuration Module is replaced:

Perform the “GEA 71B Test” on page 342502130.

4) If the Master Configuration Module is replaced:

a) Start the G1000 NXi system in configuration mode.
b) Go to the Configuration Upload Page on the PFD.
c) Press the UPDT CFG softkey.

5) If both the PFD and Master Configuration Module is replaced:

The G1000 NXi system (except GRS 79/GMU 44B and GDC 72) must be reconfigured. Proceed to “Post-Installation Procedures” on page 342502119.

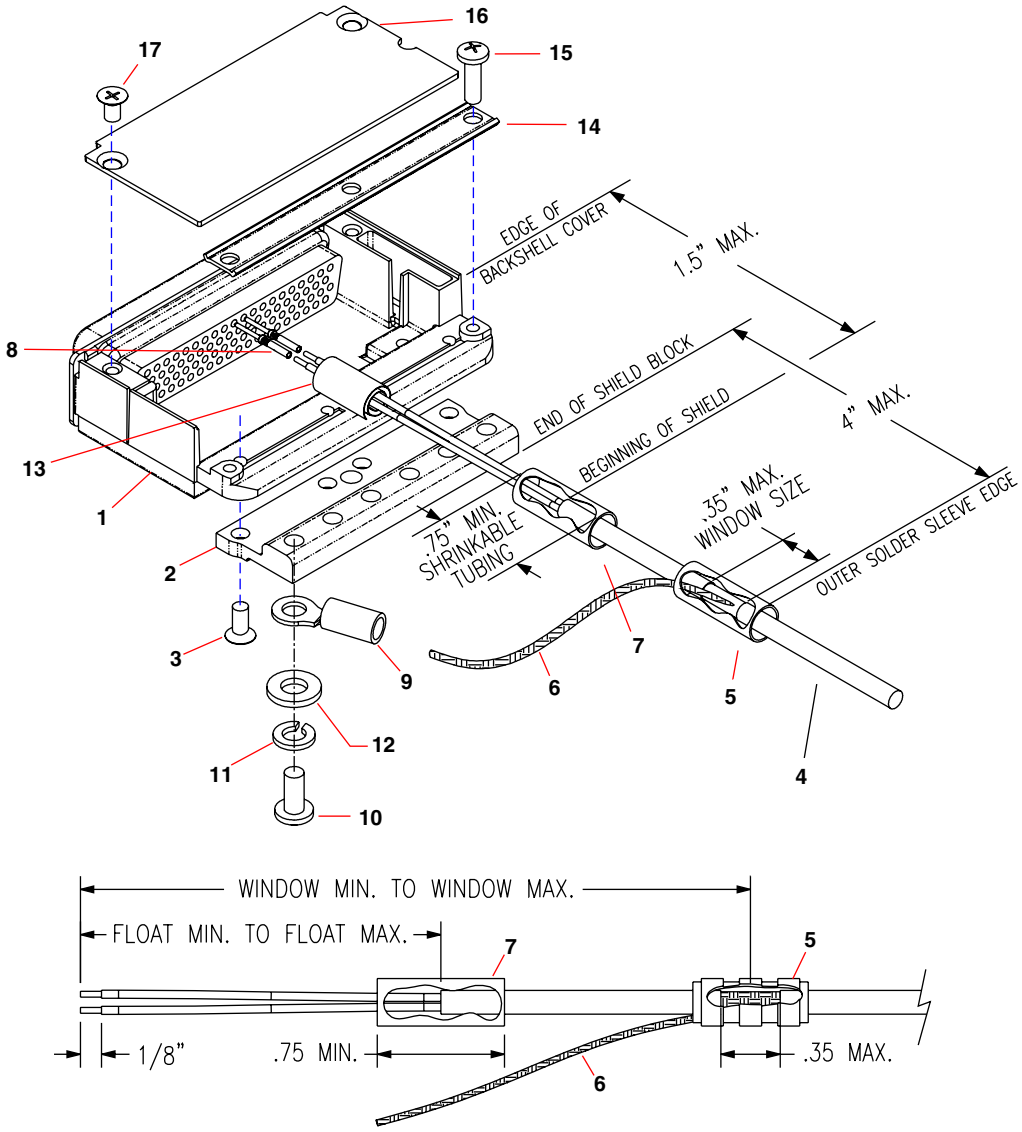
(18) Shield Block Installation

See “Figure 27” on page 34250290.

Most G1000 NXi connectors employ a shield block grounding system to provide necessary ground reference to shielding and/or transducers.

(19) Backshell Connectors

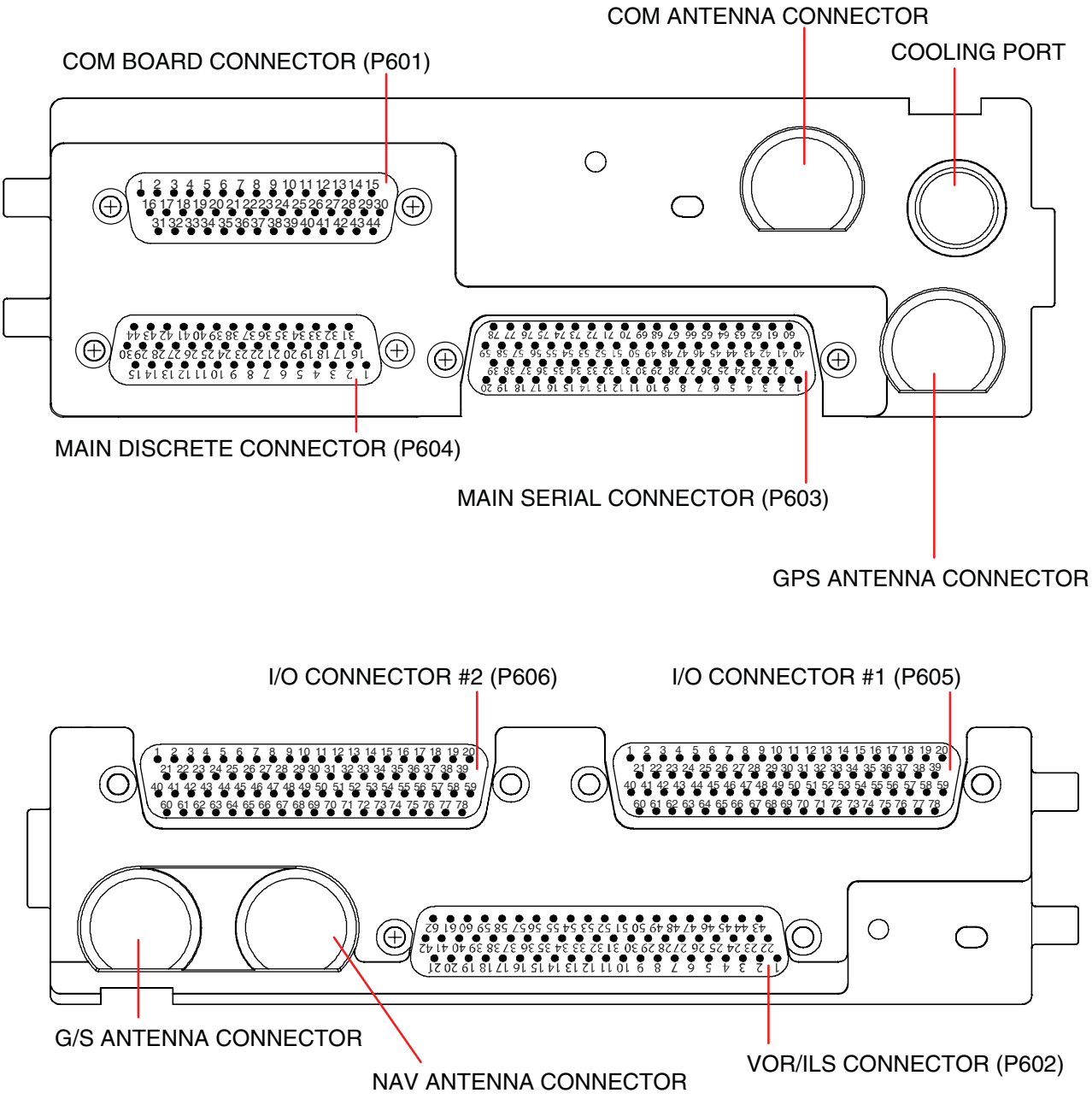
See “Figure 28” on page 34250291.



1. CAST HOUSING
2. SHIELD BLOCK(S)
3. SCREW
4. MULTIPLE CONDUCTOR SHIELDED CABLE (2–CONDUCTOR DEMONSTRATED HERE)
5. DRAIN WIRE SHIELD TERMINATION (METHOD OPTIONAL)
6. BRAID, FLAT
7. FLOATING SHIELD TERMINATION (METHOD OPTIONAL)
8. PINS
9. RING TERMINAL, #8, INSULATED
10. SCREW
11. SPLIT WASHER
12. FLAT WASHER
13. SILICON FUSION TAPE
14. STRAIN RELIEF
15. SCREW
16. LID
17. SCREW

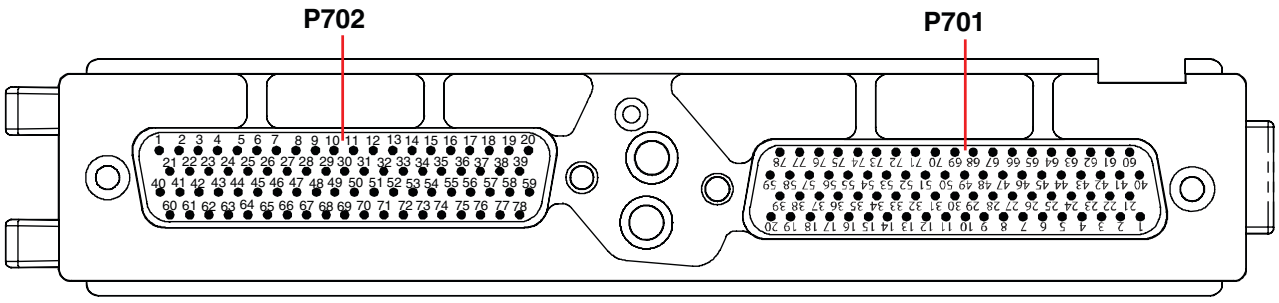
Shield Block Installation to Backshell
Figure 27

Backshell Connectors

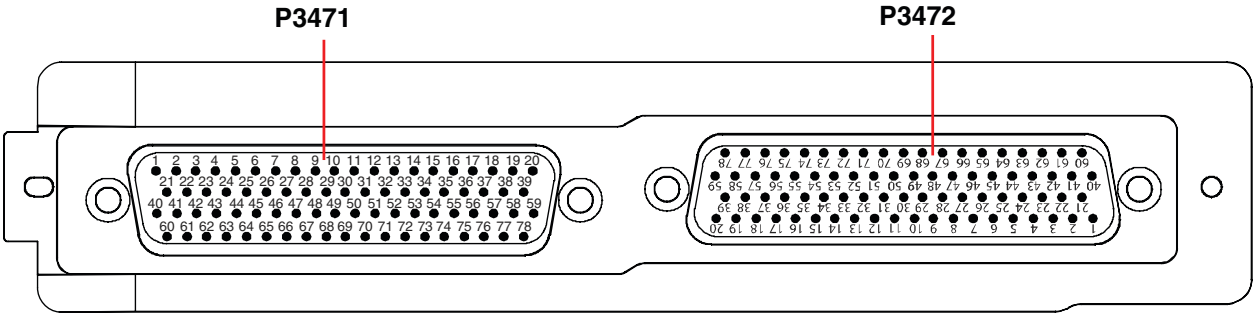


Backshell connectors as viewed with the LRU removed.

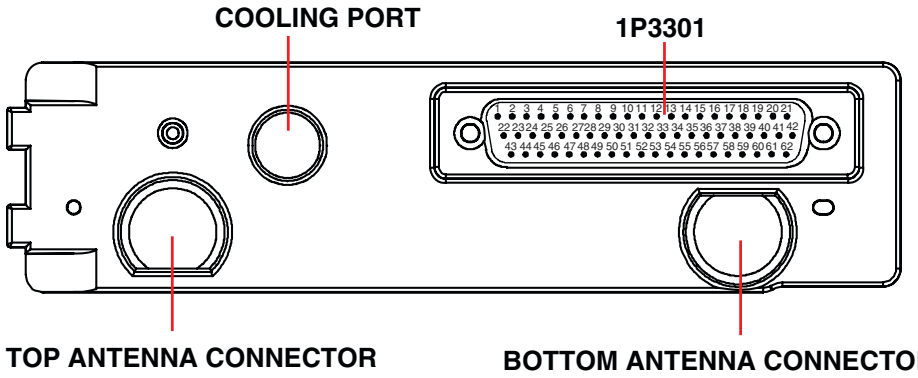
Backshell Connectors
Figure 28 (Sheet 1 of 4)



GEA 71B Backshell Connectors



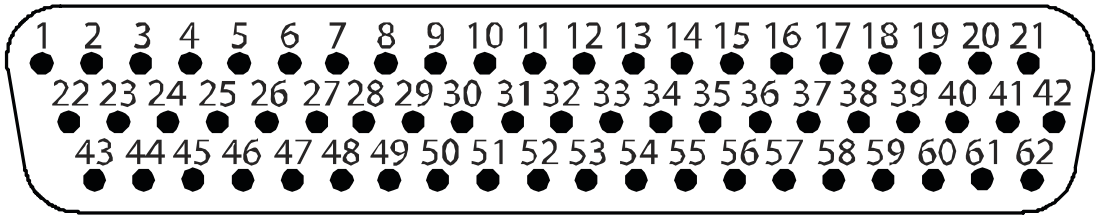
GMA 1360D Backshell Connectors



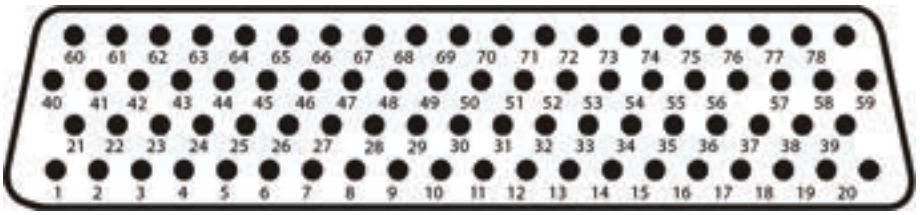
GTX 335R/345R Backshell Connectors

Backshell connectors as viewed with the LRU removed.

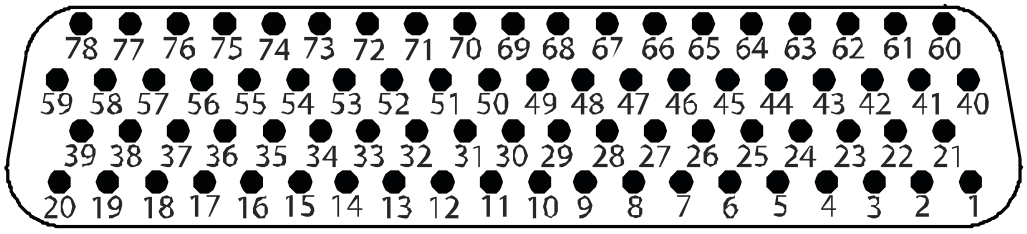
Backshell Connectors
Figure 28 (Sheet 2 of 4)



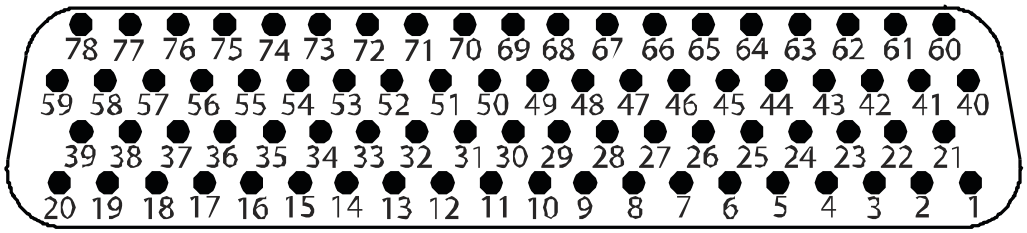
GDU 1050/1054/55 Backshell Connector (1P10001, 2P10001, and 3P10001)



GRS 79 Backshell Connector (P791)



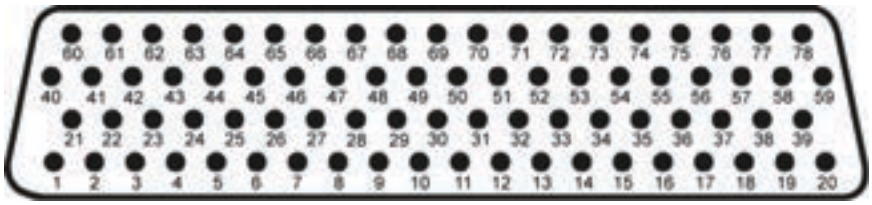
GDC 72 Backshell Connector (P721)



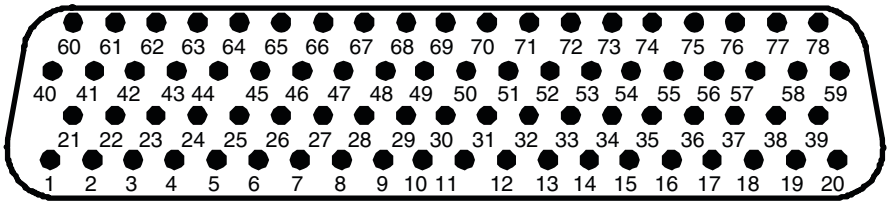
GDL 69eA Backshell Connector (P691)

Backshell connectors as viewed with the LRU removed.

Backshell Connectors
Figure 28 (Sheet 3 of 4)



GWX 75/80 Backshell Connector (P751)



GTS 825 Backplate Connector (P8001, P8002)

Backshell connectors as viewed with the LRU removed.

Backshell Connectors
Figure 28 (Sheet 4 of 4)

CHART 24
TROUBLESHOOTING / MESSAGE ADVISORIES - G1000 NXi SOFTWARE CONFIGURATION

Problem	Solutions
MFD or PFD displays do not power up:	• Ensure power is present at display backshell connector. • Troubleshoot per the “Blank Display” GDU section.
Software file load fails: 	• Ensure that LRU is reporting data on System Status page (LRU is ‘ONLINE’). Check data path wiring as needed. • Retry software file load or try using a different card. Ensure that the MFD is not touched during the loading process. • Ensure that LRU part number is compatible with software version and Card Loader. • Replace LRU.
Configuration file load fails: 	• Ensure that LRU is reporting data on System Status page (LRU is ‘ONLINE’). Check data path wiring as needed. • Retry configuration file load or try using a different card. Ensure that the MFD is not touched during the loading process. • Ensure that LRU part number is compatible with Card Loader. • Replace LRU.
GIA1 and/or GIA2 to ‘LRU’ data path not working	• Ensure GIA1 and GIA2 are configured correctly. • Check wiring, connectors & pins as needed.
Software File Mismatch Alert appears in lower right corner of PFD when started in normal mode: 	• Ensure that proper software file part number and version were loaded to LRU. To attain the current sofware P/N and version for your aircraft contact: Piper Aircraft Inc. Customer Service. • Check and ensure that correct Card Loader was used during load process. Refer to the General Arrangement Drawing. • Reload software to LRU.

CHART 25
G1000 NXi SYSTEM COMMUNICATION HIERARCHY

The following criteria must be satisfied to be able to perform the desired operation:	
Desired Operation	Criteria for Success
Load Software to MFD or PFD Displays	• G1000 NXi Loader Card must be inserted in top slot for each display to be loaded. • CLR & ENT keys must be held during power up of display. • Power on only one display at a time during software loading.
Load AIRFRAME, SYSTEM, MFD, PFD1, PFD2 and MANIFEST configuration files to MFD and PFDs	• G1000 NXi Loader Card must be inserted in top slot of PFD1. • PFD1 and MFD must be powered on. • PFD1 and MFD must have correct software.
Load Software/Configuration files to GIA 64s	• G1000 NXi Loader Card must be inserted in top slot of PFD1. • G1000 NXi system must be powered on. • PFD and MFD must have correct software. • PFD 1 and MFD must be successfully configured with AIRFRAME, SYSTEM, MANIFEST, MFD, PFD1 and PFD2 configuration files.
Load Software/Configuration files to: - GMA 1360 - GDC 72 - GEA 71B - GRS 79 (software only) - GMU 44B (software only) - GTX 3X5R	• G1000 NXi Loader Card must be inserted into PFD1 top slot. • G1000 NXi must be powered on. • PFD1 and MFD must have correct software and configuration settings. • GIA 64s must have correct software. • GIA 64s must be successfully configured with GIA1 and GIA2 configuration files. • Data path from GIA1 to each LRU must be operational.

I. Configuration Mode Overview

CAUTION: THE CONFIGURATION MODE CONTAINS CERTAIN PAGES AND SETTINGS THAT ARE CRITICAL TO AIRCRAFT OPERATION AND SAFETY. THESE PAGES ARE PROTECTED AND CANNOT BE MODIFIED, UNLESS THE TECHNICIAN IS PROPERLY AUTHORIZED AND EQUIPPED. HOWEVER, MOST PROTECTED PAGES ARE VIEWABLE TO ALLOW SYSTEM AWARENESS FOR TROUBLESHOOTING.

The Configuration Mode exists to provide the avionics technician with a means of configuring, checking, and calibrating various G1000 NXi sub-systems. Troubleshooting and diagnostics information can also be viewed in this mode.

(1) Start the system in Configuration Mode:

(a) Press and hold the ENT key on the co-pilot's PFD while applying power using the PFD2 circuit breaker.

(b) Release the ENT key after 'INITIALIZING SYSTEM' appears in the upper left corner of the PFD.

(c) Press the ENT key or the YES softkey to update system files if prompted.

(d) Repeat steps (a) thru (c) on the MFD using the MFD circuit breaker to apply power and using the far right soft key on the MFD.

(e) Repeat steps (a) thru (c) on the pilot's PFD using the PFD1 circuit breaker to apply power.

NOTE: For a complete description and breakdown of each Configuration Mode page, refer to the appropriate G1000 NXi Line Maintenance & Configuration Manual.

(2) SET>ACTV Configuration

Throughout the configuration mode pages, there are SET and ACTIVE columns for input/output settings and other parameters.

SET: Refers to a setting or group of settings that reside in PFD Internal Memory and/or the Master Configuration Module.

ACTIVE: Refers to an 'active' setting or parameter currently being used by the LRU. LRUs store the 'active' settings within internal memory.

Data can be manually copied from one column to the other (and consequently from PFD memory to the LRU memory and vice-versa) by using the following two softkeys, when available:

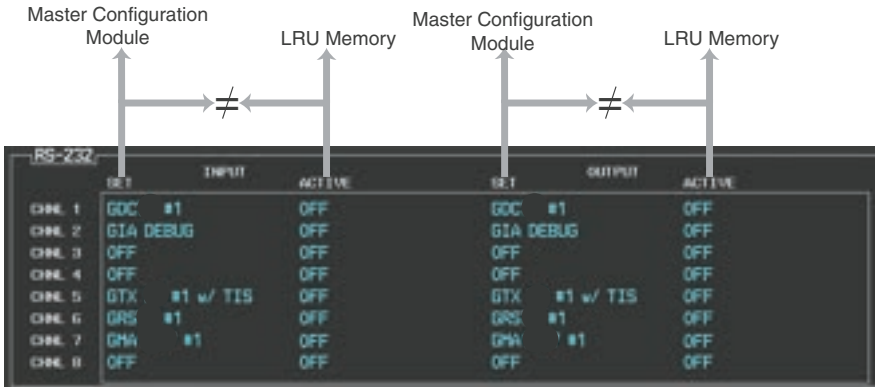
SET>ACTV (read 'Set to Active') softkey: Allows the installer to send the information in the SET column (data stored in the master config module) to the ACTV column (data used by LRU).

ACTV>SET (read 'Active to Set') softkey: Causes the LRUs current settings to be copied to the master configuration module as SET items.

CAUTION: THE ACTV>SET SOFTKEY MUST BE USED WITH CAUTION! IF AN IMPROPERLY CONFIGURED UNIT IS INSTALLED, THIS SOFTKEY CAUSES THE WRONG CONFIGURATION TO REPLACE THE CORRECT AIRCRAFT CONFIGURATION.

In the first example shown in "Figure 29" on page 34250298, the SET columns do not match the ACTIVE columns. The inequality between SET and ACTIVE indicates a configuration mismatch. By pressing the SET>ACTV softkey, this copies the SET column to the LRU unit's configuration memory. The settings then become the ACTIVE settings for the LRU being configured.

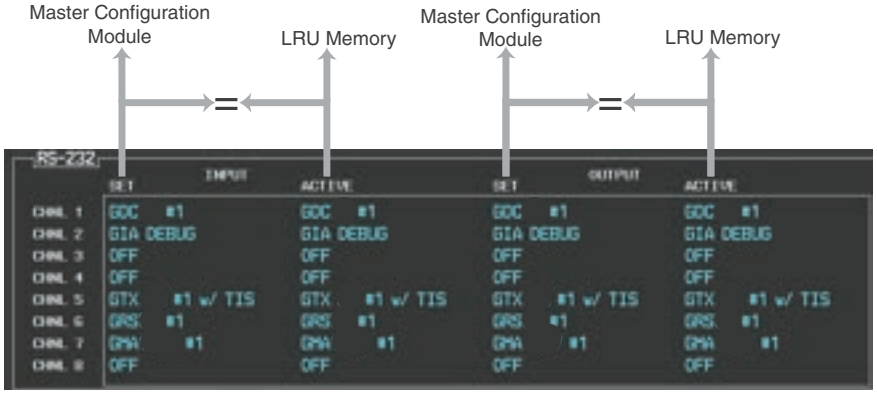
Configuration Mismatch



SET>ACTV Softkey



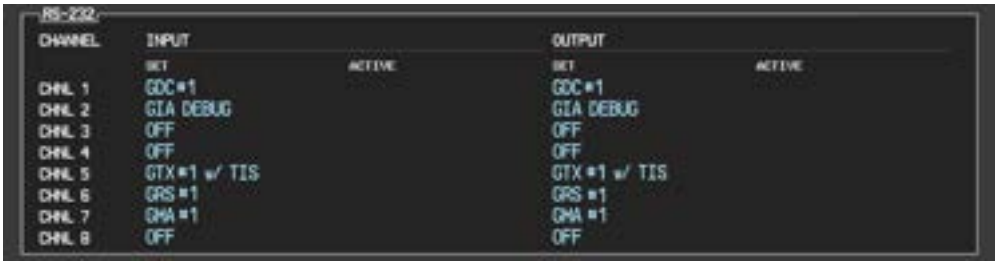
Configuration Correct



SET>ACTV Configuration
Figure 29

When troubleshooting the system, technicians can look for inequalities between SET and ACTIVE columns. Certain problems can be resolved simply by pressing the SET>ACTV softkey, which reloads settings to the specific LRU from the PFD. (Note that this can also be accomplished by reloading the configuration files for the LRU, using the software loader card.).

A blank active column, as shown in “Figure 30”, represents loss of communication between the display and the particular unit. See troubleshooting for more details.



Blank Active Columns
Figure 30

(3) Configuration Prompts

When configuration settings are changed, the technician receives on-screen prompts and/or confirmations such as those shown in “Figure 31”. Other prompts may be encountered during the configuration process.



Configuration Prompts
Figure 31

(4) Data Transmission Indicators

See “Figure 32”.

Several configuration screens utilize an indicator light system to show discrete (ON/OFF) data and/or hardware component status. Unless otherwise noted, the following applies to all such status indicators:

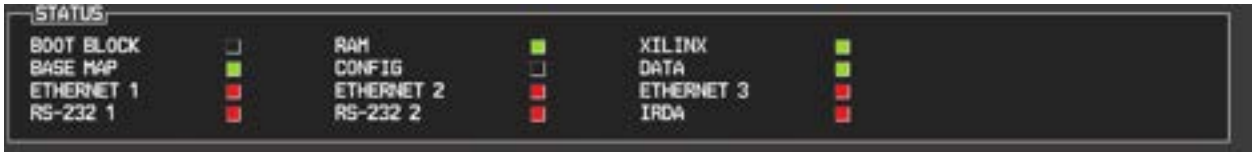
(a) Green Light: Expected data is successfully received and is ON. A green light could also indicate that the parameter/component is working correctly.

(b) Red Light: Expected data is not received. A red light could also indicate that a parameter/component is invalid.

(c) No Light (Black): Expected data is successfully received and is OFF, or no data is expected. A black light could also indicate that the parameter/component is not responding.

(5) Configuration Mode Navigation

Using the FMS knob, a user can navigate through different pages and page groups in the Configuration Mode. See “Chart 26” on page 342502101. For a complete description and breakdown of each page, refer to the G1000 NXi Line Maintenance & Configuration manual.



Data Transmission Indicators
Figure 32

CHART 26
CONFIGURATION MODE NAVIGATION

System Page Group

1. System Status
2. Date/Time Setup
3. Main Lighting
4. Audio Alert Configuration
5. System Upload
6. Diagnostics Terminal
7. Maintenance Log
8. OEM Diagnostics
9. System Configuration
10. System Setup
11. Manifest Configuration

GDU Page Group

1. RS-232 / ARINC 429 Config
2. GDU Status
3. GDU Test
4. Diagnostics
5. Serial/Ethernet I/O
6. Alert Configuration
7. Airframe Configuration

GIA Page Group

1. RS-232 / ARINC 429 Config
2. CAN / RS-485 Config
3. GIA I/O Configuration
4. COM Setup
5. GIA Status

GEA Page Group

1. Engine Data
2. GEA Status
3. GEA Configuration

GTX Page Group

1. RS-232 / ARINC 429 Config
2. Transponder Configuration

GRS Page Group

1. AHRS / Air Data Input
2. GRS / GMU Calibration

GDC Page Group

1. GDC Configuration

GFC Page Group

1. GFC Configuration
2. GFC Status

GMA Page Group

1. GMA Configuration

GDL Page Group

1. GDL 69 Config

RMT Page Group

1. Remote Controller Status

GWX Page Group

1. GWX Configuration

CAL Page Group

1. Fuel Temperature Calibration
2. Flaps and Trim Calibration
3. HSCM Calibration

J. G1000 NXi Software Information

NOTE: The following sections provide a detailed description of loading all G1000 NXi software and configuration files, which may be excessive for individual LRU removal and replacement. If removing and replacing individual LRUs, refer to “Components” on page 34250237, for the necessary steps.

(1) G1000 NXi Software Loader Card

CAUTION: IT IS CRITICAL THAT THE TECHNICIAN INSTALLS THE CORRECT SOFTWARE IMAGE PART NUMBER WHEN SERVICING THE G1000 NXI SYSTEM. APPROVED SOFTWARE DEFINITIONS FOR EACH CONFIGURATION ARE DEFINED IN THE REQUIRED EQUIPMENT LISTS (REL),“Chart 31” on page 342502165.

All software and configuration files were certified by Garmin as part of the FAA-approved Type Design data. Approved software definitions for each configuration are defined in the Required Equipment List (REL), “Chart 31” on page 342502165.

G600 software and configuration files are controlled via the approved software image part number listed in the REL. This software image is loaded into the G1000 NXi using a software loader card. The installer shall create this software loader card by downloading the approved software image in accordance with the procedure provided immediately following.

CAUTION: BE CAUTIOUS WHEN USING SOFTWARE LOADER CARDS DURING MAINTENANCE. THE G600 SYSTEM IMMEDIATELY INITIALIZES THE CARD UPON POWER-UP. ON-SCREEN PROMPTS MUST BE GIVEN CAREFUL ATTENTION IN ORDER TO AVOID POTENTIAL LOSS OF DATA. ALWAYS READ THROUGH PROCEDURES GIVEN IN THIS SECTION, BEFORE ATTEMPTING TO USE THE SOFTWARE LOADER CARDS.

(2) Software Loader Card Creation

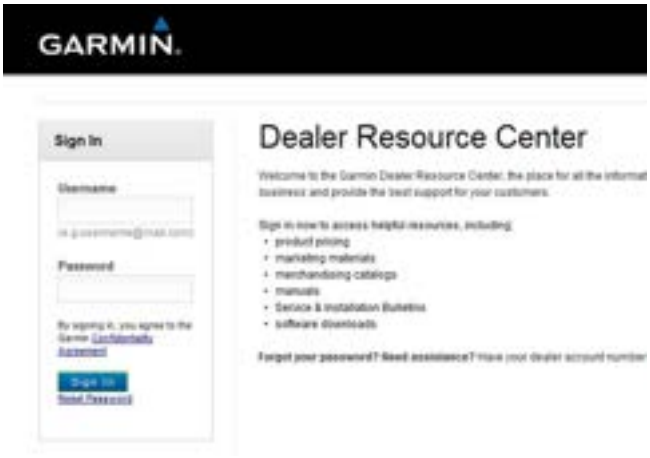
The following procedure requires a PC with an SD card reader. Piper recommends SanDisk® SDDR-99 and SDDR-93 card readers. Additionally, only SanDisk brand 16GB SD cards are recommended.

The software image is an executable self-extracting file which builds the correct file structure onto an SD card for use loading software to the G1000 NXi. To obtain the current file follow the procedures outlined below.

NOTE: In order to create a loader card, the installer completing these procedures must be an authorized Piper or Garmin service center to gain access to the necessary data via the Garmin website.

(a) Go to https://www.garmin.com and click on the ‘Dealer Resource Center’ link in the lower right hand portion of the home page. Enter username and password. See “Figure 33”.

(b) Select Support and then Software Downloads.



Dealer Login Screen
Figure 33

(c) Search for the appropriate image version (i.e., “1981.xx” from the appropriate “Chart 31” on page 342502165) in the “Search by Keyword” field.

(d) Save the file to a known location on the PC.

(e) Run the executable file that was downloaded. A window will appear to confirm software installation.

(f) Insert the SD card into the SD card reader and select the Next button in the pop-up window.

(g) In the pop-up window, select the drive that corresponds to the inserted SD card and then click the Next button.

NOTE: The selected drive will be completely erased. Click the Next button to acknowledge any warnings that appear. Another pop-up window will appear to monitor loader card creation progress. After the card is successfully completed, another pop-up window will appear.

(h) Select the Finish button.

(i) Stop the card reader using Windows stop device function.

(j) Eject the SD card from the card reader.

(k) Apply a label to the software loader card identifying the aircraft, G1000 NXi SFTW and the appropriate software image part number as shown below:

NOTE: See appropriate “Chart 31” on page 342502165 for software image part number.

PA-34-220T G1000 NXi SFWT 006-x1981-xx

(3) Handling a Software Download Card

The SD memory cards used for the Software Download Card are manufactured to the specifications of the SD Card Association for use in portable devices. As such they are reasonably robust devices that do not require special handling instructions. However, the memory cards are not unbreakable and reasonable precautions regarding their use and handling should be used including the following guidelines:

(a) Do no bend an SD memory card or subject it to violent impact.

(b) Do not leave SD cards in locations where they may be exposed to moisture or high temperatures.

(c) Avoid exposing the card to static electricity or electro-magnetic interference.

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K. Software Files

Software files are defined by part number and version number. Each G1000 NXi LRU reports the software version it currently contains to the user in two places.

(1) Normal System Mode

The AUX – SYSTEM STATUS page lists each LRU and the reported software version.

(2) Configuration Mode

The SYSTEM STATUS page (SYSTEM page group) reports more detailed LRU information, including software version, part number, and LRU status. Software files are loaded to LRUs from the SYSTEM UPLOAD page in configuration mode.

(3) Configuration File Descriptions

(a) Configuration Files are divided into groups, and are only displayed at the System Upload page during the software/configuration loading process.

(b) Configuration files contain preset selections for input/output channels, aircraft-specific settings, and LRU-specific settings. The following list describes each of the configuration files:

AIRFRAME This file contains data such as airspeed parameters, engine/airframe sensor limitations, fuel tank parameters and alerting system settings that tailor a G1000 NXi PFD or MFD to the PA34-220T.

AIRFRAME ALERTS PA34-220T-specific alerts are set when this file is loaded.

SYSTEM This file configures the G1000 NXi high-speed data bus (HSDB) to expect two PFDs, MFD, two GIAs, and a GDL.

MANIFEST This file loads a manifest checklist of all software part numbers and versions associated with an approved system configuration. The G1000 NXi performs a software check between each LRU's reported version and the version contained on the manifest. If an inequality is detected for an LRU, this LRU is then excluded from the G1000 NXi and a manifest alert is triggered to the operator.

MFD1 This file configures MFD serial/discrete communication and alert system settings.

PFD1/PFD2 These files configure PFD1/PFD2 serial/discrete communication and alert system settings.

GIA1/GIA2 These files configure GIA1/GIA2 serial/discrete communication settings.

GMA_PIL This file configures GMA 1360 audio and serial communication settings

GTX1/GTX2 These files configure GTX 1/GTX 2 transponder and serial communications settings.

GEA1/GEA2 These files configure GEA1/GEA2 engine/airframe parameters.

GDC_PIL/ GDC_COPIL This file configures GDC 72 air data values for the PA34-220T.

GDL_69 This file configures gains and cable loss for the PA34-220T installation.

GWX This file configures parameters as appropriate for the PA34-220T radar installation.

AUDIO This file configures audio alerts for the PA34-220T.

(4) Configuration File Storage

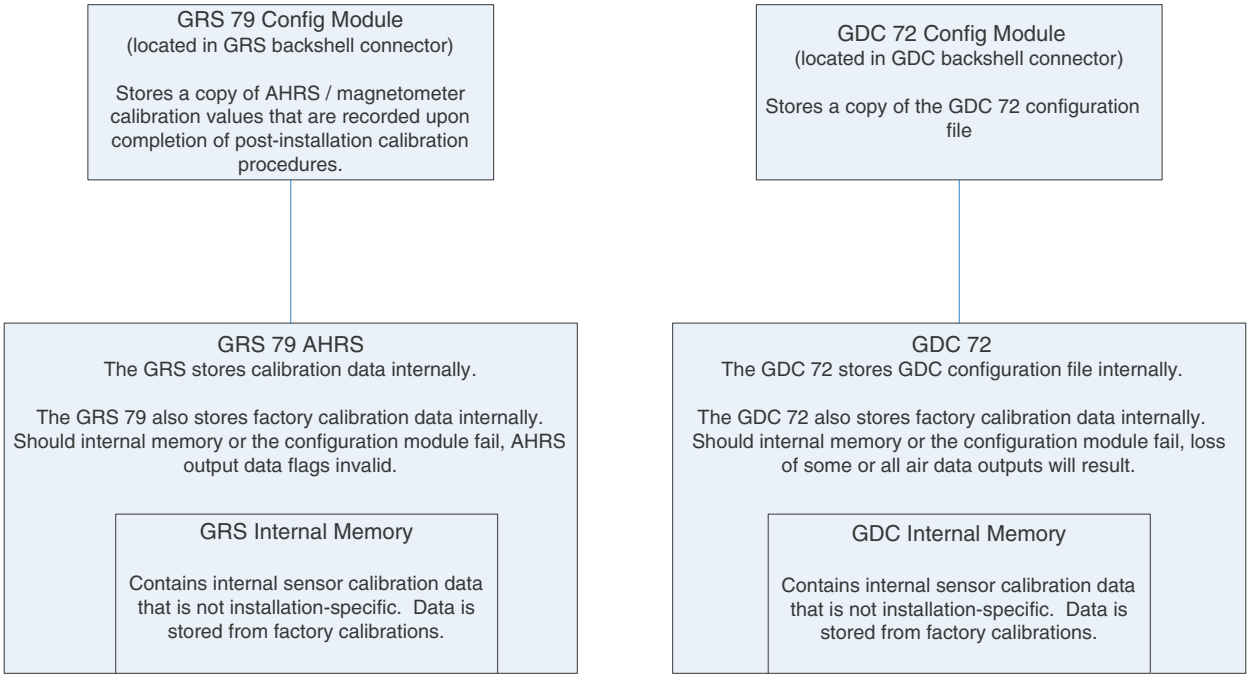
The G1000 NXi system is designed to store all configuration settings in various places so that the configuration is retained in the aircraft during maintenance of units.

During system configuration, each file is sent directly to the applicable LRU where it is stored in local LRU memory (except GRS 79 & GDC 72). Each file is also stored in the PFD internal memory. The applicable PFD also sends a copy of all configuration files to the 'Master Configuration module', located in the connector backshell (see "Configuration Mode Overview" on page 34250297). If the PFD is replaced, the configuration module retains all configuration files in the aircraft.

NOTE: The GRS 79 AHRS and GMU 44B Magnetometer do not have a configuration file. However, these LRUs do store calibration data acquired during the post installation checkout, which are characteristic to the specific installation. While performing maintenance on these units, re-calibration may be required. See "GRS/GMU Calibration Procedures" on page 342502134.

The GRS 79 and GDC 72 configuration modules, as shown in "Figure 34", function differently than the rest of the system. The GDC 72's configuration file is loaded directly to GDC internal memory. A copy of the file is stored in the GDC configuration module.

The GRS 79 configuration module does not store any configuration settings. Instead, it stores calibration data recorded during installation calibration procedures.

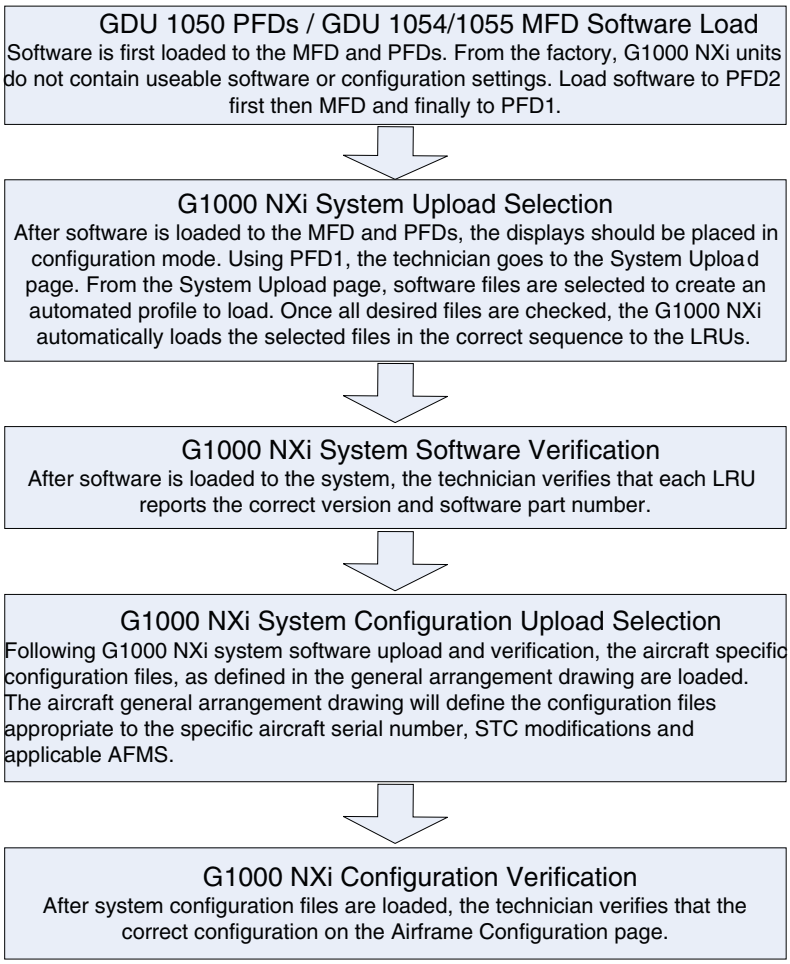


GRS/GDC Configuration Settings Storage
Figure 34

(5) G1000 NXi Software/Configuration Procedure

This and the following sections summarize the procedures required to load software and configuration files to the G1000 NXi. It is intended to work as a central guide for technicians to use while performing maintenance on the aircraft. In sections of this manual where software is required to be reloaded, these sections will make reference back to this Section for instructions. The technician should use proper judgment regarding the context of maintenance required while following this section.

The diagram in “Figure 35” depicts an overview of the software/configuration sequence for the G1000 NXi system. This applies mostly to a new G1000 NXi system which has not previously been powered up and is for informative purposes only.



Software/Configuration Overview
Figure 35

L. Complete Software Load

(PIR-34G34A020 C)

(1) MFD & PFD Software Load

CAUTION: DO NOT INTERRUPT POWER TO THE AIRCRAFT AT ANY TIME DURING THESE PROCEDURES. INTERRUPTION OF POWER COULD RESULT IN DAMAGE TO THE EQUIPMENT.

(a) Apply power to the aircraft and all avionics.

(b) Pull the MFD and both PFD circuit breakers. Remove any SD cards that were previously inserted in either PFD or MFD.

(c) Insert G1000 NXi/PA-34 Loader Card into the PFD1 top card slot.

(d) While holding the ENT key on the PFD1, restore power to the PFD1 by pushing in the PFD1 circuit breaker.

(e) When the words “INITIALIZING SYSTEM” appear in the upper left corner of the PFD1, release the ENT key.

(f) Press the YES softkey at the following prompt:

“DO YOU WANT TO UPDATE SYSTEM FILES?”

“NO WILL BE ASSUMED IN 30 SECONDS”

“UPDATING SYSTEM FILES, PLEASE WAIT.”

(g) Press the YES softkey at the following prompt (prompt may not appear):

“DO YOU WANT TO UPDATE THE CUSTOM GRAPHIC FILES (EG, SPLASH SCREEN)?”

“NO WILL BE ASSUMED IN 30 SECONDS”

(h) Press any softkey at the following prompt to confirm load completion:

“PRESS ANY KEY TO CONTINUE.”

“CONTINUING IN 10 SECONDS.”

(i) When the #1 PFD update is complete, it starts in the configuration mode. DO NOT remove power.

(j) While holding the ENT key on the MFD, restore power to the MFD.

(k) While holding the ENT key on the #2 PFD, restore power to the #2 PFD.

CAUTION: FOR THE REST OF THE SOFTWARE/CONFIGURATION PROCEDURE, DO NOT OPERATE THE MFD OR PFD2 WHILE LOADING SOFTWARE OR CONFIGURATION FILES UNLESS SPECIFICALLY INSTRUCTED TO DO SO. A FAILED OR CANCELLED LOAD MAY RESULT.

(2) System Upload

NOTE: To Select/Deselect the cursor in fields, press the small FMS knob on the PFD. The large FMS knob changes Groups, and the small FMS knob changes Pages within a Group.

(a) Activate the cursor and rotate the small FMS knob to generate a pick list and select Baseline.
(b) Press ENT.
(c) In the ITEM field, create a pick list by rotating the small FMS knob and select PA-34-220T Seneca.
(d) Press ENT.
(e) Press the LOAD softkey.
(f) Review list and verify that all items state “PASS”.
(g) Press ENT to accept the software load.
(h) Deselect cursor.

(3) Optional Equipment Installation

Perform the following steps if any of the following options are installed:

ADF RA-3504 Installation	Air Conditioning Installation
De-ice Installation	KN-63 DME Installation
Dual GDC 72 and GAS 79 Installation	Flight Stream 510 Installation
GDL 69A(SXM) Installation	GSA 56 Installation
GTS 825 Installation (No GTX345)	GTX 345R without GTS installed
GTX 345R with GTS installed	GWX 75 Installation
Hobbs Installation - Configuration (A, B, OR C)	Oxygen Installation
Pulse Oximeter	WX-500 Stormscope Installation

NOTE: One of the Hobbs configurations must be installed. If not installed, the Hobbs timer will not function.

(a) On PFD1, select the System Upload page and activate cursor.
(b) In the Group field, create a pick list by rotating the small FMS knob and select Options.
(c) Press ENT.
(d) In the ITEM field, create a pick list by rotating the small FMS knob and select the desired option, press ENT.
(e) Verify both Software and Configuration boxes are checked for Garmin options.
(f) Verify Configuration boxes checked for all other options.
(g) Press LOAD softkey.
(h) Verify the summary field lists the software and configuration are complete as required and PASS is displayed next to both appropriate boxes.
(i) Press ENT to accept.

(j) For other installed options, repeat steps (d) thru (i) by rotating the large FMS knob and highlighting the desired option in the ITEM Field.

(k) De-activate the cursor.

NOTE: Upon completion of all required baseline config, option config, and calibration loading procedures, remove the Loader card from top slot of PFD1.

(4) AFCS Installation (Optional)

The following options are available only with the GFC 700 autopilot. Perform the following steps if any of the following options are installed:

GFC 700 (without YD) Installation Option GFC 700 (with YD) Installation Option

Enhanced AFCS Support

NOTE: Enhanced AFCS Support must be loaded to any aircraft with the Enhanced AFCS option. This is required in addition to the Enhanced AFCS Unlock card to fully install the Enhanced AFCS option.

(a) On #1 PFD, rotate the FMS knob to select the SYSTEM UPLOAD page and activate cursor.

(b) In the GROUP field, create a pick list by rotating the small FMS knob and select GFC 700.

(c) Press ENT.

(d) In the ITEM field, create a pick list by rotating the small FMS knob and select the desired option, press ENT.

(e) Press the CHK ALL softkey.

(f) Press LOAD softkey.

(g) Verify the summary field lists the software and configuration are complete as required and PASS is displayed next to both appropriate boxes.

(h) Press ENT to accept.

(i) For other installed options repeat steps (d) thru (i) by rotating the large FMS knob and highlighting the desired option in the ITEM Field.

(j) De-activate the cursor.

(5) Calibration Option Load

NOTE: Only load configuration option once. Reloading of configuration option could result in a loss of system calibrations.

(a) Go to the System Upload page and activate cursor.

(b) In the Group field, create a pick list by rotating the small FMS knob and select Options.

(c) Press ENT.

(d) In the ITEM field, create a pick list by rotating the small FMS knob and select the desired calibration (Flaps Calibration Initialization or Fuel Calibration Initialization.

(e) Press LOAD softkey.

(f) Verify the Summary field lists the software and configuration are complete as required and PASS is displayed next to both appropriate boxes.

(g) Press UPDT CFG softkey.

(h) Press ENT to select YES when prompted “Update Config Module?”.

(i) Press ENT to confirm completion.

NOTE: Upon completion of all required baseline config, option config, and calibration loading procedures, remove the Loader card from top slot of PFD1.

(6) Splash Screen Update

(a) Remove power from PFD1, PFD2, and MFD.

(b) Insert PA-34 Loader Card into the PFD2 top card slot.

(c) Apply power to PFD2.

(d) Press the NO softkey at the following prompt:

“DO YOU WANT TO UPDATE SYSTEM FILES?”

(e) Press the YES softkey at the following prompt:

“DO YOU WANT TO UPDATE THE CUSTOM GRAPHIC FILES (EG, SPLASH SCREEN)?”

(f) At the prompt, press any key to continue.

(g) Remove PA-34 Loader Card from the PFD2 top card slot.

(h) Repeat steps (b) thru (g) for MFD, and then PFD1.

NOTE: Upon completion of splash screen update, remove the Loader card from top slot of PFD1.

(7) Software Load Confirmation

(a) Each PFD and MFD running in configuration mode.

(b) On PFD1, select the “Systems Status Page”. Activate the cursor and toggle to the LRU window.

(c) Highlight each of the listed items in the LRU window and verify the software part number and software version against the current numbers and versions shown in the appropriate Required Equipment List (REL), “Chart 31” on page 342502165. Some items may be within the expandable list (denoted by a “+” next to the LRU) of the corresponding item. To expand the list, press ENT with the desired LRU highlighted.

NOTE: Not all of the software P/Ns and versions on the Systems Status page need to be verified, only those recorded in the appropriate REL.

(d) De-activate the cursor.

NOTE: If any software is not successfully loaded, do not continue with post installation procedures. Troubleshoot and resolve the issue before continuing.

(e) Re-start PFD1, PFD2 and MFD.

(f) On MFD splash screen (“Figure 41” on page 342502155), verify appropriate version number (“Piper PA-34-220T System 3156.01) is displayed in the upper right corner of the MFD.

(g) Select softkey on MFD to proceed beyond splash screen.

(h) Verify that no manifest (software mismatch) alerts are shown on any of the displays.

(8) Configuration and Setup

(a) XM Satellite Radio

See “Chart 18” on page 34250279 if XM Satellite Radio is installed and requires activation.

(b) Aircraft Registration Number Entry

NOTE: The large FMS knob changes Groups and the small FMS knob changes Pages within a Group.

1) Place the G1000 NXi/PA-34 loader card into the top slot of the PFD
2) Start the #1 PFD, MFD, and #2 PFD in the configuration mode by holding the ENT key while applying power.
3) Select the SYSTEM group by rotating the large FMS knob, and then select the Aircraft Configuration page by rotating the small FMS knob.
4) Activate the cursor and highlight the AIRCRAFT REGISTRATION field. Use the small/large FMS knob to enter the aircraft registration number.
5) Ensure that an ICAO ADDRESS is automatically populated
6) Press the SET GTS softkey to configure traffic system (if installed). Press ENT to select OK at the dialog that pops up. Then press ENT again after configuration is completed.
7) Press the SET GTX1 softkey to configure transponder. Press ENT to select OK at the dialog box that pops up.

(c) Flaps Calibration

NOTE: The large ‘FMS’ knob changes Groups and the small ‘FMS’ knob changes Pages within a Group.

1) Start the PFD1, MFD and PFD2 displays in the Configuration Mode by holding the ENT key while applying power.
2) On PFD #1, navigate to the CAL page group, then to the DAT CALIBRATION page.

NOTE: The FLAPS AND TRIM CALIBRATION page is not to be used to calibrate flaps for the G1000 NXi PA-34 Seneca V.

NOTE: DAT CAL TBL 0, with the INPUT value of GEN_GAUGE_29 is the table that contains the flaps calibration data. All further references in this procedure are related to DAT CAL TBL 0. DO NOT modify any other tables in this page.

3) Ensure the flaps system is powered and functioning properly.
4) Set the flaps to the 0 degree detent position.
5) On PFD1, press the FMS knob to bring up the cursor and highlight the first point in the RAW PNTS column of the table.
6) Press the CAL softkey, and press the ENT key to acknowledge the pop-up box. The first RAW PNTS value will now be modified.
7) Verify that the RAW VAL is unchanged from before the calibration (except for noise in the sensor sample updates), and that the CAL VAL now matches the desired flap position value (except for noise in the sensor sample updates).
8) Repeat steps 4) thru 7) for each of the 2nd, 3rd and 4th rows of the table, setting the flaps position to the 10 degree, 25 degree, and 40 degree detents (respectively).

(d) GSR 56 Activation Procedure (If Installed)

Contact FlightDataServices@Garmin.com, PH: 866-739-5687 to activate Iridium services.

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(e) Options Unlock Procedures

WARNING: PRIOR TO UNLOCK OF ANY OF THE FOLLOWING OPTIONS, ENSURE THAT THE SOFTWARE IS PROPERLY LOADED AND THAT THE CONFIGURATION MODULE IS FUNCTIONAL.

NOTE: Once an unlock SD card is utilized for a particular installation the card becomes locked to the system ID for that aircraft. The unlock cards must be retained for that aircraft should there be a need to unlock the feature again in the field.

1) CHARTVIEW Unlock (Optional)

a) Remove power from both PFD's and MFD by opening the PFD and MFD circuit breakers.

NOTE: The Chartview Enable card can only enable Charts on one system (one aircraft). A new Chartview Enable card must be used for each aircraft.

b) Insert the Chart Unlock card, in the upper slot of the PFD1.

c) While holding the ENT key on both PFD's and MFD, restore power by closing the circuit breakers.

d) When the words "INITIALIZING SYSTEM" appear in the upper left corner of the MFD and PFD's release the ENT key.

e) On the PFD1, go to the System Upload page with the small FMS knob.

f) Activate the cursor and use the small FMS knob to create a pick list and select Configuration Files in the GROUP field. Press ENT.

g) Use the small FMS knob to create a pick list and select Enable Chartview in the ITEM field. Press ENT.

h) Verify there is a check mark in the box in the configuration column for Airframe.

i) Press the LOAD softkey.

j) Monitor the status of the upload. When the upload is Complete and Pass is displayed, press ENT.

k) De-activate the cursor.

l) Power down the system and remove the Chartview Enable card from the PFD.

m) After the Chart function is unlocked, the card should be stored with the aircraft records.

2) Synthetic Vision Unlock (Optional)

a) Remove power from the #1 PFD by opening the PFD1 circuit breaker.

NOTE: The Synthetic Vision Enable card can only enable Synthetic Vision on one system (one aircraft). A new Synthetic Vision Enable card must be used for each aircraft.

b) Insert the Synthetic Vision Unlock card in the upper slot of the #1 PFD.

c) While holding the ENT key on PFD1, restore power by closing the PFD1 circuit breaker.

d) When the words “INITIALIZING SYSTEM” appear in the upper left corner of the PFDs and MFD release the ENT key.

e) On the #1 PFD, go to the SYSTEM UPLOAD page with the small FMS knob.

f) Activate the cursor and use the small FMS knob to create a pick list and select Configuration Files in the GROUP field. Press ENT.

g) Use the small FMS knob to create a pick list and select Enable SVS Dual PFD in the ITEM field. Press ENT.

h) Press the LOAD softkey.

i) When the upload is Complete and Pass is displayed, press ENT.

j) De-activate the cursor.

k) Power down the system and remove the Synthetic Vision Unlock card from the PFD.

l) After the Synthetic Vision function is unlocked, the card should be stored with the aircraft records.

3) TAWS Unlock (Optional)

a) Remove power from the #1 PFD by opening the PFD1 circuit breaker.

NOTE: The TAWS Enable card can only enable TAWS on one system (one aircraft). A new TAWS Enable card must be used for each aircraft.

b) Insert the TAWS Unlock card in the upper slot of the PFD1.

c) While holding the ENT key on PFD1, restore power by closing the PFD1 circuit breaker.

d) Activate the cursor and use the small FMS knob to create a pick list and select Configuration Files in the GROUP field. Press ENT.

e) Use the small FMS knob to create a pick list and select Enable TAWS in the ITEM field. Press ENT.

f) Press the LOAD softkey.

g) Monitor the status of the upload. When the upload is Complete and Pass is displayed, press ENT.

h) De-activate the cursor.

i) Power down the system and remove the TAWS Unlock card from the PFD.

j) After the TAWS function is unlocked, the card should be stored with the aircraft records.

4) Enhanced AFCS Unlock (Optional)

The Enhanced AFCS Support option must be loaded before using the Enhanced AFCS Enable card for the system to properly operate. See “AFCS Installation (Optional)” on page 342502110.

a) Remove power from the #1 PFD by opening the PFD1 circuit breaker.

NOTE: The Enhanced AFCS Enable card can only enable TAWS on one system (one aircraft). A new Enhanced AFCS Enable card must be used for each aircraft.

b) Insert the Enhanced AFCS Enable card in the upper slot of the #1 PFD.

c) While holding the ENT key on PFD1, restore power by closing the PFD1 circuit breaker.

d) When the words “INITIALIZING SYSTEM” appear in the upper left corner of the PFDs and MFD release the ENT key.

e) On the #1 PFD, go to the SYSTEM UPLOAD page with the small FMS knob.

f) Activate the cursor and use the small FMS knob to create a pick list and select Enhanced AFCS in the GROUP field. Press ENT.

g) Use the small FMS knob to create a pick list and select Enable Enhanced AFCS in the ITEM field. Press ENT.

h) Verify there is a check mark in the box in the configuration column for Airframe.

i) Press the LOAD softkey.

j) Monitor the status of the upload. When the upload is Complete and Pass is displayed, press ENT.

k) De-activate the cursor.

l) Power down the system and remove the Enhanced AFCS Enable card from the PFD.

m) After the Enhanced AFCS function is unlocked, the card should be stored with the aircraft records.

5) Surface Watch Unlock (Optional)
   a) Remove power from the #1 PFD by opening the PFD1 circuit breaker.

      NOTE: The SurfaceWatch Enable card can only enable Surface Watch on one system (one aircraft). A new Surface Watch Enable card must be used for each aircraft.

   b) Insert the SurfaceWatch Enable card in the upper slot of the #1 PFD.
   c) While holding the ENT key on PFD1, restore power by closing the PFD1 circuit breaker.
   d) When the words “INITIALIZING SYSTEM” appear in the upper left corner of the PFDs and MFD release the ENT key.
   e) On the #1 PFD, go to the SYSTEM UPLOAD page with the small FMS knob
   f) Activate the cursor and use the small FMS knob to create a pick list and select Enable_SurfaceWatch in the GROUP field. Press ENT.
   g) Use the small FMS knob to create a pick list and select Enable SurfaceWatch in the ITEM field. Press ENT.
   h) Verify there is a check mark in the box in the configuration column for Airframe.
   i) Press the LOAD softkey.
   j) Monitor the status of the upload. When the upload is Complete and Pass is displayed, press ENT.
   k) De-activate the cursor.
   l) Power down the system and remove the SurfaceWatch Enable card from the PFD.
   m) After SurfaceWatch function is unlocked, the card should be stored with the aircraft records.

6) Ground Clutter Suppression and Turbulance Detection Unlock (Optional)
   a) Remove power from the #1 PFD by opening the PFD1 circuit breaker.

      NOTE: The GCS & TD Enable card can only enable GCS & TD on one system (one aircraft). A new GCS & TD Enable card must be used for each aircraft.

   b) Insert the GCS & TD Enable card in the upper slot of the #1 PFD.
   c) While holding the ENT key on PFD1, restore power by closing the PFD1 circuit breaker.
   d) Activate the cursor and use the small FMS knob to create a pick list and select Enable_TD_GCS in the GROUP field. Press ENT.
   e) Use the small FMS knob to create a pick list and select Doppler, TD, GCS Enablement in the ITEM field. Press ENT.
   f) Verify there is a check mark in the box in the configuration column for Airframe.
   g) Press the LOAD softkey.
   h) Monitor the status of the upload. When the upload is Complete and Pass is displayed, press Enter.
   i) De-activate the cursor.
   j) Power down the system and remove the GCS & TD Enable card from the #1 PFD.
   k) After GCS & TD function is unlocked, the card should be stored with the aircraft records.

(f) Aviation Database Loading Procedures

1) GARMIN GDU 10XX Terrain Data Card Loading Procedure

NOTE: Garmin Field Service Engineering, resident at Piper, will be responsible for supplying updates for the GDU 10XX Terrain Data Base Card.

a) Remove power from the PFD and MFD by pulling the PFD and MFD circuit breakers.

b) Install databases (SafeTaxi, Terrain 4.9 Arc Second, Airport Directory, Basemap, Obstacle D82, FliteCharts, IFR/VFR Charts, NAV Database) on Database Card utilizing TimerTW.

c) Install Database Card in the bottom slot of the MFD.

d) Restore power to the PFDs and MFD.

e) The MFD will load the databases and automatically sync them to the PFD. Status can be monitored on the Databases page of the Aux group on the MFD.

f) After database sync is completed, remove power from the PFDs and MFD and remove the card from the bottom slot of the MFD.

g) Restore power to the PFDs and MFD.

h) This completes the installation of the Terrain Database Card.

2) JEPPS CHARTS Aircraft Loading Procedure

a) Remove power from the MFD by pulling the MFD circuit breaker.

b) Install Jepps Charts Card in the bottom slot of the MFD.

c) Restore power to the MFD by closing the MFD circuit breaker.

d) Press the ENT key to confirm the database update.

e) After the update completes, the MFD starts in normal mode.

f) The Jepps Charts are now loaded in the MFD and do not need to be loaded in the PFD.

g) The Jepps Charts Card remains in the bottom slot of the MFD.

M. Post-Installation Procedures

CAUTION: ENTER “MAINTENANCE MODE” WHEN ACTIVATING THE GARMIN G1000 NXi SYSTEM ON THE GROUND TO CONDUCT TESTS, INSPECTIONS, ETC. SEE “MAINTENANCE MODE” ON PAGE 3425028.

NOTE: This entire procedure must be successfully accomplished in order for the G1000 NXi system to be airworthy in your airplane.

NOTE: The following checks MUST be performed prior to electrical power-up with the associated wiring connected to the electronic equipment. Failure to perform these checks prior to powering up may result in damage to the equipment.

(1) Bus Power Check

(a) Do not apply external power to the aircraft.

(b) Verify that all circuit breakers are set.

(c) Turn ON Battery Master switch with Avionics switch and the Emergency switch in the “OFF” position.

1) Verify Avionics Bus 1, Avionics Bus 2, and Emergency Bus are not powered.
2) Pull Non-Ess Bus 1, Non-Ess Bus 2 and Lighting Bus circuit breakers.
3) Verify Non-Ess Bus 1, Non-Ess Bus 2 and Lighting Bus are not powered.
4) Reset Non-Ess Bus 1 circuit breaker.
5) Verify Non-Ess Bus 1 is powered. Verify Non-Ess Bus 2 and Lighting Bus are not powered.
6) Pull Non-Ess Bus 1 circuit breaker and reset Non-Ess Bus 2 circuit breaker.
7) Verify Non-Ess Bus 2 is powered. Verify Non-Ess Bus 1 and Lighting Bus are not powered.
8) Pull Non-Ess Bus 2 circuit breaker and reset Lighting Bus circuit breaker.
9) Verify Lighting Bus is powered. Verify Non-Ess Bus 1 and Non-Ess Bus 2 are not powered.
10) Pull Lighting Bus circuit breaker.
11) Turn ON Avionics Master switch.
12) Verify Non-Ess Bus 1, Non-Ess Bus 2 and Lighting Bus are not powered.
13) Verify Ess Bus 1, Ess Bus 2, Av Bus 1, and Av Bus 2 are powered.
14) Pull Ess Bus 1 circuit breaker.
15) Verify Ess Bus 1 is not powered. Verify Ess Bus 2, Av Bus 1, and Av Bus 2 are powered
16) Reset Ess Bus 1 circuit breaker. Pull Ess Bus 2 circuit breaker.
17) Verify Ess Bus 2 is not powered. Verify Ess Bus 1, Av Bus 1 and Av Bus 2 are powered
18) Pull AV Bus Tie circuit breaker.
19) Verify Ess Bus 2 and Av Bus 2 are not powered. Verify Ess Bus 1 and Av Bus 1 are powered.
20) Pull Av Bus 1 and Av Bus 2 circuit breakers. Reset Ess Bus 2 circuit breaker.
21) Verify Ess Bus 1 and Ess Bus 2 are powered. Verify Av Bus 1 and Av Bus 2 are not powered.

(d) Turn the Emergency switch to ARM, then turn OFF the Battery Master switch and Avionics switches.

1) Verify Ess Bus 1, Ess Bus 2, Non-Ess Bus 1, Non-Ess Bus 2, Av Bus 1, Av Bus 2 and Lighting Bus are not powered.

2) Verify Emergency Bus is powered.

(e) Turn OFF the Emergency switch, the Battery Master switch and Avionics Master Switch.

1) Verify the Battery Bus is powered.

2) Verify the Emergency Bus and Main Bus are not powered.

(2) Wiring Harness Checkout

The following is required only when a harness has been repaired, modified, or replaced, but is recommended when troubleshooting.

Prior to installing any LRUs, the wiring harness should be checked for proper connections to the aircraft systems and other avionics systems. Point to point continuity should be checked to expose any faults such as shorting to ground. Any faults or discrepancies found should be corrected before proceeding. Refer to wiring schematics in chapter 91-00-00 of this manual.

(3) Initial Display Testing

1) Verify COM/NAV fields are valid in the top corners of the PFDs.

2) Verify altitude, airspeed, vertical speed, heading and OAT fields are valid on the PFDs.

3) Verify no BACKUP PATHs alerts on the PFDs.

4) Select Reversionary Mode on GMA 1360 Audio Panel. Verify #1 PFD, #2 PFD, and MFD display valid attitude, altitude, airspeed, vertical speed, and engine instruments. De-activate the Reversionary Mode.

5) Verify flap position indication on MFD engine page matches selected flap angles at each setting (0°, 10°, 25°, 40°).

6) BAGGAGE DOOR annunciation:

a) Open door.

b) Verify “BAGGAGE DOOR” annunciation is an advisory (White).

c) Close door.

d) Verify “BAGGAGE DOOR” annunciation is cleared from display.

7) HYDR PUMP ON and GEAR DOWN discrete I/O’s.

a) Put the PFDs in configuration mode by pressing the ENT key while applying power.

b) Pull MFD circuit breaker. Reset MFD circuit breaker. At MFD splash screen, press softkeys in the following sequence: 1, 2, 3, 1, then press the far right softkey.

c) Suspend the aircraft on jacks.

d) Select reversionary mode on GMA 1360 audio panel. Verify Maintenance Mode CAS advisory on MFD. Deselect reversionary mode on GMA 1360 audio panel.

e) Using the FMS knob on PFD1 go to the IO Configuration page of the GIA group.

f) Activate the cursor and using the large FMS knob scroll to channel “IN 1A (Right), IN 11A (Nose), and IN* 12A (Left)”. Verify the DATA box next to CHANNEL “IN 1A, IN 11A, and IN 12A” is green with the landing gear down and black with the landing gear up (this verifies the GEAR DOWN discrete I/O is functioning).

g) Verify that L GEAR DOWN, R GEAR DOWN, AND N GEAR DOWN indications (green circles) are displayed in the EIS strip on the MFD.

h) Activate the cursor and using the large FMS knob scroll to CHANNEL “IN 16”.

i) Retract the landing gear and verify the DATA box next to CHANNEL “IN 16” is green during operation of the hydraulic pump and black when the hydraulic pump is off (this verifies the HYDR PUMP ON discrete I/O is functioning).

j) Verify that L GEAR, R GEAR, and N GEAR Transition Indication is displayed as White square with crosshatching in the EIS strip on the MFD while the gear is retracting.

k) Activate the cursor and using the large FMS knob scroll to channel “IN 2A (Right), IN 10A (Nose), and IN 13A (Left)”. Verify the DATA box next to CHANNEL “IN 2A, IN 10A, and IN 13A” is green with the landing gear up and black with the landing gear down (this verifies the GEAR UP discrete I/O is functioning).

l) Verify that L GEAR, R GEAR, and N GEAR are displayed as WHITE hollow circles in the EIS strip on the MFD when the gear is up.

m) Select the gear down.

n) Use the large FMS knob to highlight the Select Unit field. Rotate the small FMS knob to generate a pick list and select GIA 2. Press ENT.

o) Repeat steps f) through m).

p) Lower aircraft to the ground.

8) Stall Annunciation

a) Defeat LH main squat switches.

b) With flaps at 0 degrees, activate the outboard stall vane.

c) Verify “Stall” aural alert.

d) Move flaps to 25 degrees.

e) Activate the outboard stall vane and verify “Stall” aural alert.

f) Remove squat switch clamps.

9) Pitot Heat Annunciations

a) With power removed from aircraft, remove connector J727.

b) Apply ground power to aircraft.

c) Verify “PITOT HT OFF” annunciation appears on display.

d) Turn the Pitot Heat switch to the ON position.

e) Verify “PITOT HT FAIL” annunciation appears on display.

f) Remove ground Power to aircraft and reconnect J727.

10) Avionics Cooling Fan Fail Annunciation Test

a) Pull the “AVIONICS FAN” circuit breaker.

b) Verify “AVI FAN FAIL” advisory message annunciates after 10 seconds.

c) Close the “AVIONICS FAN” circuit breaker.

d) Verify “AVI FAN FAIL” advisory message is cleared from the PFD display.

e) Pull the “MFD FAN” circuit breaker.

f) Verify “MFD FAN FAIL” advisory message annunciates after 20 seconds.

g) Close the “MFD FAN” circuit breaker.

h) Verify “MFD FAN FAIL” advisory message is cleared from the PFD display.

i) Pull the “PFD 2 FAN” circuit breaker.

j) Verify “PFD2 FAN FAIL” advisory message annunciates after 20 seconds.

k) Close the “PFD 2 FAN” circuit breaker.

l) Verify “PFD2 FAN FAIL” advisory message is cleared from the PFD display.

m) Pull the “PFD 1 FAN” circuit breaker.

n) Verify “PFD1 FAN FAIL” advisory message annunciates after 10 seconds.

o) Close the “PFD 1 FAN” circuit breaker.

p) Verify “PFD1 FAN FAIL” message advisory is cleared from the PFD display.

11) Hour Meter Operation Test

a) For aircraft with Hobbs Configuration B, defeat both squat switches. Hobbs shall run while aircraft is “in air”.

b) For aircraft with Hobbs Configuration C, Hobbs shall run whenever GIA 1 is powered and reports OK status.

12) Verify Main Bus Voltage Indication.

a) With external power removed and Battery Master switch set to ON,

b) Verify Main bus voltage indication.

c) With external power removed, Battery Master Switch set to OFF, and Emergency Switch set to ARM.

d) Verify Emergency bus voltage indication.

(4) G1000 NXi Connection Check

Perform the following steps to verify the following:

NOTE: Do not have External Power on the aircraft while conducting the following tests.

(a) AHRS and ADC data path check

1) In the GIA page group, go to the GIA SERIAL CONFIGURATION page.

2) With GIA1 selected in the SELECT UNIT window, verify the RS232 Channel 1 (GDC1) and RS232 Channel 6 (GRS1) indicators are green.

3) With GIA1 selected in the SELECT UNIT window, verify the ARINC 429 IN5 (GDC1) and ARINC 429 IN6 (GRS1) indicators are green.

4) With GIA2 selected in the SELECT UNIT window, verify the RS232 Channel 1 (GDC2) and RS232 Channel 6 (GRS2) indicators are green.

5) With GIA2 selected in the SELECT UNIT window, verify the ARINC 429 IN5 (GDC2) and ARINC 429 IN6 (GRS2) indicators are green.

6) In the GDU page group, go to the SERIAL CONFIGURATION page.

7) With PFD1 selected in the SELECT UNIT window, verify the ARINC 429 IN1 (GRS1) and ARINC 429 IN2 (GDC1) indicators are green.

8) With PFD2 selected in the SELECT UNIT window, verify the ARINC 429 IN1 (GRS2) and ARINC 429 IN2 (GDC2) indicators are green.

(b) PFD and MFD Ethernet connection check

1) Re-start PFD2, PFD1 and MFD in normal mode.

2) On the circuit breaker panel (Av Bus 2), open the MFD circuit breaker:

3) Verify NAV1, COM1, NAV2 and COM2 remain valid on both PFD1 and PFD2.

4) Close the MFD circuit breaker and wait for MFD to initialize:

5) On the circuit breaker panel (Av Bus 2), open the PFD2 circuit breaker:

6) Press the DISPLAY BACK UP button.

7) Verify NAV2 and COM2 are each replaced by a red X and NAV1 and COM1 remain valid on both the MFD and PFD1.

8) Press the DISPLAY BACK UP button to return to normal mode.

9) Close the PFD2 circuit breaker and wait for PFD2 to initialize:

10) On the circuit breaker panel (Ess Bus 1), open the PFD1 circuit breaker:

11) Press the DISPLAY BACK UP button.

12) Verify NAV1 and COM1 are each replaced by a red X and NAV2 and COM2 remain valid on both the MFD and PFD2.

13) Press the DISPLAY BACK UP button to return to normal mode.

14) Close the PFD1 circuit breaker and wait for PFD1 to initialize:

(5) Engine Data Check (GIA1 or GIA2 Inoperative)

NOTE: This procedure should be accomplished in conjunction with the engine run-up. Valid engine indications will be needed. Do the following to verify the engine data availability and validity, with GIA1 or GIA2 inoperative:

(a) Ensure the G1000 NXi is in normal mode.

(b) With the engine running, note the indication values for MAP, TIT and oil pressure.

(c) On the circuit breaker panel (Ess Bus 1), open the INTEG AV1 circuit breaker.

(d) Verify the values for MAP, TIT and oil pressure remain unchanged.

(e) Close the INTEG AV1 circuit breaker and wait until the COM1 and NAV1 fields are valid.

(f) On the circuit breaker panel (Av Bus 2), open the INTEG AV2 circuit breaker.

(g) Verify the values for MAP, TIT and oil pressure remain unchanged.

(h) Close the INTEG AV2 circuit breaker and wait until the COM2 and NAV2 fields are valid.

(i) Put the G1000 NXi in Reversionary mode.

(j) Repeat steps (b) thru (h), verifying engine gauge indications on PFD1, MFD and PFD2.

(k) The Engine Data Check is complete.

(6) G1000 NXi Miscompare Checks (Optional)

This procedure will check the AHRS, airspeed and altitude miscompare monitors. Accomplish the following checks with the aircraft positioned where it can receive GPS signals and magnetic heading.

NOTE: Access to AHRS1 and AHRS2 will be required during this test.

(a) Ensure the G1000 NXi is operating in normal mode.

(b) Connect a pitot/static test set to the aircraft ADC1 pitot and static ports (Pilot's side). (Do not connect the pitot/static tester to ADC2 ports at this time.)

(c) Set the baro-correction on PFD1 and PFD2 to 29.92.

(d) Slowly increase the pitot/static test set to simulate an ADC1 / ADC2 altitude miscompare of greater than 200 ft., verify ADC2 altitude does not increase and amber comparator window text "ALT MISCOMP" is displayed on PFD1 and PFD2.

(e) Reduce ADC1 altitude to ambient pressure.

(f) Repeat Step (e) with the pitot/static test set connected to the aircraft ADC2 pitot and static ports.

(g) Use the pitot/static test set to simulate an airspeed of 40 kts for ADC1 and 55 kts for ADC2. Verify the amber comparator window text "IAS MISCOMP" is displayed on PFD1 and PFD2.

(h) Use the pitot/static test set to simulate an airspeed of 55 kts for ADC1 and 40 kts for ADC2. Verify the amber comparator window text "IAS MISCOMP" is displayed on PFD1 and PFD2.

(i) Use the pitot/static test set to simulate an airspeed of 85 kts for ADC1 and 95 kts for ADC2. Verify the amber comparator window text "IAS MISCOMP" is displayed on PFD1 and PFD2.

(j) Use the pitot/static test set to simulate an airspeed of 95 kts for ADC1 and 85 kts for ADC2. Verify the amber comparator window text "IAS MISCOMP" is displayed on PFD1 and PFD2.

(k) Reduce ADC1 and ADC2 airspeeds to 0 kts.

(l) Slowly rotate AHRS1 along the lateral (pitch) axis to a pitch attitude of greater than 5 degrees, verify the following:

1) AHRS2 pitch attitude does not change.

2) An amber “PIT MISCOMP” is annunciated on PFD1 and PFD2.

(m) Replace AHRS1 to normal attitude and verify that attitude display on PFD1 displays current aircraft attitude.

(n) Slowly rotate AHRS1 along the longitudinal (roll) axis to a roll attitude of greater than 5 degrees, verify the following:

1) AHRS2 roll attitude does not change.

2) An amber “ROLL MISCOMP” is annunciated on PFD1 and PFD2.

(o) Replace AHRS1 to normal attitude and verify that attitude display on PFD1 displays current aircraft attitude.

(p) Repeat Steps (l) thru (o) for AHRS2.

(q) The G1000 NXi Miscompare checks are complete.

N. LRU and System Test Procedures

CAUTION: ENTER “MAINTENANCE MODE” WHEN ACTIVATING THE GARMIN G1000 NXI SYSTEM ON THE GROUND TO CONDUCT TESTS, INSPECTIONS, ETC. SEE “MAINTENANCE MODE” ON PAGE 3425018.

(1) PFD/MFD Test Procedure

(a) Allow displays to initialize for ~1 minute.

(b) Check that all COM/NAV display fields are valid in the top corners of both PFDs.

(c) Check that attitude, heading, altitude, airspeed, vertical speed and OAT fields are valid within 2 minutes of power up on both PFDs.

(d) Press the SENSOR softkey on each PFD and switch between ADC1 and ADC2. Verify that data from both GDC 72s is valid on both displays.

(e) Press the SENSOR softkey on each PFD and switch between AHRS1 and AHRS2. Verify that data from both GRS 79s is valid on both displays.

(f) Check that the engine instrument fields are valid on the MFD.

(g) Push the red DISPLAY BACKUP button on the pilot-side GMA 1360. Verify that the pilot-side PFD and MFD displays enter reversion mode. MFD should have valid altitude, airspeed, vertical speed, COM1, COM2, NAV1, NAV2 and engine instruments.

(h) De-activate pilot-side reversion mode by pushing the DISPLAY BACKUP button. Verify PFD1 and MFD return to normal display modes.

(i) Repeat Step (a) using the co-pilot-side PFD. Ensure that PFD2 and MFD enter reversion mode and MFD displays valid altitude, airspeed, vertical speed, COMM1, COMM2, NAV1, NAV2 and engine instruments.

(j) De-activate co-pilot’s side reversion mode by pushing the DISPLAY BACKUP button. Verify PFD2 and MFD return to normal display modes.

(k) If TAWS was activated, select the TAWS page (4th page in the MAP group) on the MFD.

1) Verify that the title at the top of the page reads “MAP – TAWS”. If TAWS has not been enabled, the title will read “MAP – TERRAIN PROXIMITY” or “MAP – TERRAIN”.

2) Press the MENU button and select “Test TAWS” from the pop-up menu.

3) After the TAWS test has completed, verify that “TAWS System Test Okay” is heard over the cockpit speaker.

(2) GMA 1360 Test

Except for marker beacon operation, an in-aircraft checkout may be performed in the aircraft with known good microphone, headset, and speaker.

(a) External power applied to aircraft. Apply power to the avionics bus.

(b) Check Pilot and Copilot Intercom System (ICS) to verify the pilot and copilot Bose jacks, and MIC and phone jack operation.

(c) Pilot side only, depress PA Key (for Passenger Address mode) on the Audio Panel, press the pilot MIC key and verify audio is heard over passenger headset.

(d) Verify operation of COM1. Conduct VSWR check per “Chart 27” on page 342502127.

(e) Failsafe Operation Check – Audio Panel to COM #1.

1) Turn the Audio Panel OFF by pulling the AUDIO MKR circuit breaker.

2) Verify COM1 transmits and receives normally over pilot headset.

3) Verify COM1 audio is controlled by the PFD COM audio control.

4) Close the AUDIO MKR circuit breaker.

(f) Verify operation of COM2. Conduct VSWR check per “Chart 27” on page 342502127.

(g) Marker Beacon Test

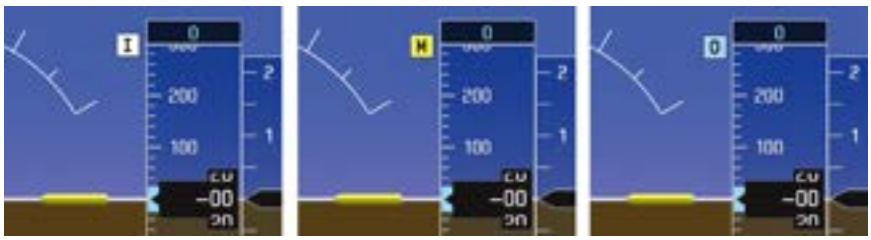
1) Verify that the outer, middle, and inner markers appear on the PFD (See “Figure 36”).

2) Verify Marker Beacon Audio and tones.

3) Verify the MKR MUTE key mutes the Marker Beacon Audio.

4) Verify Marker HI/LO Sensitivity operation.

(h) Verify Passenger ICS is working at all stations.



Marker Beacon Symbology
Figure 36

CHART 27
VSWR TEST

VOLTAGE STANDING WAVE RATIO (VSWR)

The standing wave ratio is a measure of the amount of power transmitted to the antenna, compared to the amount reflected back to the transmitter. S.W.R.'s are a function of the transmission line, the antenna and their installation.

A. Required Equipment

Thruline Wattmeter – Model 43 Bird Electronics, Cleveland, Ohio.

B. S.W.R. Testing:

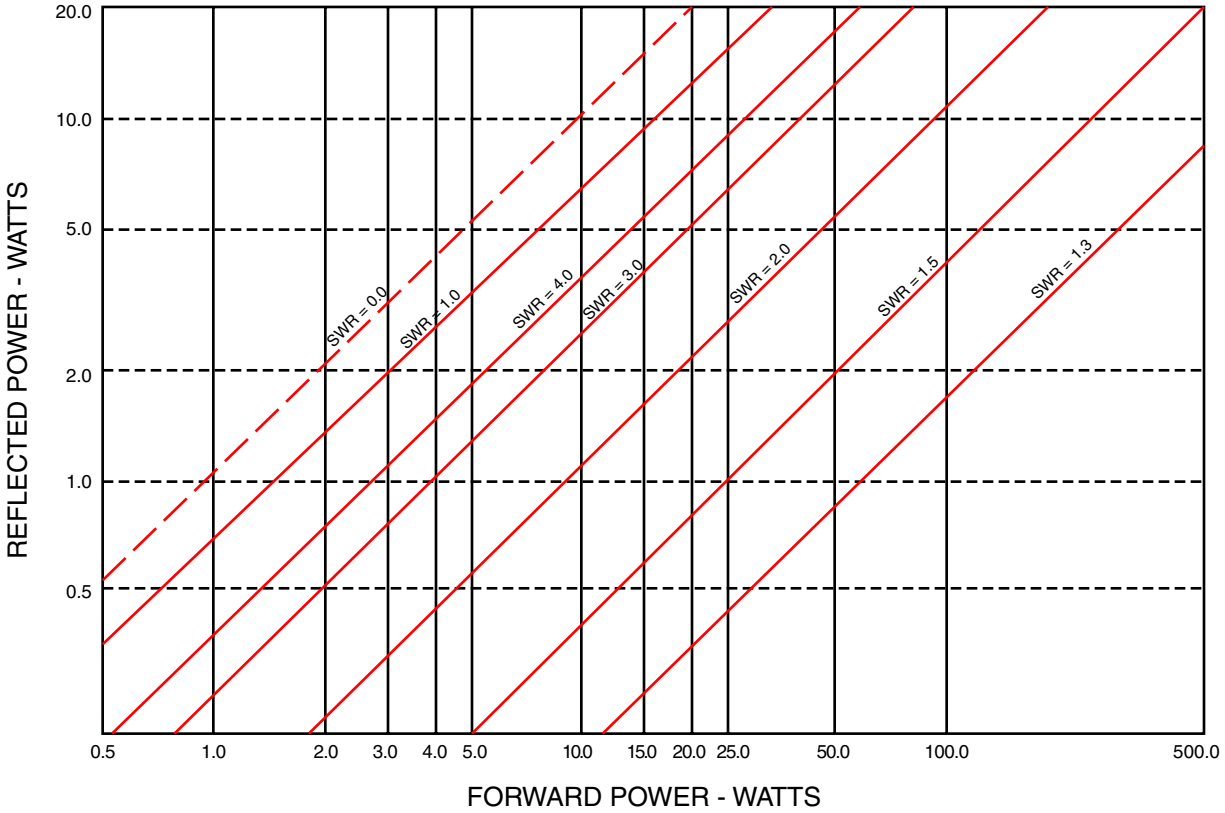
Test the antenna installation using the Thruline Wattmeter.

(1) Check VSWRs at 118.0, 123.0, 128.0, 132.0, and 135.95 MHZ by transmitting on COM 1 and then COM 2.

(2) Determine the VSWR value with the VSWR Conversion Chart below.

(3) VSWRs in excess of 2.5:1 are not acceptable.

VSWR CONVERSION CHART



(3) GIA 64 Test

(a) GPS Signal Acquisition

See “Figure 37”.

The GIA 64 units should normally acquire a GPS navigation solution within 2 minutes of startup, provided the aircraft is outside (or indoors with a GPS repeater). Select the GPS STATUS page on the MFD (4th page in AUX group). Two softkeys on the bottom of the display allow the user to toggle between GPS1 and GPS2. Verify that both receivers show 3D DIFF NAV on the MFD.

Continue to the VHF COM Interference test.

NOTE: It may be necessary to temporarily disable or move away from GPS repeaters while testing, as repeaters may adversely affect GPS receiver performance.



GPS STATUS Page (MFD)
Figure 37

(b) VHF COM Interference Test

This test must be conducted outside. Use of a GPS repeater inside a hangar may result in a failed test.

This procedure assumes that the system is currently set to 25 kHz COM channel spacing. Once the signal acquisition test has been completed successfully, perform the following steps:

1) On the MFD, monitor GPS signal strength bars on the AUX – GPS STATUS page.
2) On the PFD, ensure that the CDI is set to GPS. If it is not, press the 'CDI' softkey until GPS ENR is displayed.
3) Verify that the GPS "INTEG" flag is out of view.
4) Select 121.150 MHz on the COM1 transceiver.
5) Transmit for a period of 35 seconds while monitoring GPS1 signal strength levels.
6) During the transmit period, verify that the GPS "INTEG" flag does not come into view on the PFD and verify that GPS1 does not lose a 3-D navigation solution on the MFD.
7) Repeat steps 5) and 6) and re-transmit while monitoring GPS2 signal levels on the MFD.
8) Repeat steps 4) thru 7) for each of the following frequencies:
   - 121.175 MHz
   - 121.200 MHz
   - 131.250 MHz
   - 131.275 MHz
   - 131.300 MHz
9) Repeat steps 4) thru 8) for the COM2 transceiver (GIA2).
10) On the MFD, select the AUX – SYSTEM SETUP page.
11) Under the COM CONFIG field, change the COM channel spacing from 25 kHz to 8.33 kHz.
12) Go back to the AUX – GPS STATUS page.
13) Select 121.185 MHz on the COM1 transceiver.
14) Transmit for a period of 35 seconds while monitoring GPS1 signal strength levels.
15) During the transmit period, verify that the GPS "INTEG" flag does not come into view on the PFD and verify that GPS1 does not lose a 3-D navigation solution on the MFD.
16) Repeat steps 14) and 15) and re-transmit while monitoring GPS2 signal levels on the MFD.
17) Repeat steps 14) thru 16) for each of the following frequencies:
   - 121.190 MHz
   - 130.285 MHz
   - 131.290 Mhz
18) Repeat steps 14) thru 17) for the COM2 transceiver (GIA2).
19) On the MFD, select the AUX – SYSTEM SETUP page and change the COM channel spacing back to 25 kHz.
20) Continue to the "VOR/LOC/GS Test" on page 342502130.

(c) VOR/LOC/GS Test

1) Perform NAV 1 and NAV 2 accuracy tests.
   a) Full scale.
   b) Half scale.
   c) OBS operation.
2) Check NAV 1 and NAV 2 audio and Ident.
3) Check NAV 1 and NAV 2 Sensitivity.
4) Perform LOC 1 and LOC 2 accuracy tests.
   a) Full scale.
   b) Half scale.
   c) Check LOC 1 and LOC 2 Sensitivity.
5) Perform GS 1 and GS 2 accuracy tests.
   a) Full scale.
   b) Half scale.
6) Check GS 1 and GS 2 Sensitivity.

(4) GEA 71B Test

On the MFD (normal mode), check the indication for each of the sensor or monitor inputs with the aircraft engines off. In general, verify all engine and system instruments show valid static normal values and markings, with no red Xs or erratic indications. See “Figure 38” (baseline configuration shown) for normal engine instrument markings.

(5) GTX 3X5R Transponder

(a) Configuration

“Aircraft Registration Number Entry” on page 342502112 configures the transponder:

(b) Test

Operation of the GTX 3X5R Mode-A, C, S transponder with ADS-B out is accomplished using PFD1, PFD2 or the MFD. Refer to G1000 NXi Cockpit Reference Guide, for basic operation.

The integrated transponder/altitude reporting system must be verified in accordance with Title 14 of the Code of Federal Regulations (CFR) §§ 91.411 and 91.413, every 24 calendar months, or any time the transponder is removed. This test requires the use of a Mode S ramp generator. Specific instructions for operating the ramp tester are contained in the applicable operator’s manual. Refer to Title 14 CFR Part 43.

(c) GTX 3X5R Extended Squitter Checkout (If installed)

1) Press the XPDR softkey on the PFD.
2) Verify that the ADS-B TX softkey is available and can be toggled (softkey 10).



NORMAL MODE



WHEN ENGINE PAGE DISPLAYED

Normal Engine Instruments
Figure 38

(6) GDC 72 Tests

NOTE: If any connections in the pitot / static system are opened for maintenance, the entire system must be rechecked per Pitot and Static Systems, ”Test” on page 34108.

NOTE: Allow the unit to warm up for 15 minutes before performing the following tests.

Verification of the altimeter and airspeed must be performed using an air data test set (ADTS). The static port and altimeter must be verified in accordance with Title 14 of the Code of Federal Regulations (CFR) § 91.411 and Part 43 Appendix E. The PFD must be in Configuration mode and the MFD must be in Reversionary mode for performing the tests as outlined in Part 43 Appendix E.

(a) Part 43 Appendix E Testing

1) Preparation

CAUTION: CONFIGURATION MODE CONTAINS CERTAIN PAGES AND SETTINGS THAT ARE CRITICAL TO AIRCRAFT OPERATION AND SAFETY. THESE PAGES ARE PROTECTED AND CANNOT BE MODIFIED, UNLESS THE TECHNICIAN IS PROPERLY AUTHORIZED AND EQUIPPED. HOWEVER, MOST PROTECTED PAGES ARE VIEWABLE TO ALLOW SYSTEM AWARENESS FOR TROUBLESHOOTING.

a) Start the G1000 NXi system in normal mode.

b) Remove power to PFD1.

c) Turn PFD1 on in Configuration mode by pressing and holding the ENT key on the PFD while applying power.

d) Release the ENT key after “INITIALIZING SYSTEM” appears in the upper left corner of the PFD.

NOTE: Configuration mode contains certain pages and settings that are critical to aircraft operation and safety. These pages are protected and cannot be modified, unless the technician is properly authorized and equipped. However, most protected pages are viewable to allow system awareness for troubleshooting.

2) Testing

a) Using the FMS knob on the PFD turn to the GRS page group. Use the B ALT field for all CFR Part 43 Appendix E tests for G1000 NXi altitude.

b) Place the MFD in Reversionary mode by pressing the red “display backup” button on the GMA. The baro setting can then be read from the MFD for CFR Part 43 Appendix E tests.

NOTE: The baro setting on the MFD is controlled by PFD1. The baro setting will apply to both GDC1 and GDC2 regardless of the GDC selected on the MFD. The copilot’s display can be ignored.

(b) After completing the tests specified by § 91.411 and Part 43 Appendix E, return both the MFD and the PFD to normal mode.

NOTE: The following tests are above and beyond the requirements set forth in Appendix E, and are required only when appendix E tests are required.

(c) Pitot/Static Airspeed Test

1) Command air data test set (ADTS) to simulate air speeds shown in the table below.

Calibrated Air Speed (Knots)	Allowed tolerance (± Knots)
50	5.0
80	3.5
100	2.0
120	2.0
150	2.0

2) Wait for ADTS to report that target values have been achieved.

3) Verify that computed air speeds shown on the PFD are within the tolerances specified in the OEM maintenance documentation.

(d) Static Port Vertical Speed (Rate of Climb) Test

1) Command ADTS to change the altitude at the rates shown in the table below.

2) Wait for ADTS to report that target rates have been achieved.

3) Verify that the Rate of Climb reported by the Vertical Speed field on the PFD is within the tolerances specified in the table below:

Vertical Speed (feet/minute)	Allowed tolerance, (± feet/minute)
2000	100
0	45
-2000	100

(7) OAT Probe Check

Ensure on-side sensors for PFD1 and PFD2. Ensure the outside air temperature (OAT) probes and a calibrated thermometer stabilize at ambient temperature. Verify that the OAT measurement shown on PFD1 and PFD2, in degrees Celsius, indicate within 2°C of the ambient temperature as measured by the calibrated thermometer.

(8) GRS/GMU Calibration Procedures

See “Chart 28” to determine when various calibrations are required.The following calibration procedures are provided for the GRS 79 and GMU 44B:

- Pitch/Roll Offset:
- Magnetometer Calibration:
- Engine Run-Up Vibration Test:
- Magnetometer Interference Test:

When ready to perform the procedures, shut the PFDs and MFD off by pulling the PFD and MFD circuit breakers. Restart all displays in configuration mode. Follow the steps given for each procedure on-screen at the GRS/GMU CALIBRATION page. Note that the CALIBRATE command cannot be selected and activated until the installer acknowledges all required steps have been carried out by pressing the ENT key on each step.

CHART 28
G1000 NXi - GRS 79 AHRS/GMU 44 MAGNETOMETER CALIBRATIONS

Required GRS/GMU Calibrations			
Condition	Calibrations Required		
	Procedure A: GRS 79 Pitch/Roll Offset	Procedure B: GRS/GMU Magnetic Calibration	Procedure C: Engine Run-up Vibration Test
Either GMU 44B was removed and reinstalled. (no change in serial number)	None Required. Continue to GRS/GMU Test section.		
GMU 44B was replaced with new unit. (New serial number)		X	
GRS 79 AHRS was removed and/or replaced. The mounting tray was NOT removed and the mounting tray bolts were NOT loosened.	None Required. Continue to GRS/GMU Test section.		
GRS 79 AHRS was removed and/or replaced. The mounting tray WAS removed and/or mounting tray bolts WERE loosened.	X	X	X
GRS 79 AHRS Configuration Module was replaced.	X	X	X

(9) GRS 79 Pitch/Roll Offset Calibration (Procedure A)

This procedure must be performed for both GRS 79 units installed in the aircraft. This first procedure must be carried out with the engine OFF.

(a) Level the aircraft to within ±0.25° of zero pitch and zero roll using a digital level. (Refer to 8-20-00.)

(b) Start the PFDs and MFD in Configuration mode by holding the ENT key on each display while closing the circuit breaker.

(c) Go to the GRS Page Group on both PFDs by rotating the large FMS knob.

(d) Select the GRS/GMU Calibration page by rotating the small FMS knob. This page is protected and requires a keystroke password to perform this test. Press the following softkeys in sequence on each PFD:

1) 9
2) 10
3) 11
4) 12 (far right softkey)

(e) Activate the cursor by pressing the small FMS knob and highlight the SELECT GRS UNIT window. Select GRS 79 #1 ON PFD1 AND GRS 79 #2 ON PFD2 (if dual sensors are installed).

(f) Highlight the SELECT PROCEDURE window and select **PITCH/ROLL OFFSET** on each PFD.

(g) Press the ENT key.

(h) Follow the checklist items displayed on each PFD and press the ENT key as each step is completed or confirmed.

(i) When the CALIBRATE field is blinking on both PFDs, press the ENT key on each PFD to begin the procedure.

(j) After several seconds, a new checklist appears in the lower half of the PFD. Press the ENT key as each step is confirmed. When the CONFIRM AIRCRAFT IS LEVEL field is blinking on both PFDs, press the ENT key to continue.

(k) The result of the pitch/roll offset compensation is displayed on each PFD. If successful, the PFDs report the successful calibration and return to normal operation.

(l) Press the ENT key on each PFD to conclude this procedure.

(m) Restart all displays in normal mode.

(10) Compass Rose Evaluation of Magnetic Disturbances for Magnetometer Calibration Procedure (Optional)

NOTE: The “GRS 79/GMU 44B Magnetometer Calibration (Procedure B)” on page 342502137 must be carried out at a site that is free of magnetic disturbances. If it is unsure whether the site is ‘clean’, the technician should verify that the site is ‘clean’ by performing the following procedure. The technician may skip the Compass Rose Evaluation Procedure if the site condition is acceptable.

NOTE: Typically, a compass rose is an acceptable location to perform the magnetometer calibration procedure. However, because not all compass roses are well maintained, even an existing compass rose should be regularly evaluated using the method described here to determine if it is free of magnetic disturbances. If evaluation of an existing compass rose indicates that magnetic disturbances are present, then an alternative location must be found to perform the Magnetometer Calibration procedure.

A G1000 NXi-equipped airplane that has completed the pitch/roll offset compensation procedure can be used to evaluate a candidate site for magnetic disturbances and determine whether it is a suitable location to perform the magnetometer calibration procedure. The magnetometer calibration procedure itself contains the logic to simultaneously survey the location for magnetic cleanliness while it is computing the magnetometer calibration parameters. In order to evaluate a candidate site, the Magnetometer Calibration procedure must be performed twice: once turning clockwise around the site, and once turning counter-clockwise. Both times, the procedure should be conducted as described in “GRS 79/GMU 44B Magnetometer Calibration (Procedure B)” on page 342502137 , with the exception of the direction of turns around the site.

NOTE: “GRS 79/GMU 44B Magnetometer Calibration (Procedure B)” on page 342502137 indicates that the Magnetometer Calibration procedure should be performed by making a series of clockwise turns around the site; however, the procedure can also be performed by making counterclockwise turns for the purpose of evaluating the site for magnetic disturbances.

If, upon completion of the Magnetometer Calibration procedure in each clockwise and counter-clockwise direction, the PFD displays the “CALIBRATION SUCCESSFUL / SITE IS CLEAN” message, then the candidate site is sufficiently free of magnetic disturbances and is acceptable for performing the Magnetometer Calibration procedure. It is important to obtain successful result in both the clockwise and counter-clockwise directions to ensure that the magnetometer sweeps over a large enough area at the candidate site.

If, upon completion of the Magnetometer Calibration procedure in either of the two directions, the PFD displays either the “MAG FIELD AT SITE NOT UNIFORM”, or “MAG FIELD AT SITE DIFFERS FROM IGRF MODEL” message, then the site contains magnetic disturbances that are too large.

(11) GRS 79/GMU 44B Magnetometer Calibration (Procedure B)

NOTE: If the GRS 79 was replaced with a new, repaired, or exchanged unit the "GRS 79 Pitch/Roll Offset Calibration (Procedure A)" on page 342502135 must first be successfully accomplished before performing Procedure B.

Calibration Procedure B must be carried out on a compass rose in order to guarantee measurements free of environmental magnetic disturbances. Attempting to carry out this maneuver on a typical ramp area may not yield a successful calibration. The accuracy of the AHRS cannot be guaranteed if this calibration is not performed on a magnetically clean compass rose. If the compass rose condition is not known, it is recommended that the technician follow the guidance in "Compass Rose Evaluation of Magnetic Disturbances for Magnetometer Calibration Procedure (Optional)" on page 342502136.

(a) Start the aircraft engine following the procedures referenced in the POH.

(b) After aircraft engine startup, taxi the aircraft to a properly calibrated compass rose.

CAUTION: CALIBRATION PROCEDURE B MUST BE CARRIED OUT ON A COMPASS ROSE IN ORDER TO GUARANTEE MEASUREMENTS FREE OF ENVIRONMENTAL MAGNETIC DISTURBANCES. ATTEMPTING TO CARRY OUT THIS MANEUVER ON A TYPICAL RAMP AREA MAY NOT YIELD A SUCCESSFUL CALIBRATION. THE ACCURACY OF THE AHRS CANNOT BE GUARANTEED IF THIS CALIBRATION IS NOT PERFORMED ON A MAGNETICALLY CLEAN COMPASS ROSE OR EQUIVALENT.

(c) At the compass rose, align the aircraft to a heading of magnetic north (±5°).

NOTE: This procedure provides instructions for calibrating both GRS 79 magnetometers simultaneously by putting both PFD1 and PFD2 in configuration mode and following the procedure below, using PFD1 to calibrate GRS 79 #1 and PFD2 to calibrate GRS 79 #2 (if installed).

(d) Start the PFDs and MFD in Configuration mode by holding the ENT key on each display while closing the circuit breaker.

(e) Go to the GRS Page Group on both PFDs by rotating the large FMS knob.

(f) Select the GRS/GMU Calibration page by rotating the small FMS knob. This page is protected and requires a keystroke password to perform this test. Press the following softkeys in sequence on each PFD:

1) 9
2) 10
3) 11
4) 12 (far right softkey)

(g) Activate the cursor by pressing the small FMS knob and highlight the SELECT GRS UNIT window. Select GRS 79 #1 ON PFD1 AND GRS 79 #2 (if installed) ON PFD2.

(h) Highlight the SELECT PROCEDURE window and select **MAGNETOMETER** on each PFD.

(i) Press the ENT key.

(j) Use the cursor to highlight the BEFORE CALIBRATION window on each PFD.

(k) Follow the checklist items displayed on each PFD and press the ENT key as each step is completed or confirmed.

(l) When the CALIBRATE field is blinking on both PFDs, press the ENT key on each PFD to begin the procedure.

(m) The PFD displays advise the operator when to turn the aircraft, when to stop, and when to turn again.

(n) Upon instruction to turn, taxi the aircraft in a right turn. After approximately 25° to 30° of turn from the last heading, the PFD displays advise the operator to stop the aircraft.

NOTE: Due to the difficulties in executing smooth, accurate turns, the PFDs may incorrectly interpret a station and instruct to “HOLD POSITION” prior to full completion of a 30° turn. If this scenario is encountered, it is best for the operator to ignore the “HOLD POSITION” command and instead use outside references to complete the approximate 30° turn. Instead of using the PFD instructions to turn as a real-time indication of when to turn, simply judge the 30° (±5°) turn increments of the aircraft by using the compass rose radials. Dwelling at these 30° increments for the time recommended by the PFDs should result in successful calibration.

(o) The PFDs guide the operator to dwell at multiple headings around a complete circle.

NOTE: Due to high winds or excessive airframe vibration, the operator may encounter a condition where a PFD restarts the 18-second countdown without full completion of the previous countdown. If this is encountered more than once for a given station, the operator should begin turning to the next station (approximately 30°). A minimum of 2 successful stations per quadrant (and for each GRS 79) is required, where a successful station is a full 18-second countdown followed by instruction to move. Ensure that at least 2 stations per quadrant are completed. Thus, it may sometimes be required to dwell at a station after a countdown restart. A maximum of 20 stations is allowed for the entire calibration procedure. If too many countdown restarts are encountered, the calibration will fail with the message, “TOO MANY STATIONS.”

(p) Repeat the turn-and-stop process until the PFDs advise that a successful calibration is complete. The GRS 79 AHRS UNITS then enter the normal operational mode. Press the ENT key on each PFD to conclude this procedure.

(12) Engine Run-Up Vibration Test (Procedure C)

NOTE: Calibration Procedure C is performed in order to guarantee that the AHRS mounting is sufficiently rigid and insensitive to vibration. This procedure must be performed for both GRS 79 units installed in the aircraft. Calibration Procedures A and B are not required prior to this procedure.

NOTE: This test must be performed with the interior installed.

(a) Start the PFDs and MFD in Configuration mode by holding the ENT key on each display while closing the circuit breaker.

(b) Go to the GRS Page Group on both PFDs by rotating the large FMS knob.

(c) Select the GRS/GMU Calibration page by rotating the small FMS knob. This page is protected and requires a keystroke password to perform this test. Press the following softkeys in sequence on each PFD:

1) 9
2) 10
3) 11
4) 12 (far right softkey)

(d) Activate the cursor by pressing the small FMS knob and highlight the SELECT GRS UNIT window. Select GRS 79 #1 ON PFD1 AND GRS 79 #2 (if installed) ON PFD2.

(e) Highlight the SELECT PROCEDURE window on each PFD and select **ENGINE RUN-UP TEST**. Press ENT.

(f) Press the ENT key again on each PFD to select the first checklist item displayed.

(g) As each command is complied with and ENT is pressed, the associated box will turn green to confirm the selection has been complied with. When the CALIBRATE field is blinking on both PFDs, press the ENT key on each PFD to begin the procedure.

(h) The PFD displays instruct the operator to gradually increase power from idle to full throttle and back to idle over a period of 1–2 minutes.

(i) When the engine run-up is completed and the engine is back to an idle setting, press the ENT key on each PFD to indicate that the process is complete. When this is done, the TEST COMPLETE field stops blinking.

(j) Each PFD will state if the installation has passed or failed the vibration test. If the test fails, the specific measurements causing the failure are identified and associated numeric values are displayed on the PFD corresponding to each GRS 79 unit. Using the large FMS knob, scroll through the list and verify that each test passed.

NOTE: Should a failure occur, the technician may perform the Engine Run-up test up to three (3) times successively before corrective action must be taken. If the test does not pass after three attempts, then the installation should not be considered reliable until the source of the vibration problem is identified and remedied. In the event of repeated failure of the engine run-up test, record the values that are reported to be out of range for future reference.

The following are potential causes for failure of the engine run-up test:

1) Vibration motion of GRS 79 and/or GMU 44B caused by neighboring equipment and/or supports.
2) Mounting screws and other hardware for GRS 79 and/or GMU 44B not firmly attached.
3) GRS77 connector not firmly attached to unit.
4) Cabling leading to GRS 79 or GMU 44B not firmly secured to supporting structure.
5) An engine / propeller that is significantly out of balance.

(k) Press the ENT key on each PFD to conclude this procedure.

(13) Magnetometer Interference Test

A magnetometer interference test is available for troubleshooting and/or verifying a magnetically 'clean' installation of the GMU 44B. This test exercises various devices on the aircraft that could potentially affect the magnetic field as measured by the GMU 44B (examples include navigation lights, control servos, landing gear motors, etc).

NOTE: This test is used to validate that no electronic device interferes with the operation of the GMU 44B magnetometer. It is highly recommended that this test be performed after installation or maintenance of electrical components on the aircraft and/or for troubleshooting the GMU 44B.

NOTE: External power cart and other aircraft need to be as far away from the test aircraft as possible. Turn Battery Master ON, AV BUS Master ON, Autopilot ON (AP engaged to allow pitch servo movement.), and all circuit breakers in. Verify the Air Conditioning Condenser Blower Motor is OFF.

(a) Start the PFDs and MFD in the Configuration Mode by holding ENT on each display while closing the circuit breaker

(b) Go to the GRS Page Group on both PFDs by rotating the large FMS knob.

(c) Select the GRS/GMU Calibration page, as shown in "Figure 39" on page 342502141, by rotating the small FMS knob. This page is protected and requires a keystroke password to perform this test. Press the following softkeys in sequence on each PFD:

1) 9
2) 10
3) 11
4) 12 (far right softkey)

(d) Activate the cursor by pressing the small FMS knob and highlight the SELECT GRS UNIT window. Select GRS 79 #1 on PFD1 and GRS 79 #2 on PFD2. Highlight the SELECT PROCEDURE window and select MAGNETOMETER INTERFERENCE TEST on each PFD.

(e) Press ENT again to select the first checklist item displayed on each PFD. As each command is complied with and ENT is hit the associated box will turn green to confirm the selection has been complied with.

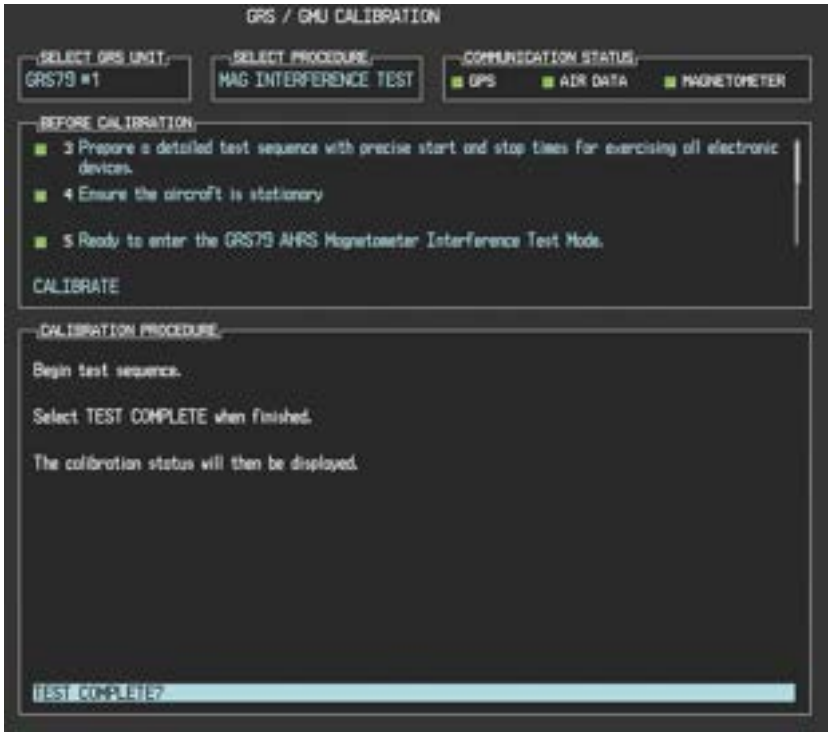
NOTE: Use a stopwatch to accurately time the tests. All actions are to be carried out in the order and at the precise elapsed time as specified in the prepared test sequence. MFD must be out of configuration mode prior to test being performed.

(f) When the CALIBRATE field is blinking on both PFDs, press the ENT keys to begin the procedure per "Chart 29" on page 342502141.

(g) When the test is complete press the ENT key on each PFD to indicate the process is complete and the TEST COMPLETE annunciation stops blinking at this time.

CHART 29
G1000 NXi - MAGNETOMETER INTERFERENCE TEST

ELAPSED TIME Since start of test (Minutes:Seconds)	ACTION
0:00	Test Begins
0:10	Turn on strobe light
0:20	Turn off strobe light
0:30	Turn on fin strobe
0:40	Turn off fin strobe
0:50	Turn on NAV lights (if optional tail light installed)
1:00	Turn off NAV lights (if optional tail light installed)
1:10	Blower Motor on Low
1:20	Blower Motor off
1:30	Blower Motor on High
1:40	Air Conditioning Switch On
1:50	Blower Motor off & AC Switch off



Magnetometer Interference Test
Figure 39

(h) The PFD informs the operator if the installation has passed or failed the magnetometer interference test. If the test passes, no further action is required for this test. If the test fails, the installation should be considered unreliable until the source of magnetic interference is identified and remedied. When the magnetometer interference test fails, record the three magnetometer maximum deviation values and their corresponding timestamps. Any maximum deviation value greater than 2.5 milliGauss indicates a problem that must be resolved. Compare the corresponding timestamps with the prepared test sequence to identify which action produced the problem. Contact Garmin for assistance in resolving the problem.

NOTE: Two common reasons for a failed magnetometer interference test are:
1) New equipment is installed in close proximity to the GMU 44B magnetometer, and
2) An existing or new electronic device has become grounded through the aircraft structure instead of via the proper ground wire in a twisted shielded pair.

(i) Press ENT on each PFD to conclude the test.

(14) GRS/GMU Test

The aircraft can now be taxied back and the engine can be shut down for final testing. Restart the displays in normal mode to conduct final system checks. When the PFDs power up in normal mode, the AHRS attitude and heading information displayed should become valid within one (1) minute of power-up, as shown in “Figure 40” (provided both GPS receivers have a valid position; if GPS is unavailable, AHRS initialization may take as long as 2 minutes).



Normal Mode AHRS Check
Figure 40

(15) GWX 75/80 Weather Radar Test (If installed)

NOTE: Before energizing the equipment, be sure microwave radiation safety precautions including both fuel and personnel safety considerations have been observed. These include clearing all personnel to an area beyond the maximum permissible exposure level (MPEL) boundary. The MPEL for the GWX 75/80 is 11 feet.

(a) Start the G1000 NXi in the Normal mode.

(b) On the MFD, turn the large FMS knob to select the MAP Group on the MFD then turn the small FMS knob to select the WEATHER RADAR page.

(c) Press the MFD FMS knob to activate the cursor in the TILT field then turn the small FMS knob to select the desired antenna tilt angle. Press the MFD FMS knob to remove the cursor.

(d) Select the MODE softkey, then select the WEATHER Softkey to energize the radar array. A caution window will come up. ONLY if you have followed the precautionary steps, select yes to accept. Press the BACK softkey to show the weather radar scan and gain control softkeys.

(e) Deselect the CAL GAIN Softkey to activate the cursor in the GAIN field. Turn the small FMS Knob to adjust the gain to a non-calibrated level then press ENTER on the MFD. Verify the gain setting is visible in the gain field as a movable horizontal bar in a flashing box with a line pointer reference depicting the calibrated position. Press the FMS Knob to remove the cursor.

(f) Select the CAL GAIN softkey. Verify CALIBRATED is displayed in the GAIN field. Deselect the CAL GAIN.

(g) Select Horizontal, Mode, and Ground Softkeys sequentially. Verify scanning of and depiction of surrounding ground features. Verify Tilt, Bearing, Sector Scan, and Gain can be adjusted from the GROUND scan mode display via the FMS knob and RANGE knob as appropriate.

(h) Press Weather key, then press Back key.

(i) If optional Ground Clutter Suppression & Turbulence Detection functions are installed, verify that the GCS button can be turned on and off with no failure annunciations.

(j) Press MODE then STANDBY softkey to turn off the weather radar.

(16) Stormscope (WX-500) Functional Check (If installed.)

(a) See “Optional Equipment Installation” on page 342502109, to configure Stormscope.

1) On the MFD, turn the large FMS knob to select the MAP group, then turn the small FMS knob to select the STORMSCOPE page.

2) During initialization, the WX-500 stormscope runs a series of self-tests to ensure that all functions are operating properly. If no faults are detected, continue on to 5).

3) If a fault is detected, consult the WX-500 installation manual for troubleshooting assistance.

4) On the MFD, verify that it is possible to enter the following modes via softkeys: cell mode, strike mode.

5) Verify that the range can be changed from 25 nm to 100 nm via the range knob.

(b) Detailed Test

The following procedure requires access to and familiarity with the set-up and operation of the 3M WX-PA Portable Analyzer.

1) Following successful power up, verify the following modes:
   a) Access to both 3600 and 1200 weather view modes.
   b) All available ranges can be displayed.
   c) Access to the cell mode and strike mode.
   d) Strike counter is displayed in all weather modes and ranges.
2) Key COM1 and COM2 several times on different frequencies representing the lower, mid and upper portion of the VHF COM frequency band. Verify keying of COM1 or COM2 does not cause strike data to appear on the MFD.
3) Operate DME, XPDR1, XPDR2 and weather radar. Verify these systems do not cause strike data to appear on the MFD.
4) Connect the WX-PA cable to the WX-PA antenna.
5) Position the WX-PA antenna on the WX-500 antenna. Ensure the connection is tight. If necessary, use tape to secure the WX-PA antenna. Ensure the “FORWARD” arrows are aligned and the WX-PA antenna suction cups are positioned forward of center along the longitudinal axis.
6) Secure the WX-PA cable to the aircraft with the attached suction cup and route the cable to the cockpit.
7) Connect the remaining cable end to the WX-PA.
8) Power up the WX-PA and verify the WX-500 is in the weather mapping mode (i.e., 3600 weather view at the 200 NM range).
9) Set the WX-500 to STRIKE mode; 100 NM range (or next highest available range).
10) Select the Continuous Out mode displayed on the WX-PA menu and press MENU/ENTER.

(17) GDL 69eA Test (If installed)

(a) If an XM Satellite Radio subscription has already been activated for the GDL 69eA, then power up the G1000 NXi and go to the AUX – XM INFORMATION page and verify that the GDL 69eA is working properly.

(b) GDL 69eA Datalink Check

NOTE: This section verifies the correct installation of the GDL 69eA.

1) Insert a headset into the audio jack on the pilot or co-pilot side of the aircraft.
2) On the MFD select AUX - XM RADIO.
3) Change the XM channel using the MFD controls.
4) Increase and Decrease the volume using the MFD controls.
5) Change the XM channel using a Bluetooth device with Garmin Pilot connected to the FS510 (if installed).
6) Verify XM audio operation at Pilot, Copilot and all passenger stations. Verify stereo operation.
7) Activate the Stall Warning and verify that the XM audio is muted.

(c) If the GDL 69eA is replaced (new unit), the owner must re-activate the subscription, using the new GDL 69eA’s Radio ID number(s). See “Chart 18” on page 34250279

(d) As a final operations check, make sure there are no MANIFEST errors shown on the PFD for the GDL 69eA.

(18) GTS 825 TAS Tests (If installed)

(a) Self-Test

1) Start the G1000 NXi in the Normal mode and select the TRAFFIC MAP page on the MFD MAP group.

2) Turn ADS-B off by pressing the ADS-B softkey if the softkey indicator is green.

3) Verify that the STANDBY, OPERATE, TEST (in Standby mode) and ALT MODE softkeys are available on the bottom of the display. Verify that a TAS Status (not TAS FAIL) is displayed in the upper left corner of the traffic map. Verify that a yellow NO DATA message is not displayed in the center of the map. If TIS is displayed instead of TAS, refer to the "(3) Optional Equipment Installation" on page 342502109 to install the GTS 825 option.

4) Press the OPERATE softkey and verify that OPERATING is displayed in the upper left corner of the traffic map.

5) Press the STANDBY softkey and verify that STANDBY is displayed in the upper left corner of the traffic map.

6) Set range to 6 NM or greater.

7) Press the TEST softkey and verigy that TEST is displayed in the upper left corner of the traffic map and that a traffic test pattern is displayed. Verify that "TAS System Test Ok" annunciation is heard over the cockpit speaker at the conclusion of the test.

8) Verify that the FAIL indicator is NOT present in the top right corner of the MFD.

9) Open the TAS circuit breaker and verify that NO DATA is displayed at the center of the traffic map after several seconds.

10) Close the TAS circuit breaker and verify that NO DATA is removed from the traffic map.

(b) Operational Test

Perform the following test to verify GTS 825 operational and surveillance functions. Use a ramp tester such as a TIC TR220 or equivalent to perform the tests.

To select a scenario that will properly converge and intercept the GTS 825, the GTS 825 must be in ground test mode. To enable ground test mode, the aircraft must be on the ground and the GTS 825 must be in normal system mode and in standby.

Position the test set directional antenna with a clear line of sight to the GTS 825 antenna at 90 degrees. With the GTS 825 powered up and in Standby mode indicated on the CDTI, cycle the GTS 825 to 'Operate'.

1) Select the following:

a) Set the intruder type as ATCRBS.

b) Intruder Start Distance: 10 nm.

c) Intruder Start Altitude: 50,000 ft.

d) Vertical Speed: 0 fpm.

e) Velocity: 360 kts.

2) Initiate the intruder scenario and observe the following:

a) Traffic should be acquired at approximately 10 NM at 90 degree bearing and co-altitude. Observe intruder closes on own aircraft at a rate of 0.1 NM/sec.

b) The intruder should transition from Other Traffic (displayed as an open diamond with 00 displayed above), to proximate traffic (displayed as a filled white diamond with 00 displayed above), to a Traffic Advisory (TA) alarm.

c) The appropriate TA symbology (yellow filled circle with 00 displayed above, and an audio annunciation of "Traffic! 3 O'clock! At Altitude! 3 Miles!"), displayed when the intruder approaches within 3 NM.

(19) GSR 56 Checkout (If Installed)

This procedure verifies correct installation of the GSR 56, not if it has been activated. If the unit needs to be activated, see "GSR 56 Activation Procedure (If Installed)" on page 342502112.

(a) On the MFD navigate to AUX group, SATELLITE PHONE page using the FMS knobs.

(b) Press the DIAL softkey.

(c) Use the softkeys or FMS knobs to enter "1911135" and press ENT to confirm.

(d) Press ENT to call.

(e) If both parties can hear each other, then the installation has been verified.

1) If the unit has an active subscription, tell the Garmin representative that you are testing the GSR 56 installation and the unit should remain active.

2) If the unit has not been activated, tell the Garmin representative that you are testing the GSR 56 installation and that the unit can be deactivated.

a) If placed on hold when calling Garmin Flight Data Services, you may call anyone for a brief verification phone call. The cost for this phone call is charged to Garmin, so be brief.

b) Send an email flightdataservices@garmin.com after the brief phone call is complete so the GSR 56 unit can be turned off. Include the GSR 56 serial number in this email.

3) Select the bottom mount antenna configuration on the WX-PA keyboard ("A" key).

4) Select a cardinal bearing and a range of 120 NM.

5) Use the F1 and F2 keys to adjust range and F3 and F4 keys to adjust heading.

NOTE: The WX-500 will plot data at one-half the range selected on the WX-PA.

6) Press MENU/ENTER to start the test.

7) Observe the MFD to ensure the proper positioning of the strikes, based on range and azimuth settings on the WX-PA. Change the cardinal bearings and verify correct test strikes. Verify the strikes are within 10 degrees of the selected azimuth and plot at 60NM (1/2 120 NM).

8) Verify after 20 seconds of operation the strike counter reads 580 +/-40.

9) After testing for all ranges and bearings indicated, press 2ND then MENU/ENTER to return to the main menu.

10) Select Circular Pattern mode on the WX-PA mode menu and press MENU/ENTER.

11) Select bottom mount antenna configuration on the WX-PA keyboard ("A" key).

NOTE: The MFD should be set at 100 NM (or next highest available range) on the 360 degree weather screen and in strike mode.

12) Use the F1 and F2 keys to select 120 NM range and press MENU/ENTER to start the test. Verify the WX-500 plots discharge points at approximately 60 NM.

13) Observe the MFD to ensure the proper positioning of the test strikes. Verify the strikes are within 10 degrees of the 30 degree azimuth increment and within 12 NM of 60 NM.

14) When complete, set WX-PA for 55 NM, set the MFD for 50 NM (or the next highest available range) and repeat test.

15) On the MFD, verify the sensor plots points just outside of 25 NM and the strikes are within 10 degrees of the 30 degree azimuth and within 5 NM of 27.5 NM.

16) Repeat above setting the MFD for 25 NM (or the next highest available range) and the WX-PA for 15 NM.

17) On the MFD, verify the positioning of test strikes are within 10 degrees of azimuth and within 2 NM of 7.5 NM.

18) Restart the G1000 NXi in configuration mode by opening the PFD and MFD circuit breakers. While holding the ENT keys on the PFDs and MFD, restore power by closing the PFDs and MFD circuit breakers.

19) Select the OTHER page group on the MFD. The STORMSCOPE page is shown by default.

(20) ADF (Optional)

(a) Functional Check

This check verifies that the ADF / G1000 NXi interface operates correctly.

1) Press the PFD softkey on PFD1 and PFD2. Toggle the BRG1 and BRG2 softkey until the ADF bearing is shown on PFD1 and PFD2.

2) Verify that the ADF window is not invalid (no red 'X').

3) Using the ADF control head select a known valid local ADF. Verify that the ADF bearing pointer moves towards and stabilizes on the correct bearing to that station.

4) Select ADF and SPKR on GMA audio panel. Using the ADF control head, select, ANT mode. Verify that the audio from the station tuned can be heard on the pilots and copilots headset and cockpit speaker.

5) Increment the ADF volume control from full low to full high. Verify the volume increases and decreases appropriately over pilots and copilots headset and cockpit speaker.

(b) Quadrantal Error Adjustment

1) Position aircraft on ramp in an area that is clear of surrounding buildings and other obstructions that may block or reflect radio signals.

2) Select ADF as the bearing source on the #1 PFD.

3) Select ADF mode on ADF receiver.

4) Tune the ADF to the appropriate known local station.

5) Orient aircraft such that the bearing indicator points to 0 degrees (aircraft is heading directly to station). Note the heading.

6) Using the heading indicator, rotate the aircraft 45 degrees to the right.

7) ADF indicator should show noted heading ±3 degrees.

8) Using the heading indicator, rotate the aircraft 90 degrees to the left.

9) ADF indicator should show noted heading ±3 degrees.

NOTE: The following steps are not required if the quadrantal error is within tolerance.

10) Adjust the variable resistor R60 on the rear side of the RMI-Converter to correct the quadrantal error. The QE adjustment potentiometer has a sensitivity of approximately 1 degree per turn.

11) Recheck the relative bearings and readjust the QE variable resistor as necessary to split the errors at the quadrantal points and obtain the lowest possible average error.

12) When QE adjustments are complete, reinstall the O-ring seal and the threaded cap in the adjustment hole to seal the antenna.

(c) Checkout

1) Tune the ADF appropriate known local station and verify the bearing to the station is correct +/- 15° and the Morse Code Identifier is clear and accurate.

2) Verify the ADF audio is available on the GMA 1360 audio panel via the AUX key.

(21) DME (Optional) Functional Check

(a) Basic Test

This check verifies that the DME-to-G1000 NXi interface operates correctly, if check does not pass, conduct Detailed Test below. It requires appropriate DME Test Equipment.

1) Perform systems check and verify the DME is tuned by NAV 1 and NAV 2
2) Verify the distance is accurate.
3) Verify the DME audio is available on the GMA 1360 audio panel.

(b) Detailed Test

This check verifies that the DME-to-G1000 NXi interface operates correctly. It requires appropriate DME Test Equipment.

1) If DME is not displayed, press the PFD softkeys on PFD1 and PFD2, then press the DME softkey. Verify the DME window is displayed next to the PFD1 and PFD2 HSI.
2) On PFD1 and PFD2, press the BACK softkey then select DME softkeys. Verify the DME TUNING screen is displayed.
3) On PFD1 and PFD2, use the large FMS knob to select the NAV1, NAV2 and HOLD modes in the DME field. Verify that the NAV1, NAV2 and HOLD modes can be selected by turning the small FMS knob.
4) Set NAV1 and NAV2 frequencies to 108.00 and 117.00 respectively.
5) On PFD1 and PFD2, select the DME NAV1 mode. Verify that the DME window display is set to the NAV1 frequency of 108.00.
6) On PFD1 and PFD2, select the DME NAV2 mode by pressing the ENT softkey. Verify that the DME window display is set to the NAV2 frequency of 117.00.
7) On PFD1 and PFD2, select the DME HOLD mode by using the FMS knob. Verify that the last selected NAV frequency of 117.00 remains the same when the NAV2 frequency is changed. Deselect DME HOLD.
8) On the NAV Test Set, set up a DME test and note the nav frequency. Tune NAV1 to the test set frequency and set PFD1 and PFD2 DME MODE to NAV1. Set NAV2 to a frequency other than the test set frequency.
9) Verify that the DME distance on PFD1 and PFD2 match the test set.
10) Select, or verify selected, DME and SPKR buttons on GMA audio panel to select the DME audio and select speaker to ON. Verify that the DME audio can be heard over the cockpit speakers.
11) On PFD1 and PFD2, set the DME mode to NAV2 and verify that the DME distance is dashed out.
12) Tune NAV2 to the test set frequency. Set NAV1 to a frequency other than the test set frequency.
13) Verify that the DME distance on PFD1 and PFD2 match the test set.

(22) FS 510 Checkout (If installed.)

NOTE: This requires the use of a Bluetooth enabled device, such as a smart phone or tablet with the Garmin Pilot app installed.

(a) Ensure the airplane has a valid GPS position.
(b) Turn on Bluetooth on the mobile device.
(c) Navigate to the Aux group, Connext Setup page on the MFD. Note the Bluetooth unique identifier at the end of the Bluetooth name field on this page.
(d) Pair to “Flight Stream 510-XXXX” on the mobile device, where XXXX is the unique identifier for this particular Flight Stream 510.
(e) Follow prompts on the device as well as on the MFD to confirm the confirmation code.
(f) Open the Garmin Pilot application on the mobile device.
(g) Tap the button in the top left corner of the mobile device.
(h) Select Connext from the list on the left of the mobile device screen, then select Flight Stream 510 from the devices list.
(i) Select Flight Stream 510-XXXX from the dropdown menu on the mobile device and then press Connect.
(j) After the device has connected, press Activate on the mobile device to receive the flight plan update.
(k) Tap the button in the top left corner of the mobile device twice and then select Flight Plan.
(l) Ensure that KVRB appears as the first airport listed.
(m) Tap “Add Waypoint” and type in KFPR and press done.
(n) Tap the blue connext icon located to the right of Flight Plan on the top row on mobile device.
(o) Tap “Send to Navigator”.
(p) Press ENT on the MFD when prompted with “pending flight plan.”
(q) Press ENT on the MFD when prompted to activate and press ENT again to click OK.
(r) Go to the Navigation Map on the MFD and verify flight path is correct.

(23) Automatic Flight Control System (AFCS) (If installed.)

See Garmin GFC 700 Automatic Flight Control System (AFCS) in 22-10-00.

(24) ChartView (Optional) Check

(a) Select the Airport Information page in the WPT group in MFD.
(b) Verify that the title at the top of the page reads “WPT- Airport Information”.
(c) Enter KVRB as the waypoint and verify that the CHRT softkey is available.
(d) Press the Chart softkey and verify the electronic chart is viewable.

(25) Synthetic Vision (Optional) Check

(a) Ensure the aircraft has a GPS position.
(b) Press the PFD Opt softkey on PFD1.
(c) Press the SVT softkey on PFD1.
(d) Press the Terrain softkey on PFD1.

NOTE: Synthetic Vision requires a valid GPS signal.

(e) Verify the traditional blue/brown attitude depiction is replaced with the Synthetic Vision rendering within 2–3 minutes of activation.

(26) TAWS (Optional) Functional Check

(a) Ensure the aircraft has a GPS position.

(b) Select the TAWS-B page in the MAP group in the MFD.

(c) Verify that the title at the top of the page reads “MAP - “TAWS-B”.

NOTE: If TAWS has not been enabled, the title will read “MAP - TERRAIN PROXIMITY” or “MAP - TERRAIN.” See “Options Unlock Procedures” on page 342502114.

(d) Press the MENU button and select “Test TAWS System” from the pop-up menu and press ENT key.

(e) After the TAWS test has completed, verify that “TAWS System Test Okay” is heard over the audio system.

(27) Terrain Proximity Check (if TAWS is not installed.)

(a) Ensure the aircraft has a GPS position.

(b) On the MFD, select the MAP group, TERRAIN page by rotating the large FMS knob to change groups and the small FMS knob to change pages.

(c) Verify that the map is colored to show terrain and no failures are annunciated on the page.

(28) Surface Watch Check (If Installed.)

(d) Navigate to the System Setup page of the Aux group on the MFD.

(e) Activate the cursor and use the large FMS knob to scroll to SurfaceWatch Alerts and set it to “Off” and verify the system message “SURFACEWATCH INHIBITED”.

(f) Set SurfaceWatch Alerts to “On” and verify “SURFACEWATCH INHIBITED” system message clears.

(29) NAV Source Selection Tests

NOTE: The CDI will not be depicted if the VOR antenna is not receiving a valid signal.

NOTE: For FMS, the word that is displayed with the FMS (i.e., FMS Term) depends on the phase of flight of the airplane. It is acceptable for FMS TERM or FMS ENR to be displayed.

(a) Start these tests with all avionics powered.

(b) Press the Direct-To-Key (--D->).

(c) Enter a Direct-To waypoint. Press ENT to accept waypoint.

(d) Enter an appropriate VOR frequency simulated with test equipment into the active NAV 1 frequency box on #1 PFD.

(e) Enter a different appropriate VOR frequency simulated with test equipment into the active NAV 2 frequency box on #1 PFD.

(f) On MFD, AUX Group, SYSTEM SETUP page, press the FMS knob to activate the cursor. Turn the large FMS knob to highlight CDI in the Synchronization window. Turn the small FMS knob to turn synchronization on. Press the FMS knob to deactivate the cursor.

(g) Verify that #1 PFD and #2 PFD synchronize (i.e. #1 PFD & #2 PFD match for the CDI).

(h) Press the CDI softkey on #1 PFD. Verify the NAV source on #1 PFD and #2 PFD changes from FMS TERM to VOR1.

(i) Press the CDI softkey on #1 PFD. Verify the NAV source on #1 PFD and #2 PFD changes from VOR1 TO VOR2.

(j) Press the CDI softkey on #1 PFD. Verify the NAV source on #1 PFD and #2 PFD changes from VOR2 TO FMS TERM.

(k) Repeat Step (f), but turn the synchronization off.

(l) By pressing the CDI softkey on #1 PFD, cycle the NAV source on #1 PFD from FMS TERM to VOR1 TO VOR2 TO GPS TERM. Verify that only the NAV source depicted on #1 PFD changes from FMS TERM to VOR1 TO VOR2 TO FMS TERM.

(m) By pressing the CDI softkey on #2 PFD, cycle the NAV source on #2 PFD from FMS TERM to VOR1 TO VOR2 TO FMS TERM. Verify that only the NAV source depicted on #2 PFD changes from FMS TERM to VOR1 TO VOR2 TO FMS TERM.

(30) Bearing Pointers and Information Windows Tests

NOTE: The bearing pointer will not be depicted if it is not receiving a valid signal.

(a) Start these tests with all avionics powered.

(b) Press the Direct-To-Key (--D->).

(c) Enter a Direct-To waypoint.

(d) Enter an appropriate VOR frequency simulated with test equipment into the NAV1 frequency box on #1 PFD.

(e) Press the PFD Opt softkey on #1 PFD to make the Bearing1 and Bearing2 softkey selections visible.

(f) Verify that the Bearing1 softkey can be pressed to sequentially cycle through NAV1, NAV2, FMS, and ADF (if installed).

(g) Verify that the Bearing2 softkey can be pressed to sequentially cycle through NAV1, NAV2, FMS, and ADF (if installed).

(h) Repeat Steps (e) thru (g) for PFD #2.

(31) BARO Selection Tests

(a) On MFD Aux Group, SYSTEM SETUP page, press the FMS knob to activate the cursor. Turn the large FMS knob to highlight BARO in the Synchronization window. Turn the small FMS knob to turn synchronization on. Press the FMS knob to deactivate the cursor.

(b) Rotate the BARO knob on the #1 PFD and verify that both PFD's BARO values change.

(c) Rotate the BARO knob on the #2 PFD and verify that both PFD's BARO values change.

(d) Set the current altimeter setting and verify that the both PFD's indicate within ±30' of current field elevation.

(e) On MFD Aux Group, SYSTEM SETUP page, press the FMS knob to activate the cursor. Turn the large FMS knob to highlight BARO in the Synchronization window. Turn the small FMS knob to turn synchronization off. Press the FMS knob to deactivate the cursor.

(f) Rotate the BARO knob on the #1 PFD and verify that only #1 PFD BARO values change and a delta indication of at least 0.01 in. Hg turns the indication yellow.

(g) Rotate the BARO knob on the #2 PFD and verify that only #2 PFD BARO values change and a delta indication of at least 0.01 in. Hg turns the indication yellow.

(h) On MFD Aux Group, SYSTEM SETUP page, press the FMS knob to activate the cursor. Turn the large FMS knob to highlight BARO in the Synchronization window. Turn the small FMS knob to turn synchronization on. Press the FMS knob to deactivate the cursor.

(32) Post-Interior Installation Checks

The following checks should be accomplished each time the interior is removed and reinstalled.

(a) Audio Checks

1) Check Passenger ICS. See “GMA 1360 Test” on page 342502126.
2) If GDL 69eA installed, perform “GDL 69eA Datalink Check” on page 342502144.

(b) Lighting Tests

Increase and Decrease lights to ensure operation.

NOTE: A tarp must be used to cover the fuselage, and a flashlight is required to test photocell operation.

1) Courtesy Lights
   a) Right FWD Headliner
   b) Left AFT armrest switch
2) Reading Lights

   Passenger Switch panels (four)
3) Cockpit Dome Lights
4) Avionics dimmer switch controls the following:
   a) PFD #1 & #2
   b) MFD
   c) Audio Panel
5) Rotate the Avionics dimmer to full dim ensure the following switch to photocell:
   a) PFD #1 & #2
   b) MFD
6) Switch lights dimmer switch controls the following:
   a) Overhead Panel Switch Lights
   b) Instrument Panel Switch Lights
   c) Air Cond Switch (If Installed)
   d) Pitot/Stall Warn Heat Switch
   e) Surface De-Ice Switch (If Installed)
   f) Prop Heat Switch (If Installed)
   g) Windshield Heat Switch (If Installed)
   h) Ice Light Switch (If Installed)
   i) Heat Fan Switch
   j) Defrost Switch
   k) ADF (If Installed)
   l) Circuit Breaker Panel Lights
   m) Main Bus Circuit Breaker Lights
   n) Instrument panel lighted placards
7) Panel lights dimmer controls the following:
   a) Lights under glare shield
   b) Pedestal Light

8) With engine off and no external power applied to aircraft select Emergency Switch “ARM” then select Battery Master Switch and Avionics Switch “OFF”. The following items should remain lit:
   a) Aspen EBD Standby Instrument
   b) Master Caution/Warning Reset Switch Lights (if they are active)
   c) PFD #1
   d) Audio Panel
   e) All other flood lights and equipment lights extinguish.
   f) All Overhead Switch Panel and Instrument Panel switch lights extinguish.

(c) USB Checks

Verify functionality of all USB ports (cockpit and cabin).

(33) Emergency Bus Operation Verification

Verify all the following functions are operational when EMERGENCY switch is “ON” (in ARM position), Battery Master SWITCH is “OFF“, and Engine is “OFF”.

(a) PFD #1
(b) COM #1
(c) NAV #1
(d) AHRS #1
(e) Audio Panel #1
(f) Air Data Computer #1
(g) GEA 71B #1 and #2 (Engine Instruments)
(h) Standby Instrument
(i) Avionics Lights (i.e., Dimmer)

NOTE: MFD, PFD2, AHRS2, GDL 69eA, Transponder #1, Air Data Computer 2, Radar, ADF, DME, GTS 825 and Stormscope are not available on Emergency Power. Also note that NAV2, GPS2, COM2, Deice (optional), and O2 (optional) display invalid.

(34) Engine Run Checks

(a) Annunciation Verification

WARNING: UTILIZE ALL SAFETY PROCEDURES. NEVER WORK IN THE ENGINE AREA WHILE THE ENGINE IS RUNNING.

1) Remove fuses F102 and F202 in the engine compartments.
2) Start both engines.
3) With left engine running above 1100 RPM, turn on L ALTR switch. Verify L ALTR FAIL warning message. Reset L ALTR switch.
4) With right engine running above 1100 RPM, turn on R ALTR switch. Verify R ALTR FAIL warning message. Reset R ALTR switch.
5) Turn off engines.
6) Reinstall fuses F102 and F202.

(b) Various Systems Verification

1) Start PFD1 in config mode by holding ENT while applying power.

2) Using the FMS knob on the #1 PFD go to GIA I/O Configuration page on GIA group.

3) During left engine start, verify the data box next to DISCRETE IN 19A is green while left engine start switch is pressed.

4) During right engine start, verify the data box next to DISCRETE IN 20A is green while right engine start switch is pressed.

5) Verify the following with the engines ON:

a) Verify Hour Meter operation:

For aircraft with Hobbs Configuration A, run the engines. Hobbs shall run while engine is running.

b) Verify MAP, RPM, TIT, Fuel Flow, CHT, Oil Pressure, Oil Temp, Oxygen (if installed), and Vacuum (if installed) are operating in normal range.

c) Verify left alternator and right alternator current indications.

1] With the Battery Master Switch ON and left alternator and right alternator on line, turn the Battery Master and left alternator off.

2] Note right alternator current.

3] Turn the left alternator ON and right alternator off.

4] Note left alternator current.

5] Verify that current readings are equal (within +/- 3 amps).

O. Return to Service Procedure

CAUTION: ENTER “MAINTENANCE MODE” WHEN ACTIVATING THE GARMIN G1000 NXI SYSTEM ON THE GROUND TO CONDUCT TESTS, INSPECTIONS, ETC. SEE MAINTENANCE MODE UNDER TROUBLESHOOTING, ABOVE.

(1) Software Verification

After reinstalling any G1000 NXi LRU, use the checkoff list in “Chart 30” on page 342502164 to verify correct LRU software part numbers and versions against the current numbers and versions shown in the appropriate Required Equipment List (REL), “Chart 31” on page 342502165.

(a) Start the G1000 NXi system in configuration mode.

(b) On MFD splash screen (“Figure 41”) verify appropriate version “PA-34-220T System XXXX.XX” is displayed in the upper right corner of the MFD (where XXXX.XX is the system version number for the appropriate Required Equipment List (REL) in “Chart 31” on page 342502165).



MFD Splash Screen
Figure 41

(c) Select softkey on MFD to proceed beyond splash screen.

(d) Verify that no manifest (software mismatch) alerts are shown on any of the displays.

(e) Re-start the G1000 NXi system in configuration mode.

(f) The PFD System Status Page (“Figure 42”) displays a list of LRUs in the LRU window.

1) Verify that the following are valid. (Green annunciations indicate that the system is healthy): MFD1, PFD1, PFD2, GIA1, GIA2, GDL 69 (optional), GTS (optional), and GWX (optional).

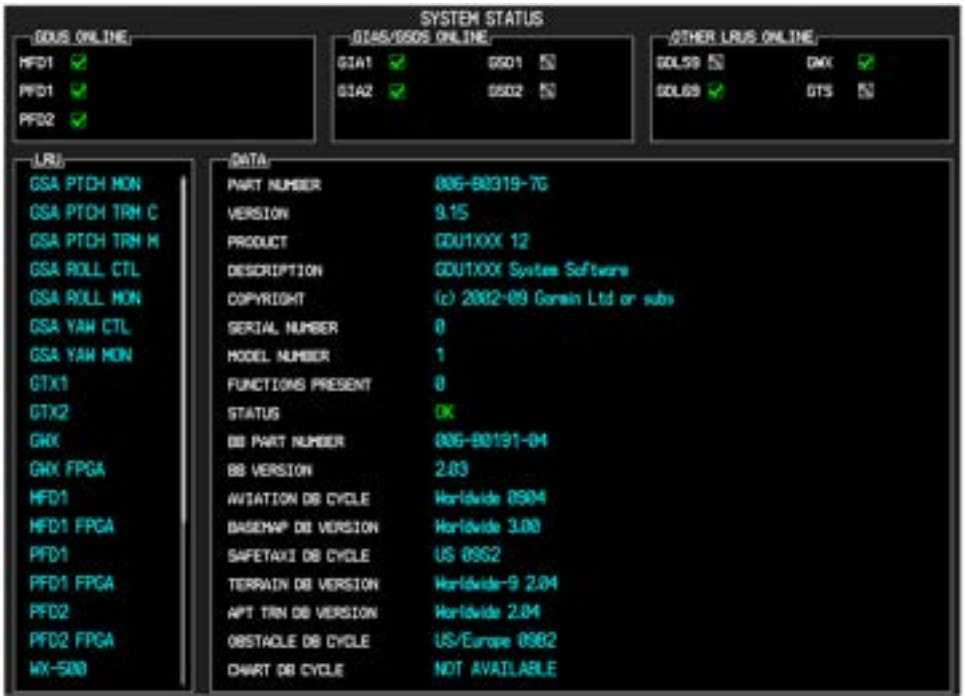
2) Rotate the small FMS knob to change to the “System Configuration Page” and verify that the following are valid. (Indicated by green annunciations): MFD1, PFD1, PFD2, GIA1, GIA2, GDL 69 (optional), GTS (optional), and GWX (optional).

3) Return to the “System Status Page.”

(g) Activate the cursor and highlight the LRU window.

(h) Use the FMS knob to scroll through the list, selecting each LRU in turn.

(i) The software part number and version is displayed in the DATA window. Compare this to the current numbers and versions shown in the appropriate Required Equipment List (REL).



System Status Page (Configuration Mode)
Figure 42

(2) GPS Failure Test

(a) Start these tests with all avionics powered.

(b) Valid GPS reception. Press Direct-To-Key (--D->). Enter a Direct-To waypoint.

(c) Place a shroud over the GPS 1 antenna to prevent signal reception. Verify loss of GPS 1 signal on MFD AUX group, GPS STATUS page. The following should remain valid on the PFD throughout the procedure.

- ✈ Attitude
- ✈ Heading
- ✈ Airspeed
- ✈ Altitude
- ✈ Vertical Speed
- ✈ OAT
- ✈ GPS CDI remains valid on PFD.

(d) Remove shroud from the GPS 1 antenna.

(e) Place a shroud over the GPS 2 antenna to prevent signal reception. Verify loss of GPS 2 signal on MFD AUX group, GPS STATUS page. The following should remain valid on the PFD throughout the procedure.

- ✈ Attitude
- ✈ Heading
- ✈ Airspeed
- ✈ Altitude
- ✈ Vertical Speed
- ✈ OAT
- ✈ GPS CDI remains valid on PFD.

(f) Leave the shroud over the GPS 2 antenna and replace the shroud over the GPS 1 antenna. Verify loss of GPS 1 and 2 signals on MFD AUX group, GPS STATUS page. The following should remain valid on the PFD throughout the procedure.

- ✈ Attitude
- ✈ Heading
- ✈ Airspeed
- ✈ Altitude
- ✈ Vertical Speed
- ✈ OAT
- ✈ Loss of GPS integrity annunciated on the PFD.

(3) GIA Failure Test

(a) Start these tests with all avionics powered.

(b) Valid GPS reception. Valid NAV signal.

(c) Remove power from GIA 1 by pulling the INTEG AV 1 and COM 1 circuit breakers.

(d) Verify the following system messages:

1) AHRS1 GPS – AHRS1 USING BACKUP GPS SOURCE
2) FAILED PATH
3) AHRS2 GPS – AHRS2 NOT RECEIVING BACKUP OF GPS INFORMATION (if installed)
4) ESP FAIL (If Enhanced AFCS installed)

(e) Verify the following flag invalid:

1) COM/NAV 1 field
2) NAV 1 CDI loses deviation bar.

(f) Verify BOTH ON GPS2 Sensor Reversion annunciation on the PFDs.

(g) Restore power to GIA1 by resetting the INTEG AV 1 and COM 1 circuit breakers.

(h) Verify that all invalid flags, system messages, and annunciations are removed. (#1 GPS will take time to reinitialize).

(i) Remove power from GIA 2 by pulling the INTEG AV 2 and COM 2 circuit breakers.

(j) Verify the following system messages:
  1) AHRS2 GPS – AHRS2 USING BACKUP GPS SOURCE (if installed)
  2) STRMSCP FAIL (if installed)
  3) AHRS1 GPS – AHRS1 not receiving backup GPS information.
  4) FAILED PATH
  5) ESP FAIL (If Enhanced AFCS installed)
  6) CO DET FAIL (If installed)

(k) Verify the following flag invalid:
  1) COM/NAV 2 field
  2) NAV 2 CDI loses deviation bar.
  3) STORMSCOPE FAILED yellow text (on MFD Stormscope Map page) (if installed)

(l) Verify BOTH ON GPS1 Sensor Reversion annunciation on the PFDs.

(m) Restore power to GIA2 by resetting the INTEG AV 2 and COM 2 circuit breakers.

(n) Verify that all invalid flags, system messages, and annunciations are removed. (#2 GPS will take time to reinitialize).

(o) Remove power from GIA 1 and GIA 2 by pulling the INTEG AV 1, INTEG AV 2, COM 1 and COM 2 circuit breakers.

(p) Verify the following system messages:
  1) GPS NAV LOST
  2) TRAFFIC FAIL
  3) XPDR1 FAIL
  4) GMA1 FAIL
  5) TRN AUD FAIL (If TAWS is installed)
  6) SURFACEWATCH FAIL (If installed)

(q) Verify the following flags invalid:
  1) COM/NAV 1 field
  2) COM/NAV 2 field
  3) XPDR field
  4) Engine Instruments

(r) Verify the following:
  1) All AHRS and ADC fields remain valid.
  2) NAV 1 & NAV 2 CDI lose deviation bar.
  3) “GEAR SYS” CAS warning

(s) Close all circuit breakers.

(4) Display Failure Test

(a) Start these tests with all avionics powered.

(b) Remove power from the MFD by pulling the MFD circuit breaker.

(c) Verify the following system messages on the PFDs:

1) GWX FAIL system message (If Installed).

2) MFD1 VOLTAGE (This may or may not appear depending on power failure rate vs. MFD update rate).

3) FAILED PATH

4) ESP OFF (If Enhanced AFCS installed)

5) GDL 69 FAIL

(d) Verify the following:

1) PFDs switch to reversion mode.

2) Attitude and Heading remain valid from AHRS.

3) Airspeed, Altitude, Vertical Speed and OAT remain valid.

4) Engine Instrumentation appears on PFDs.

(e) Restore power to the MFD. The displays should return to Normal operation.

(f) Select reversionary mode on GMA 1360 Audio Panel.

(g) Remove power from the #1 PFD by pulling the #1 PFD circuit breaker.

(h) Verify the following system messages are displayed on #2 PFD and the MFD:

1) FAILED PATH

2) TRAFFIC FAIL (If GTS 825 installed)

(i) Verify the following:

1) MFD and PFD2 switch to reversion mode.

2) Attitude and Heading remain valid from AHRS.

3) Airspeed, Altitude, Vertical Speed and OAT remain valid.

4) MFD retains engine instrumentation.

(j) Restore power to the #1 PFD and deselect reversionary mode on GMA 1360 Audio Panel. The displays should return to normal operation.

(k) Select reversionary mode on GMA 1360 Audio Panel.

(l) Remove power from the #2 PFD by pulling the #2 PFD circuit breaker.

(m) Verify the following system messages are displayed on #1 PFD and the MFD:

1) FAILED PATH

2) XPDR 1 (if GTX 345 installed)

(n) Verify the following:

1) MFD and PFD1 switch to reversion mode.

2) Attitude and Heading remain valid from AHRS.

3) Airspeed, Altitude, Vertical Speed and OAT remain valid.

4) MFD retains engine instrumentation.

(o) Restore power to the #2 PFD and deselect reversionary mode on GMA 1360 Audio Panel. The displays should return to normal operation.

(5) AHRS Failure Tests

(a) Single AHRS

NOTE: Allow for the AHRS to realign between tests.

1) Start these tests with all avionics powered.
2) Remove power from the #1 AHRS by pulling the #1 AHRS circuit breaker.
3) Verify Heading, Pitch, and Roll flag invalid on #1 PFD & #2 PFD.
4) Restore power to the #1 AHRS.
5) Verify normal operation and failure annunciations are removed.

(b) Dual AHRS (If Installed)

NOTE: Allow for the AHRS to realign between tests.

1) Start these tests with all avionics powered.
2) Remove power from the #1 AHRS by pulling the #1 AHRS circuit breaker.
3) Verify the following Comparator Annunciations are displayed on the PFDs:
   a) HDG
   b) ROL
   c) PIT
4) Verify Heading, Pitch, and Roll flag invalid on #1 PFD.
5) Press the PFD Opt key on #1 PFD and press Sensors softkey.
6) Press the AHRS2 key on #1 PFD.
7) Verify the following:
   a) #2 AHRS parameters are displayed on #1 PFD without invalid flags.
   b) BOTH ON AHRS2 sensor reversion annunciation is displayed on the PFDs.
   c) White Comparator Annunciations remain on the PFDs.
8) Restore power to the #1 AHRS.
9) Select AHRS1 key on #1 PFD.
10) Verify normal operation, failure and Comparator Annunciations are removed.
11) Repeat steps 2) thru 10) for the #2 AHRS and #2 PFD.

(c) Cross-Side AHRS Test (If Dual AHRS Installed)

1) Start this test with all avionics powered.
2) Press the PFD Opt key on the each PFD and press Sensors softkey.
3) Press the AHRS2 Key on #1 PFD.
4) Press the AHRS1 Key on #2 PFD.
5) Verify the USING AHRS 2 annunciation on #1 PFD and USING AHRS 1 annunciation on #2 PFD.
6) Press the AHRS1 Key on #1 PFD.
7) Press the AHRS2 Key on #2 PFD.
8) Verify AHRS annunciations are removed.

(6) Air Data Computer (ADC) Failure Tests

(a) Single ADC

1) Start these tests with all avionics powered.
2) Remove power from the #1 ADC by pulling the #1 ADC circuit breaker.
3) Verify the following flag invalid #1 PFD & #2 PFD.
   a) Altitude
   b) Indicated Airspeed
   c) Vertical Speed
   d) Static Air Temperature (OAT) and ISA
4) Restore power to the #1 ADC.
5) Verify normal operation and invalid flags are removed.

(b) Dual ADC (If Installed)

1) Start these tests with all avionics powered.
2) Remove power from the #1 ADC by pulling the #1 ADC circuit breaker.
3) Verify the following Comparator Annunciations are displayed on the PFDs:
   a) IAS NO COMP
   b) ALT NO COMP
4) Verify the following flag invalid #1 PFD.
   a) Altitude
   b) Indicated Airspeed
   c) Vertical Speed
   d) Static Air Temperature (OAT) and ISA
5) Press the PFD Opt key on #1 PFD and press Sensors softkey.
6) Press the ADC2 key on #1 PFD.
7) Verify the following:
   a) #2 ADC parameters are displayed on #1 PFD without invalid flags.
   b) BOTH ON ADC2 sensor reversion annunciation is displayed on the PFDs.
   c) White Comparator Annunciations remain on the PFDs.
8) Restore power to the #1 ADC.
9) Select ADC1 key on #1 PFD.
10) Verify normal operation, invalid flags and Comparator Annunciations are removed.
11) Repeat steps 2) thru 10) for the #2 ADC and #2 PFD.

(7) AHRS/ADC Backup Path Tests

(a) Start these tests with all avionics powered.

(b) Remove power from the #1 PFD by pulling the #1 PFD circuit breaker.

(c) On #2 PFD, press the SENSOR soft key and select ADC1 and AHRS1 (if dual sensor is installed). This test verifies the secondary AHRS/ADC 1 paths. The following shall occur during this test:

1) Attitude, Heading, Airspeed, Altitude, Vertical Speed and OAT remain valid.

2) Engine Instrumentation remains valid.

(d) Restore power to the #1 PFD.

(e) Remove power from the #2 PFD and GIA 1 by pulling the #2 PFD and INTEG AV 1 circuit breakers.

(f) On #1 PFD, press the SENSOR soft key and select ADC1 and AHRS1 (if dual sensor is installed). This test verifies the primary AHRS/ADC 1 paths. The following shall occur during this test:

1) Attitude, Heading, Airspeed, Altitude, Vertical Speed and OAT remain valid.

2) COM/NAV1 and COM/NAV2 fields flag invalid.

3) All engine instruments flag invalid.

(g) Restore power to the #2 PFD and GIA 1.

(8) Flight Test

A flight test is recommended after the installation is complete to ensure satisfactory performance of the G1000 NXi.

(9) VHF Comm Tests

Verify the communications capability on both the high and low ends of the VHF COM spectrum.

(10) VOR/ILS Tests

Select a VOR channel within a 40 nautical mile range. Listen to the VOR audio and verify that no electrical interference such as magneto noise is present. Check the tone identifier filter operation. Fly inbound or outbound on a selected VOR radial and check for proper LEFT / RIGHT, TO / FROM flag indications on the CDI. Check the VOR accuracy.

(11) Maintenance Records

After conducting the function check flight, the airplane may be returned to service. Record the following information in the appropriate airplane maintenance logs:

(a) Part number of the loader card used to perform software loading or software updates.

(b) Any other applicable information related to the maintenance work performed on the airplane.

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CHART 30
G1000 NXi - RETURN TO SERVICE CHECK LIST

Group System	SW Ver. OK	System	SW Ver. OK	System	SW Ver. OK
Control & Display:					
PFD 1	_____	PFD 2 FPGA	_____		
PFD 1 FPGA	_____	MFD1	_____		
PFD 2	_____	MFD1 FPGA	_____		
CNS:					
GIA1	_____	GPS2	_____	GDL69	_____
GIA2	_____	GTX1	_____	GWX	_____
GMA1	_____	GIA1 AUDIO	_____	GWX FPGA	_____
GPS1	_____	GIA2 AUDIO	_____	GTS	_____
Sensors:					
GDC1	_____	GMU2	_____	GEA1	_____
GDC1 FPGA	_____	GMU2 FPGA	_____	GEA2	_____
GDC2	_____	GRS1	_____		
GDC2 FPGA	_____	GRS1 FPGA	_____		
GMU1	_____	GRS2	_____		
GMU1 FPGA	_____	GRS2 FPGA	_____		

CHART 31
REQUIRED EQUIPMENT LISTS (REL)

G1000 NXi VER. 3360.02 (Sheet 1 of 2)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
COM1	006-B2371-02	2.02	
COM2	006-B2371-02	2.02	
GDC1	006-B1838-55	2.05	
GDC1 RGN LIST	006-D5080-50	2.00	
GDC2	006-B1838-55	2.05	Opt., Verify if installed.
GDC2 RGN LIST	006-D5080-50	2.00	Opt., Verify if installed.
GDL69	006-B1902-06	5.21	Opt., Verify if installed.
GEA1	006-B2139-03	2.11	
GEA1 REGION LIST	006-D6358-00	2.00	
GEA1 XHTML	006-D6358-H6	2.05	
GEA2	006-B2139-03	2.11	
GEA2 REGION LIST	006-D6358-00	2.00	
GEA2 XHTML	006-D6358-H6	2.05	
GIA1	006-B2548-0B	2.05	
GIA1 AUDIO	006-D0425-25	4.05	
AFCS GAINS	006-D4801-01	2.01	
GIA2	006-B2548-0B	2.05	
GIA2 AUDIO	006-D0425-25	4.05	
AFCS GAINS	006-D4801-01	2.01	
GMA1	006-B2212-09	5.26D	
GMA1 AUDIO CONFIG	006-D3035-EF	5.27D	
GMA1 AUX	006-B2210-08	5.26D	
GMU1	006-B2679-00	2.00	
GMU2	006-B2679-00	2.00	
GPS1	006-B1827-10	7.0	
GPS2	006-B1827-10	7.0	
GRS1	006-B1838-06	2.06	
GRS1 REGION LIST	006-D5080-00	2.00	
GRS2	006-B1838-06	2.06	Opt., Verify if installed.
GRS2 REGION LIST	006-D5080-00	2.00	Opt., Verify if installed.

CHART 30
REQUIRED EQUIPMENT LISTS (REL)

G1000 NXi VER. 3360.02 (Sheet 2 of 2)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
GSA PTCH CTL	006-B0398-41	3.51	Opt., Verify if installed.
AFCS GAINS	006-D4801-01	2.01	Opt., Verify if installed.
GSA PTCH MON	006-B0398-41	3.51	Opt., Verify if installed.
AFCS GAINS	006-D4801-01	2.01	Opt., Verify if installed.
GSA PTCH TRM C	006-B0398-41	3.51	Opt., Verify if installed.
AFCS GAINS	006-D4801-01	2.01	Opt., Verify if installed.
GSA PTCH TRM M	006-B0398-41	3.51	Opt., Verify if installed.
AFCS GAINS	006-D4801-01	2.01	Opt., Verify if installed.
GSA ROLL CTL	006-B0398-41	3.51	Opt., Verify if installed.
AFCS GAINS	006-D4801-01	2.01	Opt., Verify if installed.
GSA ROLL MON	006-B0398-41	3.51	Opt., Verify if installed.
AFCS GAINS	006-D4801-01	2.01	Opt., Verify if installed.
GSA YAW CTL	006-B0398-41	3.51	Opt., Verify if installed.
AFCS GAINS	006-D4801-01	2.01	Opt., Verify if installed.
GSA YAW MON	006-B0398-41	3.51	Opt., Verify if installed.
AFCS GAINS	006-D4801-01	2.01	Opt., Verify if installed.
GTS	006-B1244-59	3.13	Opt., Verify if installed.
GTS FPGA	006-C0081-20	2.0	Opt., Verify if installed.
GTX1	006-B1607-09	2.12	
GWX	006-B2386-05	2.21	Opt., Verify if installed.
MFD1	006-B1177-6A	20.87	
NAV1	006-B2253-00	2.00	
NAV1 REGION LIST	006-D5761-00	2.00	
NAV2	006-B2253-00	2.00	
NAV2 REGION LIST	006-D5761-00	2.00	
PFD1	006-B1177-6A	20.87	
PFD2	006-B1177-6A	20.87	
PA-34 IMAGE	006-B3360-02	3360.02(1-2)	

(1) PIR-34G34010 B.
(2) See “Figure 41” on page 342502155.

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INDEPENDENT POSITION DETERMING

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

1. Stormscope WX-500

A. Description and Operation

The optional Stormscope WX-500 Weather Mapping Sensor detects electrical discharges associated with thunderstorms within a 200 mile radius of the aircraft. Information is sent to an external multifunction display (MFD) that shows the location of the thunderstorms.

WX-500 is a passive sensor that listens for electromagnetic signals with an antenna. There is no transmitter or harmful transmissions. Stormscope operates on the ground or in the air, giving the pilot important planning information before flying. The two major components of the WX-500 are:

(1) WX-500 Processor

Processor houses data acquistion circuitry plus circuitry to process heading and communicate with the MFD. Processor is tray mounted on the aft shelf just forward of F.S. 244.44.

(2) Antenna

Combined croos-loop and sense antenna is sealed in an aerodynamic flat-pack and mounte on the outside of the aircraft at F. S. 238.75. Stormscope detection systems can correlate elctric and magnetic signatures of lightning strikes better than other systemsdue to its patened sense channel technology. Antenna is designed to help filter out pulsed noise from sources other than atmospheric electrical discharges.

B. Troubleshooting

Stormscope WX-500 detects most common faults and sends error messages to the MFD indicating the fault and which functions may be inoperative. See the manufacturers user's manual for an updated list of error messages and recommended actions.

C. WX-500 Processor

See "Figure 1" on page 34404

CAUTION: BEFORE REMOVING PROCESSOR FROM MOUNTING TRAY, DE-ENERGIZE OR DISCONNECT ALL POWER AND SIGNAL SOURCES AND LOADS USED WITH THE WX-500 SENSOR.

(1) Removal

NOTE: Removal of components is on condition of failure.

(a) Remove safety wire from locking mechanism.
(b) Loosen hold down mechanism securing processor to mounting tray.
(c) Slide the processor out of mounting tray.

(2) Installation

CAUTION: BEFORE INSTALLING PROCESSOR FROM MOUNTING TRAY, DE-ENERGIZE OR DISCONNECT ALL POWER AND SIGNAL SOURCES AND LOADS USED WITH THE WX-500 SENSOR.

(a) Align the processor with guide pins at rear of the mounting tray and Slide the processor into place.

NOTE: Substantial hand pressure must be used when seating processor into the tray. When properly seated the processor front surface will be flush with front edge of the tray, or slightly recessed.

(b) Place the front hold-down retainer cup over the processor J-hook and tighten the hold down mechanism securing the processor to the mounting tray.

(c) Secure locking mechanism with aircraft safety wire.

(d) After installation is complete, conduct Stormscope Post-Installation Test. See “Stormscope Post-Installation Test” on page 342366.

D. Antenna

See “Figure 1” on page 34404.

CAUTION: BEFORE REMOVING ANTENNA FROM AIRCRAFT, DE-ENERGIZE OR DISCONNECT ALL POWER AND SIGNAL SOURCES AND LOADS USED WITH THE WX-500 SENSOR.

(1) Removal

NOTE: Removal of components is on condition of failure.

(a) Remove two antenna mounting screws.

(b) Carefully break seal and move antenna away from aircraft skin enough to access the electrical connector.

(c) Disconnect electrical connector.

(2) Installation

CAUTION: BEFORE INSTALLING ANTENNA TO AIRCRAFT,, DE-ENERGIZE OR DISCONNECT ALL POWER AND SIGNAL SOURCES AND LOADS USED WITH THE WX-500 SENSOR.

(a) Align antenna and gasket, connect electrical connector.

(b) Secure antenna using two mounting screws and washers. Do Not Over Tighten mounting screws.

NOTE: The antenna should not be sealed until the sensor performance tests are successfully completed.

(c) Seal around the entire perimeter of the antenna.

1) For non-pressurized aircraft use a non-corrosive sealant that meets the physical requirements of MIL-A-4461466 such as General Electric RTV 162.

2) For pressurized aircraft, use a sealant that meets the requirements of SAE AMS-S-8802 such as Flamemaster® CS3204 Class B.

(d) After installation is complete, conduct Stormscope Antenna Test. See “Stormscope Antenna Test” on page 342367.

E. Maintenance

At regular intervals, perform periodic maintenance as described below:

(1) Processor

Check to ensure unit is properly seated and secured in the mounting tray.

(2) Antenna

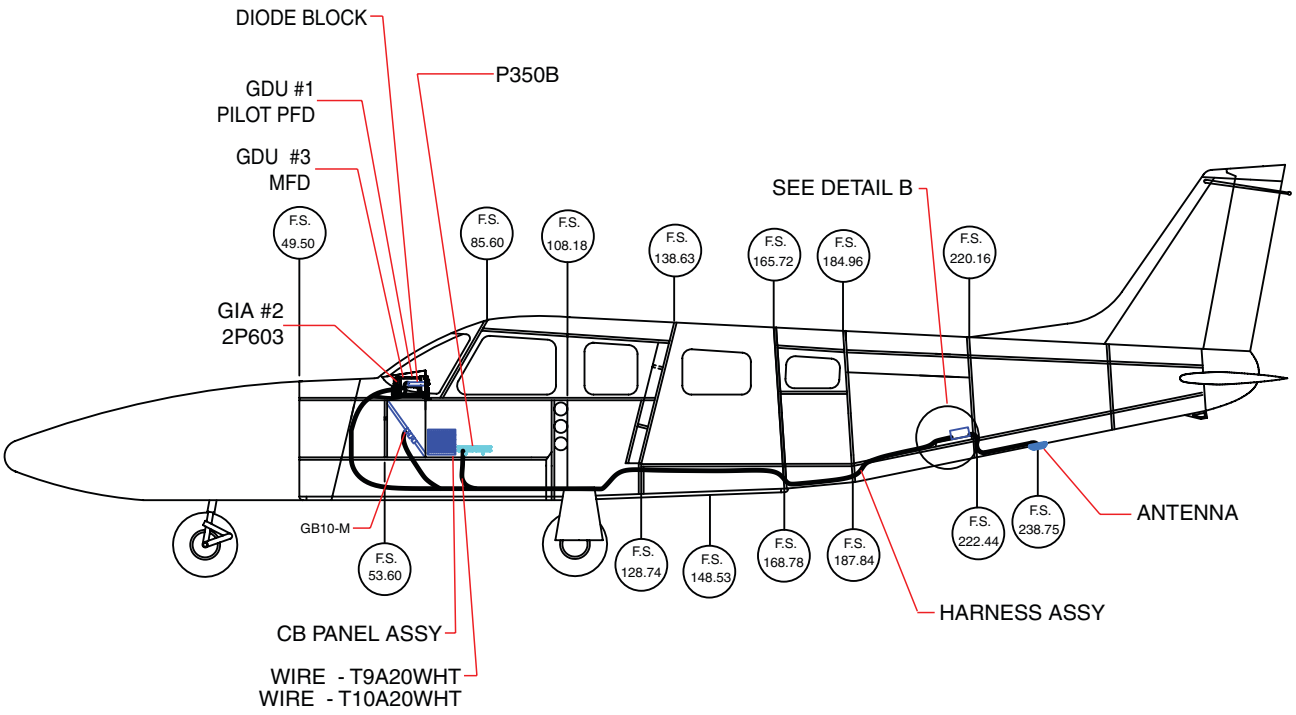
(a) Check for dents, cracks and punctures.

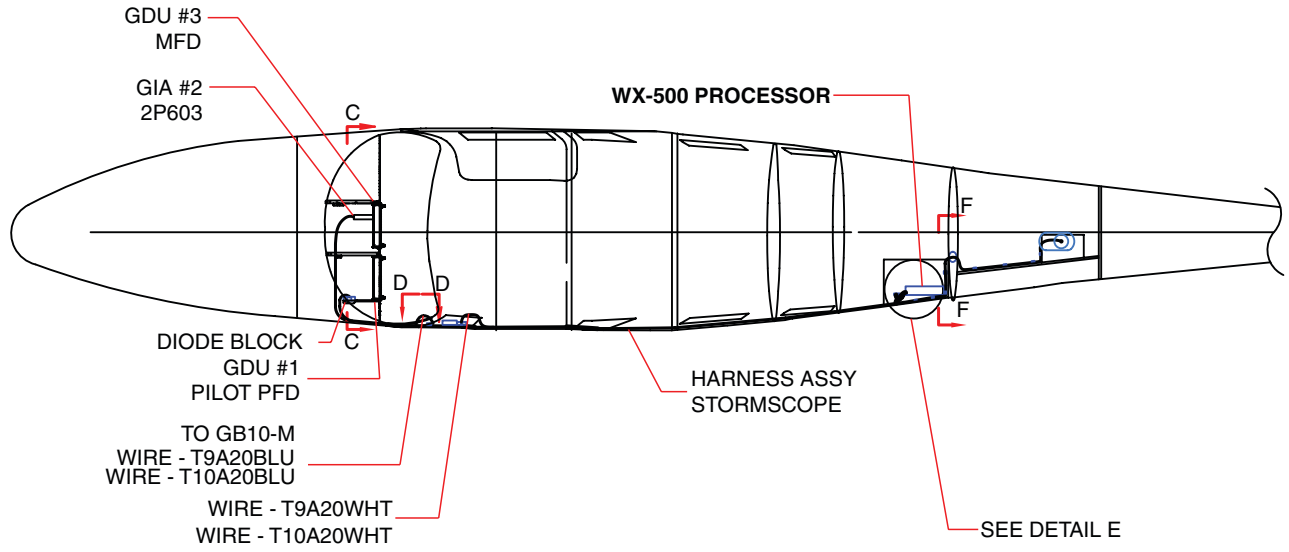
CAUTION: DO NOT PAINT ANTENNA. DO NOT USE CLEANING SOLVENTS ON ANTENNA.

(b) Remove all dirt and grease from surface areas. Clean with a soft cloth moistened with mild soap and water.

(c) Visually inspect sealant around antenna base. Reapply sealant if needed.

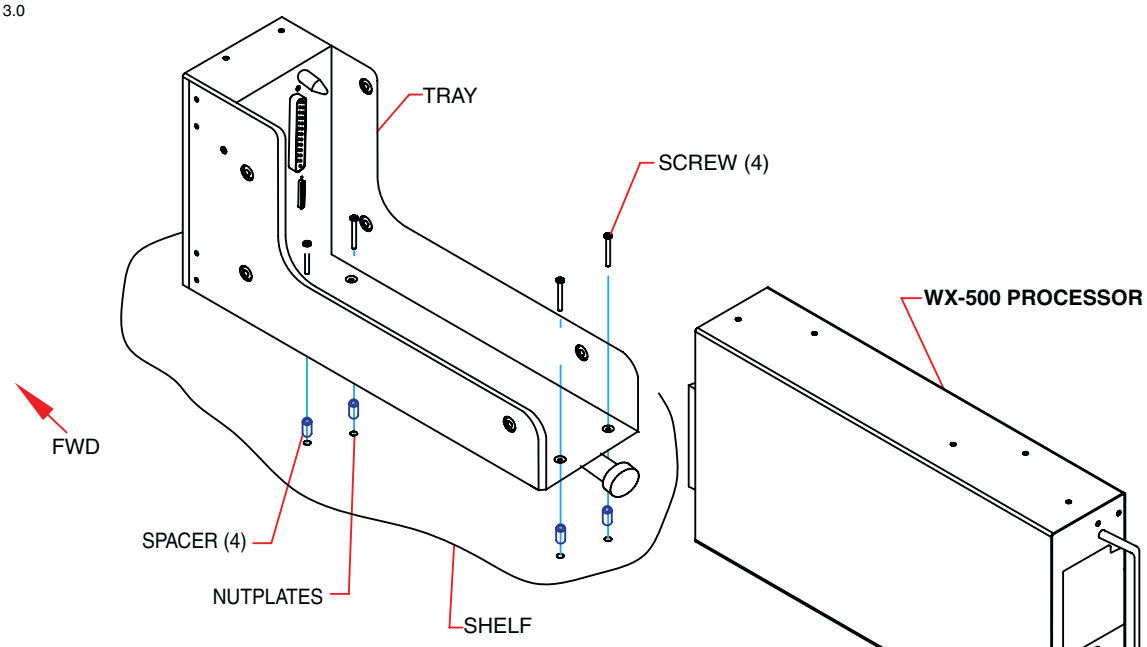
107326 NEW 2.0



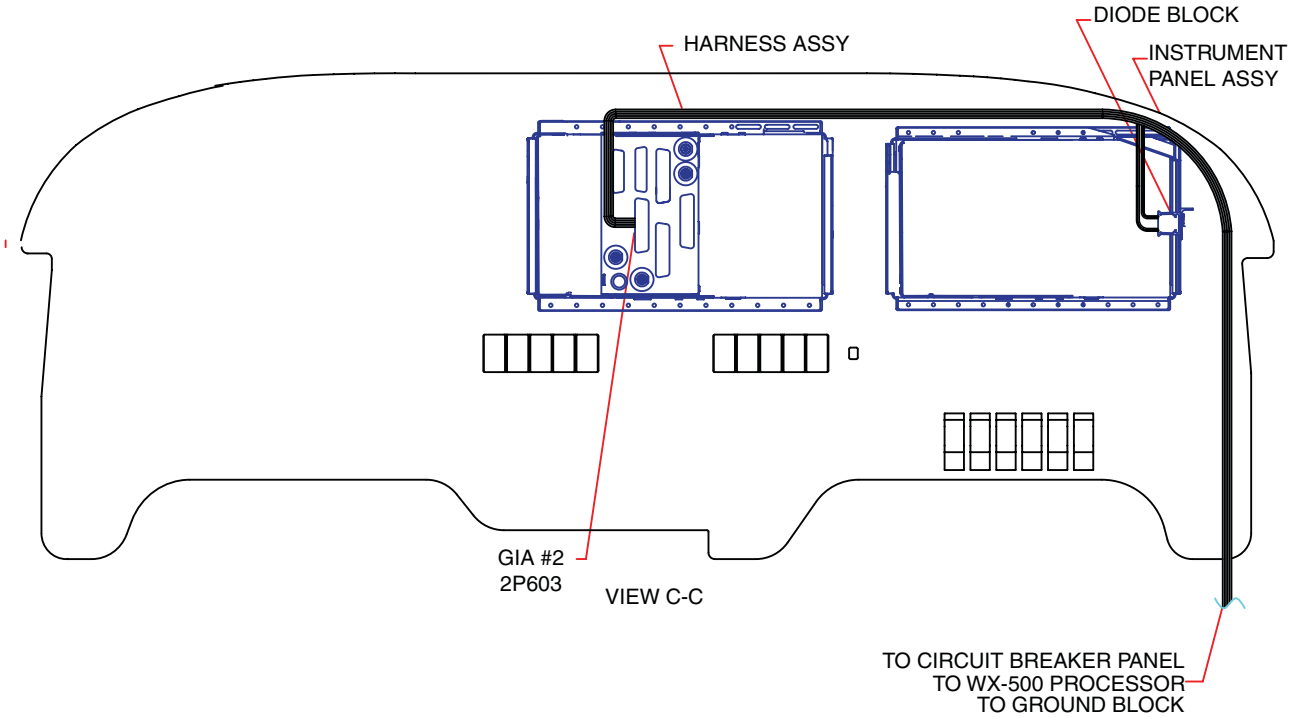


WX-500 Stormscope Installation
Figure 1 (Sheet 1 of 4)

107326 NEW 3.0

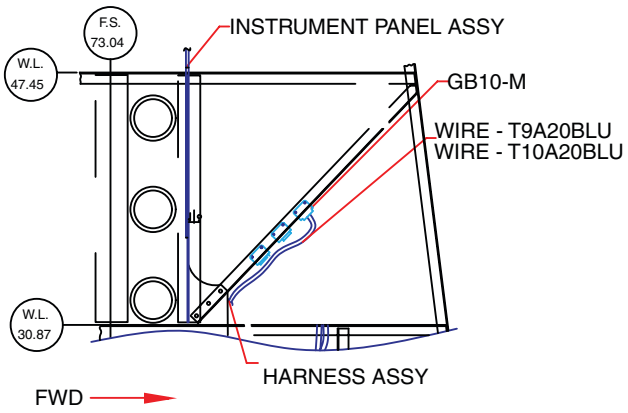


DETAIL B
STORMSCOPE PROCESSOR INSTALLATION
LH SIDE LOOKING INBOARD

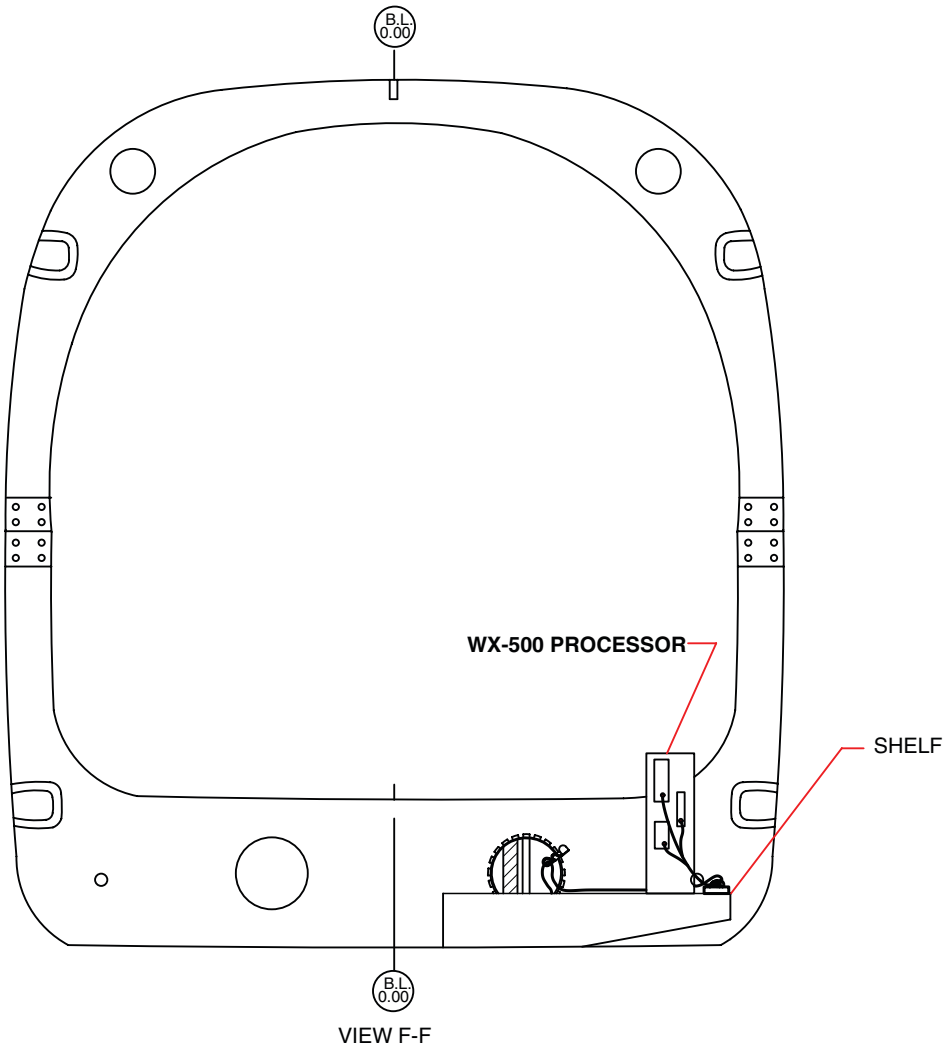


WX-500 Stormscope Installation
Figure 1 (Sheet 2 of 4)

107326 NEW 1.0
3.0

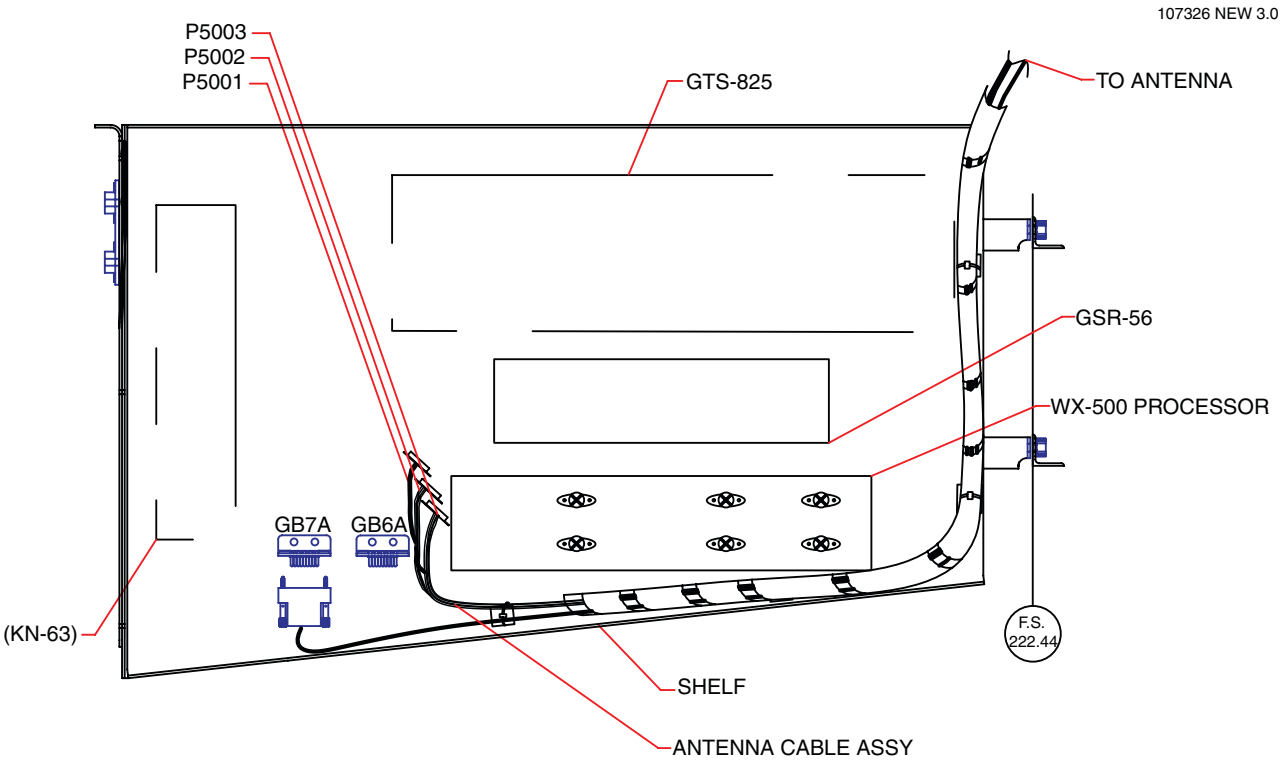


VIEW D-D
HARNESS GROUNDING DETAIL
LH SIDE LOOKING OUTBOARD

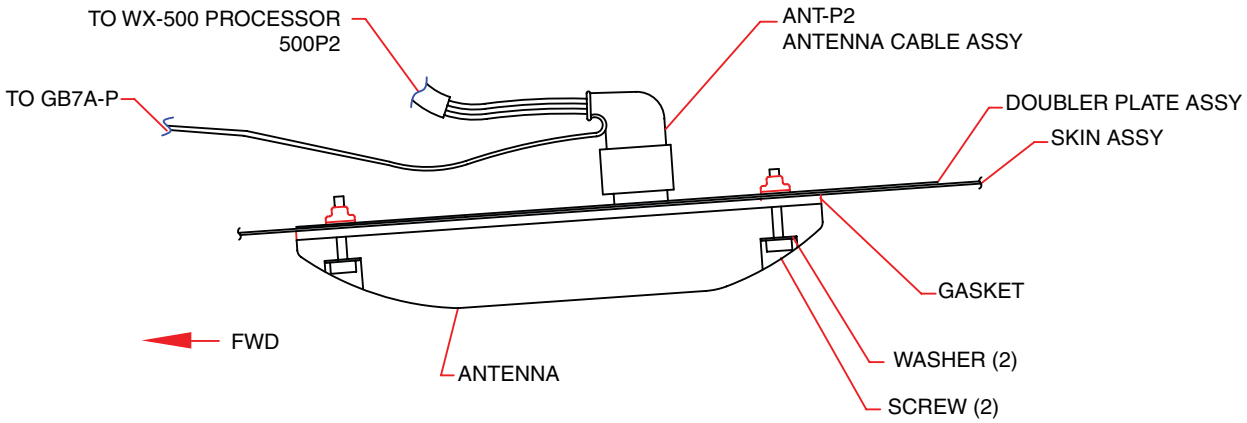


VIEW F-F

WX-500 Stormscope Installation
Figure 1 (Sheet 3 of 4)



DETAIL E
AFT AVIONICS SHELF GROUNDING & CONNECTOR DETAIL
LOOKING DOWN



VIEW H-H
WX-500 ANTENNA DETAIL
LH SIDE LOOKING INBOARD

WX-500 Stormscope Installation
Figure 1 (Sheet 4 of 4)

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DEPENDENT POSITION DETERMINING

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

1. COM/NAV/GPS

S/N's 3449269 and up.

A. Description

Dual Garmin GNS COM/NAV/GPS systems are installed as standard equipment. Earlier airplanes have dual GNS 430s. Later airplanes have a GNS 430(W) and a GNS 530 installed. Airplanes equipped with Garmin G600 have dual GNS 430Ws. The installation consists of the GNS transceiver/display units, associated wiring, and antennas.

Maintenance of the Garmin GNS 400/500 series is "on condition" only and, with the exception of swapping complete units, should be performed only by a qualified avionics shop in accordance with the appropriate GARMIN Maintenance Manual.

Information provided in this manual is intended solely to aid the removal and installation of the GNS 430(W)/530 transceiver/display units, their associated wiring and antennas.

B. Removal and Installation

See 39-10-00.

C. Post Installation Setup Procedure

(PIR-PPS60199-2, Rev. K/PPS60199-6, Rev. New / PPS55028, Rev. E.)

(1) Aircraft Preparation and Configuration

Setup and preliminary checks may be made using the aircraft battery or external power supplied by an external power cart. Final checks (OBS/CDI or HSI accuracy) should be made with the engine running and the aircraft configured as in "Chart 1".

(2) Configuration Setup Procedure

Access the Configuration Mode of the unit by depressing and holding the ENT key while applying power to the unit. Release the ENT key when the display activates. After the database pages, press the ENT key twice to display the MAIN ARINC 429 CONFIG page. Pages may be selected by ensuring the cursor is off and rotating the inner concentric knob on the right side of the unit.

CHART 1
FINAL CHECK CONFIGURATION

Engine running, alternators	On	Avionics	On
Instrument Panel Lights	Full Bright	Day/Night Switch	Day
Navigation Lights	On	Strobe Lights	On
Pitot Heat	Off	Air Conditioning	Off
Cabin Blower Fan	Low	Vent Blower	Off

To change data on the displayed Configuration Page, the cursor must be selected. Press the inner concentric knob on the right side of the unit to activate the cursor. Rotating the outer concentric knob on the right side of the unit changes the selected data field. Rotating the inner concentric knob changes the data within the selected field. To accept entry of the desired selection, press the ENT key.

(a) Main ARINC 429

Configure the MAIN ARINC 429 page on GNS #1 and GNS #2 as follows:

1) Dual GNS-430 with Avidyne Entegra IFDS

	GNS UNIT #1		GNS UNIT #2	
BUS	**SPEED**	**DATA**	**SPEED**	**DATA**
IN 1	LOW	SANDEL EHSI	LOW	SANDEL EHSI
IN 2	LOW	“EFIS/AIR DATA”	LOW	“EFIS/AIR DATA”
OUT	LOW	GAMA 429 GRAPHICS	LOW	GAMA 429 GRAPHICS
SDI	LNAV 1		LNAV 2	
VNAV *	ENABLE LABELS		ENABLE LABELS	* with GNS-430W only.

2) GNS-530 (Unit #1) and GNS-430 (Unit #2) with Meggitt EFIS

	GNS UNIT #1		GNS UNIT #2	
BUS	**SPEED**	**DATA**	**SPEED**	**DATA**
IN 1	HIGH HIGH	GARMIN GTX 330 “AIRDATA” W/IHAS	HIGH HIGH	GARMIN GTX 330 “AIRDATA” W/IHAS
IN 2	HIGH	SANDEL EHSI	HIGH	SANDEL EHSI
OUT	HIGH	GAMA 429 GRAPHICS with INT	HIGH	GAMA 429 GRAPHICS with INT
SDI	LNAV 1		LNAV 2	

3) GNS-530 (Unit #1) and GNS-430 (Unit #2) with Mechanical Indicators

	GNS UNIT #1		GNS UNIT #2	
BUS	**SPEED**	**DATA**	**SPEED**	**DATA**
IN 1	HIGH HIGH	GARMIN GTX 330 “AIRDATA” W/IHAS	HIGH HIGH	GARMIN GTX 330 “AIRDATA” W/IHAS
IN 2	HIGH	OFF	HIGH	OFF
OUT	HIGH	GAMA 429 GRAPHICS with INT	HIGH	OFF
SDI	LNAV 1		LNAV 2	

4) Dual GNS-430W with Garmin G600 Integrated System

	GNS UNIT #1		GNS UNIT #2	
BUS	**SPEED**	**DATA**	**SPEED**	**DATA**
IN 1	LOW	GARMIN GDU	LOW	GARMIN GDU
IN 2	LOW	OFF	LOW	OFF
OUT	HIGH	GAMA 429	HIGH	GAMA 429
SDI	LNAV 1		LNAV 2	
VNAV	ENABLE LABELS		ENABLE LABELS	

(b) Main RS232 Config

Deselect the cursor and rotate the inner concentric knob on the right side of the unit to display the MAIN RS232 CONFIG page. Configure GNS #1 and #2 as follows:

NOTE: Channels 5 and 6 (not shown) are always “OFF”; except with GNS-430W - in that case, not used.

1) Dual GNS-430 with Avidyne Entegra IFDS

BUS	GNS UNIT #1 INPUT	GNS UNIT #1 OUTPUT	GNS UNIT #2 INPUT	GNS UNIT #2 OUTPUT
CHNL 1	OFF	OFF	OFF	OFF
CHNL 2	OFF	OFF or HW EGPWS (W/IHAS)	OFF	OFF
CHNL 3	CROSSFILL	CROSSFILL	CROSSFILL	CROSSFILL
CHNL 4	OFF	OFF	OFF	OFF

2) GNS-530 (Unit #1) and GNS-430 or GNS-530 (Unit #2) with Meggitt EFIS

BUS	GNS UNIT #1 INPUT	GNS UNIT #1 OUTPUT	GNS UNIT #2 INPUT	GNS UNIT #2 OUTPUT
CHNL 1	OFF	“AVTN NO ALT”	OFF	OFF
CHNL 2	OFF	OFF or HW EGPWS (W/IHAS)	OFF	OFF
CHNL 3	CROSSFILL	CROSSFILL	CROSSFILL	CROSSFILL
CHNL 4	OFF	OFF	OFF	OFF

3) GNS-530 (Unit #1) and GNS-430 or GNS-530 (Unit #2) with Mechanical Indicators

BUS	GNS UNIT #1 INPUT	GNS UNIT #1 OUTPUT	GNS UNIT #2 INPUT	GNS UNIT #2 OUTPUT
CHNL 1	OFF	“AVTN NO ALT”	OFF	OFF
CHNL 2	OFF	OFF or HW EGPWS (W/IHAS)	OFF	OFF
CHNL 3	CROSSFILL	CROSSFILL	CROSSFILL	CROSSFILL
CHNL 4	OFF	OFF	OFF	OFF

4) Dual GNS-430W with Garmin G600 Integrated System

BUS	GNS UNIT #1 INPUT	GNS UNIT #1 OUTPUT	GNS UNIT #2 INPUT	GNS UNIT #2 OUTPUT
CHNL 1	OFF	MapMX	OFF	MapMX
CHNL 2	OFF	AVTN NO ALT	OFF	OFF
CHNL 3	CROSSFILL	CROSSFILL	CROSSFILL	CROSSFILL
CHNL 4	WX-500 *	WX-500 *	WX-500 *	OFF

* Set to “OFF” if Stormscope not installed.

(c) Main System Config

Deselect the cursor and rotate the inner concentric knob on the right side of the unit to display the MAIN SYSTEM CONFIG page. Configure fuel, terrain and discretes as follows:

1) Dual GNS-430 with Avidyne Entegra IFDS or GNS-530 (Unit #1) and GNS-430 or GNS-530 (Unit #2) with Meggitt EFIS or with Mechanical Indicators

	GNS UNIT #1 (or single unit)	**GNS UNIT #2**
CONFIGURE	FUEL	FUEL
FUEL TYPE	AV GAS	AV GAS
CONFIGURE	TERRAIN	
TERRAIN TYPE	“TERRAIN” (W/O TAWS B) “TAWS” (W/ TAWS B)	TERRAIN
TEST CARD?	PASS	PASS
HW CONFIGURE	“TERRAIN” (W/O TAWS B) “TAWS” (W/ TAWS B)	TERRAIN

2) Dual GNS-430W with Garmin G600 Integrated System

	GNS UNIT #1	**GNS UNIT #2**
CONFIGURE	FUEL	FUEL
FUEL TYPE	AV GAS	AV GAS

NOTE: For the next steps, to change the item in the configure window on the MAIN SYSTEM CONFIG page, it may be necessary to first exit the MAIN SYSTEM CONFIG page, then select it again, then enable the cursor.

CONFIGURE	TERRAIN	TERRAIN
TERRAIN TYPE	NONE	NONE
CONFIGURE	DISCRETES	DISCRETES
GPS SELECT	AUTO	AUTO
COM PRESETS	DISABLED	DISABLED

(d) Instrument Panel Self-Test

On this page, verify that each GNS unit communicates properly with other units. Compare the on-screen indications with information depicted on connected units.

(e) Main Lighting

Deselect the cursor and rotate the inner concentric knob on the right side of the unit to display the MAIN LIGHTING page.

1) Dual GNS-430 with Avidyne Entegra IFDS or GNS-530 (Unit #1) and GNS-430 or GNS-530 (Unit #2) with Meggitt EFIS or with Mechanical Indicators

MAIN LIGHTING	DISPLAY	KEY	
LIGHTING	NO SETTING	NO SETTING	
SOURCE	PHOTO	PHOTO	
RESPONSE TIME/ MINIMUM	4 / 80	4 / 40 *	* 4 / 99 with GNS-430W only.
SLOPE/OFFSET	50 / 50	50 / 50 *	* 50 / 90 with GNS-430W only.

2) Dual GNS-430W with Garmin G600 Integrated System

MAIN LIGHTING	DISPLAY	KEY
LIGHTING	NO SETTING	NO SETTING
SOURCE	28V DC	28V DC
RESPONSE TIME/ MINIMUM	4 / 80	4 / 99
SLOPE/OFFSET	25 / 50	70 / 65
PHOTO TRANS %	25	
PHOTO SLP/OFST	50 / 22	

(f) Main CDI/OBS or HSI

1) Dual GNS-430 with Avidyne Entegra IFDS or GNS-530 (Unit #1) and GNS-430 or GNS-530 (Unit #2) with Meggitt EFIS or with Mechanical Indicators

Deselect the cursor and rotate the inner concentric knob on the right side of the unit to display the MAIN CDI/OBS CONFIG page, which allows you to verify the MAIN CDI outputs, both lateral (LAT) and vertical (VERT); and allows you to verify and calibrate the MAIN OBS input. (GNS Unit #1 only with GNS-430W.)

a) Adjust the OBS or HSI to indicate a selected course of 150°.

b) On GNS, the SELECTED COURSE field should indicate near to 150° and a “Calibrate to 150°?” field should appear.

c) Select this field to calibrate the GNS to match the input source.

d) Verify OBS (or HSI course) operation by checking that the course displayed on the GNS unit is within 2° of the selected course on the OBS or HSI.

e) Repeat steps a) thru d) at 30° intervals around the OBS card.

2) Dual GNS-430W with Garmin G600 Integrated System

Deselect the cursor and rotate the inner concentric knob on the right side of the unit to display the MAIN CDI/OBS CONFIG page.

a) Select MENU pushbutton, highlight “Ignore CDI Key?” and press the ENT key.

b) Verify that “GPS only” is displayed in the CDI window.

c) Verify that “No Input” is displayed for Selected Course.

d) Select “Always GPS” for OBI Source.

(g) COM Setup

Deselect the cursor and rotate the inner concentric knob on the right side of the unit to display the COM SETUP page. Change only the SPACING setting to display “25.0 KHz” (frequency spacing).

(h) VOR/LOC/GS CDI Config

Deselect the cursor and rotate the inner concentric knob on the right side of the unit to display the VOR/LOC/GS CDI CONFIG page.

1) For airplanes with Mechanical Indicators, Meggitt EFIS or with Avidyne Entegra installed

Change only the DME CHNL MODE setting, to display “Parallel 2x5,” when DME is installed.

2) For airplanes with Garmin G600 and optional GI-106 installed,

a) Electrical power applied to Avionics, GNS #2 running in configuration mode.

b) On GNS #2, select the VOR/LOC/GS CDI Page.

c) Select OBS to 150° on the GI-106.

d) On GNS #2, the SELECTED COURSE field should indicate near to 150° and a “Calibrate to 150°?” field should appear.

e) Select this field to calibrate the 430W to match the input source and press ENT key.

f) Verify that the course displayed on the GNS #2 is within 2° of the selected course on the GI-106 at 30-degree intervals around the course card.

g) Repeat steps c) thru f) at 30° intervals around the OBS card.

3) For airplanes with Garmin G600 and optional DME installed,

a) Electrical power applied to Avionics, GNS #1 and GNS #2 running in configuration mode.

b) Set “Parallel 2X5” for DME CHNL MODE on GNS #1 and GNS #2.

(i) VOR/LOC/GS ARINC 429 Config

GNS #1 and GNS #2 are to be as follows:

1) For airplanes with Mechanical Indicators or with Meggitt EFIS,

	GNS UNIT #1	**GNS UNIT #2**
SPEED	HIGH (RX & TX)	HIGH (RX & TX)
SDI	VOR/ILS 1	VOR/ILS 2
DME MODE	DIRECTED FREQ 1	DIRECTED FREQ 2

2) For airplanes with Avidyne Entegra installed,

	GNS UNIT #1	**GNS UNIT #2**
SPEED	LOW (RX & TX)	LOW (RX & TX)
SDI	VOR/ILS 1	VOR/ILS 2
DME MODE	DIRECTED FREQ 1	DIRECTED FREQ 2

3) For airplanes with Garmin G600 installed,

	GNS 430W UNIT #1	**GNS 430W UNIT #2**
SPEED	LOW (RX & TX)	LOW (RX & TX)
SDI	VOR/ILS 1	VOR/ILS 2
DME MODE	DIRECTED FREQ 1 (W/ DME INSTALLED)	DIRECTED FREQ 2 (W/ DME INSTALLED)

(j) GPS Vertical Offset

GPS antenna height is 6.0 above ground level.

(k) Stormscope Config

NOTE: Perform this procedure only if the optional Stormscope is installed.

1) Electrical power applied to Avionics, GNS #1 running in configuration mode.
2) On GNS #1, select the STORMSCOPE CONFIG Page.
3) Verify the following settings:
   a) HDG FORMAT is “SERIAL” (RS-232).
   b) ANT MOUNT is BOTTOM.
   c) HDG FLAG SENSE is High Valid.
   d) Status of HDG items on bottom left side of page are valid (green).

D. VOR/LOC/GS Ramp Checks

Once the GNS units have been configured, proper navigation receiver and indicator operation (VOR/LOC/GS) should be verified using the IFR NAV-401L ramp test set or equivalent. This procedure assumes familiarity with proper operation of the test equipment and is not intended to be a tutorial on test set use. Refer to the appropriate test set operating instructions for proper operation of the ramp test equipment.

(1) OBS/CDI Checks

(a) Check that the proper OBS indicator is connected to the appropriate VOR receiver – #1 OBS connected to GNS #1; #2 OBS connected to GNS #2.

(b) Check the lateral and vertical deviation needles and/or lateral and vertical flags for deflection while keying the communication transmitter on a representative sampling of frequencies (118.00 MHz to 136.975 MHz).

(c) Tune the Navigation receiver and the adjust the test set to 108.0 MHz. Using the ramp test set, check the accuracy of the OBS or Course Selector needle (HSI) at cardinal increments (30°, 60°, … , 330°, 360°).

(2) Localizer Indicator Checks

(a) Tune the navigation receiver and adjust the test set to 108.1 MHz. Using the ramp test set, check that the course deviation needle (CDI) centers within one-half (½) dot with zero (0) DDM applied at the test set.

(b) Check left and right deviation of the CDI at the following test points:

CDI (LOC)	DDM	DEFLECTION
Centered	0.000	0% F.S.
Standard Deviation	0.093	60% F.S.
Full Scale Deflection	0.155	100% F.S.
More than F.S.	0.200	100%+ F.S.
Full Tone (single mod)	0.400	FLAG

(3) Glide Slope Indicator Checks

(a) Set Navigation receiver and the test set to 108.1 MHz. Glide Slope generator defaults to proper G/S frequency (334.7 MHz). Using the ramp test set, check that the needle (G/S) centers within one-half (½) dot with zero (0) DDM applied at the test set.

(b) Check up and down deviation of the glide slope indicator at the following points:

G/S	DDM	DEFLECTION
Centered	0.000	0% F.S.
Standard Deviation	0.091	52% F.S.
Full Scale Deflection	0.175	100% F.S.
More than F.S.	0.400	100%+ F.S.
Full Tone (single mod)	0.800	FLAG

E. G600 GNS-430W NAV/COM/GPS Ramp Checks

(1) GNS X-TALK

(a) Apply electrical power to Avionics.

(b) On GNS #1, select the FLIGHT PLANNING Page from the AUX page group.

(c) Select CROSSFILL.

(d) Verify that the displayed status is READY.

(e) Repeat steps (b) thru (d) at 30° intervals around the OBS card.

(2) VOR Post-Installation Test

(a) Apply electrical power to Avionics.

(b) Select NAV1 as source on GDU #1, NAV 2 as source on GDU #2.

(c) Adjust test set to 108.0 MHz (if applicable).

(d) Tune the GNS#1 and GNS #2 to same frequency as test set.

(e) On GDU #1 and GDU #2, set the selected course to 30 degrees.

(f) Using the ramp test set, transmit a test signal for a 30 degree radial.

(g) Verify the course deviation indication (CDI) centers within ±4 degrees course selection.

(h) Repeat steps (c) thru (g) at 30° increments (30°, 60°, … , 330°, 360°).

(3) Localizer Post-Installation Test

(a) Apply electrical power to Avionics.

(b) Adjust the test set to 108.1 MHz (if applicable). Apply zero (0) DDM.

(c) Select LOC1 as the NAV source on GDU #1 and LOC2 as the NAV source on GDU #2.

(d) Tune GNS #1 and GNS #2 to same frequency as test set.

(e) Verify that the course deviation indicator (CDI) centers within 1/2 dot on each GDU.

(f) Verify proper CDI deflection LEFT and RIGHT for LOC1 and LOC2, using the DDM for the following test points:

CDI (LOC)	DDM	DEFLECTION
Centered	±0.000	Centered
Standard Deviation	±0.093	60% F.S.
Full Scale Deflection	±0.155	Full Scale
More than F.S.	±0.200	100%+ F.S.
Full Tone (single mod)	±0.400	FLAG

(4) VHF NAV Audio Test

(a) Apply electrical power to Avionics.

(b) Select NAV1 as receive input on Audio panel.

(c) Verify receive audio at pilot headset using NAV1.

(d) Select NAV2 as receive input on Audio panel.

(e) Verify receive audio at pilot headset using NAV2.

(5) Glide Slope Post-Installation Test

(a) Electrical power applied to Avionics.

(b) Adjust the test set to 108.1 MHz (if applicable). Glide Slope generator defaults to proper G/S frequency (334.7 MHz.).

(c) Select LOC1 as the NAV source on GDU #1 and LOC2 as the NAV source on GDU #2.

(d) Tune GNS #1 and GNS #2 to same frequency as test set.

(e) Verify that the glideslope deviation indicator centers within 1/2 dot with 0 DDM applied at the test set.

(f) Verify proper glideslope deviation indicator deflection UP and DOWN for GNS#1 and GNS#2, using the DDM for the following glide slope test points:

G/S	DDM	DEFLECTION
Centered	±0.000	0% F.S.
Standard Deviation	±0.091	52% F.S.
Full Scale Deflection	±0.175	100% F.S.
More than F.S.	±0.400	100%+ F.S.
Full Tone (single mod)	±0.800	FLAG

(6) GPS Operation

(a) Apply electrical power to Avionics.

(b) On GDU #1, select GPS1 as the navigation source for the PFD.

(c) On GDU #2, select GPS2 as the navigation source for the PFD.

(d) Re-set power on each GNS.

(e) While on the power-up self-test page of each GNS, verify that the correct lateral and vertical deviation information is displayed on the correspondent PFD.

(f) Wait until both navigators acquire a position before proceeding.

(g) Review the active alerts on each MFD (if any) and verify that the GPSx PPS FAIL alert is not present (where “x” is the source number).

(h) Create/activate a flight plan on the GPS navigator.

(i) Review the active alerts on each MFD (if any) and verify that the NO GPS POSITION is not present.

(j) Verify that the active waypoint displayed on each PFD is the same as the active waypoint on the sourced navigator.

(k) Review the active alerts on each MFD (if any) and verify that the “AHRS1 GPS –AHRS1 not receiving any GPS information” is not present.

(l) Verify that the flight plan is displayed on each MFD using the flight plan (FPL) function.

(m) On each GPS navigator, enter OBS mode.

(n) Press the CRS button on each PFD and adjust the course using the PFD knob.

(o) Verify that the course to the active waypoint changes as the PFD course pointer is rotated.

(p) Exit OBS mode on each navigator.

(q) On GDU #1, select GPS2 as the NAV source.

(r) On GDU #2, select GPS1 as the NAV source.

(s) Repeat steps (d) thru (p).

(7) Fuel Computer Interface

(a) Apply electrical power to Avionics.

(b) On the DDMP, select the first FUEL page. Enter a fuel load and confirm.

(c) Using the Down arrow and Select button on the DDMP, select the 2nd Fuel management page.

(d) On GNS #1, create a flight plan to a known destination (KMIA, for example). Select an arrival procedure. Select a transition waypoint, if prompted. Ensure that the GNS shows a valid distance and time to destination.

(e) Engines running (for valid fuel flow).

(f) Verify that a valid value is displayed for fuel TO Destination.

(8) VHF NAV Source

(a) Pull NAV2/GPS2 circuit breaker.

(b) Apply electrical power to Avionics.

(c) Select NAV1 as the source on GDU #1.

(d) Tune NAV1 to a localizer frequency.

NOTE: For this check it is not necessary that a valid localizer signal is being received.

(e) Verify that the CDI on the PFD displays LOC1.

(f) Pull NAV1/GPS1 circuit breaker. Close NAV2/GPS2 circuit breaker.

(g) Repeat steps (c) thru (e) for NAV2. Source should indicate LOC2.

(h) Repeat steps (c) thru (e) using GDU #2 and NAV2. Source should indicate LOC2.

(i) Pull NAV2/GPS2 circuit breaker. Close NAV1/GPS1 circuit breaker.

(j) Repeat steps (c) thru (e) using GDU #2 and NAV1. Source should indicate LOC1.

(k) Close NAV2/GPS2 circuit breaker.

(9) VHF COM Operation

(a) Verify side-tone audible from pilot and co-pilot headsets when transmitting with COM1.

(b) Verify transmit audio at remote receiver using COM1 from pilot Mic.

(c) Verify receive audio at pilot headset using COM1.

(d) Pull AUDIO/MKR circuit breaker.

(e) Verify transmit audio at remote receiver using COM1 from pilot Mic.

(f) Verify receive audio at pilot headset using COM1.

(g) Push in AUDIO/MKR circuit breaker.

(h) Verify side-tone audible from pilot and co-pilot headsets when transmitting with COM2 from co-pilot Mic.

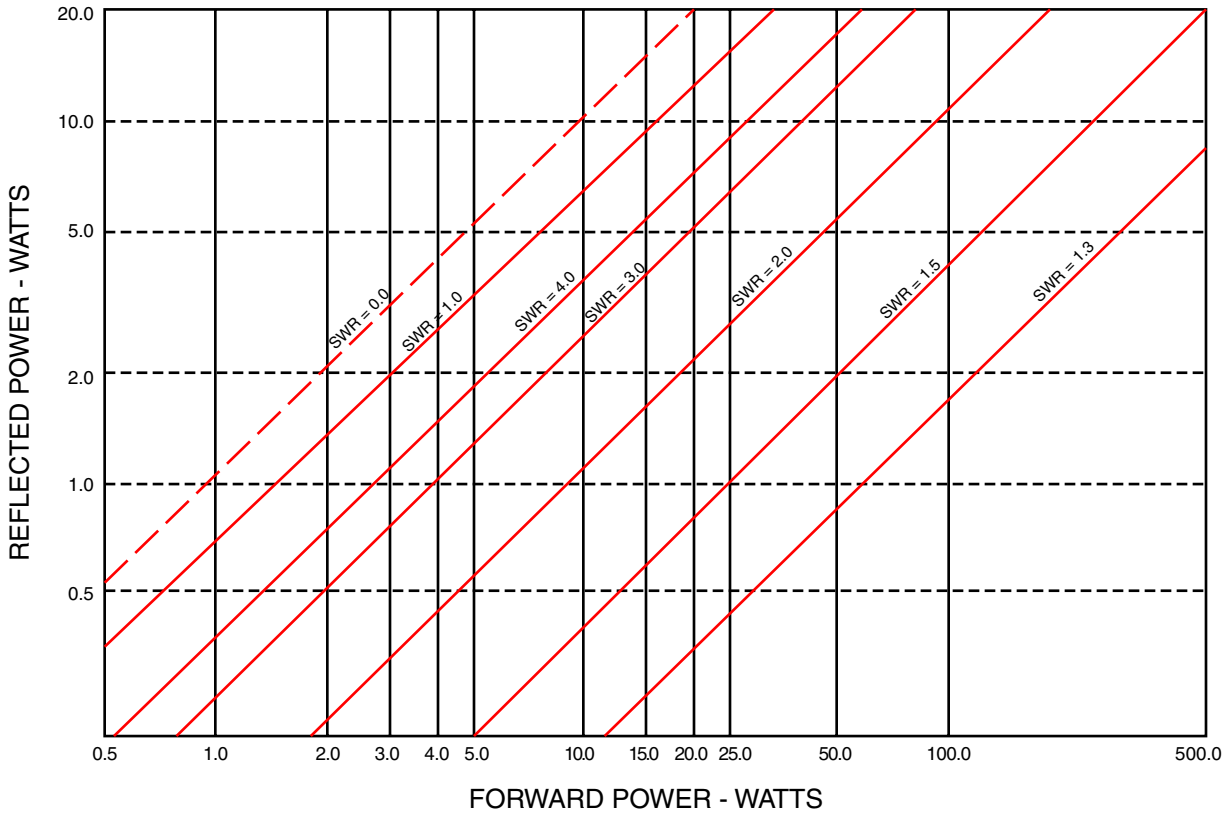
(i) Verify transmit audio at remote receiver using COM2 from co-pilot Mic.

(j) Verify receive audio at co-pilot headset using COM2.

(k) Conduct Voltage Standing Wave Ratio (VSWR) check for COM1 per “Figure 1”.

(l) Conduct VSWR check for COM2 per “Figure 1”.

VSWR CONVERSION CHART



Voltage Standing Wave Radion Conversion
Figure 1

(10) GI-106 Post-Installation Test

(a) Apply electrical power to Avionics, GNS #2 running in configuration mode.
(b) On GNS #2, select VOR/LOC/GS CDI page.
(c) Select LAT CDI to Full Left.
(d) Verify that CDI on GI-106 indicates full scale left.
(e) Select LAT CDI to Full Right.
(f) Verify that CDI on GI-106 indicates full scale right.
(g) Select LAT CDI to Center.
(h) Verify that CDI on GI-106 indicates centered.
(i) Select VERT CDI to Full Up.
(j) Verify that vertical deviation indicator on GI-106 indicates full scale up.
(k) Select VERT CDI to Full Down.
(l) Verify that vertical deviation indicator on GI-106 indicates full scale down.
(m) Select VERT CDI to Center.
(n) Verify that vertical deviation indicator on GI-106 indicates centered.
(o) Select LAT Flag to View.
(p) Verify that lateral flag is in view.
(q) Select LAT Flag to Hide.
(r) Verify that lateral flag is hidden.
(s) Select Vert Flag to View.
(t) Verify that vertical flag is in view.
(u) Select VERT Flag to Hide.
(v) Verify that vertical flag is hidden.
(w) Select TO-FR to FROM.
(x) Verify that FROM indicator is in view.
(y) Select TO-FR to TO.
(z) Verify that TO indicator is in view.
(aa) Select TO-FR to HIDE.
(ab) Verify that the FROM and TO indicators are hidden.

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2. Transponder

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

A. GTX 330

S/N's 3449301 and up

A Garmin GTX 330 Transponder is installed as standard equipment. Maintenance of the GTX 330 is "on condition" only and, with the exception of swapping complete units, should be performed only by a qualified avionics shop in accordance with the GARMIN GTX 330 Maintenance Manual (Garmin P/N - 190-00207-02).

Information provided in this manual is intended solely to aid the removal and installation of the GTX 330 transceiver/display unit, its associated wiring and antenna.

(1) Removal and Installation

See 39-10-00

(2) Post Installation Setup Procedure (PIR-PPS60206, Rev. O / PPS55028, Rev. E.)

Access the Configuration Mode of the unit by depressing and holding the FUNC key while applying power to the unit. Release the FUNC key when the display activates.

The FUNC key sequences forward through the configuration pages. The START/STOP key reverses through the pages, stopping at the Menu page. The CRSR key highlights selectable fields on each page. When a field is highlighted, the 0-9 keys enter numeric data and the 8 or 9 keys move through list selections. Press the CRSR key to accept changes. When a field is highlighted, pressing the FUNC key moves to the next configuration page without saving the changes.

To exit the configuration pages, turn the power off and then turn the unit on again without holding the FUNC key for normal operation.

NOTE: When the unit it turned on for the first time, or an invalid address is recognized, the unit will prompt the user to enter a valid aircraft address. See Mode S Address paragraph, below.

(a) Configuration Menu

The JUMP TO menu page provides the capability to select a configuration mode starting page without having to step through all of the pages. Press the CRSR key and sequence through to the desired section with the 8 and 9 keys. Jump to the selection by pressing the CRSR key again with the desired selection highlighted. The FUNC key steps to the next configuration page, after which the START/STOP key reverses until stopping at the JUMP TO menu page. Following is a list of GTX-330 JUMP TO menu page selections and their descriptions:

SELECTION	DESCRIPTION
DIAGNOSTICS	Jumps to Gray Code Input Page
DISPLAY/AUDIO	Jumps to Audio Volume Page
I/O CONFIG	Jumps to ARINC INPUT #1 Page
ACFT CONFIG	Jumps to Operation Configuration #1 Page

(b) See "Chart 2" on page 345016 thru "Chart 9" on page 345018 for required settings on the configuration pages.

CHART 2
AUDIO MODE AND TRAFFIC INFORMATION PAGES
(TIS Installation Only - No Other Traffic System Installed.)

Function	Selection	Description
VOLUME	MAX	
VOICE	MALE	
MESSAGE (0-5)		Selected Audio Tones and Messages 0= Toggle a continuous tone on and off 1= Attention Tone, precedes voice messages to attract the pilot's attention. 2= "Leaving Altitude", when altitude monitor is active and the altitude deviation is exceeded. 3= "Traffic", when a TIS traffic alert is received. 4= "Time Expired", when the countdown expires. 5= "Traffic Not Available", when TIS service is not available or out or range of an operating TIS MODE S site. 6-9 are not used.
ALTITUDE MONITOR	OFF TONE	When Altitude Pre-Select is installed. When Altitude Pre-Select is not installed.
PAGE CHANGE	ENABLE DISABLE	When G600 System is not installed. When G600 System is installed.
COUNTDOWN TIMER	TONE OFF	When G600 System is not installed. When G600 System is installed.
TRAFFIC MESSAGES	TONE	

CHART 3
DISPLAY MODE/BACKLIGHT, KEY LIGHTING AND CONTRAST PAGES

DISPLAY MODE	AUTO	
LEVEL	75	
BKLT	AUTO	(LVL - Not Selectable)
RSP TIME	4	
MIN	8	
BKLT SRCE	PHOTO (28V W/G600 INSTALL)	
SLOPE	50	
OFFSET	50	
KEY	AUTO	(LVL - Not Selectable)
RSP TIME	4	
MIN	5 (30 W/G600 INSTALL)	
KEY SRCE	28V	
SLOPE	20 (30 W/G600 INSTALL)	
OFFSET	30 (90 W/G600 INSTALL)	
CONTRAST MODE	AUTO	(LVL - Not Selectable)
VFR KEY	ENABLE	

CHART 4
ARINC 429 CONFIGURATION PAGE - With Mechanical Indicators

	GTX 330	
429 INPUT	**SPEED**	**DATA**
CHANNEL 1	HIGH	GPS/FMS
CHANNEL 2	HIGH	OFF
CHANNEL 3	LOW	OFF
CHANNEL 4		OFF
429 OUTPUT		**DATA**
CHANNEL 1		OFF
CHANNEL 2		GARMIN W/TIS GARMIN W/IHAS

CHART 5
ARINC 429 CONFIGURATION PAGE - With Meggitt EFIS

	GTX 330	
429 INPUT	**SPEED**	**DATA**
CHANNEL 1	HIGH	GPS/FMS
CHANNEL 2	HIGH	ADC W/ALT
CHANNEL 3	LOW	OFF
CHANNEL 4		OFF
429 OUTPUT		**DATA**
CHANNEL 1		OFF
CHANNEL 2		GARMIN W/TIS GARMIN W/IHAS

CHART 6
ARINC 429 CONFIGURATION PAGE - With Avidyne Entegra

	GTX 330	
429 INPUT	**SPEED**	**DATA**
CHANNEL 1	LOW	GPS/FMS
CHANNEL 2	HIGH	ADC W/ALT
CHANNEL 3	LOW	OFF
CHANNEL 4		OFF
429 OUTPUT		**DATA**
CHANNEL 1		OFF
CHANNEL 2		“GARMIN W/TIS” “OFF” W/IHAS

CHART 7
ARINC 429 CONFIGURATION PAGE - With G600 System

	GTX 330	
429 INPUT	**SPEED**	**DATA**
CHANNEL 1	LOW	EFIS w/ALT
CHANNEL 2		OFF
CHANNEL 3		OFF
CHANNEL 4		OFF
429 OUTPUT		**DATA**
CHANNEL 1		OFF
CHANNEL 2	HIGH (W/TIS INSTALL) Blank (W/O TIS INSTALL	GARMIN W/TIS (W/TIS INSTALL) OFF (W/O TIS INSTALL

CHART 8
RS-232 INPUT AND OUTPUT

RS-232 CONFIG Page	**GTX**	
232 INPUT	INPUT	OUTPUT
CHANNEL 1	OFF	OFF
CHANNEL 2	OFF	OFF

CHART 9
OPERATION CONFIGURATION

VS RATE	500
FORMAT	FEET
VFR ID	1200
ALTITUDE ALERT DEVIATION	200
SQUAT SWITCH	“YES”
“SENSE”	“LOW”
DELAY TIME	24
AUTO FLIGHT TIMER	MANUAL

(c) On the TEMPERATURE Page, select “NO” for SENSOR INSTALLED.

(d) MODE S Address (ICAO Aircraft Address Code) Entry Page

When the unit is turned on for the first time, or an invalid address is recognized, the unit will prompt the user to enter a valid aircraft address.

NOTE: U.S. registered aircraft ICAO Aircraft Address Code (ADDRESS HEX) is the “N number” and is displayed as such on the ATC-601. On Non-U.S. registered aircraft, verify only that the hexadecimal code displayed on ATC-601 matches the “ADDRESS HEX” code programmed on the GTX 330.

4) U.S. Registered Aircraft

e) To highlight the “U.S. Tail #” address field, press the CRSR key one time.

f) Enter the registration number using the number keys. Press the CRSR key to select the numeric entry field. Press a key repeatedly to scroll through the digit/alpha characters for that entry field.

g) Repeat step b until the complete number is entered.

h) When finished, press the CRSR key to accept the number entry.

i) Press the FUNC key to leave the page, then the START/STOP key to re-access the MODE S Address page.

j) Verify the MODE S Address is correct.

5) Non-U.S. Registered Aircraft

a) For entering Non-U.S. Aircraft Registration, press the CRSR key one time, then 8 or 9 to select ADDRESS HEX.

b) Non-U.S. Aircraft Registry normally supplies a 24 bit binary code, known as the ICAO Aircraft Code for the Mode S address in lieu of directly entering the N# for U.S. registered aircraft.

c) The GTX 330 only accepts a hexadecimal format for the Non-U.S. Registered Aircraft, so the 24-bit binary code must be converted to a hexadecimal format.

d) Convert the 24-bit binary code to hexadecimal as follows:

1] 24-bit binary code: 111001000111111101101101

2] Separate for hex conversion: 1110/0100/0111/1110/1110/1101

3] Apply the hexadecimal values found in “Chart 10” on page 345020.

E	4	7	E	E	D
1110/	0100/	0111/	1110/	1110/	1101

e) Enter the hexadecimal address using the number keys. Press the CRSR key to select the numeric entry field. Press a key repeatedly to scroll through the digit/alpha characters for that entry field.

f) Repeat step e until the complete number is entered.

g) When finished, press the CRSR key to accept the number entry.

h) Press the FUNC key to leave the page, then the START/STOP key to re-access the MODE S Address page.

i) Verify the MODE S Address is correct.

CHART 10
HEXADECIMAL CONVERSION

Binary	Hexadecimal	Decimal	Binary	Hexadecimal	Decimal
/0000	0	0	/1000	8	8
/0001	1	1	/1001	9	9
/0010	2	2	/1010	A	10
/0011	3	3	/1011	B	11
/0100	4	4	/1100	C	12
/0101	5	5	/1101	D	13
/0110	6	6	/1110	E	14
/0111	7	7	/1111	F	15

(e) On the “MODE S Aircraft Type” Page:

Select “<15.5k Lb” for AC TYPE, “<=300 kt” for MAX AIRSPEED, and “1E-5” for GPS INTEGRITY.

(f) On the “Aircraft Size” Page:

Select “<=15 MT” for AC LENGTH TYPE LENGTH, and “<=11.5 MT” for AC WIDTH TYPE WIDTH.

(g) On the “EHS” Page:

Select “ENABLE” for EHS function.

NOTE: Advance through remaining display screens that cannot be changed.

(h) When entries are complete, cycle unit power and check for entries.

B. GTX-33/33 ES

S/N’s 3449459, 3449467-3449508.

A Garmin GTX-33/33 ES Transponder is installed as standard equipment. Maintenance of the GTX-33/33 ES is “on condition” only and, with the exception of swapping complete units, should be performed only by a qualified avionics shop in accordance with the GARMIN appropriate GTX-33/33 ES Maintenance Manual.

The GTX-33/33 ES is an integral part of the Garmin G1000 Integrated Avionics System. Any configuration, set-up, removal and installation information is provided in “GTX 33 Mode S Transponder” on page 34250164.`

C. GTX-3X5R

S/N’s 3449509 and up.

A Garmin GTX-3X5R Transponder is installed as standard equipment. Maintenance of the GTX-3X5R is “on condition” only and, with the exception of swapping complete units, should be performed only by a qualified avionics shop in accordance with the GARMIN appropriate GTX 3X5R Maintenance Manual.

The GTX-3X5R is an integral part of the Garmin G1000 Integrated Avionics System. Any configuration, set-up, removal and installation information is provided in “GTX 3X5R Mode A, C, S Transponder” on page 34250264.`

D. Mode S Transponder Ground Tests

(PIR-PPS60207, Rev. F.)

These tests enable compliance with FAR 91.413 and FAR Part 43 Appendixes E and F. Perform these tests in conjunction with Post Installation Set-up, above.

(1) Requirements

(a) Tests to be accomplished by using a Transponder Ramp Tester capable of testing to requirements stated in FAR Part 43 Appendix F.

(b) Transponder must be properly configured per aircraft setup specification prior to starting this process.

(c) Controllable vacuum source.

(d) During all tests specified herein, the aircraft altimeter shall be set a 29.92 inches of mercury (1013.2 Millibars).

(e) Altitude encoders shall be tested up to and including the maximum operating altitude for that aircraft.

(2) Auto Test Requirements

(a) Perform the AUTO TEST function and verify that MODES A, C, and S PASS.

(b) Record the following results:

1) FREQ: 1090 +/-1 MHz (1089.0 MHz to 1091.0 MHz)

2) ERP: 125 Watts minimum to 500 Watts maximum

3) MTL: -73 dBm +/- 4 (-69 dBm to -77 dBm)

4) DIVERSITY ISOLATION: Greater than 20db (Only on aircraft equipped with diversity transponder.)

(3) Individual Test Requirements

U.S. Registered Aircraft ICAO aircraft address code (address hex) is actually the “N number”. On foreign registered aircraft verify only that the hexadecimal code displayed matches the “address hex” code programmed on the transponder.

(a) SLS Level

NOTE: This can be verified via ATC 601 TEST 4 or IFR 6000 A/C DECDR/SLS.

(b) Test Set interrogates UUT with ATCRBS Modes A and C. It verifies a reply is received when P2 pulse is 9dB less than P1 pulse and no reply is received when P2 pulse amplitude is equal to P1 pulse. Transponder response to Mode 3/A interrogations not to exceed 1% when amplitude of the P2 pulse = P1 pulse and the transponder is interrogated at a pulse repetition rate of 235 per second.

(c) ATCRBS Only All-Call

NOTE: This can be verified via ATC 601 TEST 5 or IFR 6000 A/C F1/F2 SPACE/WIDTH.

Test Set interrogates UUT with ATCRBS Only ALL-CALL (0.8 microsecond P4 pulse) and verifies that Mode S transponders do not reply.

(d) MODE S All-Call Interrogations

NOTE: This can be verified via ATC 601 TEST 6 or IFR 6000 S. ALL-CALL.

Test Set interrogates UUT with Mode A/S All-Call (1.6 microsecond P4 pulse). Address received from Mode S reply is sent in UF4 interrogation and verified with DF4 address.

(e) Invalid Address: Mode S Address

NOTE: This can be verified via ATC 601 TEST 7 or IFR 6000 S REPLY.

Test Set interrogates UUT with MODE S interrogations using correct address and two incorrect addresses (Nominal rate of 50 interrogations per second). These addresses are different from the address determined from Mode S All-Call. The Test Set verifies that no reply is received.

(f) Mode S UF0

NOTE: This can be verified via ATC 601 TEST 9 or IFR 6000 UF0.

Test Set interrogates UUT with Mode S UF0 and verifies that reply has correct altitude (Compared to ATCRBS Mode C), correct address (Compared to DF11) and correct format. UFO AQ bit is “1” for requesting airspeed information in DFO RI field.

(g) Mode S UF4

NOTE: This can be verified via ATC 601 TEST 10 or IFR 6000 UF4.

Test Set interrogates UUT with Mode S UF4 and verifies reply has correct altitude (Compared to ATCRBS Mode C), correct address (Compared to DF11) and correct format.

(h) Mode S UF5

NOTE: This can be verified via ATC 601 TEST 11 or IFR 6000 UF5.

Test Set interrogates UUT with Mode S UF5 and verifies that reply has correct ID Code (Compared to ATCRBS mode A), correct address (Compared to DF11) and correct format.

(i) Mode S UF11

NOTE: This can be verified via ATC 601 TEST 12 or IFR 6000 UF11.

Test Set interrogates UUT with Mode S UF16 and verifies that reply has correct address and format.

(j) Mode S UF16

NOTE: This can be verified via ATC 601 TEST 13 or IFR 6000 UF16.

Test set interrogates UUT with UF16 and verifies that reply has correct altitude (Compared to ATCRBS Mode C), correct address (Compared to DF11) and correct format. UF16 AQ bit is “0” for requesting TCAS capability information in DF16 RI field.

(k) Mode S UF20

NOTE: his can be verified via ATC 601 TEST 14 or IFR 6000 UF20.

Test Set interrogates UUT with Mode S UF20 and verifies that reply (if received) has correct ID code (Compared to ATCRBS Mode A), correct address (Compared to DF11) and correct format.

(l) Mode S UF21

NOTE: This can be verified via ATC 601 TEST 15 or IFR 6000 UF21.

Test Set interrogates UUT with Mode S UF21 and verifies that reply (if received) has correct ID code (Compared to ATCRBS Mode A), correct address (Compared to DF11) and correct format.

(m) Squitter

NOTE: This can be verified via ATC 601 TEST 15 or IFR 6000 S REPLY.

Test Set verifies that squitters are received from UUT every 0.8 to 2.4 seconds.

(n) Radio Reply Frequency

NOTE: This can be verified via ATC 601 TEST 17 or IFR 6000 POWER/FREQ.

Test Set verifies that UUT transmit frequency is 1090 MHz (+/- 1 MHz).

(o) Mode S Diversity Transmission Channel Isolation

NOTE: This can be verified via ATC 601 TEST 18 or IFR 6000 S REPLY. (Test only applicable to aircraft equipped with Diversity Transponder)

Test set verifies that power level difference is ≥ 20dB between “on “ antenna squitters and “off” antenna squitters.

NOTE: To insure ≥ 20dB dynamic range, Diversity Test must be run within 50 ft of UUT Antenna being tested.

(p) MTL Difference (Receiver Sensitivity)

NOTE: This can be verified via ATC 601 TEST 19 or IFR 6000 POWER/FREQ.

Test Set verifies that receiver sensitivity difference between Mode A and Mode C is ≤ 1.0 dB.

(q) Power and MTL

NOTE: This can be verified by ATC 601 Power Test or IFR 6000 POWER/FREQ.

(r) Test Set verifies that UUT power output is between the following parameters:

- 48.5 dBm < UUT ERP < 57.0 dBm
- UUT MTL = -73.0 dBm (+/- 4.0 dB)

NOTE: Power Test must be run at a distance established in Setup procedure. The minimum RF peak output power (ERP on Test Set) is at least 125 watts and maximum RF peak output power does not exceed 500 watts.

(4) Accuracy Test

(a) Set-up

1) Apply power to the Altitude Reporter and the Transponder Test Set.
2) Set primary flight altimeter to 29.92 in hg.
3) Allow a minimum of 15 minutes warm up time.
4) The Transponder Test Set reading should match with flight altimeter reading, not with the static pressure system tester.

(b) Testing

Select ten or more evenly spaced altitude test points between sea level and the altitude designated in 3.6. Increase the vacuum, causing the aircraft altimeter indicated altitude to increase. Approach each test point slowly. Tap the altimeter lightly before each reading to minimize friction effects. If a check point is missed, do not go back, go on to the next test point. For each test point check the aircraft altimeter dial reading at the encoding altimeter/ digitizer transition point. (The transition point is the point at which the altitude encoder readout displayed on the ATC-600A or equivalent test set, changes to the proper code for the test point).

(c) Accuracy Requirements

Accuracy requirements shall be as follows: The altitude encoder shall transition to the selected test point within ±125 ft. at each test point, and within ±87 ft. at no less than 70 percent of the test points.

EXAMPLE: Approaching the 6,000 foot transition point, the aircraft altimeter dial shall read 6,000 ±125 ft. (5,875 to 6,125 ft.) when the transition to the ±87 ft. (5,913 to 6,087 ft.) if the 6,000 foot transition point is one of 70% noted above).

(d) Out of Limits

If either the aircraft altimeter of the aircraft encoder fails the accuracy tests, replace the out-of-limits device and repeat the test.

(e) Post Test Cleanup

After testing is completed, the aircraft transponder switch and battery switch shall be placed in the off position, the test set shall be disconnected. The static system shall be reconnected, and the static system then tested for leaks.

(f) ADS-B Out Functionality Tests

Perform Advisory Circular 20-165 test and verify the following:

1) NACp ≥ 8
2) NIC ≥ 7
3) NACv ≥ 1
4) Verify ADS-B barometer altitude matches transponder barometer altitude (Alt. Air= 0).

(5) Test Complete

Restore airplane to operational configuration and secure.

3. ADF

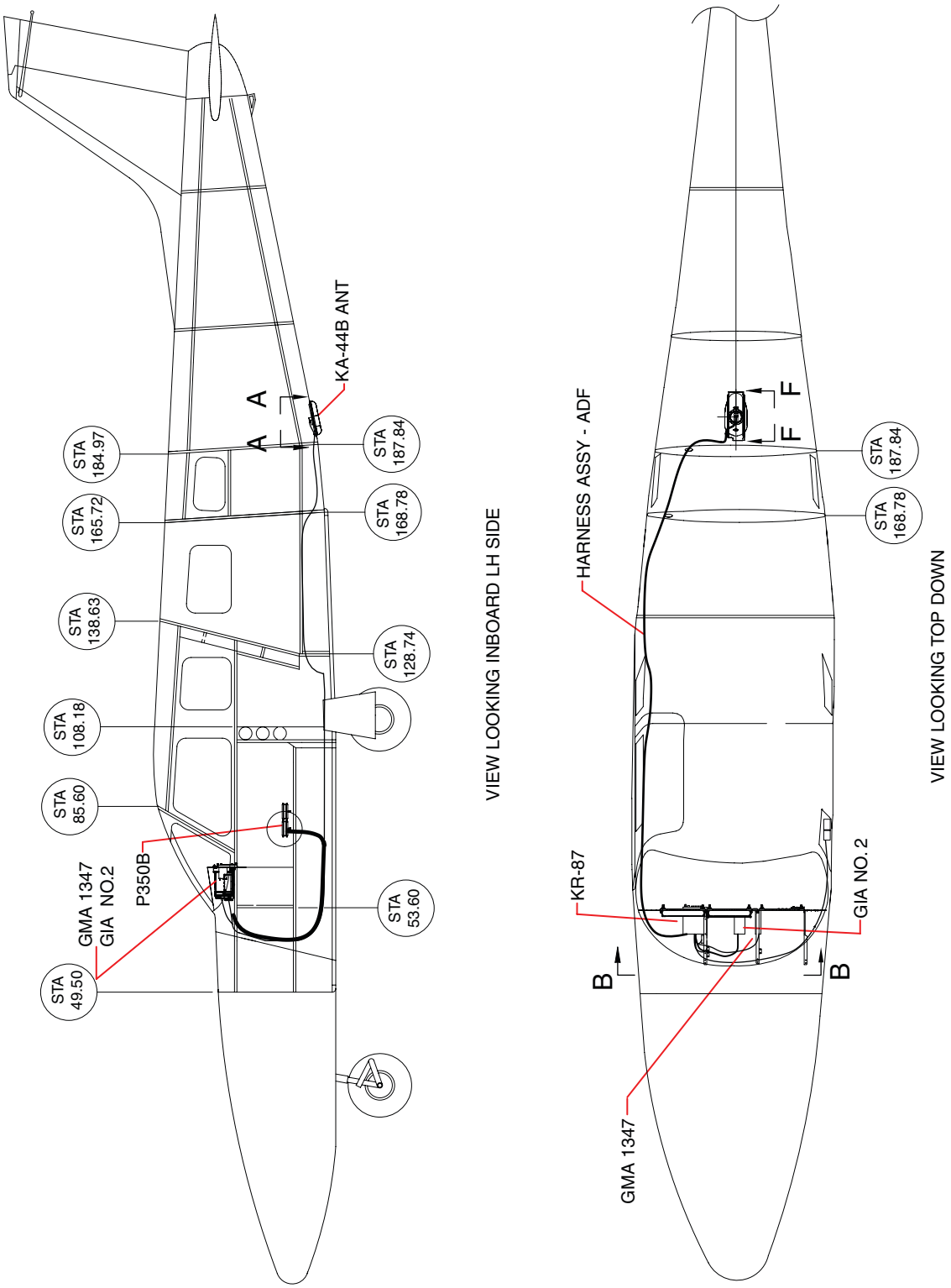
A. Honeywell KR-87

S/N's 3449459, 3449467–3449508. See "Figure 2" on page 345025.

B. Becker ADF 3500

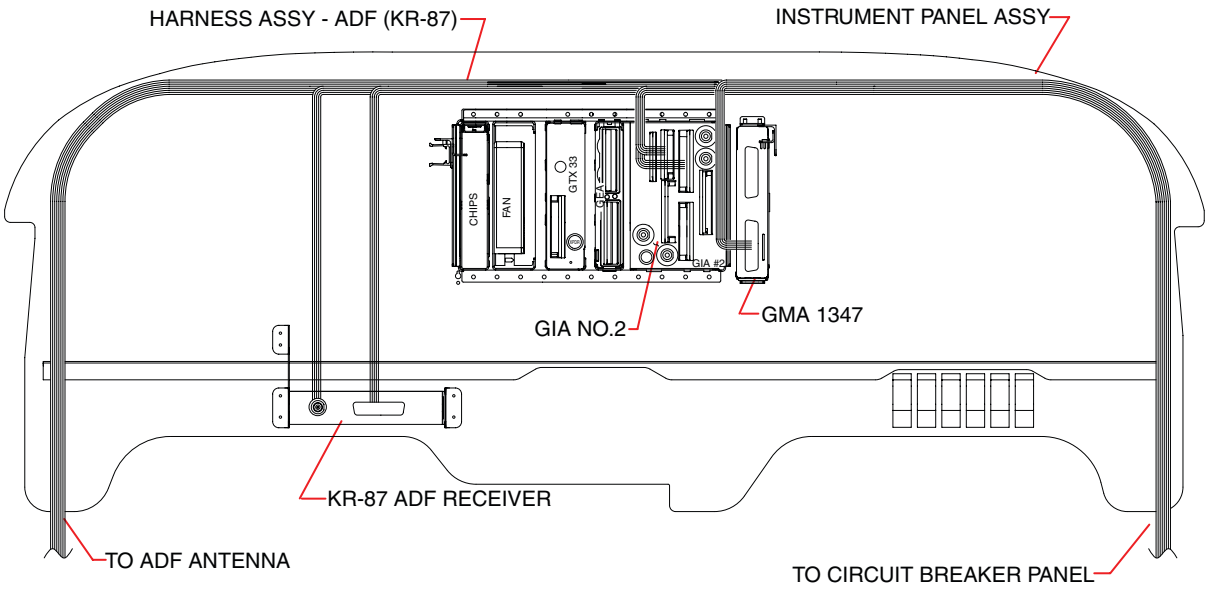
S/N's 3449509 and up. See "Figure 3" on page 345027.

107262 NEW 2.0

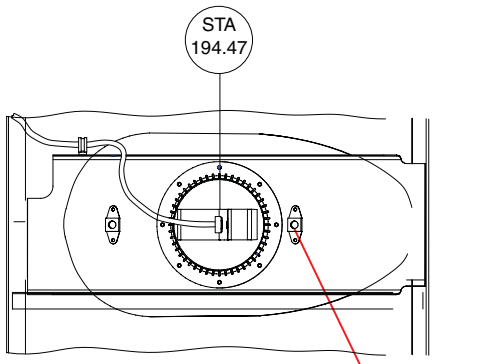


KR-87 ADF Installation
Figure 2 (Sheet 1 of 2)

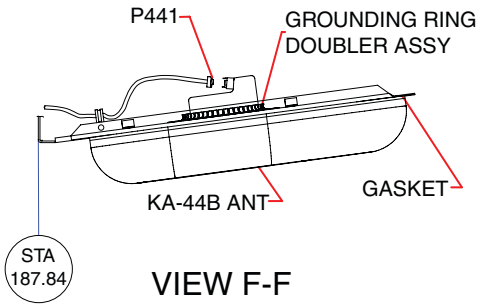
107262 NEW 3.0



VIEW B-B
INSTRUMENT PANEL LOOKING AFT



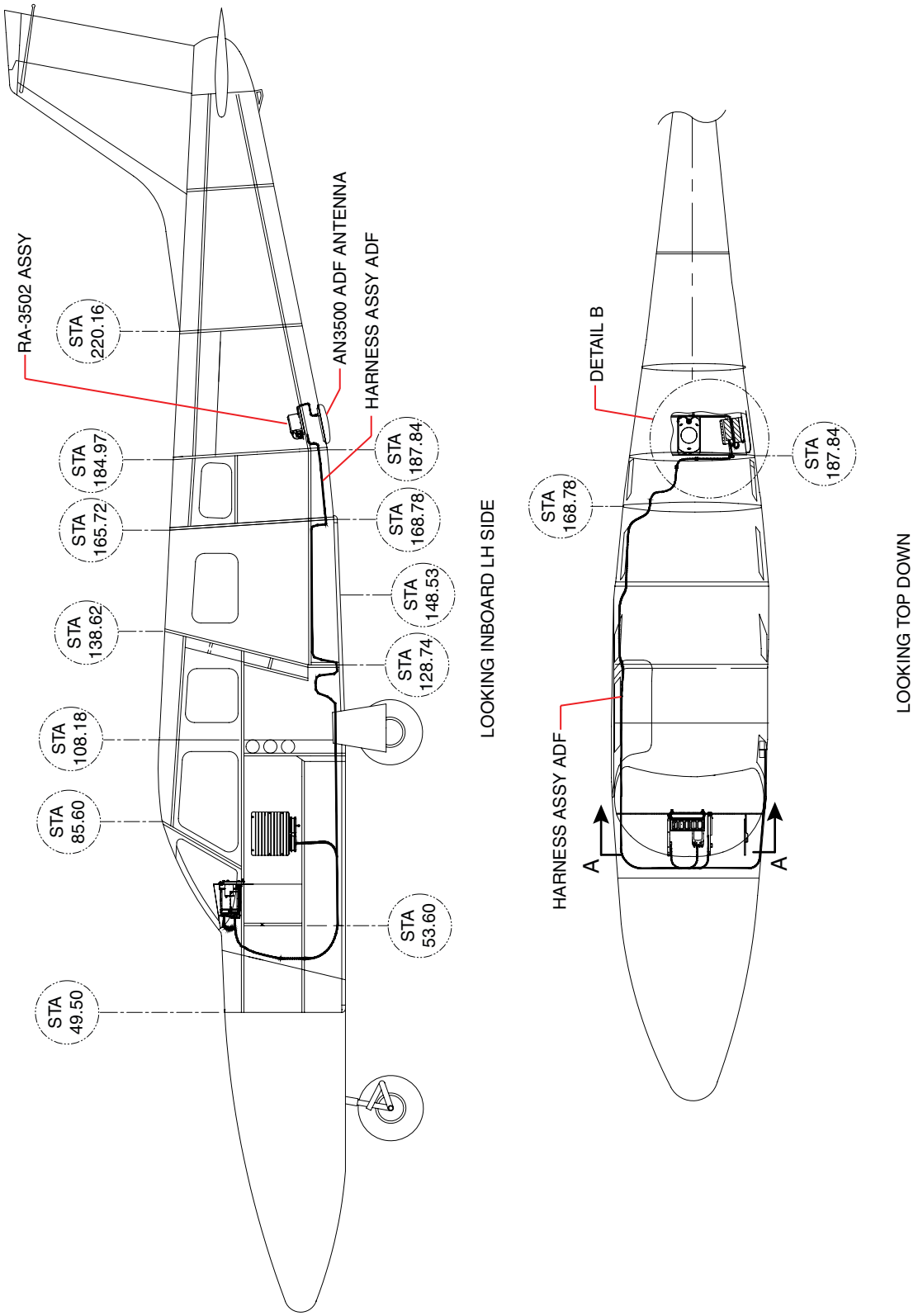
VIEW A-A
FWD



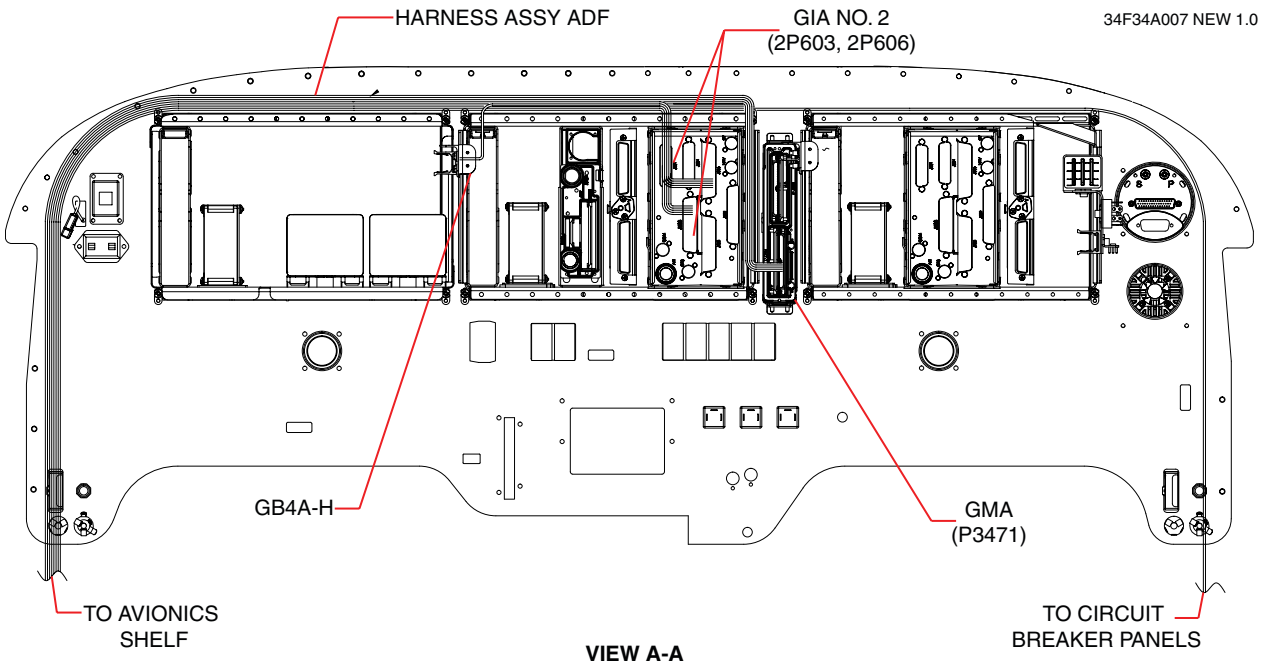
VIEW F-F
FWD

KR-87 ADF Installation
Figure 2 (Sheet 2 of 2)

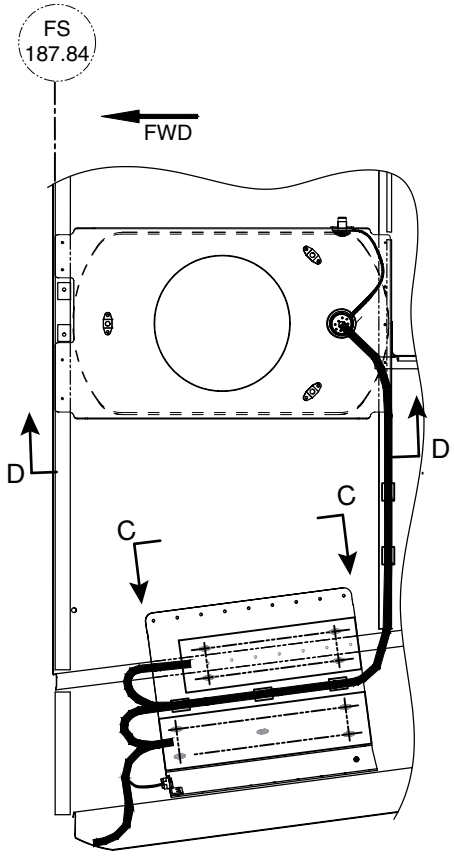
34F34A007 NEW 2.0



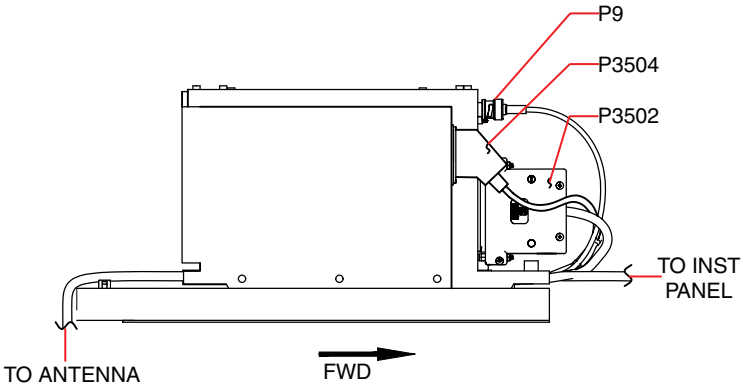
Becker ADF 3500 Installation
Figure 3 (Sheet 1 of 2)



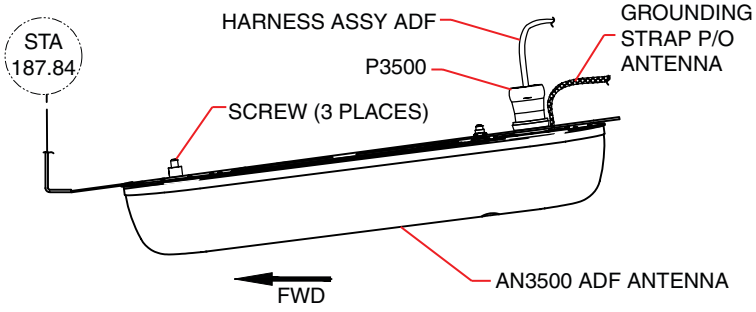
VIEW A-A



DETAIL B



VIEW C-C
LHS LOOKING OUTBD
(MOUNTING SURFACE SLOPE OMITTED FOR CLARITY)



VIEW D-D

Becker ADF 3500 Installation
Figure 3 (Sheet 2 of 2)

WIRING

Airplanes equipped with Garmin GFC 700 Automatic Flight Control Systems have additional EMI shielding on some harness assemblies. If repair is required, see EMI Shielding in 20-97-00.

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CHAPTER

35

OXYGEN

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CHAPTER 35

LIST OF EFFECTIVE PAGES

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	2	Sep 30/18			
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	2	Sep 30/18			
35-10-00	1	Oct 10/23			
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CHAPTER 35 - OXYGEN

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CREW/PASSENGER SYSTEMS

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

The following information is provided as supplemental information for the servicing of the oxygen systems. Major repairs to the oxygen system should be accomplished by an approved shop.

When refilling any oxygen cylinder make sure to use only aviation breathing oxygen as specified in MIL-PRF-27210. The moisture content of aviation oxygen cannot exceed 0.005 milligrams of water vapor per liter of gas at 70°F (21°C) and 29.92 inches of mercury (760 mm Hg.).

1. Description and Operation

WARNING: DO NOT USE GREASE OR ANY TYPE OF GREASE FITTING ON ANY OXYGEN SYSTEM. WHEN WORKING WITH AN OXYGEN SYSTEM MAKE SURE HANDS, CLOTHING, TOOLS, AND IMMEDIATE AREA ARE FREE OF GREASE.

A fixed oxygen system is available. The major components for the fixed oxygen system are manufactured by Avox Systems (previously Scott Aviation). Accordingly, contact Avox Systems (See Supplementary Publications in the Introduction), as well as your Piper Dealer, for any required information not covered herein.

The fixed oxygen system cylinder is installed in the left side of the nose section aft of the closeout panel at F.S 31.04.

The ON-OFF push-pull control knob and pressure gauge are installed on the pilot's instrument panel below, and slightly to the right of, the control wheel. See "Figure 1" on page 35102.

NOTE: The oxygen pressure gauge on the Seneca V G1000 is located on a bracket next to the fill valve attached to the cylinder.

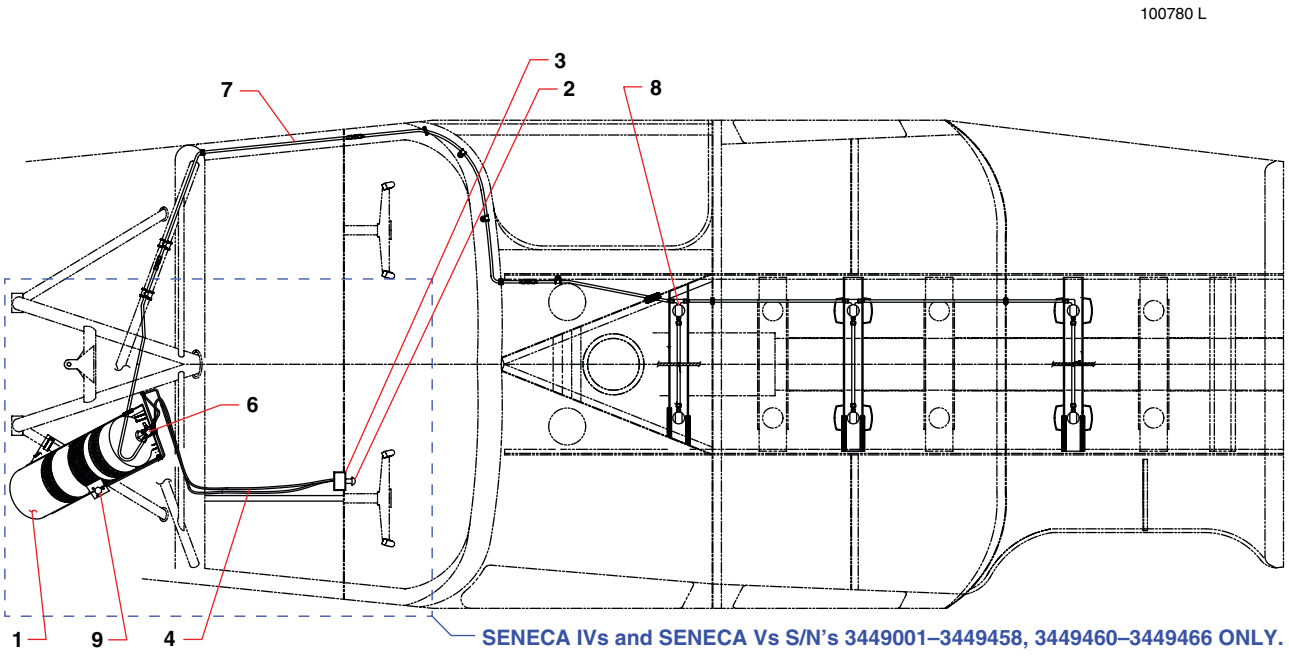
The low pressure feed for the outlets is arranged with the tank feed line entering the right side of the cabin at approximately F. S. 53.6, from where it is routed behind the right front window trim cover, windshield trim cover and overhead cover panel, and attached to tee fittings on the right side of each outlet from where the left outlets are also fed. See "Figure 1" on page 35102.

2. Troubleshooting

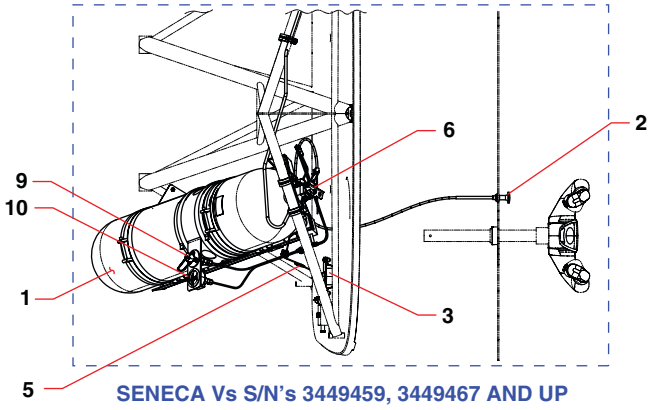
See Chart 1.

CHART 1
TROUBLESHOOTING OXYGEN SYSTEM

Trouble	Cause	Remedy
No indication of pressure on pressure gauge.	Cylinder empty or leak in system has exhausted pressure.	Purge, charge, and check system for leaks.
	Pressure gauge or regulator defective.	Replace gauge.
Pressure indication normal but no oxygen flowing.	Oxygen cylinder regulator assembly defective.	Remove tank and have regulator removed.
Offensive odors in oxygen.	Cylinder pressure below 50 psi. Foreign matter has entered the system during previous servicing.	Purge the oxygen system.



1. OXYGEN BOTTLE
2. ON - OFF CONTROL
3. PRESSURE GAUGE (SENECA IVs and SENECA Vs S/N's 3449001–3449458, 3449460–3449466 ONLY)
   TRANSDUCER (SENECA Vs S/N's 3449459, 3449467 AND UP)
4. HIGH PRESSURE OXYGEN LINE
5. HIGH PRESSURE CAPILARY LINE
6. REGULATOR VALVE
7. LOW PRESSURE OXYGEN LINE
8. OXYGEN OUTLETS (3 PAIRS)
9. EXTERNAL FILLER VALVE
10. FILL VALVE PRESSURE GAUGE (SENECA V WITH G1000/G1000 NXi ONLY)



Fixed Oxygen System Installation
Figure 1

3. Precautions

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

CAUTION: DO NOT ATTEMPT TO TIGHTEN ANY CONNECTIONS WHILE THE SYSTEM IS CHARGED.

CAUTION: BOTTLES WHICH HAVE BEEN EVACUATED TO 200 PSI FOR A SIGNIFICANT LENGTH OF TIME, OR THOSE THAT DO NOT PRODUCE AN AUDIBLE HISSING SOUND WHEN THE VALVE IS CRACKED, SHOULD BE REMOVED AND HYDROSTATICALLY TESTED. IF EITHER OF THESE CONDITIONS HAS EXISTED FOR A SIGNIFICANT LENGTH OF TIME IT IS ALSO RECOMMENDED THAT THE SYSTEM BE PURGED.

CAUTION: MAKE SURE THERE IS NO OIL, GREASE, HYDRAULIC FLUID, OR FUEL IN THE VICINITY OF ANY FITTINGS BEING SERVICED.

CAUTION: DO NOT USE THREAD LUBRICANTS OF ANY KIND. USE TEFLON TAPE (3M NO. 48) ON TAPERED PIPE THREADS, WITHOUT TAPE EXTENDING BEYOND THE FIRST THREAD. REFER TO AFFECTIVE INFORMATION IN THIS CHAPTER.

CAUTION: BEFORE WORKING WITH THE SYSTEM, MAKE SURE AIRCRAFT IS ELECTRICALLY GROUNDED AND YOUR HANDS TOOLS, AND CLOTHES ARE FREE OF OIL, GREASE AND DIRT.

4. Inspection and Maintenance

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

Due to the nature of the process used to test compressed gas tanks, servicing and hydrostatic tests must be conducted by a DOT or manufacturer (Avox Systems previously Scott Aviation) approved shop. The following material gives recommended inspection and maintenance information for the various parts of the oxygen systems.

NOTE: Oxygen cylinders are identified by the US ICC or DOT (or Transport Canada) identification stamped on the cylinder. The month and year of the last test is stamped beneath the ICC/DOT/ Transport Canada identification.

On Seneca IV models, the standard weight cylinder (i.e., ICC or DOT "3AA") must be hydrostatically tested every five (5) years, or if marked with a five pointed star every ten (10) years, (or as specified in the latest applicable national regulations).

On Seneca V models, the lightweight cylinder of composite construction (DOT E 8162 or SP 8162 or Transport Canda SU 4237-127 or SU 9209), must be hydrostatically tested every 5 years (or as specified in the latest applicable national regulations), and the service life may not exceed 15 years.

A. Check the outlets for leakage both in the use and non-use condition and for leakage around an inserted connector. For leak testing information, refer to the appropriate subject in this chapter.

B. Check the high pressure gauge for accuracy by comparing its indicated pressure with that of a gauge of known accuracy connected to the fill port.

C. Inspect tank for dents, bulges, corrosion, and major strap chaffing marks. Should any of these problems exist, the tank should be removed and hydrostatically tested.

D. System Flow Check

(PIR-PPS60087, Rev. Y.)

WARNING: DO NOT OPERATE ANY ELECTRICAL EQUIPMENT DURING THE FLOW CHECK. EXCESS OXYGEN FLOWS INTO THE AIRFRAME AND SPARK IGNITION IS POSSIBLE. OBSERVE ALL SAFETY PRECAUTIONS RELATED TO OXYGEN HANDLING.

An operational check of the regulator can be accomplished as follows: (Refer to “Figure 1” on page 35102 and “Figure 2” on page 35105.)

(1) Using the test pressure gauge (or equivilent) shown in “Figure 2” on page 35105, connect the apparatus to the pilot’s outlet in the overhead panel.

(2) Using the flow test apparatus (or equivilent) shown in “Figure 2” on page 35105, connect the flow apparatus to the copilot’s outlet.

(3) Connect additional flow test apparatus (or equivilent) to each of the other outlets and pull the oxygen control knob to the ON position. The pressure and flow at sea level should be 55 to 80 psi and 3.3 to 5.3 liters per minute respectively.

(4) Replace the test pressure gauge (or equivilent) at the pilot’s outlet with a flow test apparatus (or equivilent) and check the pilot’s outlet for flow. Similarly, exchange the flow test apparatus (or equivilent) at each of the other outlets with the pressure test apparatus (or equivilent) and check pressure at each outlet.

E. Check airframe logbook for last maintenance on oxygen system and perform as required per “Chart 2” on page 35105.

F. Test the oxygen for odor. Pure oxygen is odorless and tasteless. Any system having a significant odor present in the gas should be purged and the bottle replaced or removed and purged.

G. Any fittings, connectors, and tubes which have imperfect threads, pitted or disfigured cones, or other damage should be replaced.

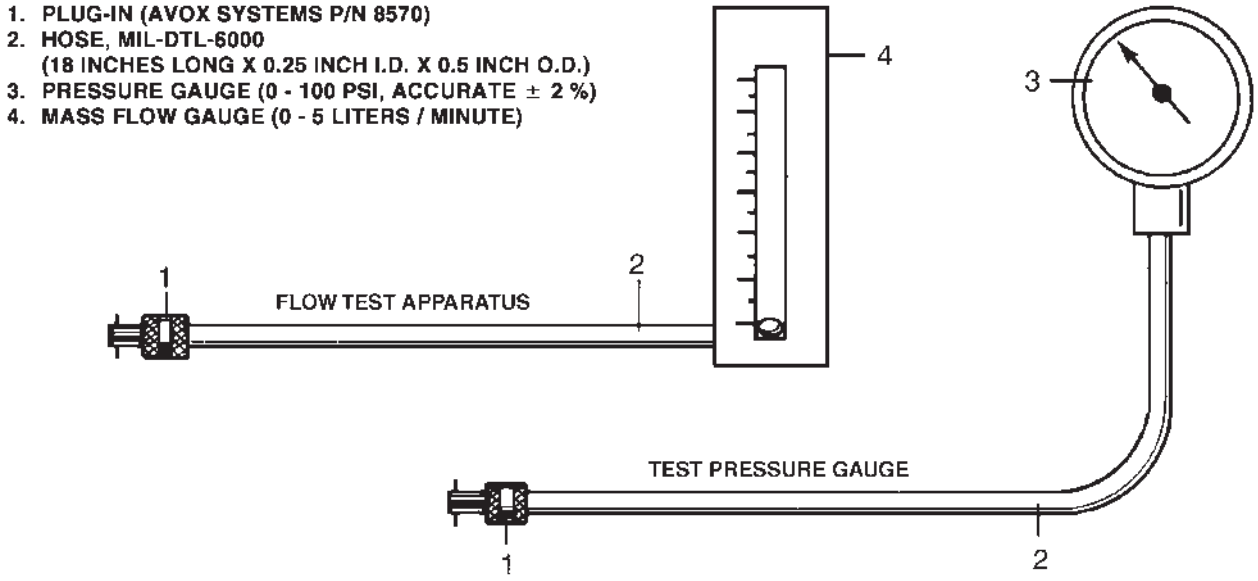
CAUTION: OXYGEN TUBES MUST NOT BE CLAMPED TO, OR SUPPORTED BY ELECTRICAL WIRE BUNDLES, HYDRAULIC, PNEUMATIC OR OTHER LINES.

H. Check plumbing for kinking, cracks, gouges, dents, deep scratches, or other damage. Replace as necessary.

CHART 2
FIXED OXYGEN SYSTEM COMPONENT LIFE LIMITS

Component	Inspection	Overhaul
Cylinder	Each 90 Days [1]	Each 5 Years (Sen V)
		Each 5 or 10 [2] Years (Sen IV)
Regulator	On Condition / Each Use [3]	Each 5 Years
Pressure Gauge	On Condition / Each Use [3]	Replace on Condition
High Pressure Lines	On Condition / Each Use [3]	Replace on Condition
Low Pressure Lines	On Condition / Each Use [3]	Replace on Condition
Outlets	On Condition / Each Use [3]	Each 5 Years [4]
External Recharge Valve	On Condition / Each Use [3]	Replace on Condition [5]
Masks	On Condition / Each Use [3]	Each 5 Years

1. Visual inspection for dents, bulges, corrosion, or chafing.
2. 3AA cylinders marked with a five pointed star are overhauled (hydrostatically tested) each ten years.
3. Visual inspection in the normal course of use.
4. On condition, replace the rubber components in the assembly or replace assembly.
5. If the screen in front of valve is dirty, replace valve. Valve replacement is recommended every 5 years.



Oxygen System Test Apparatus
Figure 2

I. Oxygen Lines Clearance Check

(PIR-PPS60087, Rev. Y.)

Make sure to check the oxygen lines for proper clearance as follows: (Refer to “Figure 3” on page 35107.)

(1) Two inch minimum between oxygen tubes and all flexible moving parts of the aircraft (flexible control cables, etc.). If enough space cannot be attained, protection from abrasion must be provided.

(2) At least 1/2 inch minimum between oxygen tubes and all rigid moving parts of the aircraft such as levers and rigid control rods.

(3) Six inch minimum separation between oxygen tubes and hydraulic, fuel and electrical system lines and components.

NOTE: When the six inch requirement cannot be complied with, one inch is allowed as long as electrical cables and other lines are supported at least every two inches; and, the oxygen tube(s) is protected by rubber neoprene hose fastened in place with cable ties at the location the specific item crosses or is near the oxygen tube(s). If an item is near the oxygen tube for a certain distance the oxygen tube for that distance must be covered. (See “Figure 3” on page 35107, Sketch C.)

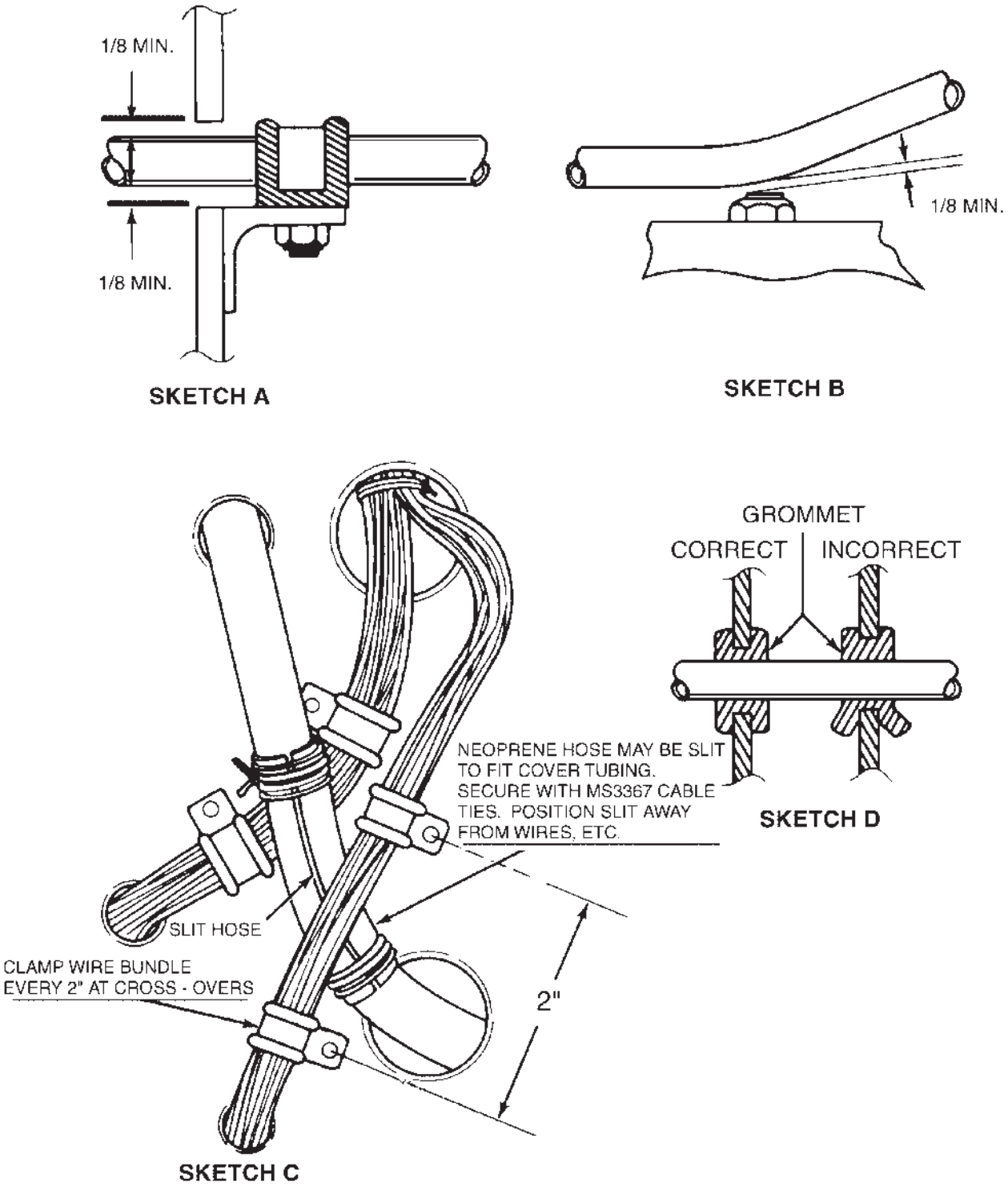
(4) A minimum of 1/8 inch between tubing and structure adjoining the supporting clamp as shown in “Figure 3” on page 35107, Sketch A.

(5) Where a tube passes through a grommet, the tube must not bear on the grommet in any way that might cause cutting of the grommet in service as shown in “Figure 3” on page 35107, Sketch D.

(6) While in service, items may receive vibrations causing them to come in contact with other parts of the aircraft. With this in mind, low pressure tubing that is supported well enough to prevent relative motion must have at least a minimum clearance of 1/8 inch from a projection (bolt, nut, etc). Low pressure tubing that cannot be supported well enough to prevent motion must have a minimum clearance of 1/8 inch allowed after the maximum travel of the tube. High pressure lines are affected similarly but require 1/2 inch minimum clearances. (See “Figure 3” on page 35107, Sketch B.)

J. Perform any other required maintenance as directed in AC43.13-1, latest revision, Chapter 8.

K. Clean components as necessary per Oxygen System Components, “Cleaning and Purging” on page 351011.



Oxygen Tubing Installation
Figure 3

5. Swagelok Fitting Installation

See “Figure 4” on page 35109.

NOTE: The high pressure line fitting at the regulator should be tightened until it bottoms. Make sure to use teflon tape on all male pipe threads.

A. For swagelok fittings not preswaged or for in-aircraft installation, proceed as follows:

(1) Turn the fitting nut onto the fitting finger tight and insert the tube until it bottoms firmly on the shoulder in the fitting.

(2) Tighten the nut with a wrench until the tube will not turn by hand.

(3) Mark the nut at the six o’clock position.

(4) Hold the fitting body steady with a backup wrench and tighten as follows:

(a) On tubing with a diameter bigger than 3/16 inch, tighten 1 1/4 turns (to the nine o’clock position).

(b) On tubing of 1/16, 1/8, and 3/16 inch diameter, tighten only 3/4 turn.

(5) If nut and tube must be disconnected from the fitting, reconnect by seating the tube on the shoulder of the fitting and tightening the nut finger tight. Follow up by tightening the nut with a wrench, one quarter turn (if absolutely necessary the original 1 1/4 or 3/4 tight position) and then snug with wrench.

B. Preswaged swagelok fittings are fabricated and installed as follows:

(1) Assemble the nut and ferrules finger tight on the preswaging tool and insert the tube until it firmly bottoms on the shoulder in the tool. The preswaging tool can be attained from Swagelok Company.

(2) Tighten the nut on the fitting just enough that the tube within the fitting will not turn by hand.

(3) With a wrench, tighten the nut as follows:

(a) On tubing with diameters over 3/16 inch, tighten 1 1/4 turns.

(b) On tubing with 1/16, 1/8, or 3/16 inch diameter, tighten 3/4 of a turn.

(4) Unscrew the nut to release the ferrule-tube assembly from the tool.

(5) The assembly is installed on the fitting as follows:

(a) Slide tube in fitting until it bottoms, turn nut to finger tight position and tighten one quarter turn with wrench.

(b) Snug slightly with wrench.

6. Teflon Tape Thread Sealant

All male pipe (tapered) threads of the oxygen system should be sealed with 3M No. 48 teflon tape. Teflon tape should not be used on straight threads. Do not use any other lubricants in place of the teflon or on any other threads.

A. Wrap tape on the threads, starting with those farthest from the opening, in the direction of the thread spiral. Circle the threads, making sure that each side of the tape has a slight overlap.

B. Wrap the tape such that it does not extend beyond the last thread on the fitting at the opening. The tape should then be pulled till it separates. Do not cut tape, it will not stick properly.

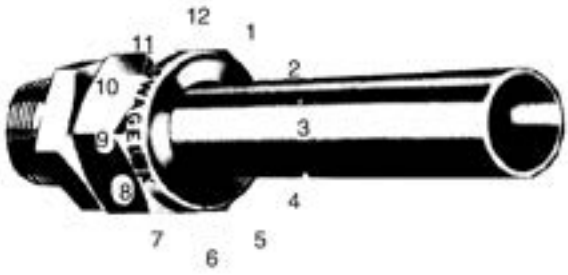


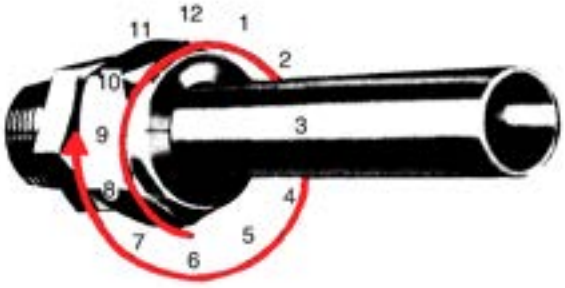
STEP 1

TURN THE FITTING NUT ONTO THE FITTING FINGER TIGHT AND INSERT THE TUBE UNTILL IT BOTTOMS FIRMLY ON THE SHOULDER IN THE FITTING

STEP 2

MARK THE NUT AT THE SIX O' CLOCK POSITION





STEP 3

HOLD THE FITTING WITH A WRENCH AND TIGHTEN THE FITTING NUT AS FOLLOWS:

A. TUBING WITH A DIAMETER GREATER THAN 3/16 INCH SHALL BE TIGHTENED 1 - 1/4 TURNS (THE NINE O' CLOCK POSITION)

B. TUBING WITH A DIAMETER OF 1/16, 1/8, OR 3/16 INCH SHALL BE TIGHTENED ONLY 3/4 TURN.

Swagelok Fitting Installation
Figure 4

7. Leak Test

(PIR-PPS60087, Rev. Y.)

Leak-Tec Formula #16-OX is recommended for leak testing. Refer to 91-10-00, Consumable Materials charts.

A. Remove the royalite covers in the baggage compartment and, with oxygen system turned off, disconnect the low pressure supply line and connect it to a regulated cylinder charged with dry nitrogen.

NOTE: Whenever a leak check is performed, all fitting connections as well as other questionable areas, should be inspected.

B. Apply the leak detector solution to the test surface and watch for indication of leakage.

C. Large leaks will produce bubbles immediately, but small leaks will form a white foam in 5 to 60 seconds.

D. With outlets vacated of masks, connect a test pressure gauge (“Figure 2” on page 35105) to the copilot’s outlet (“Figure 1” on page 35102) as described in Inspection and Maintenance, above.

E. Adjust the regulator on the dry nitrogen cylinder for 100 psi and check for leakage at the outlets.

F. Correct any leaks and wipe off excess leak detector solution.

G. Close the valve on the nitrogen gas tank and insert a flow test apparatus (or equivilent) (“Figure 2” on page 35105) to relieve system pressure.

H. Disconnect pressure test gauge, flow test apparatus, and nitrogen tank.

I. If the oxygen cylinder is not to be hooked up or installed immediately, cap and cover the exposed fittings with new clean plastic bags. Temporarily support lines as needed to prevent damage. Make sure caps and coverings are as clean as possible.

8. Oxygen System Components

Keeping in mind the effect of compressed oxygen on materials, oxygen system components must be handled carefully. Ports on regulators, indicators, and other opened components must also be kept capped or plugged to prevent ingestion of foreign material. Adjustments or modifications should only be initiated under the auspices of the FAA, Piper, or Avox Systems (previously Scott Aviation).

NOTE: See required component “Inspection and Maintenance” on page 35103. Component life limits are listed in “Chart 2” on page 35105.

A. Cleaning and Purging

CAUTION: CARE MUST BE EXERCISED TO PREVENT CONTAMINATION OF COMPONENTS BY OIL, GREASE, WATER, OR FOREIGN MATTER. COMPRESSED AIR USED IN CLEANING AND FLUSHING TUBES MUST BE CLEAN, DRY, FILTERED (OIL FREE) AIR ONLY.

Three methods are recommended for cleaning oxygen system components:

(1) Method I
   (a) Vapor degrease part(s) with trichlorethylene.
   (b) Blow part(s) dry with a stream of compressed air or dry nitrogen. Refer to previous caution.

(2) Method II
   (a) For tubing, flush with naphtha per specification TT-N-95.
   (b) Blow clean and dry off all solvent with clean, dry, filtered air. Refer to previous caution.
   (c) Flush with isopropyl alcohol.
   (d) Rinse thoroughly with fresh water.
   (e) Dry with air as described in previous caution or by heating at a temperature of 250° to 300°F for one-half hour.

      NOTE: Solvents can be reused provided they do not become badly contaminated with oil. This condition can be determined by thoroughly evaporating 100 millimeters of the liquid in a glass dish of a determined weight. Evaporation may be accomplished by heating the dish at 200°F (93°C) for one-half hour. If after evaporation and cool down, the residue exceeds 100 milligrams in weight, the solvent cannot be used for this purpose.

(3) Method III
   (a) Flush with hot inhibited alkaline cleaner until free from oil and grease.
   (b) Rinse thoroughly with fresh water.
   (c) Dry thoroughly with a stream of clean air as described in the previous caution or by heating 250°F to 300°F (121°C to 149°C) for one-half hour minimum.

      CAUTION: DO NOT USE ADHESIVE TAPE FOR ATTACHING OR SECURING PROTECTIVE COVERINGS ON OXYGEN COMPONENTS. USE WAXED LACING TWINE OR TIE RAPS.

(4) After cleaning, all tubing must be protected by caps, plugs and/or plastic bags.

(5) Before installation, make sure fitting, tube, and fixture threads are in good condition and that the cones do not exhibit pitting or disfigurement.

B. Oxygen Cylinder / Regulator Assembly

The fixed oxygen system cylinder / regulator assembly is installed in the left side of the nose section, aft of the closeout panel at F.S. 31.04. See “Figure 5” on page 351013 and “Figure 5” on page 351013.

(1) Removal

(a) From inside of of forward baggage compartment:

1) Remove hydraulic reservoir cover located in rear top center of compartment by removing four attachment screws.

NOTE: Two of these screws also secure closeout panel.

2) Remove left rear closeout panel by removing 13 remaining attachment screws.

(b) Disconnect the low pressure ON–OFF control cable from the lever on the regulator by removing screw from cable support bracket and cotter pin attaching cable to regulator lever.

CAUTION: OPENING ON–OFF CONTROL VALVE DURING REMOVAL OF OXYGEN LOW PRESSURE LINE FROM CYLINDER WILL RESULT IN AN UNCHECKED FLOW OF OXYGEN INTO BAGGAGE COMPARTMENT UNTIL VALVE IS CLOSED.

(c) Safety valve lever on regulator in the OFF position.

(d) Disconnect low pressure line from the regulator. Cap line immediately after removal.

(e) Disconnect high pressure line to fill valve and gauge from the regulator. Cap line immediately after removal.

NOTE: Continuous pressure is applied to high pressure line until it is disconnected from the regulator. A check valve in the regulator will close when high pressure line is disconnected. Closing of this valve is frequently accompanied by a loud popping sound.

(f) If installed, disconnect high pressure (capillary) line to the transducer from the regulator. Cap line immediately after removal.

(g) Remove two fill valve clamps and two cylinder hold down clamps.

(h) Remove cylinder / regulator assembly from airplane.

(2) Installation

(a) Position cylinder / regulator assembly in airplane as shown in “Figure 5” on page 351013 or “Figure 5” on page 351013, as appropriate. Check to be sure that regulator valve lever is free to move and does not contact surrounding area.

(b) Install two cylinder hold down clamps.

(c) Install two fill valve hold down clamps.

(d) Connect high pressure line from fill valve and gauge to regulator. Insert tubing into fitting until ferrule seats in fitting. Tighten the nut by hand and then one quarter turn with a wrench. If fitting is relatively new the nut might be turned 3/4 of a turn. Follow up by snugging the nut slightly with a wrench.

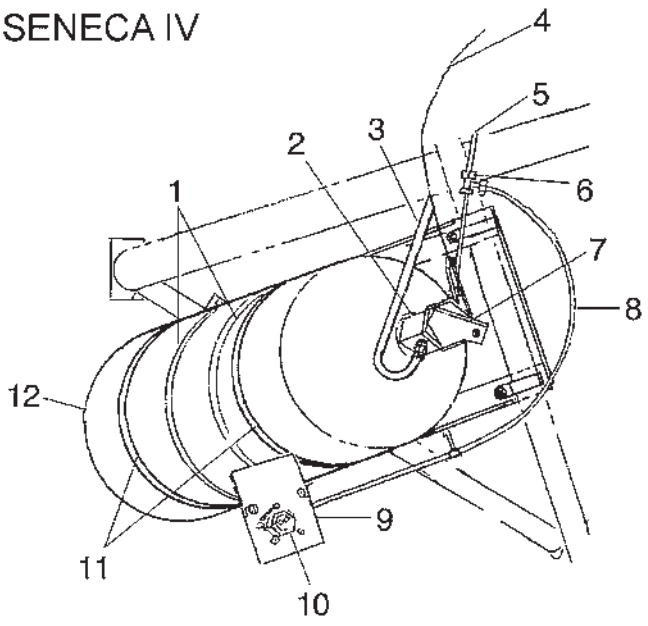
(e) If installed, connect high pressure (capillary) line from transducer to the regulator. Insert tubing into fitting until ferrule seats in fitting. Tighten the nut by hand and then one quarter turn with a wrench. If fitting is relatively new the nut might be turned 3/4 of a turn. Follow up by snugging the nut slightly with a wrench.

(f) Connect low pressure line to regulator. Insert tubing into fitting until ferrule seats in fitting. Tighten the nut by hand and then one quarter turn with a wrench. If fitting is relatively new the nut might be turned 3/4 of a turn. Follow up by snugging the nut slightly with a wrench.

(g) Remove safety from regulator valve lever. Check that lever remains in OFF position.

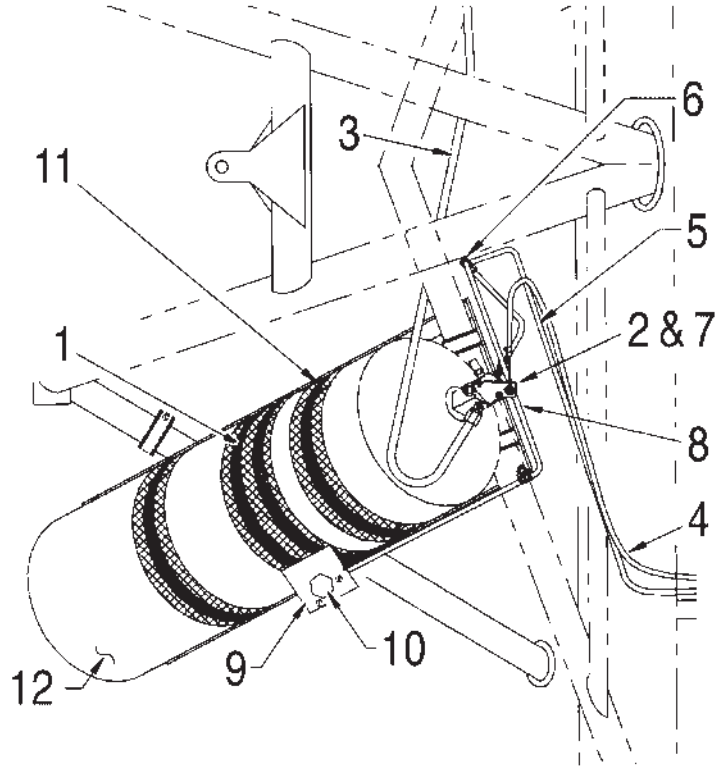
SENECA IV

87441 E



1. FILL VALVE CLAMPS
2. REGULATOR VALVE
3. DISTRIBUTION LINE (LP)
4. ON - OFF VALVE CABLE
5. GAUGE LINE (HP)
6. HIGH PRESSURE LINE "T"
7. ON - OFF LEVER
8. FILL LINE (HP)
9. FILL VALVE MOUNTING BRACKET
10. FILL VALVE
11. CYLINDER CLAMPS
12. OXYGEN CYLINDER

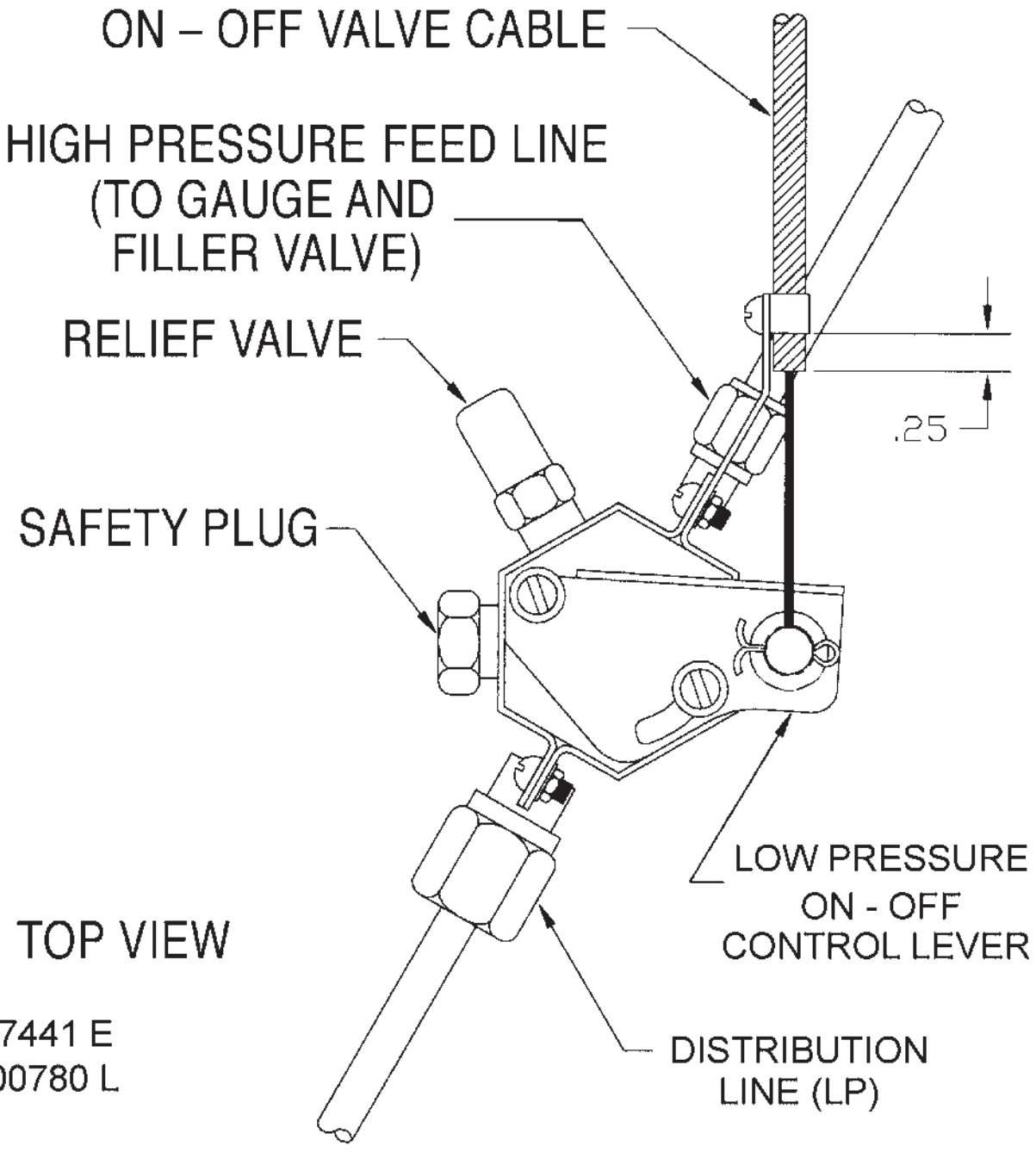
SENECA V



HP = HIGH PRESSURE
LP = LOW PRESSURE

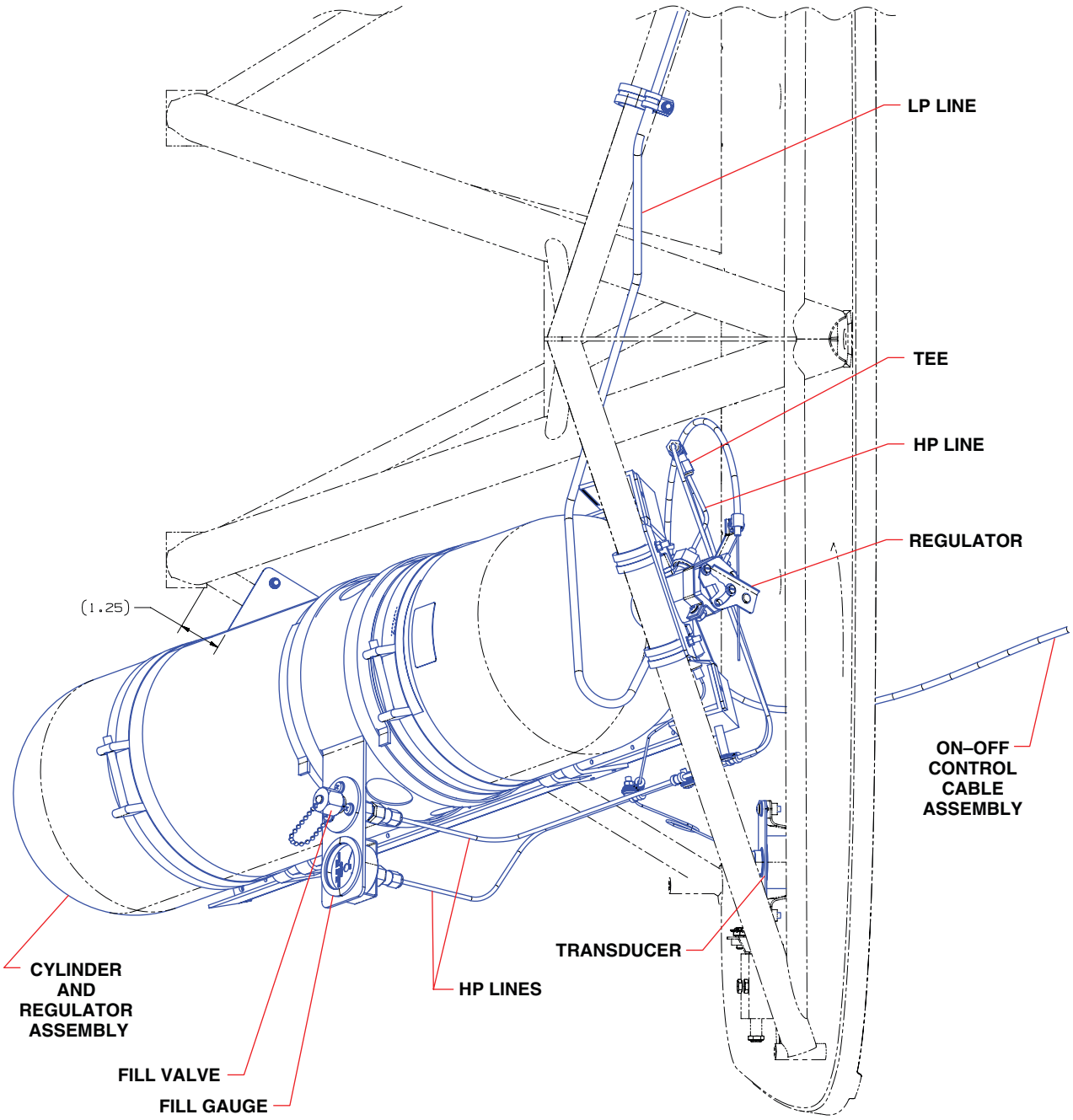
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Oxygen Cylinder and Regulator Valve
Figure 5 (Sheet 1 of 2)



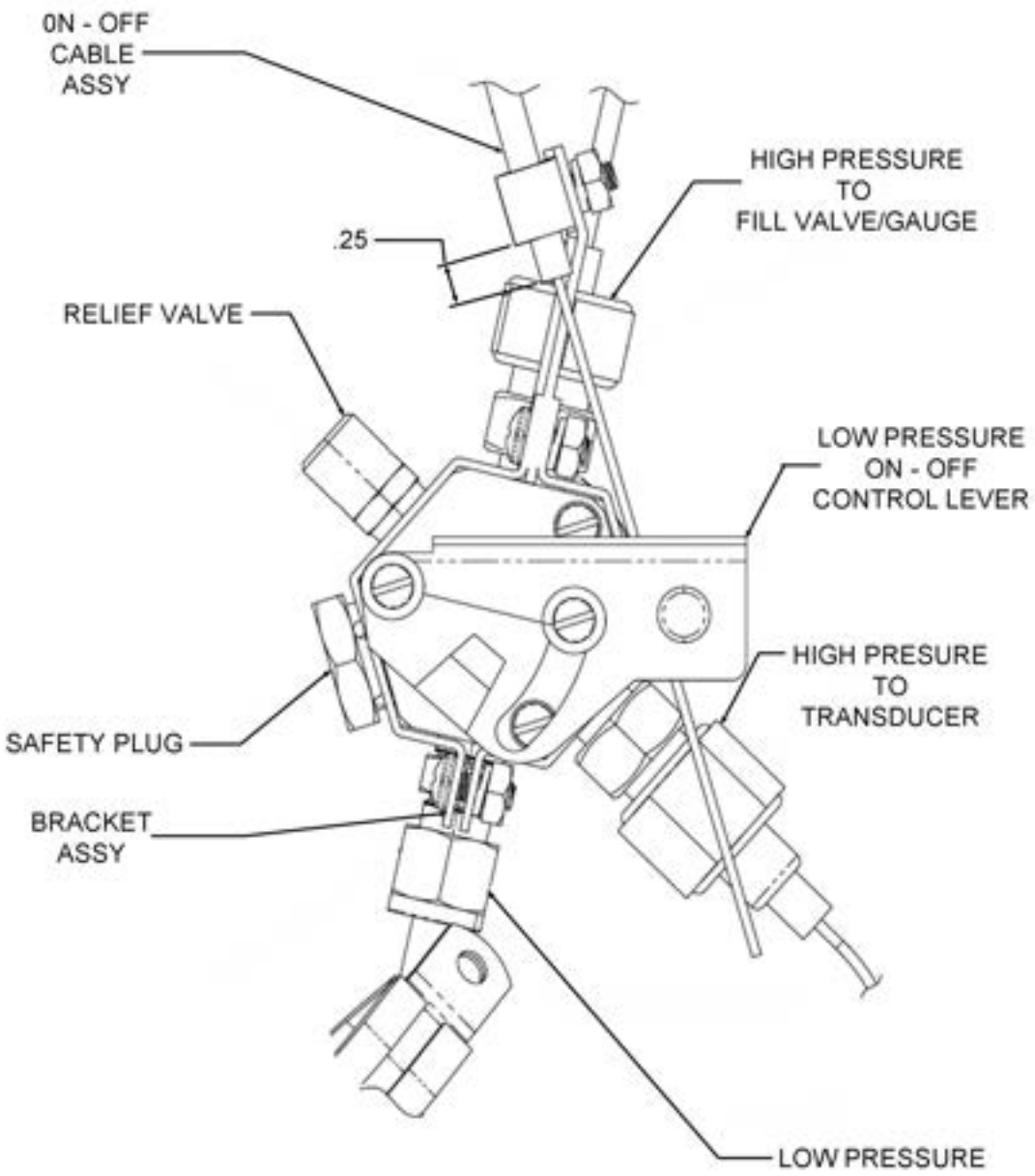
Oxygen Cylinder and Regulator Valve
Figure 5 (Sheet 2 of 2)

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Oxygen Cylinder and Regulator Valve
Figure 6 (Sheet 1 of 2)

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REGULATOR CONNECTIONS

Oxygen Cylinder and Regulator Valve
Figure 6 (Sheet 2 of 2)

(h) Connect ON-OFF cable to regulator valve lever. Secure with a new cotter pin.

(i) Position and attach cable support bracket.

(j) Check pressure and refill cyliner as necessary.

(k) Inspect for leaks, especially at fittings that have been separated.

(l) Install forward baggage compartment left rear closeout panel. Secure by installing 13 of the 15 attachment screws.

(m) Ensure cylinder and regulator valve ON–OFF control lever are clear of headliner and floor panel.

(n) Install hydraulic reservoir cover and secure with four screws.

NOTE: Two of these screws also secure closeout panel.

(o) Close and secure forward baggage door.

C. Cockpit Pressure Indication

(1) Pressure Gauge

See “Figure 7” on page 351018.

In Seneca IVs and Seneca Vs S/N’s 3449001–3449458, 3449460–3449466, the oxygen pressure gauge and ON–OFF control knob are installed in the bottom center of the pilot’s instrument panel below, and slightly right of, the control wheel. Access to either one is obtained from beneath the instrument panel.

NOTE: In Seneca Vs S/N’s 3449459, 3449467 and up, oxygen pressure is indicated in the cockpit by the Garmin G1000 / G1000 NXI displays, so the mechanical oxygen pressure gauge has been removed from the instrument panel. The ON–OFF control knob remains.

The pressure gauge is tied into the same high pressure line as the recharge valve, through a tee fitting near the tank regulator-control valve. Continuous pressure is applied to high pressure line until it is disconnected from cylinder. A check valve will close when high pressure line is disconnected from cylinder. The closing of this valve is frequently accompanied by a loud popping sound.

(a) Removal

1) From inside of of forward baggage compartment:

a) Remove hydraulic reservoir cover located in rear top center of compartment by removing four attachment screws.

NOTE: Two of these screws also secure closeout panel.

b) Remove left rear closeout panel by removing the 13 remaining attachment screws.

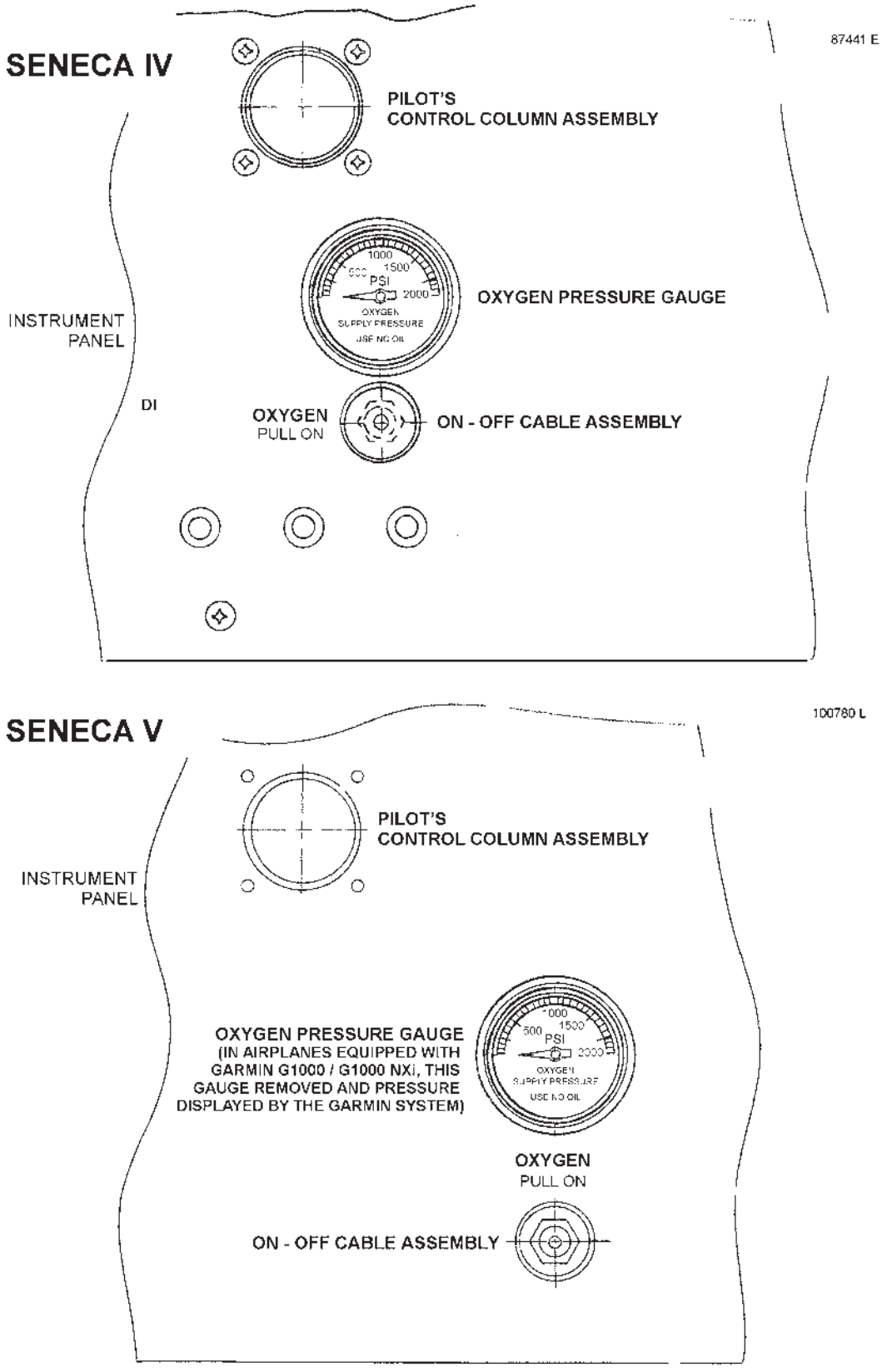
2) Disconnect high pressure line from regulator. Cap line immediately after disconnect.

NOTE: Continuous pressure is applied to high pressure line until it is disconnected from the regulator. A check valve in the regulator will close when high pressure line is disconnected. Closing of this valve is frequently accompanied by a loud popping sound.

3) Disconnect high pressure line from gauge and cap immediately.

4) Behind the instrument panel, remove two nuts from brass studs securing gauge to bracket.

5) Remove gauge aft through the instrument panel.



Pressure Gauge and ON-OFF Control Knob
Figure 7

(b) Installation

1) Insert pressure gauge into instrument panel.
2) Behind the instrument panel, secure gauge to mounting bracket by installing nuts on the two brass studs extending from gauge. Finger tighten, then snug with wrench. Do not over-torque; the studs break off easily.
3) Connect high pressure line to gauge.
4) Connect high pressure line to regulator. Insert tubing into fitting until ferrule seats in fitting. Tighten the nut by hand and then one quarter turn with a wrench. If fitting is relatively new the nut might be turned 3/4 of a turn. Follow up by snugging the nut slightly with a wrench.
5) Inspect fittings that have been separated for leaks.
6) Install forward baggage compartment left rear closeout panel. Secure by installing 13 of the 15 attachment screws.
7) Ensure cylinder and regulator ON–OFF control lever are clear of headliner and floor panel.
8) Install hydraulic reservoir cover and secure with four screws.

NOTE: Two of these screws also secure closeout panel.

9) Close and secure forward baggage door.

(2) Transducer

See “Figure 8” on page 351020.

In Seneca Vs S/N’s 3449459, 3449467 and up, a transducer is mounted to a bracket on the F.S. 49.50 bulkhead just inboard of the parking brake valve. It feeds oxygen cylinder pressures to the GEA for display on the Garmin G1000 / G1000 NXi MFD.

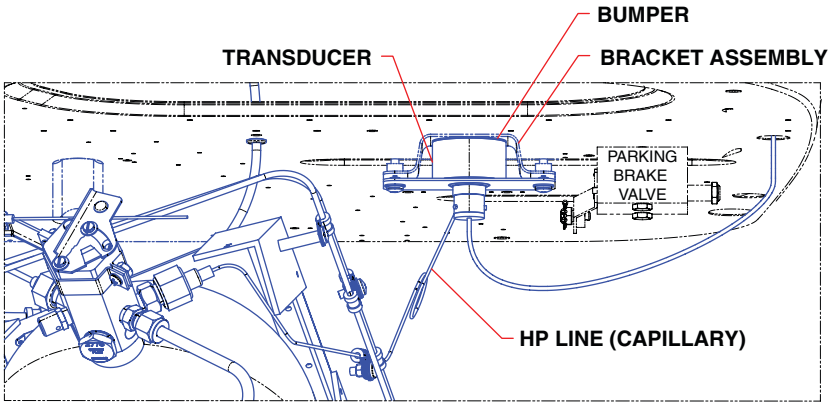
(a) Removal

1) Ensure oxygen and electrical system is off.
2) Disconnect the HP Capillary Line form the bottom of the transducer and cap line.
3) Disconnect the Harness Assembly P805 connector at the front of the transducer.
4) Remove the mounting screws, washers and plate assembly.
5) Remove the transducer.

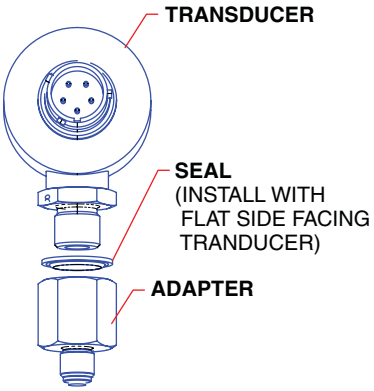
(b) Installation

1) Install the transducer into the bracket assembly.
2) Install the Plate Assembly and secure mounting screws and washers.
3) Connect the Harness Assembly P805 connector to the front of the transducer.
4) Remove cap from the HP Capillary Line and connect to the bottom of the transducer and tighten.
5) Turn on oxygen system and check for leaks.
6) Turn on electrical system and check MFD correct oxygen information is displayed.
7) Power down aircraft.

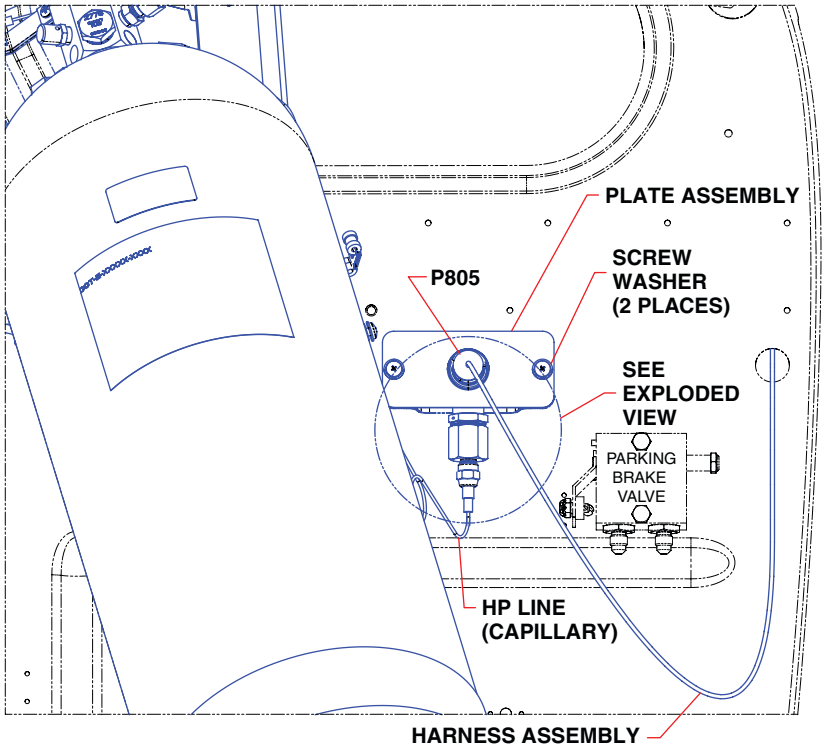
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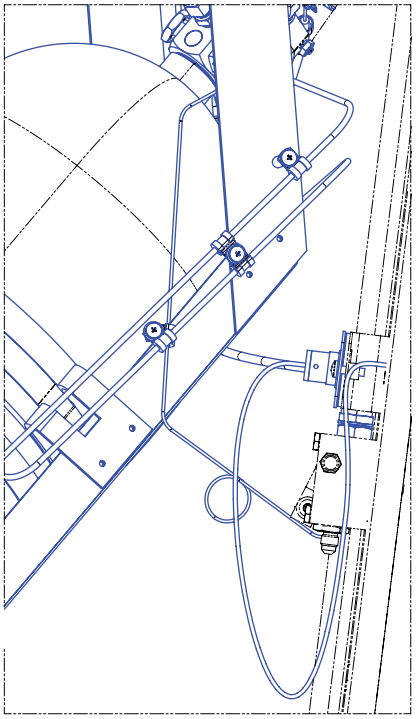
LOOKING DOWN



EXPLODED VIEW



LOOKING AFT
AT OXYGEN CYLINDER / REGULATOR ASSEMBLY
AND TRANSDUCER INSTALLATION
(COVERS AND CLOSEOUTS REMOVED)



LOOKING RIGHT

Transducer Installation
Figure 8

D. Oxygen On–Off Control

See “Figure 7” on page 351018.

The oxygen ON–OFF control knob is installed in the bottom center of the instrument panel below, and slightly right of, the pilot’s control wheel. Access is from beneath the instrument panel.

(1) Removal

To remove control knob and cable:

(a) From inside of of forward baggage compartment:

1) Remove hydraulic reservoir cover located in rear top center of compartment by removing four attachment screws.

NOTE: Two of these screws also secure closeout panel.

2) Remove left rear closeout panel by removing the 13 remaining attachment screws.

(b) Safety the regulator ON–OFF control lever in the OFF position.

(c) Disconnect cable from regulator control lever by removing cotter pin and washer.

(d) Release cable from all clamps and cut the tie wrap securing cable to high pressure oxygen line. Note position of tie wraps for reinstallation.

(e) Behind the instrument panel, remove retaining nut from rear of control knob.

(f) Cut loop off end of cable core. Pull cable from airplane through instrument panel. When cable has cleared grommet in left side of F.S. 44.5 bulkhead, slide retainer nut off of cable.

(2) Installation

NOTE: When rigging in the OFF position, the knob is not to be separated more than 1/8 inch away from the instrument panel.

To install new control knob and cable:

(a) On Seneca IV models, trim cable shield to 33.0 inches (83.82 cm) long. On Seneca V models, trim cable so that 13.7 inches (34.80 cm) remains forward of bulkhead when cable is rigged to instrument panel.

(b) Trim core to allow sufficient material to make a two turn loop, two inches (5.08 cm) from the end of the shield.

(c) Insert cable through instrument panel. Slide retainer nut onto cable and secure control knob to instrument panel.

(d) Route cable through grommet in F.S. 44.5 bulkhead and along high pressure line to regulator.

(e) Secure cable to high pressure line with same number of tie wraps, CR-2M ring connectors and clamps installed at same location as those cut or loosened during removal.

(f) Bend core wire end for 1 1/2 to 2 turns with 0.188 inch (0.478 cm) inside diameter.

(g) Place loop over pin on regulator control lever and secure with washer and cotter pin.

(h) Check operation before installing closeout panel.

(i) Install forward baggage compartment left rear closeout panel. Secure by installing 13 of the 15 attachment screws.

(j) Ensure cylinder and regulator ON–OFF control lever are clear of headliner and floor panel.

(k) Install hydraulic reservoir cover and secure with four screws.

NOTE: Two of these screws also secure closeout panel.

(l) Close and secure forward baggage door.

E. Outlets

Six oxygen outlets are installed in the cockpit/cabin overhead. One for each pilot or passenger. They are fed from the low pressure line and controlled by the Oxygen Pull ON knob.

(1) Removal

(a) Check that the Oxygen Pull ON knob is in the closed position – i.e., pushed forward against the instrument panel.

(b) Insert an oxygen mask into any outlet to release pressure and ensure the system is OFF.

(c) With a suitable spanner wrench, remove the outer half of the outlet.

(d) If removing one or more of the four aft cabin outlets, also remove two sheet metal screws from bezel.

(e) Remove or drop overhead panel sufficiently to gain access to low pressure line connections.

(f) If removing right side outlet(s), disconnect two unions on low pressure oxygen supply feed line, and one union connecting outlet to left side feed line.

(g) If removing left side outlet(s), disconnect one union connected to oxygen supply line form right side outlet.

(h) Remove outlet from airplane.

(2) Installation

(a) Position outlet in airplane.

(b) If installing left side outlet, connect union to oxygen supply line form right side outlet

(c) If installing right side outlet(s), connect two unions on low pressure oxygen supply feed line, and one union on outlet to left side feed line.

(d) Inspect fittings that have been separated for leaks.

(e) Replace overhead paneling and secure in place.

(f) If installing one or more of the four aft cabin outlets, install two bezel sheet metal screws. Check that word OXYGEN on bezel ring faces aft if supplying forward facing seats; forward if supplying aft facing seats.

(g) With a suitable spanner wrench, install the outer half of the outlet. If installing either one of the forward outlets, Check that word OXYGEN on bezel ring faces aft.

(h) Insert an oxygen mask into the outlet(s) that was (were) replaced and Pull ON the Oxygen knob to operationally test the outlet.

F. Fill Valve

See “Figure 9” on page 351024.

(1) Valve Assembly

(a) Removal

1) From inside of of forward baggage compartment:

c) Remove hydraulic reservoir cover located in rear top center of compartment by removing four attachment screws.

NOTE: Two of these screws also secure closeout panel.

d) Remove left rear closeout panel by removing 13 remaining attachment screws.

NOTE: Continuous pressure is applied to the high pressure line until it is disconnected from the regulator. A check valve in the regulator will close when the high pressure line is disconnected from cylinder. The closing of this valve is frequently accompanied by a loud popping sound.

2) Disconnect the high pressure line at the regulator. Cap the line immediately after removal.

3) Disconnect high pressure line from fill valve. Cap line immediately after removal.

4) Remove three screws from refill valve mounting plate. Capture cap chain with information plate when bottom screw is removed.

5) Remove valve from airplane.

(b) Installation

NOTE: Apply teflon tape to all tapered male threads as described in “Precautions” on page 35103.

1) Insert valve assembly into support.

2) Align screw holes in valve mounting plate with those in support.

3) Install the three mounting screws. Attach cap chain, with information plate attached, to bottom screw.

CAUTION: CONNECT HIGH PRESSURE LINE TO REFILL VALVE ASSEMBLY BEFORE CONNECTING TO REGULATOR.

4) Connect high pressure line to refill valve.

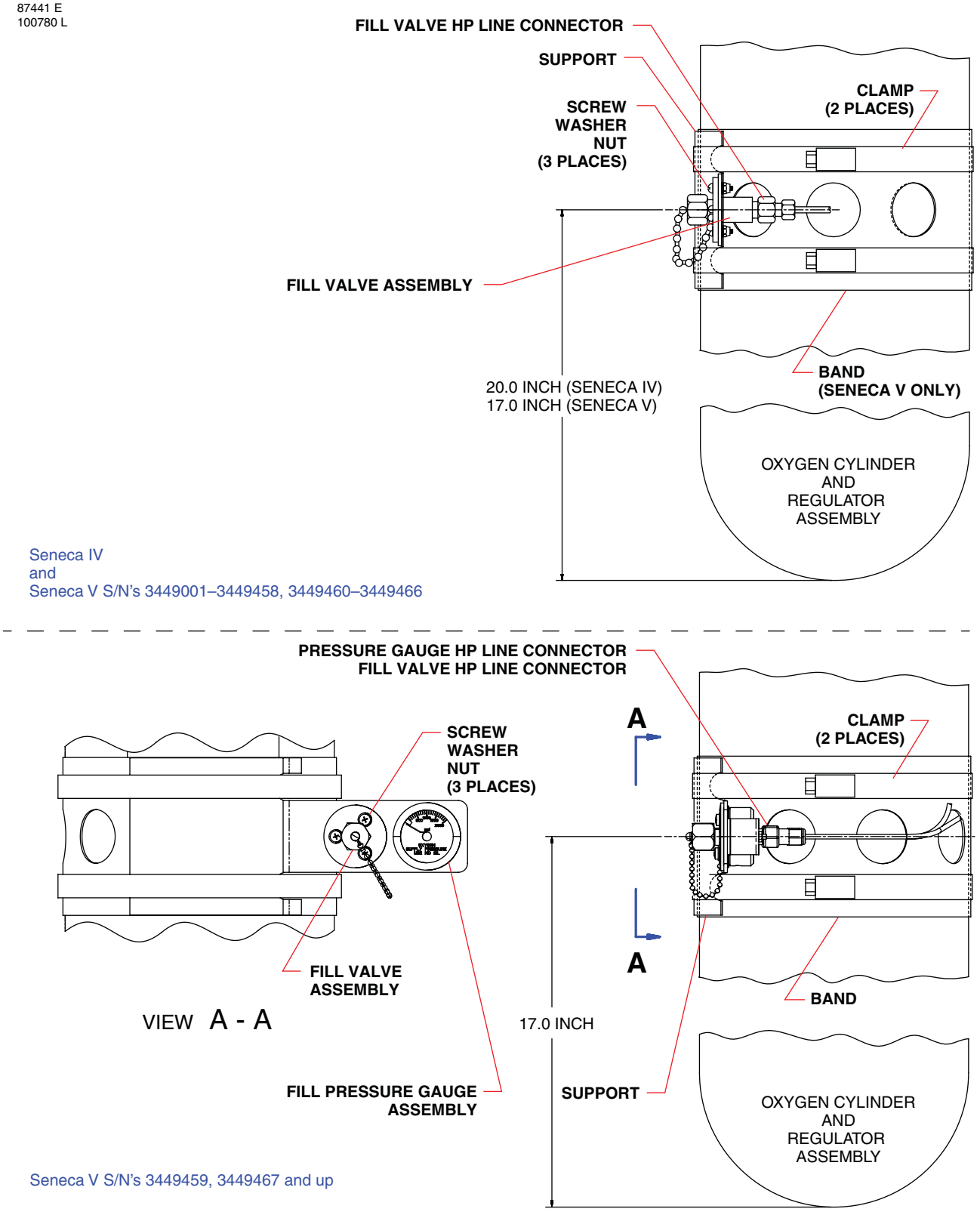
5) Connect high pressure line to regulator. Torque fitting 30–50 inch·pounds.

6) Check all connections that had been separated for leaks.

7) Install closeout panel and hydraulic reservoir cover.

8) Close and secure nose baggage compartment door.

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100780 L



Oxygen System Fill Valve Installation
Figure 9

(2) Pressure Gauge

In Seneca Vs S/N's 3449459, 3449467 and up, a pressure gauge is colocated with the fill valve on the support attached to the cylinder. See "Figure 9" on page 351024.

The pressure gauge is tied into the same high pressure line as the recharge valve, through a tee fitting near the tank regulator-control valve. Continuous pressure is applied to high pressure line until it is disconnected from cylinder. A check valve will close when high pressure line is disconnected from cylinder. The closing of this valve is frequently accompanied by a loud popping sound.

(a) Removal

1) From inside of of forward baggage compartment:

a) Remove hydraulic reservoir cover located in rear top center of compartment by removing four attachment screws.

NOTE: Two of these screws also secure closeout panel.

b) Remove left rear closeout panel by removing 13 remaining attachment screws.

2) Disconnect high pressure fitting at the regulator. Cap line immediately after removal.

3) Disconnect high pressure line from gauge and cap immediately.

4) Remove two nuts from brass studs securing gauge to support.

5) Remove gauge from support.

(b) Installation

1) Insert pressure gauge into support.

2) Secure to support by installing nuts on the two brass studs extending from gauge. Finger tighten, then snug with wrench. Be careful not to over torque; studs break off easily.

3) Connect high pressure line to gauge.

4) Connect high pressure fitting to regulator.

5) Inspect fittings that have been separated for leaks.

6) Install forward baggage compartment left rear closeout panel. Secure by installing 13 of the 15 attachment screws.

7) Ensure ON-OFF control cable is clear of headliner and floor panel.

8) Install hydraulic reservoir cover and secure with four screws.

NOTE: Two of these screws also secure closeout panel.

9) Close and secure forward baggage door.

9. Refilling

CAUTION: BEFORE SERVICING THE OXYGEN SYSTEM, MAKE SURE THE AIRCRAFT IS SECURELY GROUNDED ELECTRICALLY.

CAUTION: DO NOT OPERATE ELECTRICAL EQUIPMENT WHILE SERVICING OXYGEN SYSTEM.

CAUTION: DO NOT ATTEMPT TO TIGHTEN ANY CONNECTIONS WHILE THE SYSTEM IS CHARGED.

Refilling of oxygen systems should be done by qualified personnel. For comparison of filling pressures to ambient temperatures refer to “Chart 3”. The following are parameters to be followed for filling.

A. Only aviators breathing oxygen (MIL-PRF-27210) and appropriate filling equipment should be used to fill the system.

B. If a cylinder has less than 5 psi pressure or has insufficient pressure to produce an audible hissing sound when the valve is cracked, it should be removed and/or purged, and if the condition has existed for a significant length of time, hydrostatically test cylinder.

C. Make sure both the charge valve and recharge cart fittings are clean and free of contamination.

WARNING: BE CERTAIN THERE IS NO OIL OR OTHER PETROLEUM BASED MATERIAL ON THE FITTINGS OR NEAR THE IMMEDIATE VICINITY.

D. Attach service cart hose to recharge port. Fill the system at a rate not exceeding 200 psig per minute proceeding as follows:

(1) To obtain the correct filling pressure for the oxygen system at various ambient temperatures, a table is included for your convenience. The pressures given are not exact, but sufficiently accurate for practical purposes of working pressures between 1800 and 2400 psig cylinders. The cylinder should be allowed to cool to a stabilized temperature after filling before checking against the values in “Chart 3”.

CHART 3
FILLING PRESSURES FOR CERTAIN AMBIENT TEMPERATURES

Ambient Temperature °F/°C	Filling Pressure	Ambient Temperature °F/°C	Filling Pressure
0/-17.78	1650 (PSI)	70/21	1975 (PSI)
10/-12.22	1700	80/27	2000
20/-6.67	1725	90/32	2050
30/-1.11	1775	100/38	2100
40/4.44	1825	110/43	2150
50/10	1875	120/49	2200
60/15.56	1925	130/54	2250

NOTE: Filling pressures are for 1850 PSI at 70°F (21.11°C). Table assumes 25°F (11.8°C) rise due to heat of compressor with max. fill rate.

(2) When using a recharge unit consisting of one supply cylinder, slowly open the valve of the supply unit and allow the oxygen to transfer.

(3) When using a recharge unit consisting of two or more supply cylinders (cascade storage system), it is recommended that the following procedure be used:

(a) Before opening any valves, check the pressure remaining in the airplane's oxygen cylinder. If it is still partly charged, note the pressure indicated on the cylinder gauge. Then open and close each valve on the cascade storage system and determine which cylinder has the lowest pressure. When found, if this cylinder has a pressure lower than the oxygen cylinder in the aircraft, do not attempt using it for filling; use the storage cylinder that has a pressure higher than the aircraft's cylinder but lower than the others.

(b) Open the valve on only the one storage cylinder with the lowest pressure. When the pressure indicated on the aircraft's oxygen gauge and charging gauge has become equal, close the valve of the storage cylinder, then go to the storage cylinder with the next higher pressure and repeat the procedure.

(c) If after using the last storage cylinder the aircraft's oxygen system is still not fully charged, a full storage cylinder should be put in place of a cylinder with the lowest pressure and used in the same manner.

(d) A good amount of oxygen will remain in the large cylinders used in the cascade system after filling only one of the cylinders. This remaining oxygen will be at a pressure something less than the 1850 psi. This is not sufficient pressure to completely refill another aircraft cylinder, although it will refill several small cylinders.

(e) It is not economical, even on a three or four cylinder cascade system, to begin recharging with oxygen at less than 300 psi pressure in the 300 cubic foot bank of cylinders. So use 300 cubic foot cylinders down to approximately 300 psi; then return for refilling. In two cylinder systems use to approximately 100 psi; then return for filling.

(4) When the pressure gauge on the recharge unit or in the aircraft reaches 1800 to 1850 psi, close the pressure regulator valve on the recharge unit. Disconnect the filler hose from the filler valve; replace the protective cap on the filler valve and close the access cover. Check the cylinder pressure according to "Chart 3" on page 351026 after the cylinder temperature stabilizes.

E. After detaching the service cart, cap hose and fittings to prevent contamination.

F. Perform a leak check of the high pressure lines and clean off solution afterwards. If solution is not properly cleaned off, corrosion may result.

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CHAPTER

37

VACUUM

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CHAPTER 37

LIST OF EFFECTIVE PAGES

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CHAPTER 37 - VACUUM

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GENERAL

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

1. Description and Operation

   See “Figure 1” on page 37004.

   The vacuum system uses two regulators, gyro inlet filters, and a manifold check valve. The regulators are mounted on the left side of their respective firewalls and are accessible after removing the nacelle hatch cover. The gyro filters and manifold are located in front of the instrument panel mounted to the Fuselage Station 49.50 bulkhead. When the optional copilot gyros are installed two gyro filters are utilized. The gyro filters, mounted to the 49.50 bulkhead, must be replaced regularly. The filters clean inlet air to the gyros. Each regulator, with an inlet filter, is connected in line between its pump and the manifold to control the vacuum pressure applied to the system, by permitting a metered amount of air to enter the system at that point.

   The pneumatic deice installation is optional. Refer to 30-10-00 for more information.

   A vacuum gauge installed in the instrument panel reads the amount of negative pressure being provided to the gyroscopic flight instruments in inches of mercury. Lights on the annunciator panel will illuminate to show if either pump fails.

2. Troubleshooting

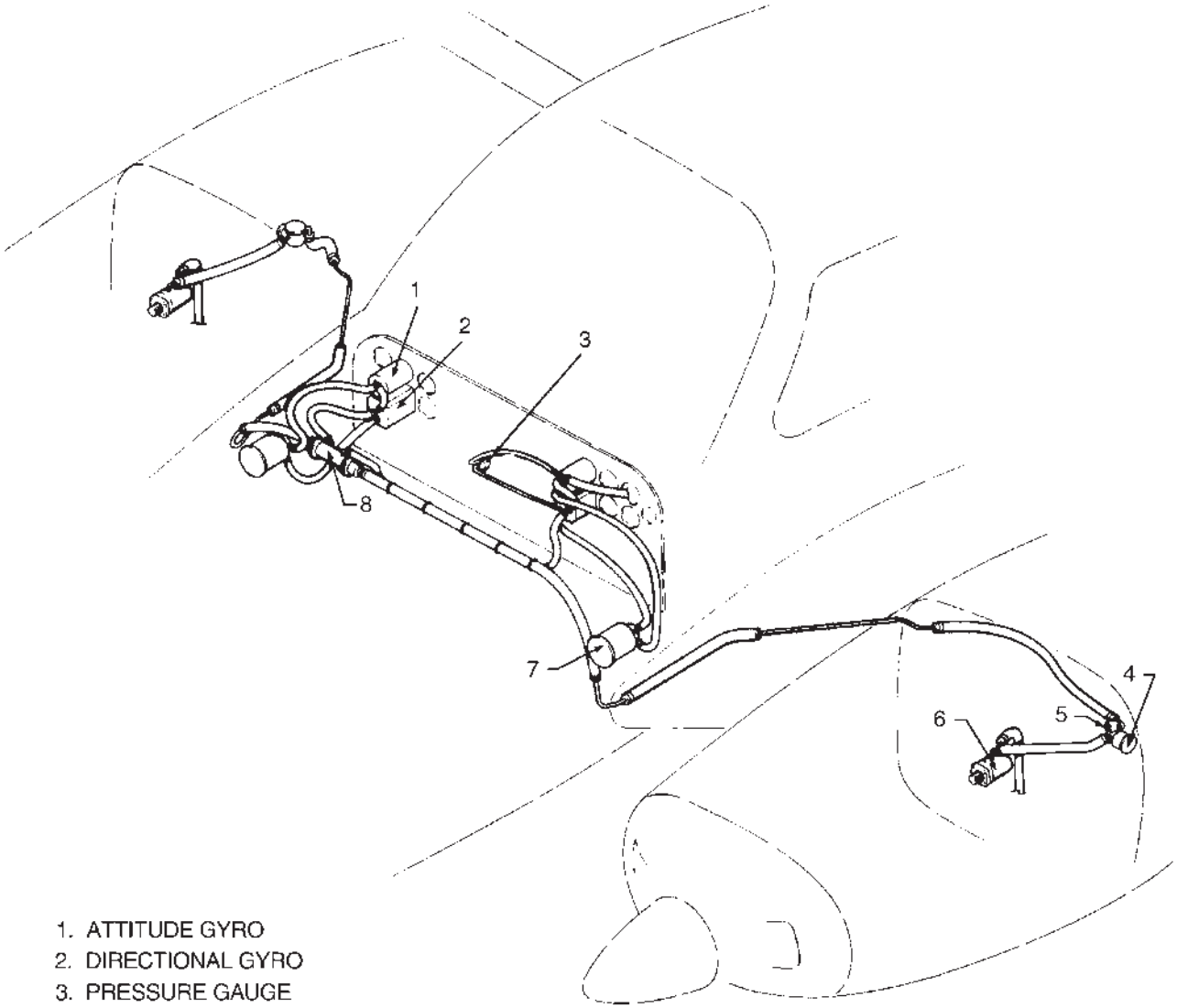
   See “Chart 1” on page 37002.

CHART 1 (Sheet 1 of 2)
TROUBLESHOOTING VACUUM SYSTEM

Trouble	Cause	Remedy
No vacuum gauge indication at instrument.	Gyro filter(s) clogged or dirty.	Clean or replace filter(s).
	Line(s) from gyro to gyro filter(s) restricted.	Check lines.
	Faulty gauge.	Replace gauge.
	Faulty transducer (Seneca V)	Replace transducer.
No vacuum gauge indication at instrument and/or source.	Malfunctioning pump(s).	Replace pump(s).
Low vacuum system pressure.	Gyro filter(s) dirty.	Clean or replace filter(s).
	Faulty air pump(s).	Replace pump(s).
	Vacuum regulator valve(s) incorrectly adjusted.	Adjust regulator valve(s) in accordance with adjustment in this section.
	Line from gyros to gyro filter(s) restricted.	Repair line.
	Lines between pumps and gyros leaking	Check all lines and fittings.
	Faulty transducer (Seneca V)	Replace transducer.
Normal vacuum indication but sluggish operation. of instrument.	Faulty instrument.	Replace instrument.
	Faulty gauge.	Replace gauge.
	Faulty transducer (Seneca V)	Replace transducer
High system vacuum.	Vacuum regulator(s) incorrectly adjusted.	Adjust regulator(s).
	Vacuum regulator(s) sticking or dirty regulator filter(s).	Clean and check operation of regulator(s) and filter(s).
Regulator(s) cannot be adjusted to produce correct pressure. (Too low)	Leaking lines, fittings, instruments.	Check hardware to instruments.
	Air pump malfunctioning.	Replace pump.
Regulator(s) cannot be adjusted to produce correct pressure. (Too high)	Dirty or clogged regulator filter(s).	Clean or replace regulator filter(s).
Vacuum correct on ground but will not maintain pressure at altitude.	Air pump malfunctioning.	Replace pump.

CHART 1 (Sheet 2 of 2)
TROUBLESHOOTING VACUUM SYSTEM

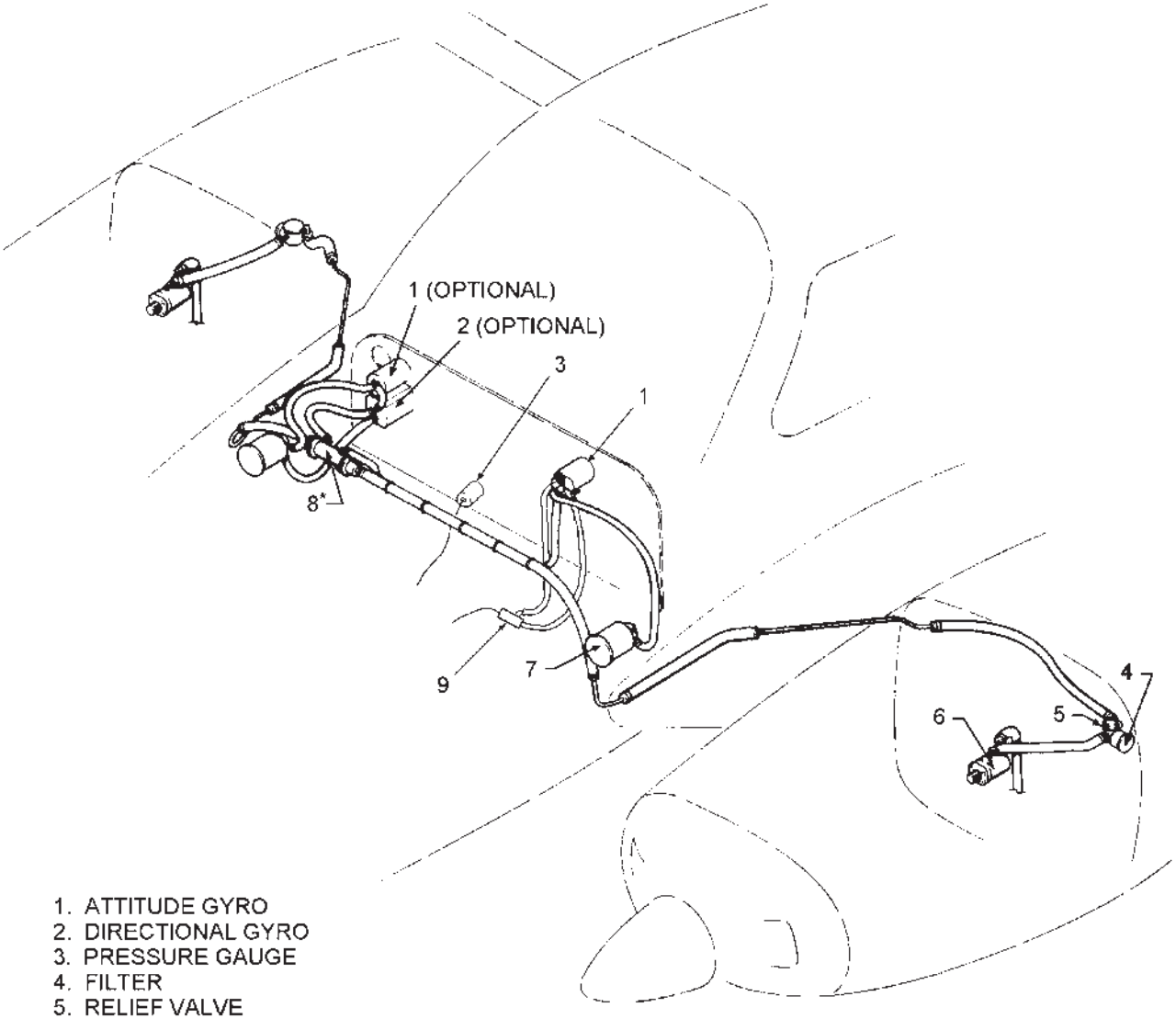
Trouble	Cause	Remedy
Vacuum correct but pilot reports pressure erratic or shows complete loss in flight.	Regulator sticky.	Clean regulator.
	Oil in pump due to leaky engine seal, or cleaning fluid blown into pump while cleaning engine.	Replace pump.
	Faulty transducer (Seneca V)	Replace transducer.
Pressure can only be maintained at full throttle on ground.	Leak in system.	Repair or replace lines.
	Worn pump.	Replace pump.
	Stuck regulator(s).	Clean or replace regulator.
Vacuum system inoperative during single engine operation.	Leaking manifold check valve.	Check operation of valve and replace if necessary.



1. ATTITUDE GYRO
2. DIRECTIONAL GYRO
3. PRESSURE GAUGE
4. FILTER
5. RELIEF VALVE
6. PUMP
7. FILTER
8. MANIFOLD

Effectivity
Seneca IV

Vacuum System Installation
Figure 1 (Sheet 1 of 4)

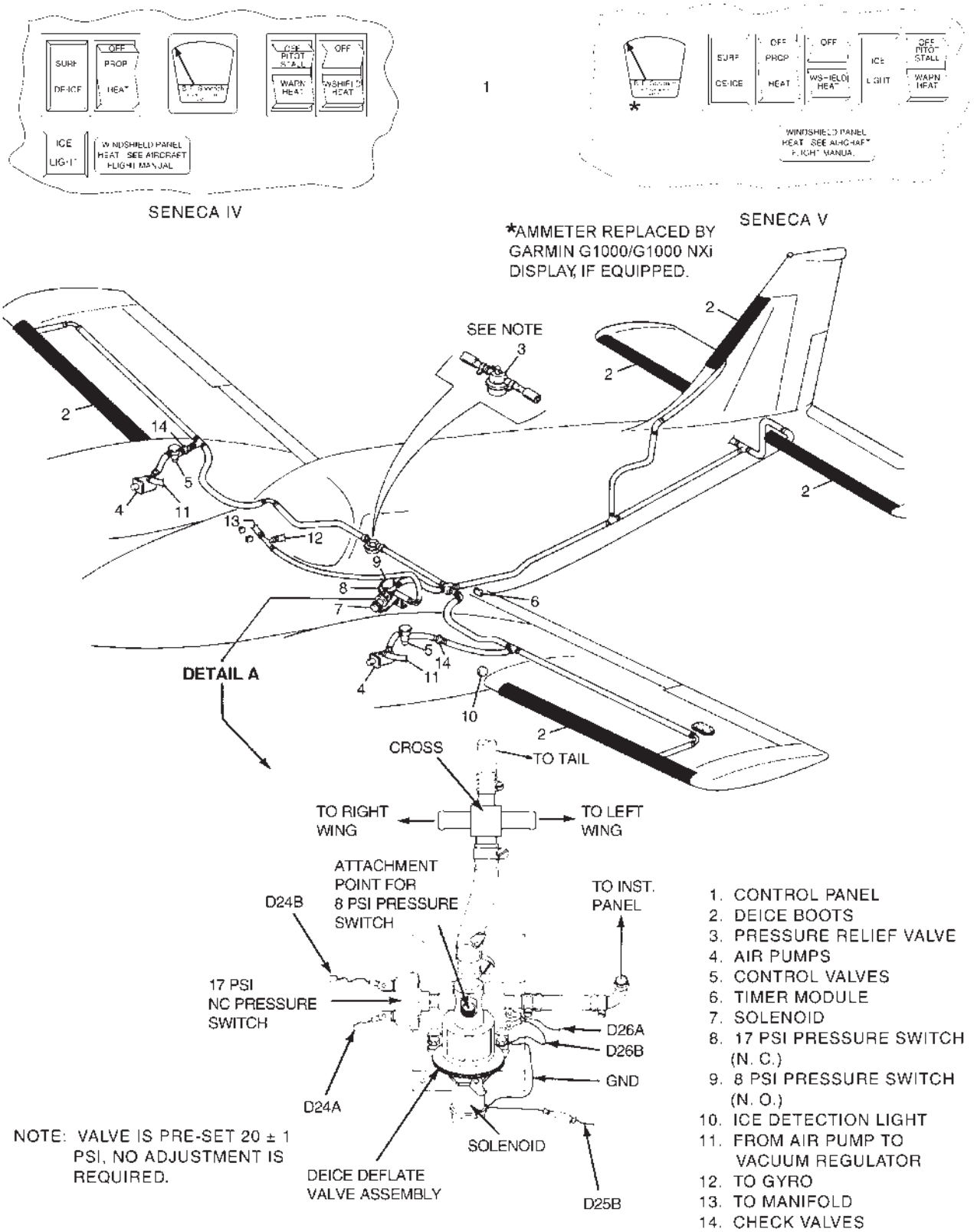


1. ATTITUDE GYRO
2. DIRECTIONAL GYRO
3. PRESSURE GAUGE
4. FILTER
5. RELIEF VALVE
6. PUMP
7. FILTER
8. MANIFOLD
9. TRANSDUCER

* 1998 AND LATER MAINIFOLD LOCATION SHOWN
1997 AND EARLIER SAME AS SENECA IV

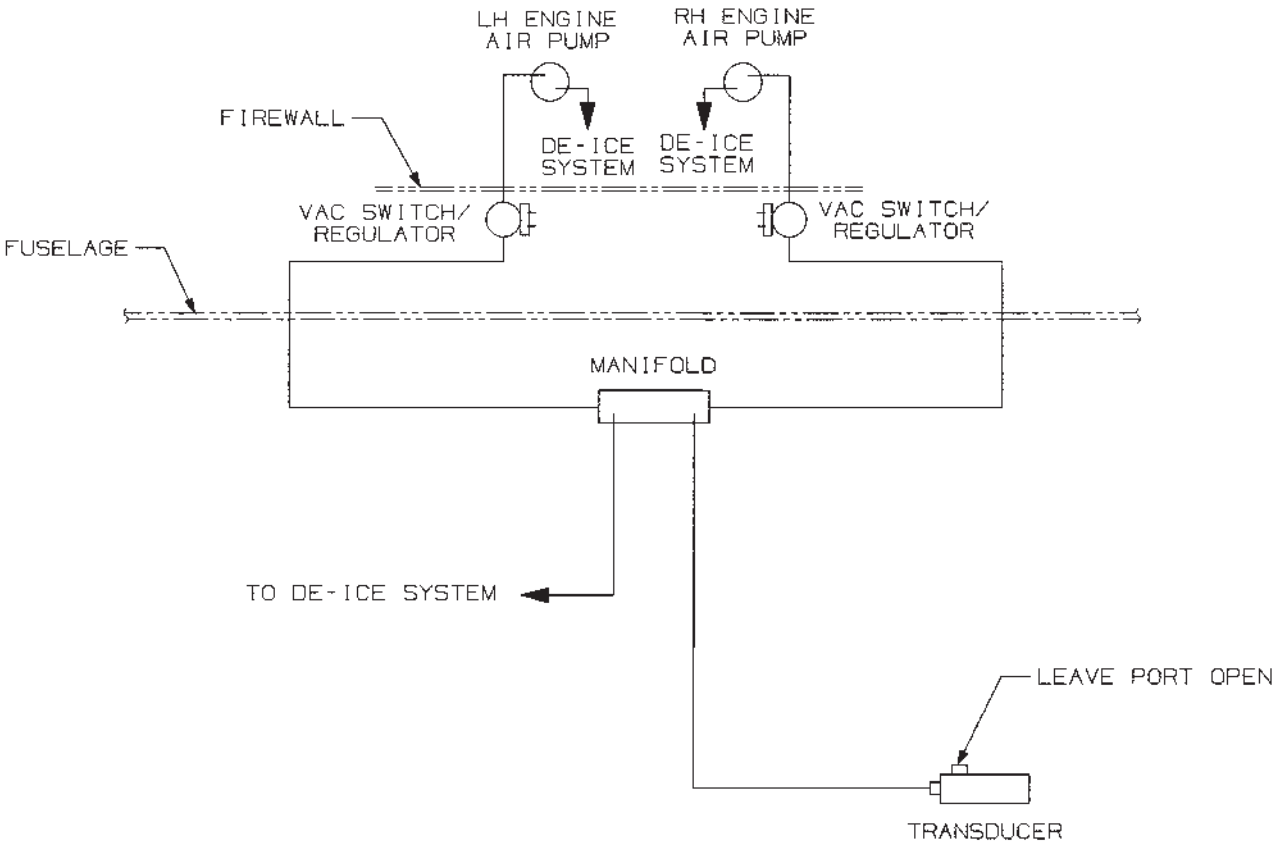
Vacuum System Installation
Figure 1 (Sheet 2 of 4)

Effectivity
Seneca V



Vacuum System Schematic
Figure 1 (Sheet 3 of 4)

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Vacuum System
Figure 1 (Sheet 4 of 4)

Effectivity
Seneca V
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DISTRIBUTION

The following information is intended to aid in diagnosing vacuum system service symptoms on those components which are serviced by removal and replacement - i.e. - hoses, clamps, gyro filters, vacuum pumps, and vacuum regulators.

1. Hoses and Clamps

   A. These items should be examined periodically and inspected carefully whenever maintenance activities cause hose disconnections to be made at the pumps, manifold, regulators, tube assemblies, gyros and/or vacuum gauge.

   B. Ends of hoses should be examined for rubber separation and slivers of rubber on inside diameter of hoses. These slivers can and do become detached. If this happens, the vacuum pump(s) will suck in the loose particles and eventually ingest them. This can cause pump failure.

   C. Replace old, hard, cracked or brittle hose. Sections of the inner layers may separate, causing pump failure.

   D. Ensure hoses are clear and clean by blowing them out with shop air. Remove from aircraft as required.

      CAUTION: DO NOT WIGGLE HOSE FROM SIDE TO SIDE DURING INSTALLATION. WIGGLING COULD CAUSE PARTICLES TO BE CUT FROM INNER WALL OF HOSE WHICH WOULD DAMAGE THE PUMP.

   E. Where hose clearance is tight, making it difficult to reinstall it onto a fitting or barb, spray the fitting or barb with silicone. Let dry, then install hose by pushing it straight on.

      CAUTION: WHEN REPLACING ANY OF THE THREADED FITTINGS, DO NOT USE PIPE DOPE, THREADLUBE, OR TAPE. PIPE DOPE / TAPE PARTICLES INGESTED BY THE VACUUM PUMP COULD CAUSE THE PUMP TO FAIL. USE ONLY SILICONE SPRAY, LETTING IT DRY BEFORE ASSEMBLY.

   F. Hose clamps and fittings should be replaced when broken, damaged or corroded.

2. Gyro Filter

   See “Figure 1” on page 37004.

   A. Gyro filters must be serviced on a scheduled basis, not to exceed 100 hours, and on condition.

   B. The system installation employs a large central filter (mounted on the upper left rear of the forward bulkhead (behind instrument panel)) and differential vacuum gauge that continuously monitors filter condition while indicating vacuum readings. A second filter will be installed on the right side, if optional co-pilot’s instruments are installed.

      NOTE: A decline in panel gauge reading indicates the filter is becoming clogged. Filters should be replaced when gauge reading declines; DO NOT adjust regulator(s).

3. Vacuum Pumps

One (1) engine-driven dry-air pump is mounted on the accessory section at the rear of each engine. Each vacuum pump is a rotary vane, positive displacement type. These units consist of an aluminum housing, a carbon rotor and carbon vanes. They are driven by means of a coupling mated to an engine-driven gear assembly.

WARNING: GROUND THE MAGNETO PRIMARY CIRCUIT (P LEAD), BEFORE PERFORMING ANY MAINTENANCE OPERATION ON THE ENGINE.

A. Inspection

See “Figure 1” on page 37102.

Aero Accessories Tempest dry air pumps are factory installed in S/N’s 3449273 and up; and available for service replacement in Seneca IV’s and earlier Seneca V’s. If so equipped;

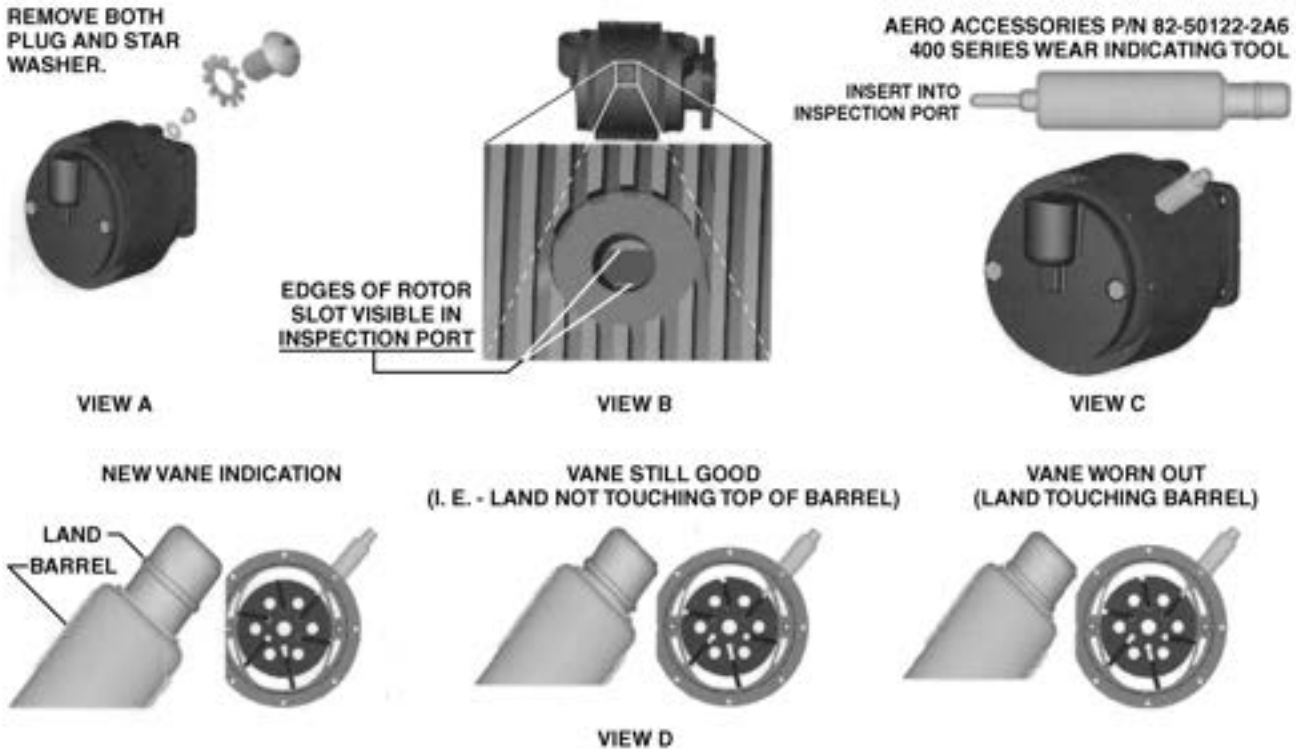
- for airplanes with frequent de-ice cycles, beginning at 200 hours time-in-service, and each 100 hours thereafter;
- for airplanes with normal de-ice cycles, beginning at 300 hours time-in-service, and each 100 hours thereafter;

inspect the vacuum pumps as follows:

(1) Remove engine cowling. (See 71-10-00.)

(2) Ground magnetos and turn the fuel selector to OFF.

CAUTION: DIRT, GREASE, OR DEBRIS IN THE AREA OF THE INSPECTION PORT PLUG COULD CONTAMINATE THE PUMP IF THE PLUG IS REMOVED PRIOR TO CLEANING.



Vane Wear Measurement
Figure 1

(3) Ensure the area around the inspection port plug is clean.

CAUTION: IF THE STAR WASHER IS NOT REMOVED WITH THE PLUG, THE SUBSEQUENT WEAR INDICATION READING WILL BE ERRONEOUS.

(4) Remove the inspection port plug and star washer (View A).

(5) While looking into the inspection port, have an assistant slowly turn the propeller by hand in the normal direction of rotation until a vane slot is centered in the port (View B).

NOTE: If you go too far, just keep turning the propeller until the next vane slot appears. DO NOT turn the propeller backwards.

CAUTION: DO NOT ROTATE PROPELLER OR PUMP SHAFT WITH THE INDICATING TOOL INSERTED INTO THE INSPECTION PORT. IF THE PUMP IS INADVERTENTLY ROTATED WHILE THE INDICATING TOOL IS INSERTED INTO THE INSPECTION PORT, PUMP REPLACEMENT IS REQUIRED - EVEN IF YOU CAN DETECT NO DAMAGE.

(6) Insert the vane wear indicating tool into the inspection port as illustrated in View C.

(a) Hold the barrel securely and squarely against the pump body.

(b) With your finger tip, gently push the plunger into the pump. When the plunger touches the vane you may feel the vane move slightly if it is not at the bottom of its slot.

NOTE: If the plunger does not slip easily into the slot, DO NOT force it. Remove the indicating tool and check the alignment of the vane slot to the inspection port.

(7) Compare the relative positions of the plunger land and barrel top to the illustrations in View D to determine vane wear.

(a) If vane is “worn out,” replace pump.

(b) If vane is “still good” or “new,” proceed to next step.

(8) If vanes are within service limits and pump is otherwise serviceable, clean the threads on the inspection port plug, install a new star washer and torque the port plug to 45 to 50 in.-lbs.

B. Removal

(1) Remove engine cowling. (See 71-10-00.)

(2) Loosen hose clamp and remove hose from pump fittings.

(3) Remove four retaining nuts, lock washers and plain washers used to secure pump to engine; then remove pump.

C. Installation

NOTE: Change the vacuum system filter when installing a new pump.

(1) If required, install fittings on pump per Replacing Pump Fittings, below.

CAUTION: ONLY PUMP MOUNTING GASKET AUTHORIZED AND APPROVED FOR USE ON AIRBORNE VACUUM PUMP IS AIRBORNE GASKET B3-1-2, PIPER PART NUMBER 751-859. USE OF ANY OTHER GASKET MAY RESULT IN OIL SEEPAGE OR LEAKAGE AT MOUNTING SURFACE.

(2) Place pump gasket in its proper place and align spline on pump drive with spline on engine drive assembly.

(3) Secure pump to engine with four plain washers, lock washers and retaining nuts. Torque nuts 50 to 70 inch-pounds.

(4) Connect hoses to pump and secure with hose clamps.

(5) Reinstall engine cowling.

D. Replacing Pump Fittings

CAUTION: WHEN REPLACING ANY OF THE THREADED FITTINGS, DO NOT USE PIPE DOPE, THREADLUBE, OR TAPE. PIPE DOPE / TAPE PARTICLES INGESTED BY THE VACUUM PUMP COULD CAUSE THE PUMP TO FAIL. USE ONLY SILICONE SPRAY, LETTING IT DRY BEFORE ASSEMBLY.

(1) Before installing any fittings on pump, check for any external damage. A pump that has been damaged or dropped should not be installed.

CAUTION: DO NOT APPLY VISE PRESSURE TO OUTSIDE DIAMETER OR OVERALL LENGTH OF PUMP.

(2) When a vise is used to hold pump while installing fittings, suitable caution must be exercised to avoid pump damage. Square mounting flange must be held between soft wood blocks and only at right angles to vise jaws. Use only enough vise pressure to hold pump firmly.

(3) The ports of AIRBORNE pumps have been treated with a dry film lubricant and AIRBORNE fittings are cadmium plated thus eliminating any need for thread lubricants. If thread lubricant is required, use only a silicone spray. Apply sparingly to external threads of fittings only and let dry before assembly.

4. Vacuum Regulators

See “Figure 2” on page 37105 and “Figure 1” on page 37004.

Two (2) vacuum regulators are incorporated in the system to control vacuum pressure to gyro instruments. The regulators are located aft of each engine, on the rear of the firewall.

A. Removal

(1) Remove inlet hose from aft side of valve.
(2) Remove outlet hose from forward side of valve.
(3) Remove nut that secures valve to firewall.
(4) Pull valve rearward out of firewall.

B. Installation

(1) Insert valve into hole in rear of firewall.
(2) Secure valve to firewall with nut.
(3) Attach inlet hose to forward fitting of valve.
(4) Attach outlet hose to aft fitting of valve.

C. Service Tips

(1) The vacuum regulating valve seldom needs replacement. Symptoms that suggest replacement are:
(a) Chatter as indicated by rapid fluctuation of the vacuum gauge needle or an audible sound.
(b) Non-repeatability of the vacuum gauge reading when the panel gauge is not suspect or has been checked against a known test gauge (cruise rpm only).

(2) All modes of regulator malfunction tend to increase the vacuum power applied to the gyros. Thus, although excess vacuum is applied, a loss of vacuum does not occur.

(3) The gyros themselves act as a limiting device to keep the vacuum power applied from exceeding safe levels.

NOTE: If the panel gauge has been checked and found “OK” and the vacuum gauge reading does not repeat within the normal operating range as marked on the gauge, then the regulating valve should be changed. Observe the usual precautions for maintaining system cleanliness to avoid premature pump service.

D. Adjustment

WARNING: DO NOT ATTEMPT ADJUSTMENT OF THIS VALVE WITH THE ENGINE IN OPERATION.

(1) Bend locking tabs up to rotate adjustment screw.

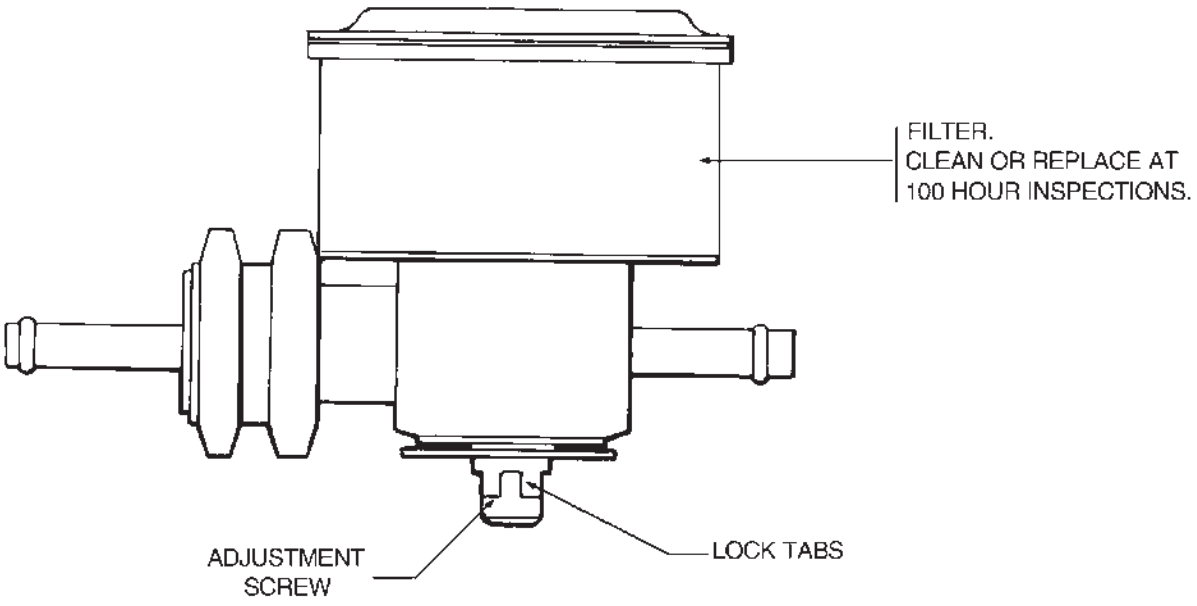
(2) Start the respective engine, after allowing time for warm-up, run the engine at medium rpm.

NOTE: Operation of engine at medium rpm is considered to be at magneto check rpm.

(3) With appropriate engine running at medium rpm, the suction gauge should indicate 5.0 inches of mercury ± 0.2 inches of mercury (Seneca IV); or within the normal operating range as marked on the vacuum gauge (Seneca V). If the pressure reading fails to fall within this range, shut down the engine and adjust the regulator valve by moving the valve adjustment screw clockwise to increase pressure, and counterclockwise to decrease pressure. Start the engine and repeat the check. Continue process until desired reading is obtained.

(4) Restart the engine and repeat the check.

(5) After the system pressure has been adjusted to recommended settings, bend locking tabs down to lock adjustment screw in place.



Vacuum Regulator
Figure 2

5. Vacuum Manifold

See “Figure 3” on page 37107, and “Figure 1” on page 37004.

The vacuum manifold is located within the nose section, installed forward of the forward bulkhead (F.S. 49.50), to the right of the airplane centerline.

A. Removal

(1) Disconnect all hoses from the manifold body.

(2) Remove the hardware that secures the manifold bracket to the forward bulkhead.

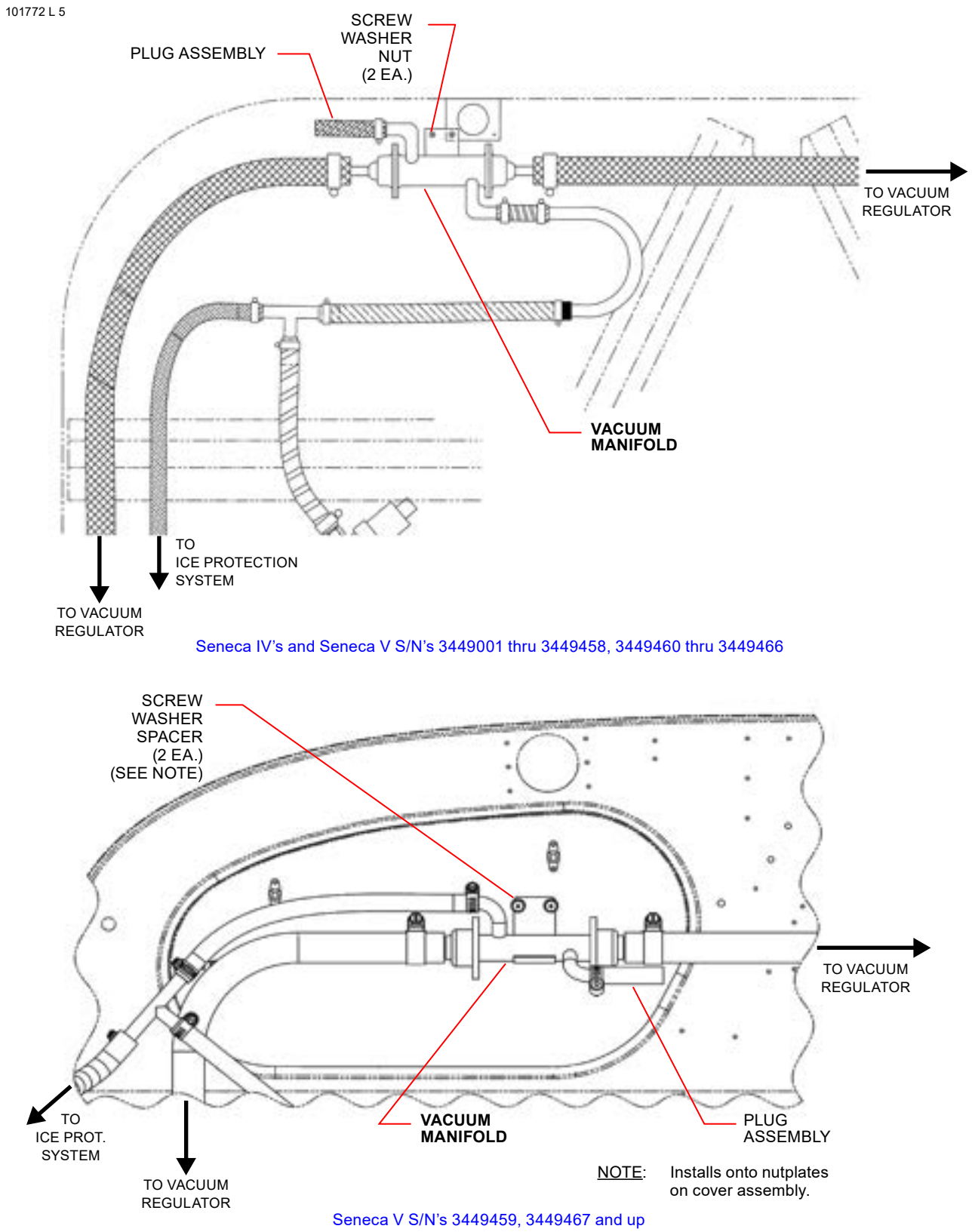
(3) Remove the vacuum manifold.

B. Installation

(1) Position the vacuum manifold in place.

(2) Secure the vacuum manifold with the hardware previously removed. (For the vacuum manifold installation on S/Ns 3449459, 3449467 and up, spacers are included, and the bracket is secured to nutplates on a cover assembly on the bulkhead. All other aircraft use nuts to install on the bulkhead.)

(3) Reconnect all hoses, removed previously, to the vacuum manifold.



Vacuum Manifold Installation
Figure 3

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INDICATING

The following information is intended to aid in diagnosing vacuum system service symptoms on those components which are serviced by removal and replacement - I.E. - vacuum gauges and switches.

1. Vacuum (Suction) Gauge

See “Figure 1” on page 37004.

A. Seneca IV

The vacuum gauge is mounted in the upper left section of the pilot’s instrument panel above the ELT switch. The gauge is calibrated in inches of mercury and is connected across the pilot’s gyros. The gauge indicates the differential pressure or actual pressure being applied to the gyro instruments. As the gyro filter(s) becomes clogged or lines obstructed, the gauge will show a decrease in pressure. Do not reset the regulator until the gyro filter(s) and lines have been checked.

B. Seneca V

The vacuum gauge is mounted in the lower right section of the pilot’s instrument panel, just above the landing gear down annunciator lights. The gauge is calibrated in inches of mercury and measures the differential pressure across the gyros by use of a transducer. As the gyro filter(s) becomes clogged or lines obstructed, the gauge will show a decrease in pressure. Do not reset the regulator until the gyro filter(s) and lines have been checked.

C. Troubleshooting

See “Chart 1” on page 37002.

D. Service Tips

(1) Vacuum gauges seldom require service and usually are replaced when malfunctions occur.

NOTE: Vacuum gauge failure in a properly operating vacuum system does not impair safety of flight.

(2) If the vacuum gauge malfunctions in a manner to cause an incorrect reading in normal cruise conditions, the gauge must be checked by comparing its reading with a gauge of known accuracy. If the gauge is confirmed to be indicating correct values and system vacuum level is not in accordance with specified vacuum, then and only then should the vacuum regulators be reset/readjusted.

(3) Visual examination of gauge performance should cover the following steps:

(a) With engine stopped and no vacuum supplied to gauge, its pointer should rest against the the internal stop in the 9 o’clock position. Any other displacement from this position suggests need for replacement.

(b) A slight overshoot during engine startup, not to exceed half an inch of mercury, is normal and is not cause to replace gauge.

(4) With engine operating at normal cruise RPM, gauge should read within the normal operating range as marked on the gauge.

(5) At 1200 rpm, vacuum gauge reading should be more than four inches of mercury.

E. Removal and Installation

The vacuum gauge is a face-mounted instrument. See 39-10-00.

2. Vacuum Switches

See “Figure 1” on page 37004.

A vacuum switch is located on each vacuum regulator at the rear of each firewall. Each activates its associated VACUUM INOP annunciator light when the suction from the associated pump drops below 4 ± .25 psig.

A. Removal

(1) Disconnect the two electrical leads.
(2) Unscrew switch unit from vacuum regulator assembly boss.
(3) Cap tube assembly boss to prevent foreign matter from entering system.

B. Installation

(1) Uncap tube assembly boss.
(2) Screw switch unit into vacuum regulator assembly boss.
(3) Reconnect the two electrical leads.
(4) Perform vacuum system operational check.

CHAPTER

39

ELECTRICAL / ELECTRONIC PANELS

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CHAPTER 39

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CHAPTER 39 - ELECTRICAL / ELECTRONIC PANELS

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INSTRUMENT AND CONTROL PANELS

1. General

A. Face-Mounted Instruments

Most instruments are face-mounted and secured to the instrument panel by screws from the front of the panel. Most instruments are removed out the back of the panel, but a few are removed through the front of the panel. Take special care when any operation pertaining to the instruments is performed.

(1) Removal

(a) Disconnect the plumbing and/or electrical connectors from the back of the instrument. Where two or more lines connect to an instrument, identify and tag each line to facilitate installation. Attach a dust cap to each fitting.

NOTE: For those instruments which remove through the front of the panel, disconnecting and tagging plumbing and/or electrical connectors can be done after the instrument retaining screws are removed and the instrument is slid gently forward to expose the connections at the rear.

(b) Remove the screws that secure the instrument in the panel cutout.

(c) Remove the instrument from the panel.

(2) Installation

(a) Place the instrument in its proper panel cutout and secure with screws.

NOTE: For those instruments which install through the front of the panel, connecting plumbing and/or electrical connectors can be done from the front of the panel before the instrument retaining screws are installed. After the connections are secure, slide the instrument into place and install the retaining screws.

(b) Connect the plumbing and/or electrical connectors to back of instrument.

(c) Check instrument operation.

B. Rack-Mounted Avionics

Most avionics are rack-mounted front-removable units generally secured to the instrument panel tray/rack by a single jackscrew located in the center of their faceplate.

(1) Removal

(a) Insert an appropriate size (generally 3/32 inch) allen wrench into the jackscrew access hole in the faceplate.

(b) Unscrew the jackscrew in a counterclockwise direction.

(c) Slide the avionics unit aft and out of the instrument panel tray/rack.

(2) Installation

(PIR-PPS60237, Rev New)

NOTE: Inspect the front of the panel-mounted avionics tray/rack to verify it is not significantly inset from the panel. If so, correct the tray/rack installation before proceeding.

NOTE: The high insertion forces required to seat a unit with “high density” connectors tend to limit the effectiveness of the first seating attempt. Accordingly, the following procedure requires sequential applications of force, and subsequent tightening of the jackscrew, to ensure all connectors seat properly.

(a) Slide the avionics unit into the instrument panel rack and forward applying a moderate insertion force.

(b) Insert an appropriate size (generally 3/32 inch) allen wrench into the jackscrew access hole in the faceplate and tighten to remove any slack, but do not try to “pull” unit into place with the jackscrew.

(c) Apply additional insertion force to front of unit.

(d) Tighten jackscrew again.

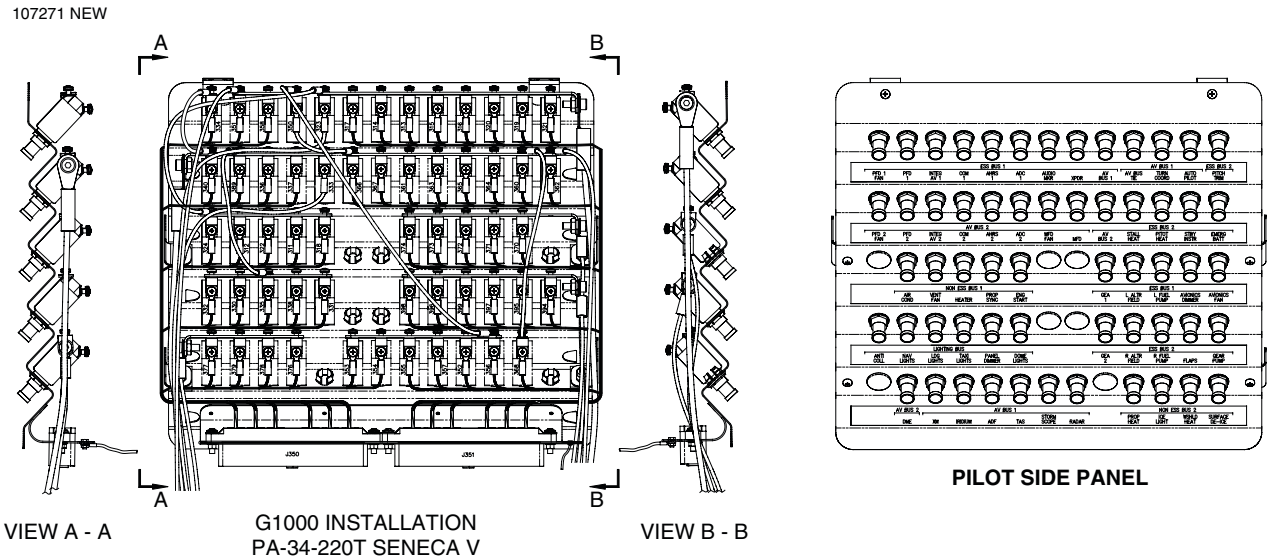
(e) Apply additional insertion force to front of unit.

(f) Finish tightening jackscrew.

(g) Ensure that unit bezel is “tight” against panel.

2. Circuit Breaker Panel

In Seneca Vs with Garmin G1000/G1000 NXi installed, circuit breakers are installed on the pilot’s side panel. In earlier Seneca IVs and Seneca Vs, circuit breakers are installed on the lower right side of the instrument panel. Circuit breakers are of the single hole mounting, push button type, with manual reset.



Effectivity
Seneca V with Garmin G1000

Circuit Breaker Panel Installation (Typical)
Figure 1

Should a circuit breaker be replaced or added, exercise extreme caution ensuring the breakers are in proper mechanical alignment, any insulators that are called out are installed correctly, and all electrical wiring and connections meet aviation standards. Do not deviate from the parts manual requirements when replacing circuit breakers.

NOTE: This type of circuit breaker can be used as a method of turning a system on and off.

A. Removal

(1) In airplanes equipped with Garmin G1000/G1000 NXi, disconnect the emergency battery P615 connector.

(2) Disconnect ground (negative) battery cable.

(3) Remove knurled nut from circuit breaker face plate on front of instrument panel.

(4) From behind instrument panel, disconnect electric bus bar from circuit breaker.

(5) From behind instrument panel, remove circuit protector from instrument panel.

NOTE: Record placement of electrical leads to aid installation.

(6) Disconnect electrical connections fastened with screws to circuit breaker.

B. Installation

(1) Check circuit breaker amperage is correct.

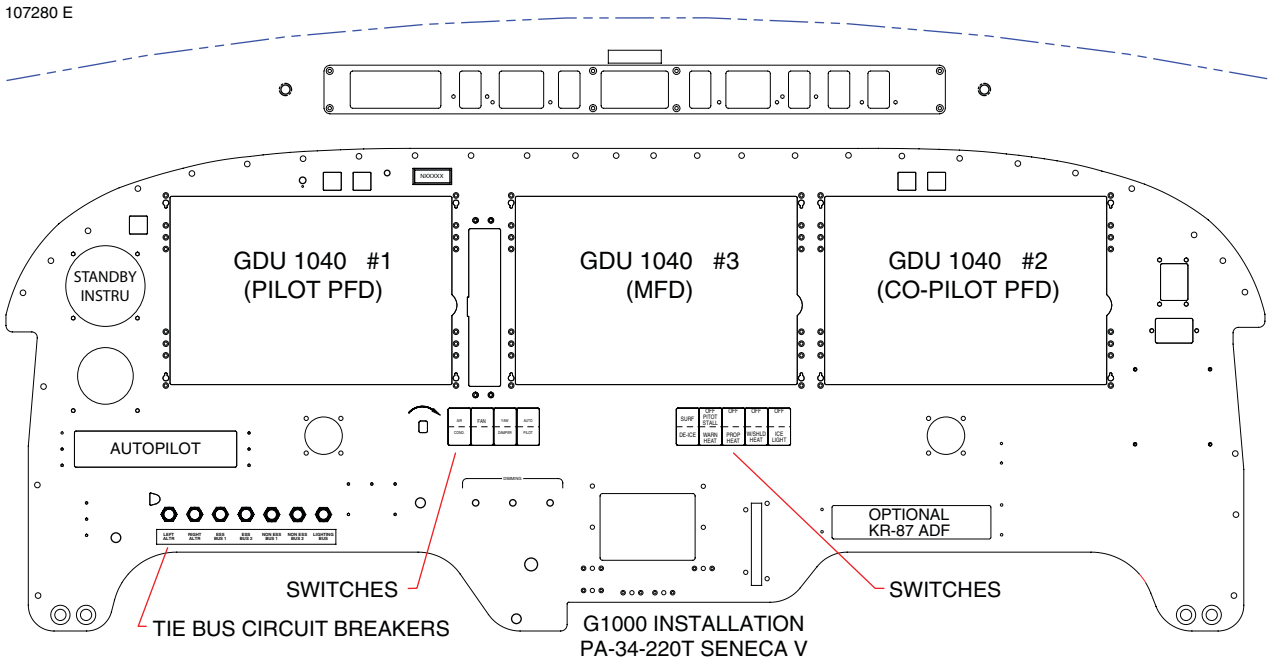
(2) Connect electrical leads to their proper screws on new breaker and secure.

(3) From behind instrument panel, insert circuit protector into its proper hole on instrument panel.

(4) From behind instrument panel, install electric bus bar to circuit breakers.

(5) Reconnect ground (negative) battery cable.

(6) In airplanes equipped with Garmin G1000/G1000 NXi, reconnect the emergency battery P615 connector.



Instrument Panel Installation (Typical)
Figure 2

Effectivity
Seneca V with Garmin G1000

3. Switches

In the Seneca IV, switches are of the rocker type and are mounted in the instrument panel.

In the Seneca V, switches are of the rocker-type and are located centrally in the instrument panel and in overhead switch panels.

A. Overhead Switch Panel

When working on the overhead switch panel, remove it from the aircraft first.

(1) Removal

(a) Remove panel retaining screws (8).

(b) Slide the switch panel down to gain access to the electrical connectors at the back of the panel.

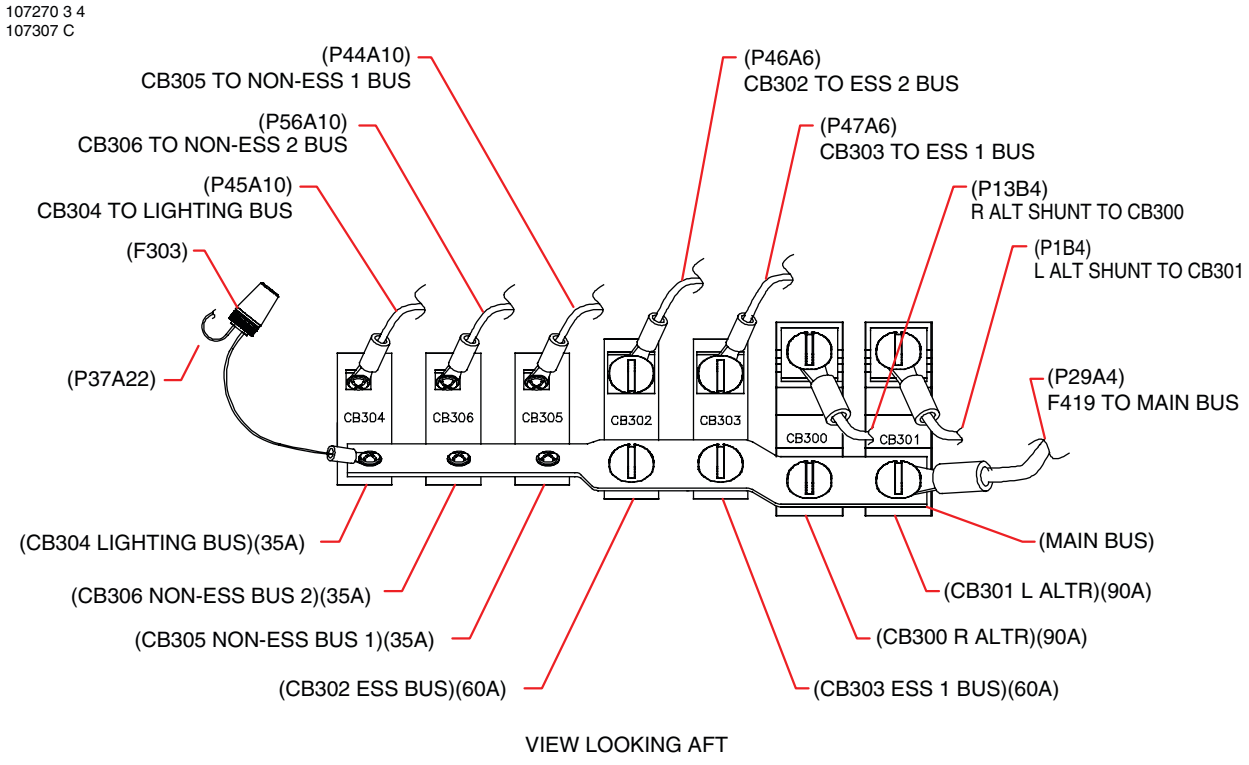
(c) Disconnect the electrical connectors and remove panel from aircraft.

(2) Installation

CAUTION: USE A MAXIMUM OF 10 ± 2 IN.-LBS. OF TORQUE WHEN INSTALLING THE SWITCH PANEL RETAINING SCREWS.

(a) Holding the switch panel below the opening in the headliner, connect the electrical connectors to the rear of the panel.

(b) Slide the panel into position and secure with screws (8). Torque screws to 10 ± 2 in.-lbs.



Tie Bus Panel Connections
Figure 3

Effectivity
Seneca V with Garmin G1000

B. Rocker-type Electrical Switches

CAUTION: ALTHOUGH SMALL SWITCH ASSEMBLIES ARE EASIER TO REMOVE IF WIRING IS FIRST DISCONNECTED, THE LIMITED WORK SPACE BEHIND THE PANEL CAN RESULT IN BURNED WIRE INSULATION. DO NOT ATTEMPT TO UNSOLDER THESE SMALL ELECTRICAL CONNECTIONS BEHIND THE INSTRUMENT PANEL UNDER ANY CONDITIONS. IF NECESSARY, CUT WIRES AT POINT OF CONNECTION. IN ANY CASE, IT'S BETTER TO DAMAGE THE SWITCH AND REPLACE IT, RATHER THAN DAMAGE THE WIRING HARNESS LEADS.

(1) Removal

(a) In airplanes equipped with Garmin G1000/G1000 NXi, disconnect the emergency battery P615 connector.

(b) Disconnect ground (negative) battery cable.

(c) Gain access to the switch from behind the instrument panel, or, in the Seneca V, by removing the overhead switch panel, see Removal above.

(d) Squeeze retainer blades on top and bottom of the switch together and push switch from the panel.

(e) Make note of the placement of, and/or tag, wires on the switch to facilitate installation.

(f) Disconnect wires from the switch. Remove switch.

(2) Installation

(a) Connect wires to the switch.

(b) Squeeze retainer blades on top and bottom of the switch together and push switch into panel until retainer blades engage the panel.

(c) Reconnect ground (negative) battery cable.

(d) In airplanes equipped with Garmin G1000/G1000 NXi, reconnect the emergency battery P615 connector.

C. Push ON - Push OFF Switches

A few Push ON - Push OFF switches are used in the instrument panel (Master Warn/Caution).

(1) Removal

(a) Gently pry off the switch cap.

(b) With a small screwdriver, rotate the two retaining tabs until they disengage.

(c) Slide the switch out.

(2) Installation

(a) Slide the switch into place.

(b) With a small screwdriver, rotate the two retaining tabs until they engage.

(c) Press on the switch cap.

4. Linear Modular Connectors (LMD/LMS)

Linear Modular Connectors (LMD/LMS) are used in some instrumentation and avionics installations to simplify assembly and reduce production costs. Special tools are needed to remove some modules in some installations. See Amphenol / Pyle publication LM-300 (LM-300-1) available from:

http://www.amphenol-aerospace.com/ServiceInstructions.asp

5. EMI and RFI Check

(PIR-PPS65119, Rev A.)

A. Background

ElectroMagnetic Interference (EMI) and Radio Frequency Interference (RFI) are a growing concern with the installation of advanced digital avionics. The following test procedure is recommended before return-to-service each time any electrical / avionics work is performed.

B. Procedure

(1) Position the aircraft at least 50 yards from buildings or any other large structures.

(2) Ensure all aircraft closeout panels (excluding interior trim panels) and engine cowling are installed on the aircraft being tested.

(3) With the aircraft running and all avionics, exterior lights and equipment ON, verify that:

(a) There are no unusual needle or display fluctuations on any display or gauge;

(b) There is no objectionable background noise in the headsets.

(c) For troubleshooting purposes, if either is present, systematically turn OFF equipment until the offending system is identified.

(4) RFI interference is typically generated by energy from the aircraft communication radios bouncing around the airframe. Verify that transmitting on the communication system does not cause RFI interference by performing the following test on each communication system.

(a) On each of the following frequencies key the microphone for 3 to 5 seconds:

121.150 MHZ	127.000 MHZ
131.250 MHZ	121.175 MHZ
121.200 MHZ	131.275 MHZ
123.000 MHZ	131.300 MHZ

(b) Verify that this does not cause any unusual needle or display fluctuations on any display or gauge.

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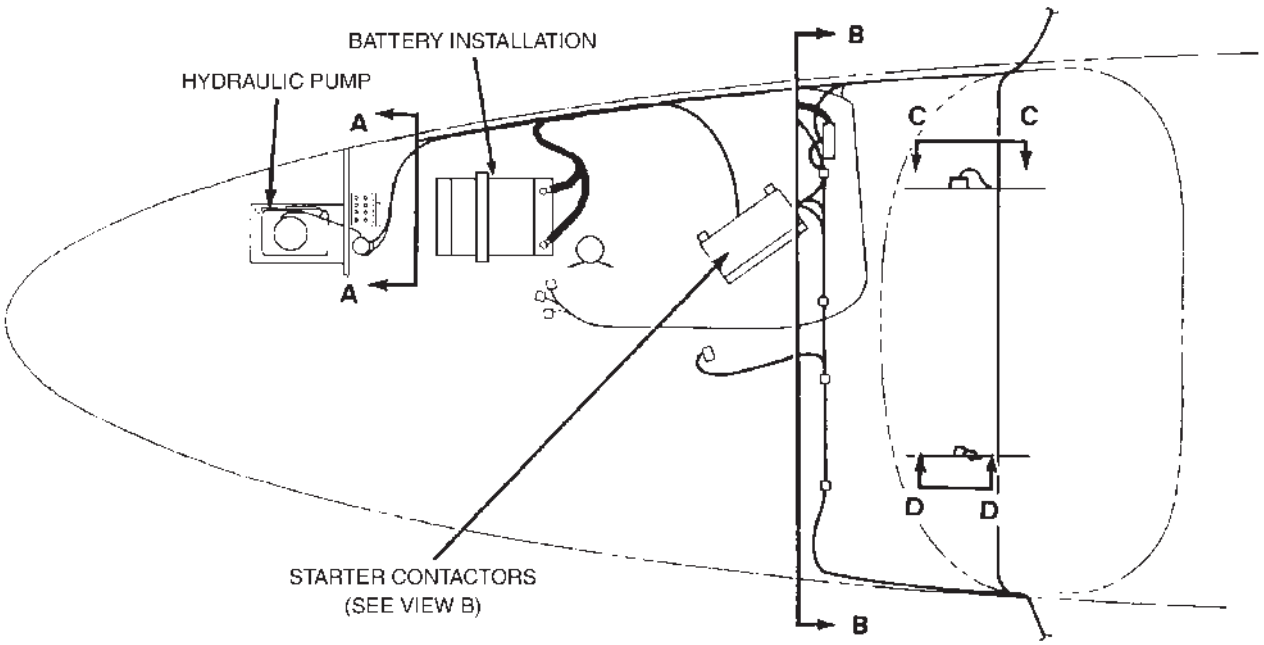
ELECTRICAL AND ELECTRONIC EQUIPMENT – SENECA IV

The following illustrations provide electrical and electronic equipment component locating information for the Seneca IV. See Section 39-22-00 for the Seneca V.

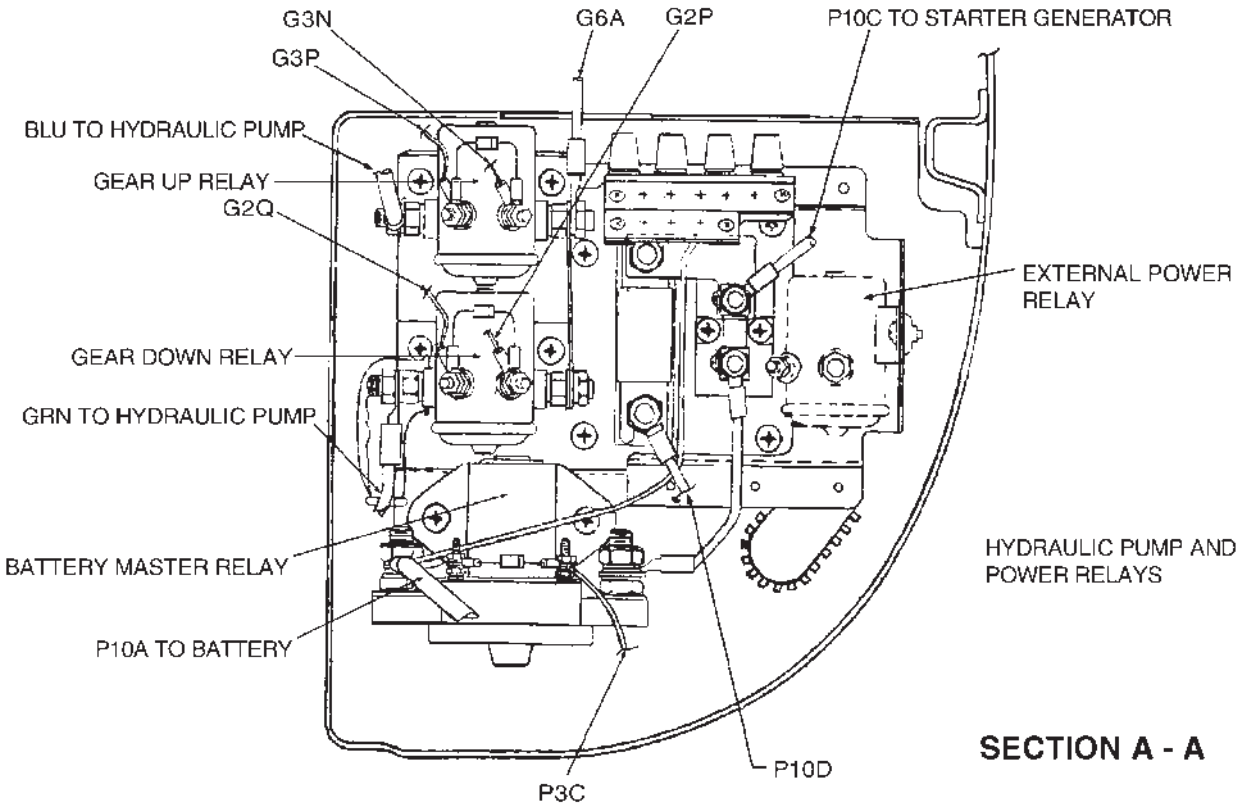
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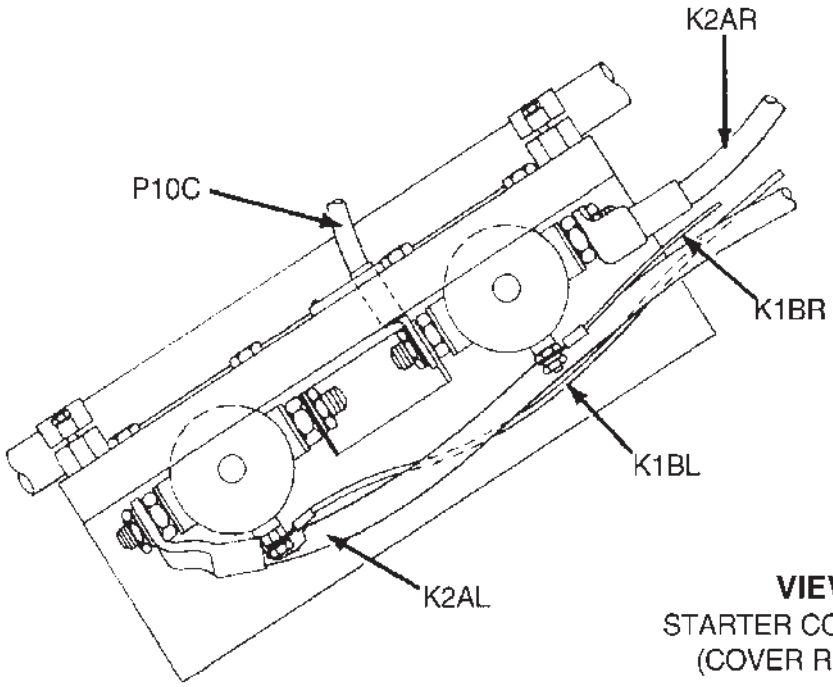
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Component Locator - Seneca IV
Figure 1

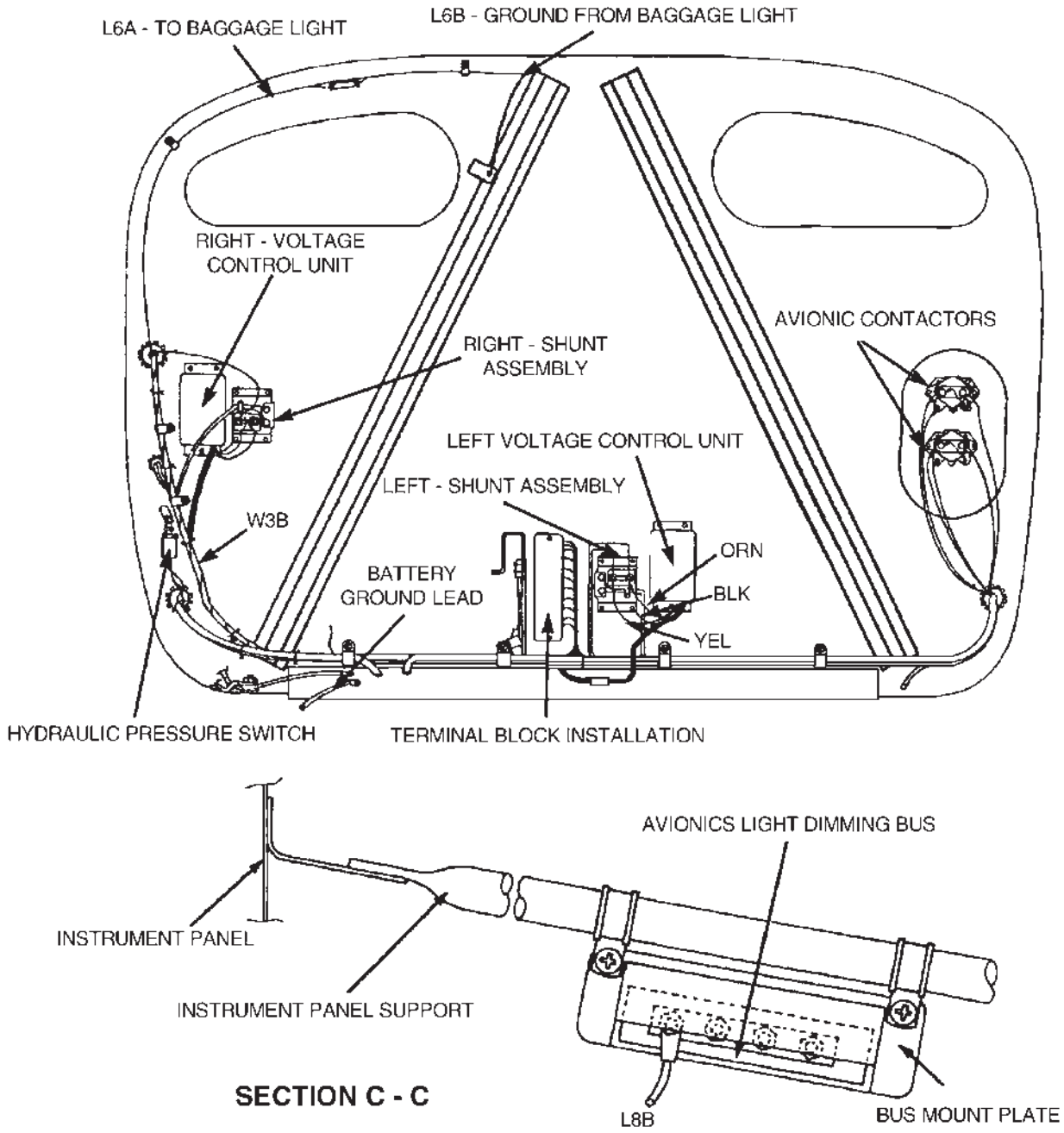


Component Locator - Seneca IV
Figure 2

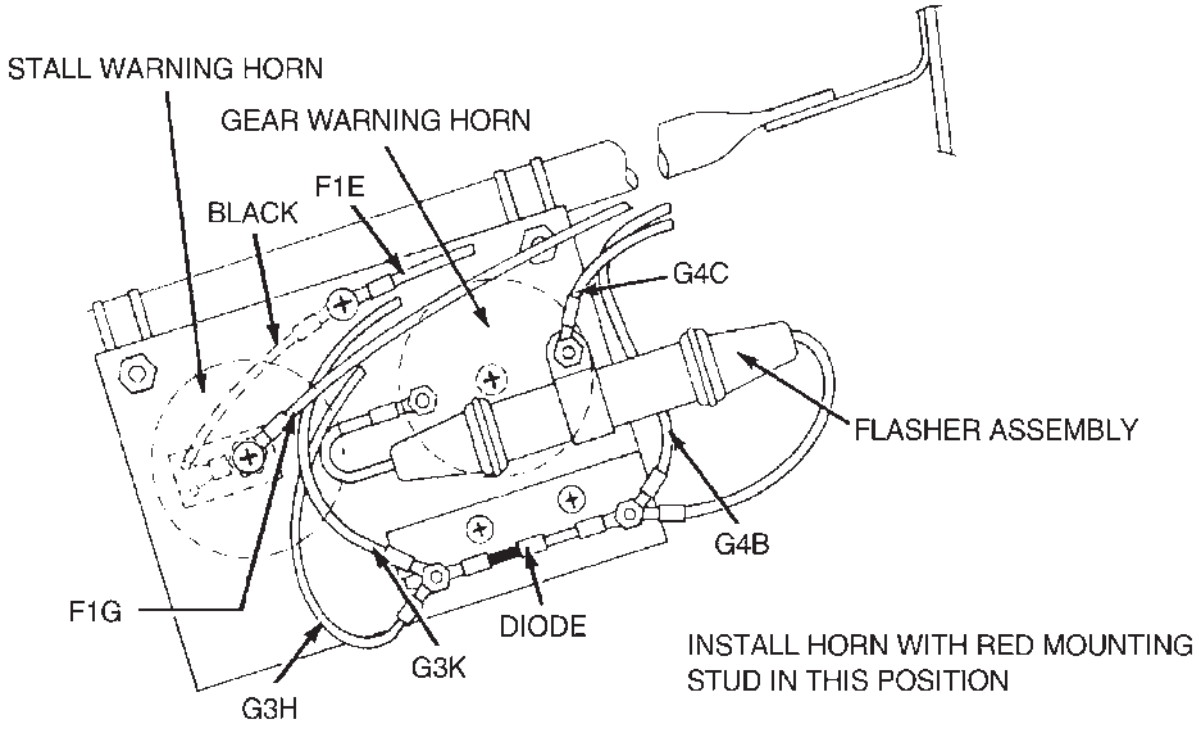


VIEW B
STARTER CONTACTORS
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Component Locator - Seneca IV
Figure 3



Component Locator - Seneca IV
Figure 4



SECTION D - D

Component Locator - Seneca IV
Figure 5

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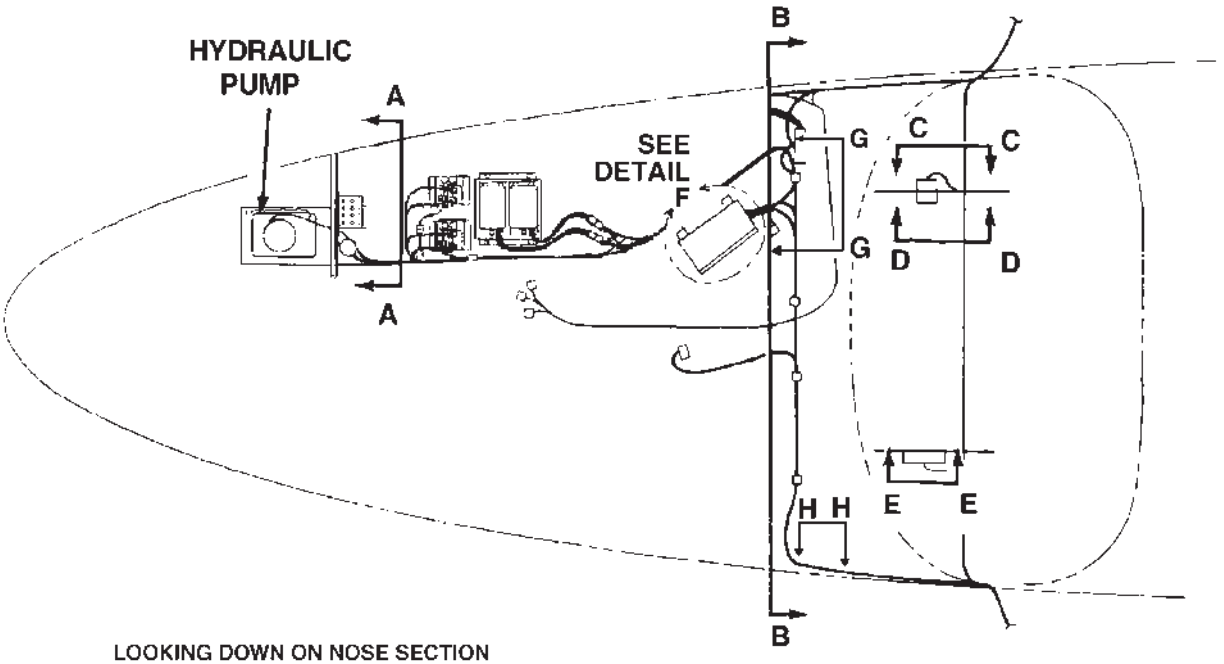
ELECTRICAL AND ELECTRONIC EQUIPMENT – SENECA V

The following illustrations provide electrical and electronic equipment component locating information for the Seneca V. See Section 39-21-00 for the Seneca IV.

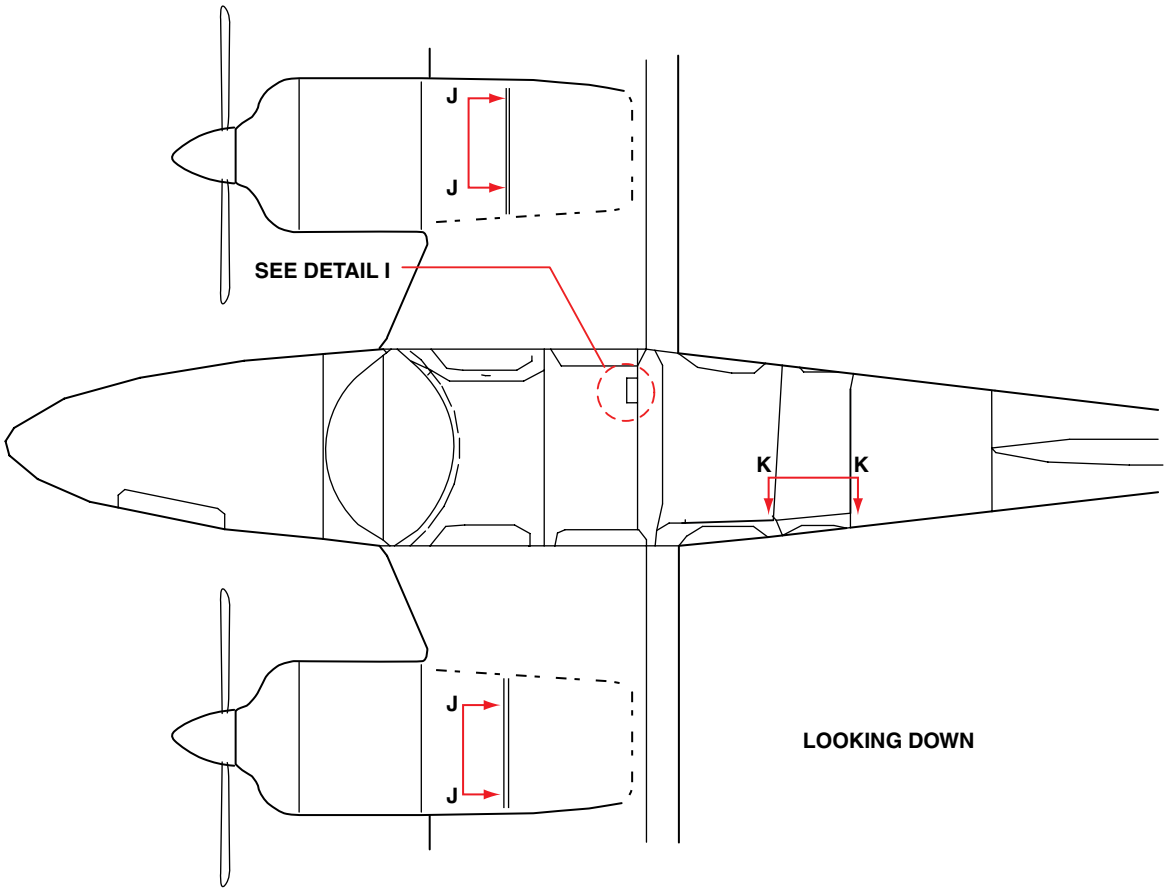
Index

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Avionics Dimming Bus	4
Avionics Light Dimmer	8
Battery Master Relay	6
Dual (Stall / Gear) Warning Horn	5
Exceedance Horn	5
External Power Relay	6
Fuel Pump Relay	9
Ground Clearance Relay	6
Hydraulic Pressure Switch	3
Hydraulic Pump	1
Landing Gear Relays	2
Panel Lights Dimmer	7
Pitot Heat Sensor Assembly	9
Post Lights Dimming Bus	4
Switch Lights Dimming Bus	4
Switch Lights Dimmer	7
Starter Contactors	5
Terminal Block	3

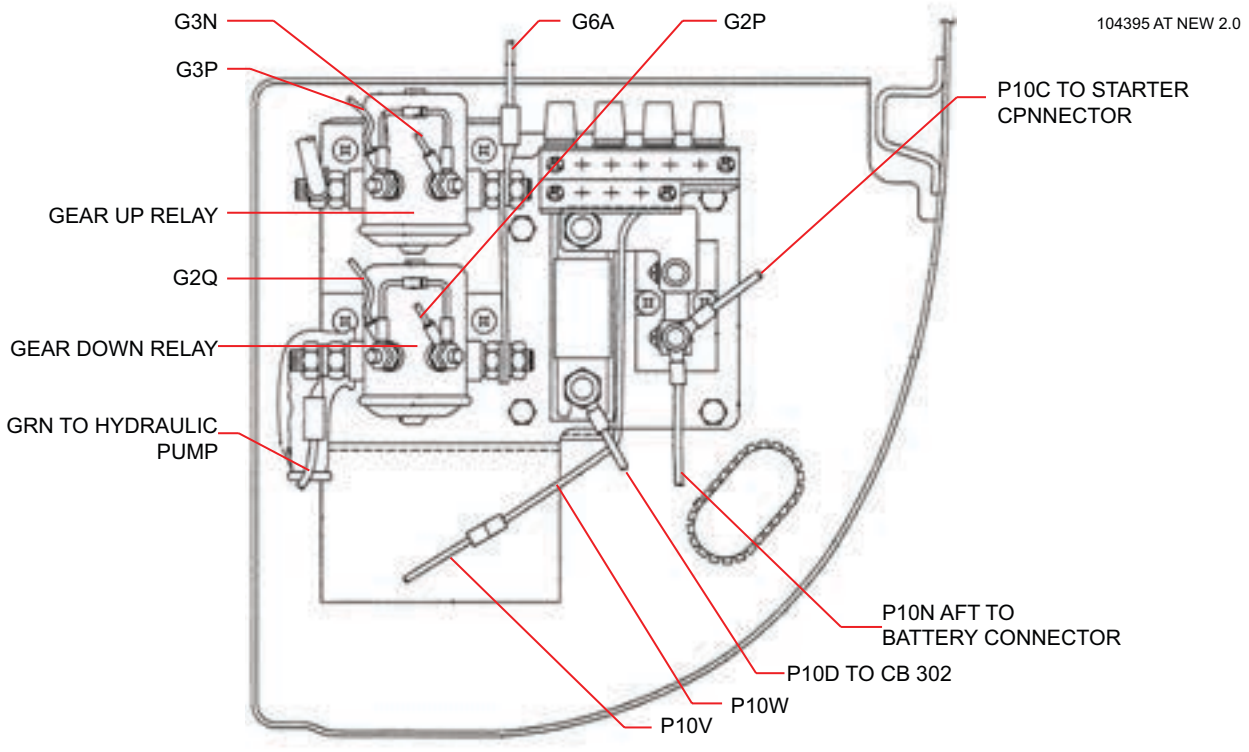
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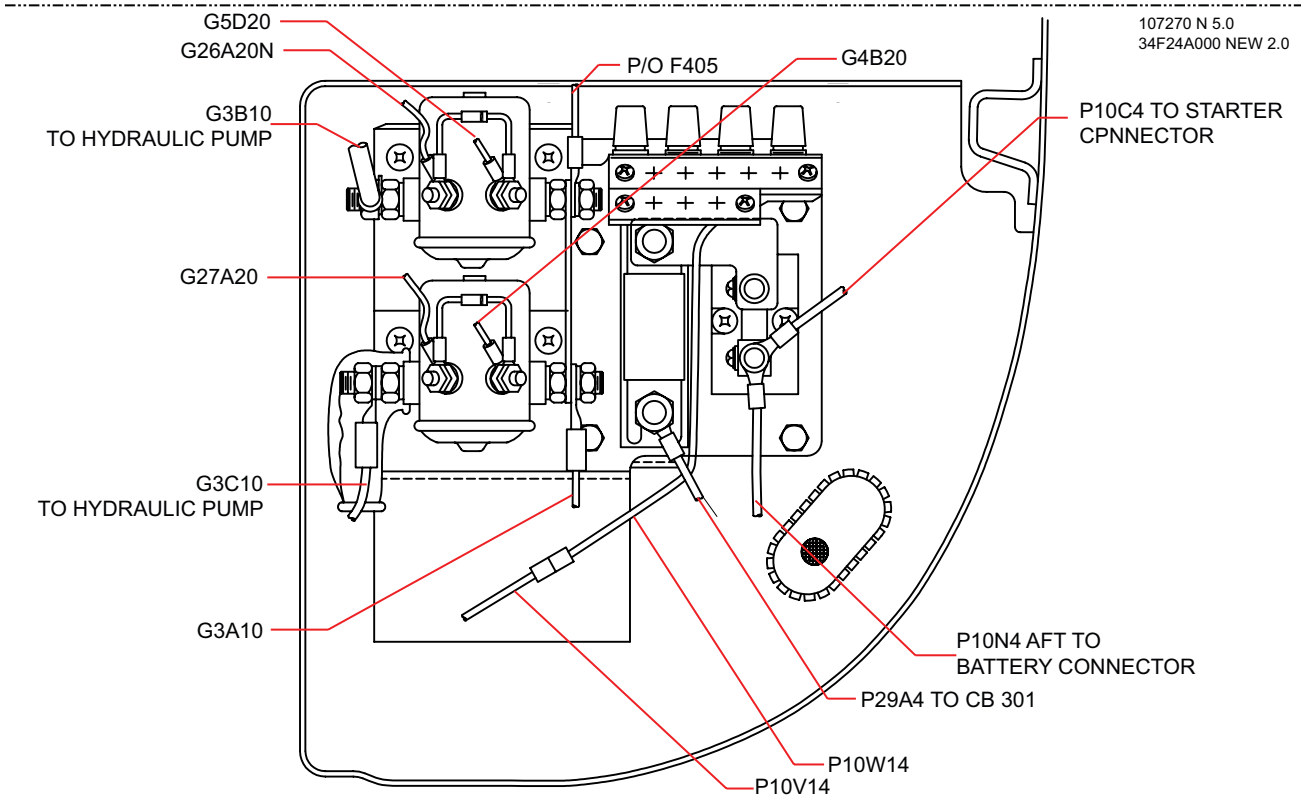
LOOKING DOWN ON NOSE SECTION



Component Locator - Seneca V
Figure 1



Effectivity S/N's 3449001-3449458, 3449460-3449466.

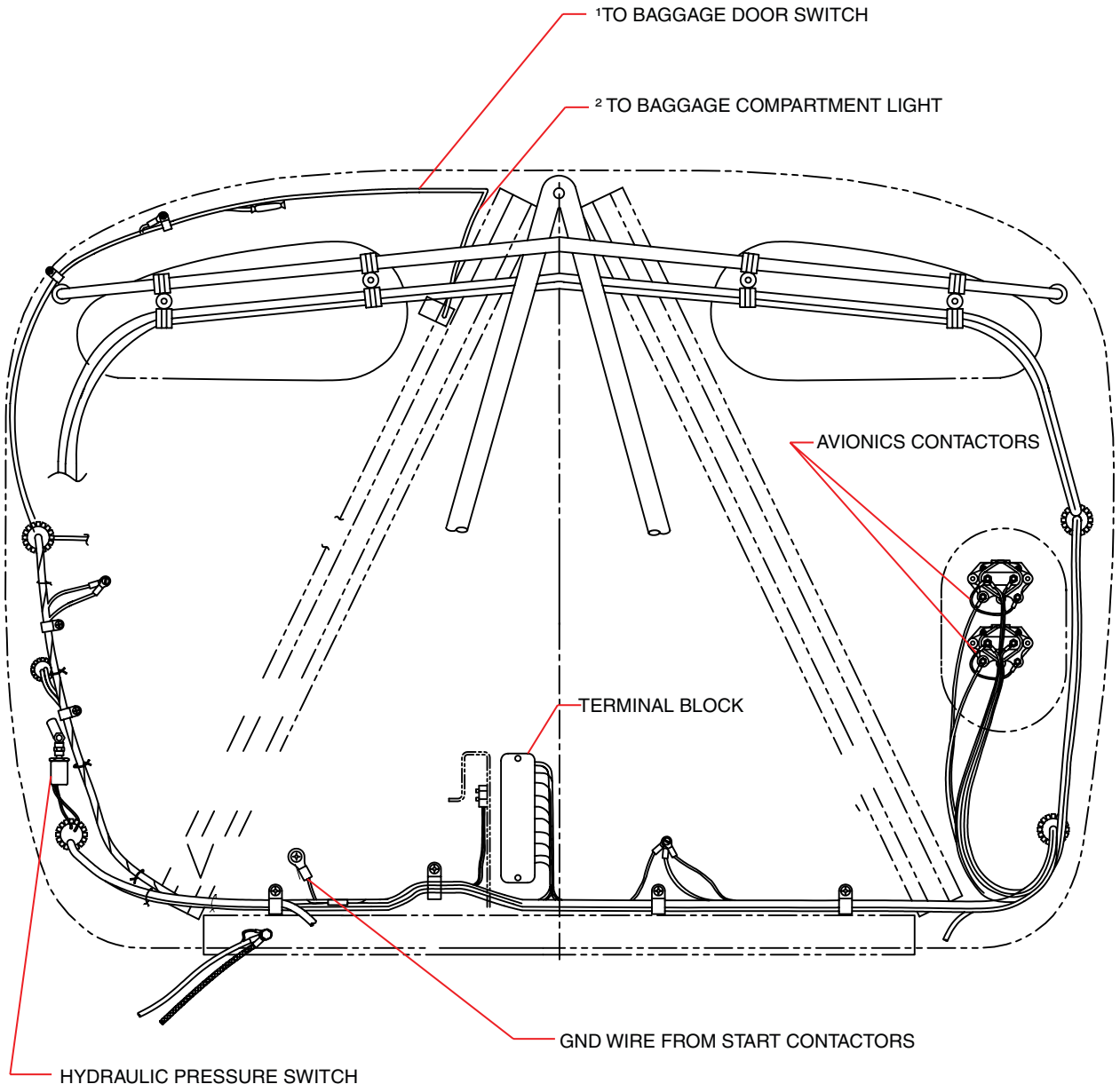


Effectivity Garmin G1000 or G1000 NXi, S/N's 3449459, 3449467 and up.

VIEW A - A

Component Locator - Seneca V
Figure 2

104395 AT 2.0
107270 N 5.0
34F24A000 NEW 2.0

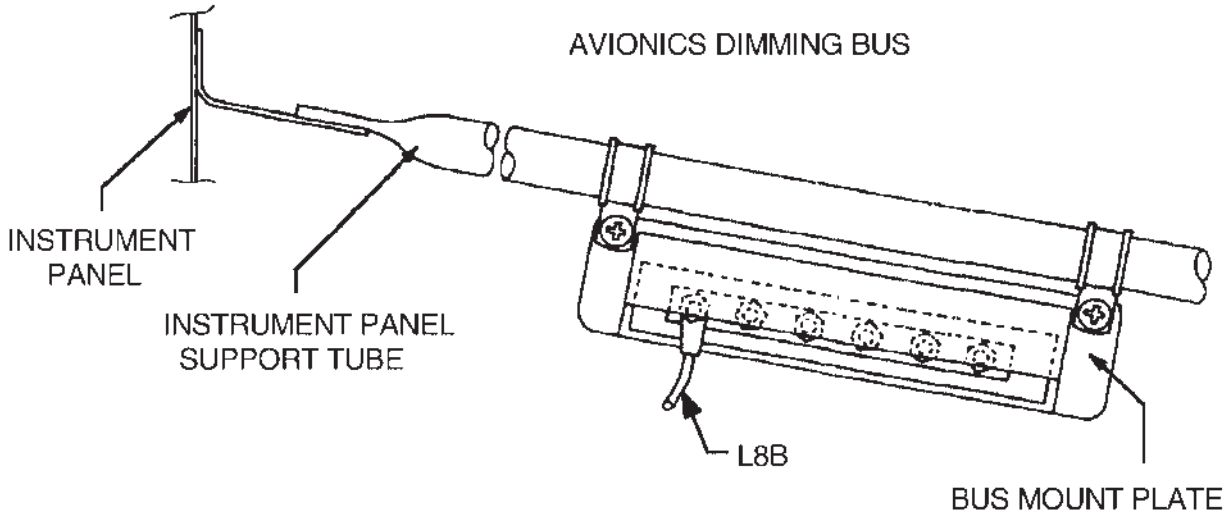


[1]L6B - S/N's 3449001-3449458, 3449460-3449466
[1]L35C20 - Garmin G1000 or G1000 NXi, S/N's 3449459, 3449467 and Up

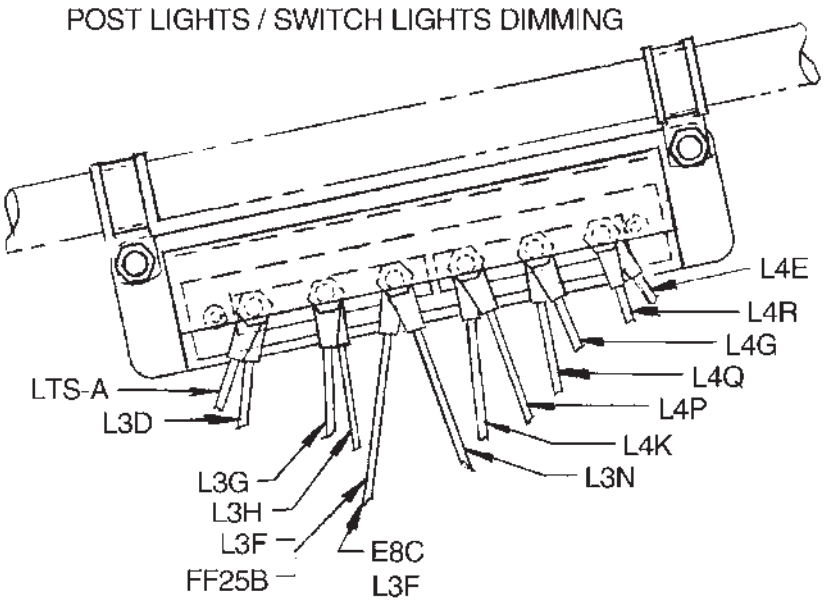
[2] L6A - S/N's 3449001-3449458, 3449460-3449466
[2] L36A20N - Garmin G1000 or G1000 NXi, S/N's 3449459, 3449467 and Up

VIEW B - B

Component Locator - Seneca V
Figure 3

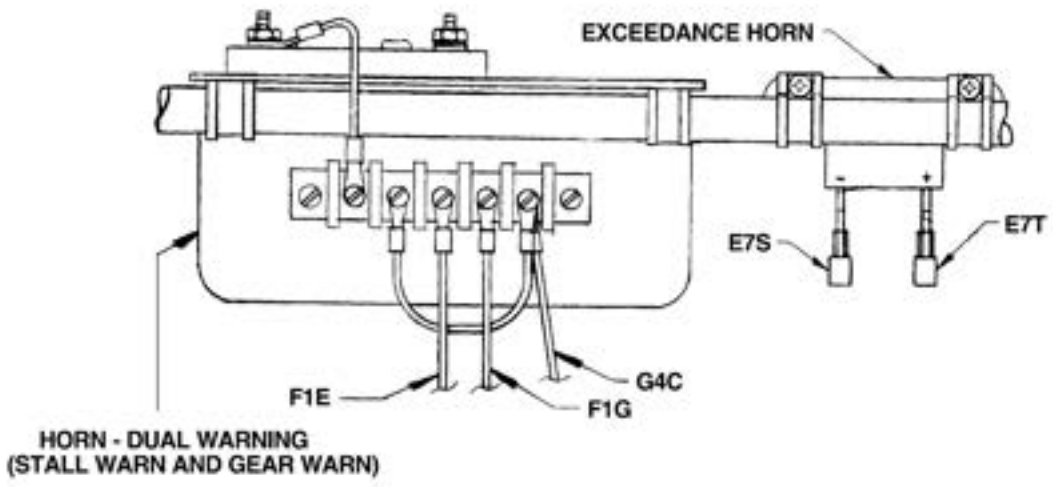


VIEW C - C

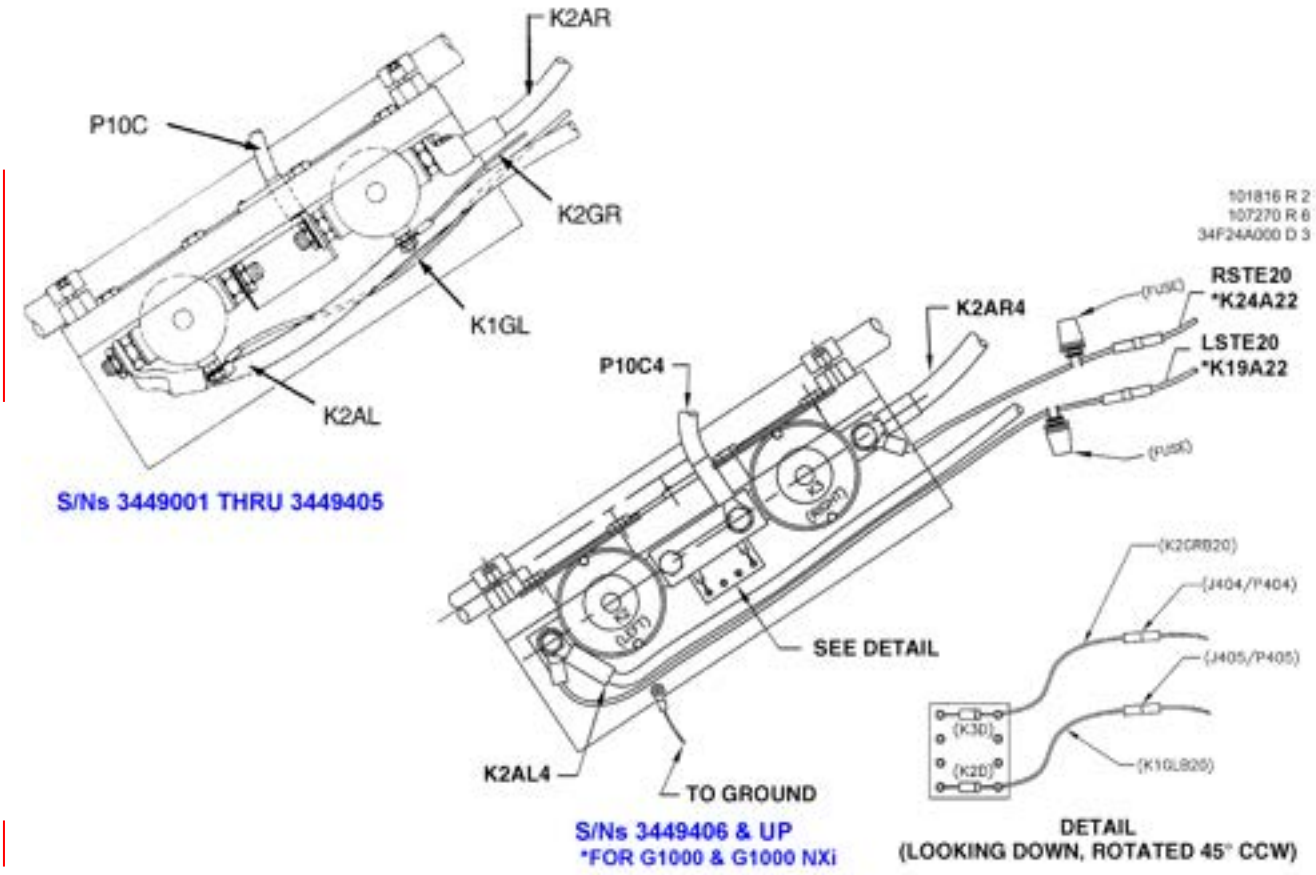


VIEW D - D

Component Locator - Seneca V
Figure 4



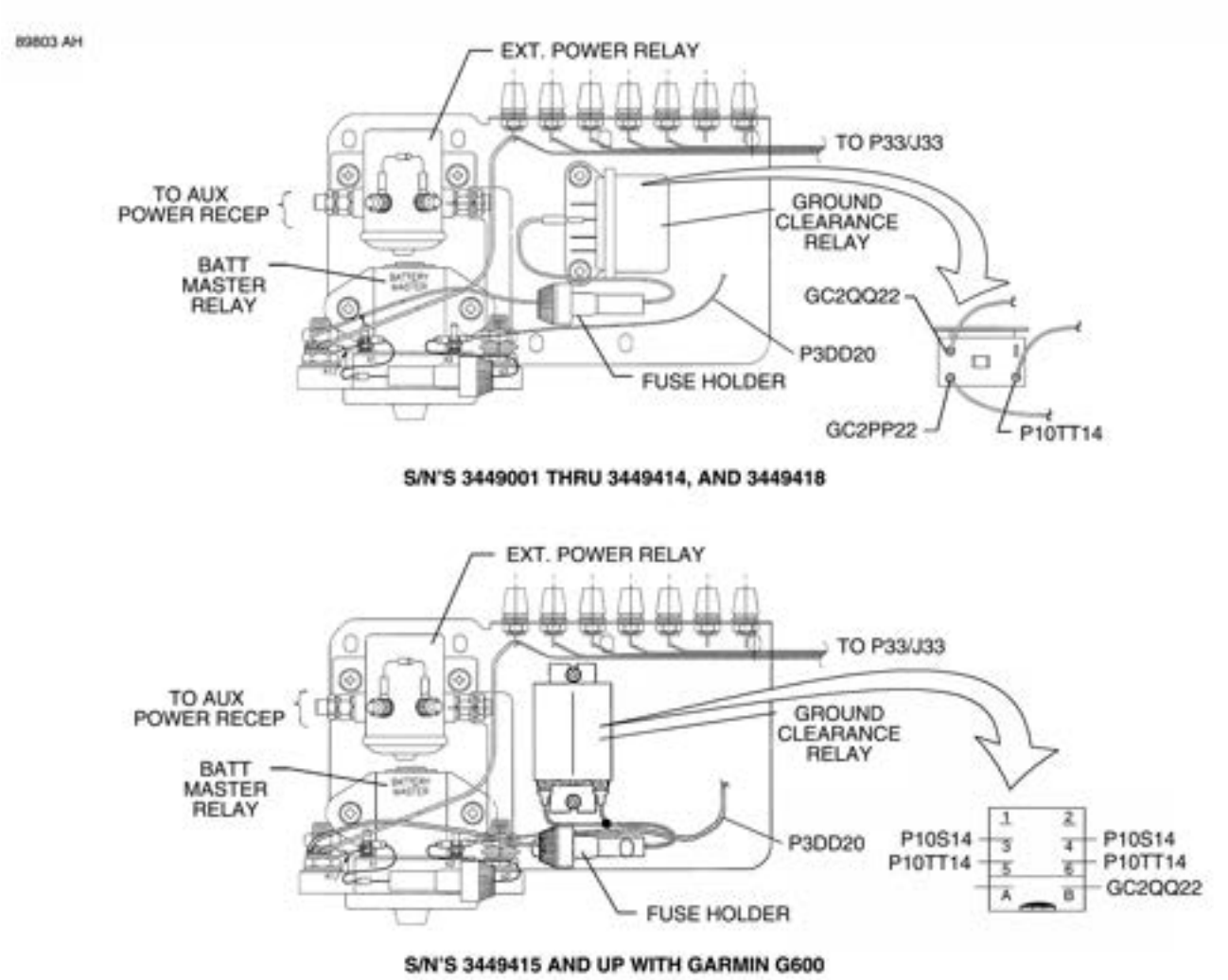
VIEW E - E



STARTER CONTACTORS
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DETAIL F

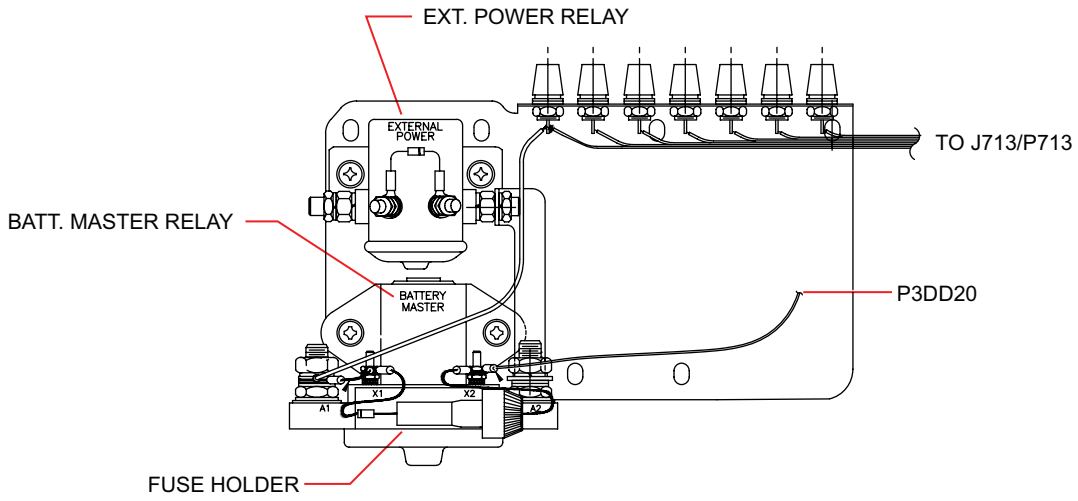
Component Locator - Seneca V
Figure 5



VIEWS K - K

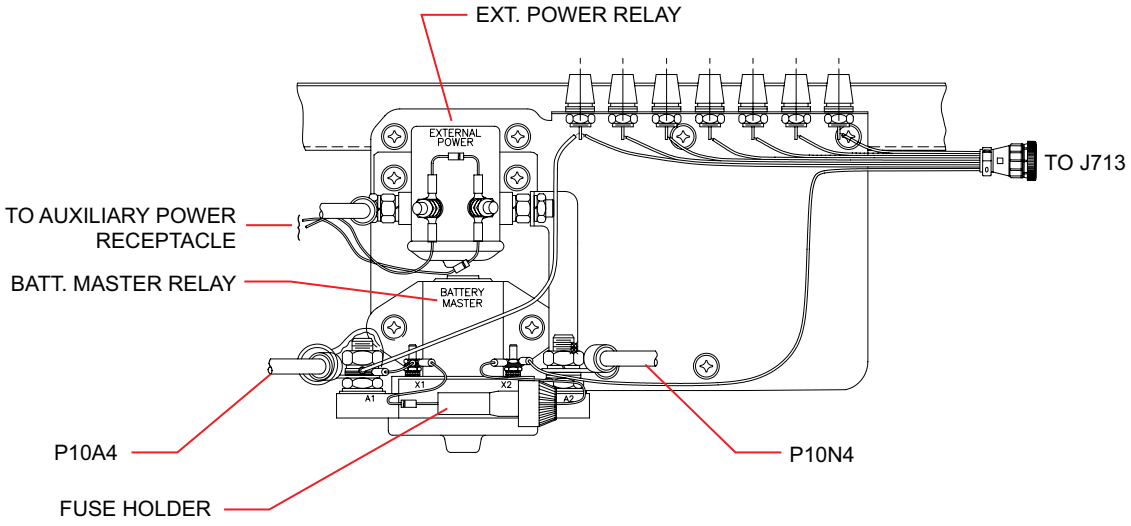
Component Locator - Seneca V
Figure 6 (Sheet 1 of 2)

101839 B 1



WITH AVIDYNE ENTEGRA ONLY

107270 N 6.0
34F24A000 NEW 3.0

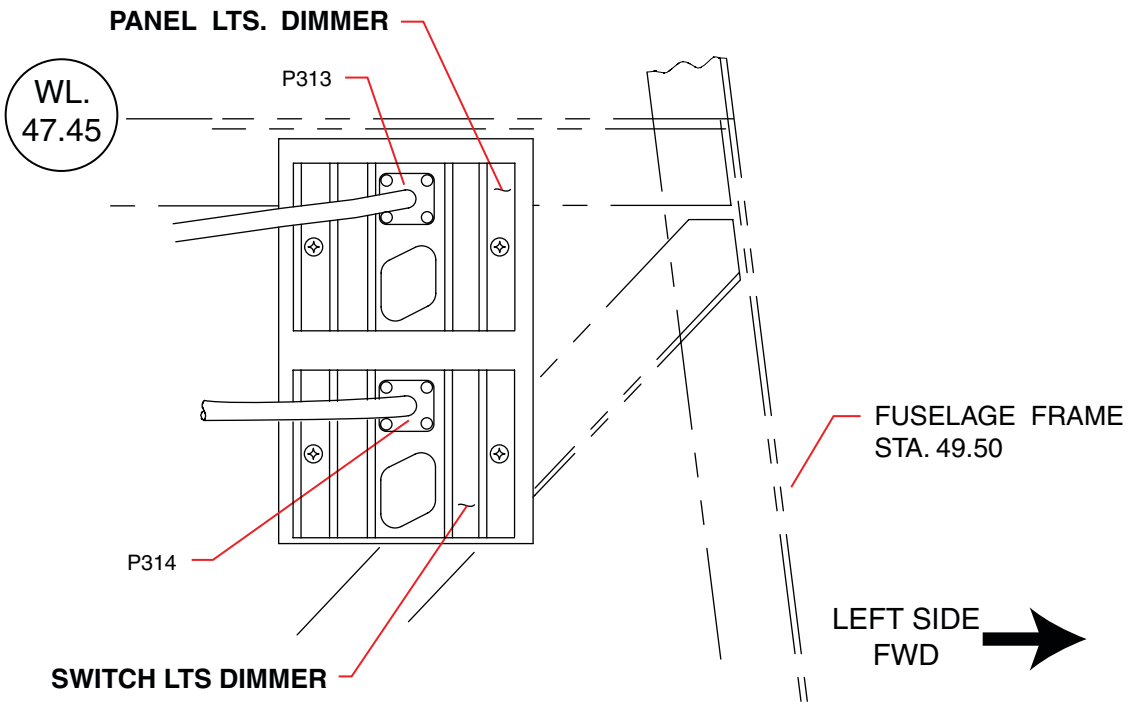


LHS LOOKING OUTBD
ROTATED 180°
Garmin G1000 / G1000 NXi, S/N's 3449459, 3449467 and Up

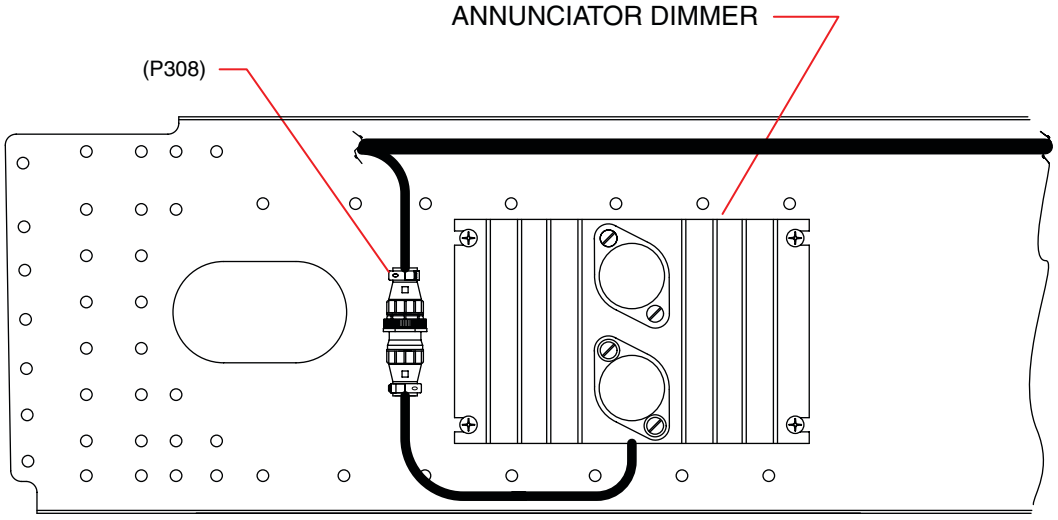
VIEWS K - K
(cont.)

Component Locator - Seneca V
Figure 6 (Sheet 2 of 2)

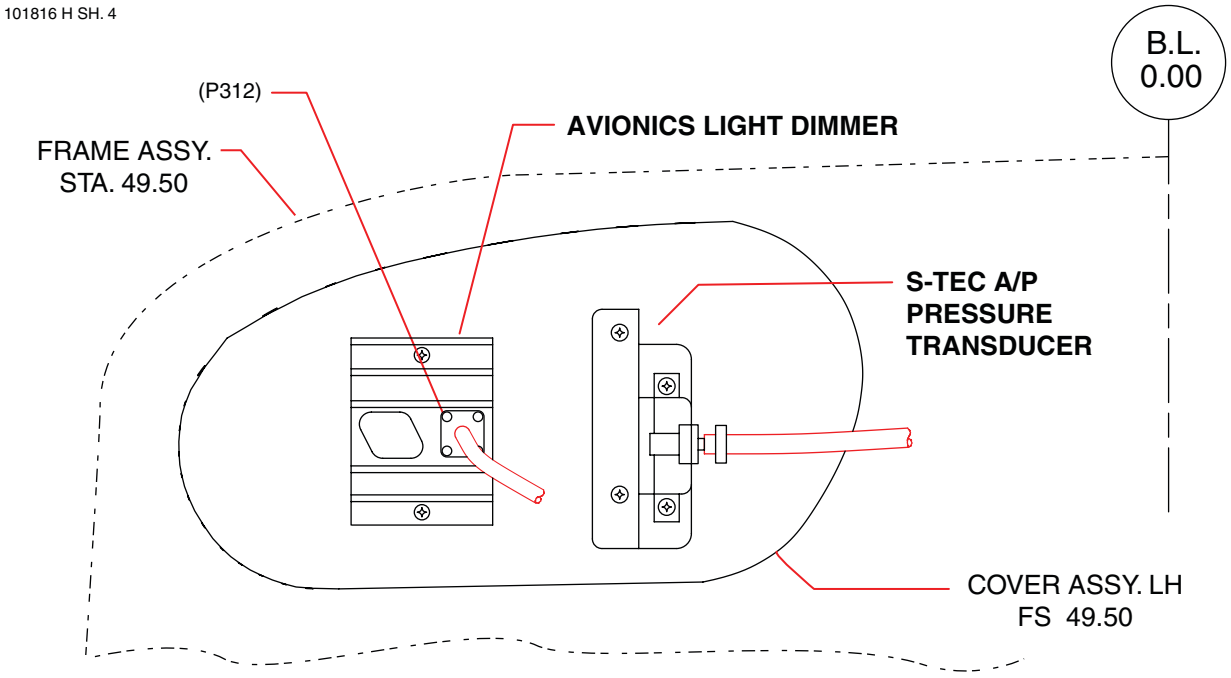
101816 H SH. 4



VIEW H-H



DETAIL I
RH WING SPAR LOOKING AFT



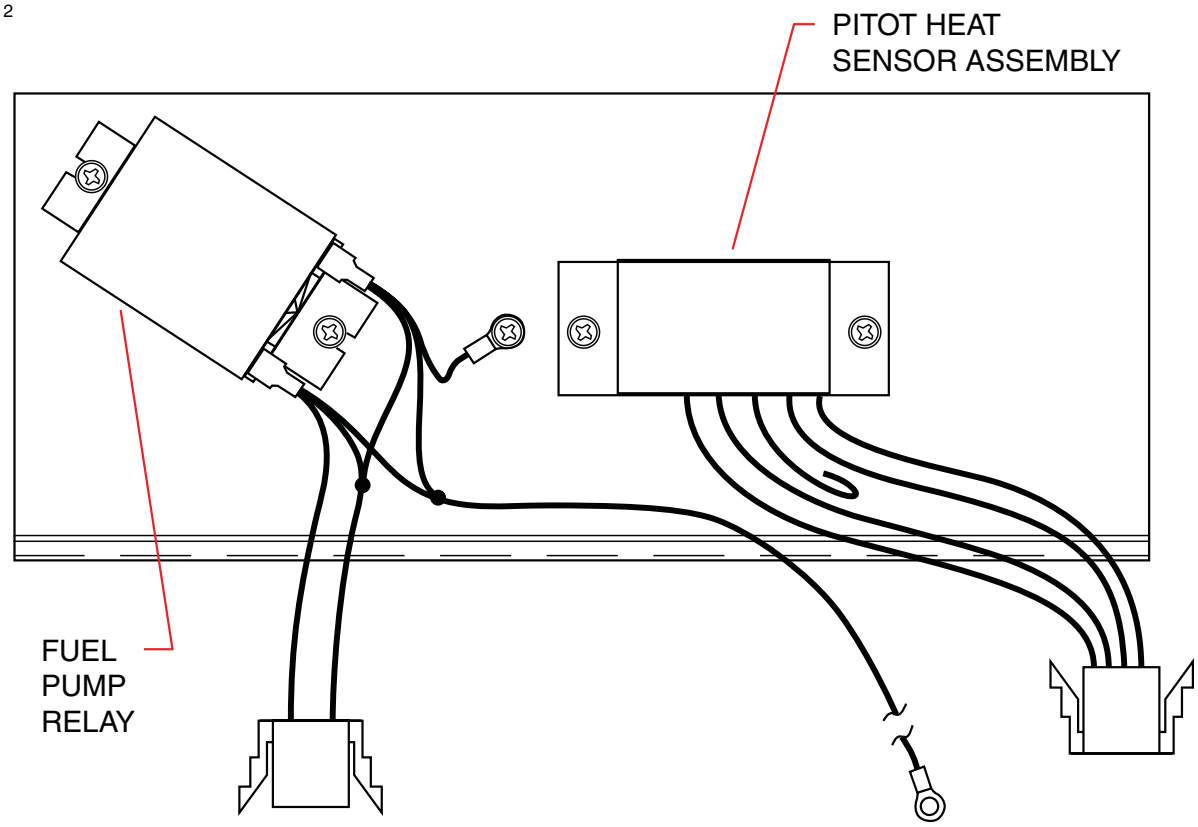
VIEW G - G
(LOOKING FORWARD)

Component Locator - Seneca V
Figure 8

Effectivity
with Avidyne Entegra

Effectivity
Seneca V

104395 AO SH. 4
85582 H SH. 2



VIEW J - J

Effectivity
3449161, 3449301 and up,
less Avidyne Entegra
equipped airplanes.

Component Locator - Seneca V
Figure 9

CHAPTER

51

STRUCTURES

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CHAPTER 51

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
51-List of Effective Pages	1	Oct 10/23			
	2	Sep 30/18			
51-Table of Contents	1	Oct 10/23			
	2	Sep 30/18			
51-00-00	1	Sep 30/18			
	2	Sep 30/18			
	3	Sep 30/18			
	4	Sep 30/18			
51-10-00	1	Sep 30/18			
	2	Sep 30/18			
	3	Sep 30/18			
	4	Sep 30/18			
	5	Sep 30/18			
	6	Sep 30/18			
51-70-00	1	Sep 30/18			
	2	Sep 30/18			
	3	Sep 30/18			
	4	Sep 30/18			
	5	Sep 30/18			
	6	Sep 30/18			
	7	Oct 10/23			
	8	Sep 30/18			
	9	Sep 30/18			
	10	Sep 30/18			
	11	Sep 30/18			
	12	Oct 10/23			
	13	Sep 30/18			
	14	Oct 10/23			
	15	Oct 10/23			
	16	Oct 10/23			
51-80-00	1	Sep 30/18			
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	3	Sep 30/18			
	4	Sep 30/18			

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CHAPTER 51 - STRUCTURES

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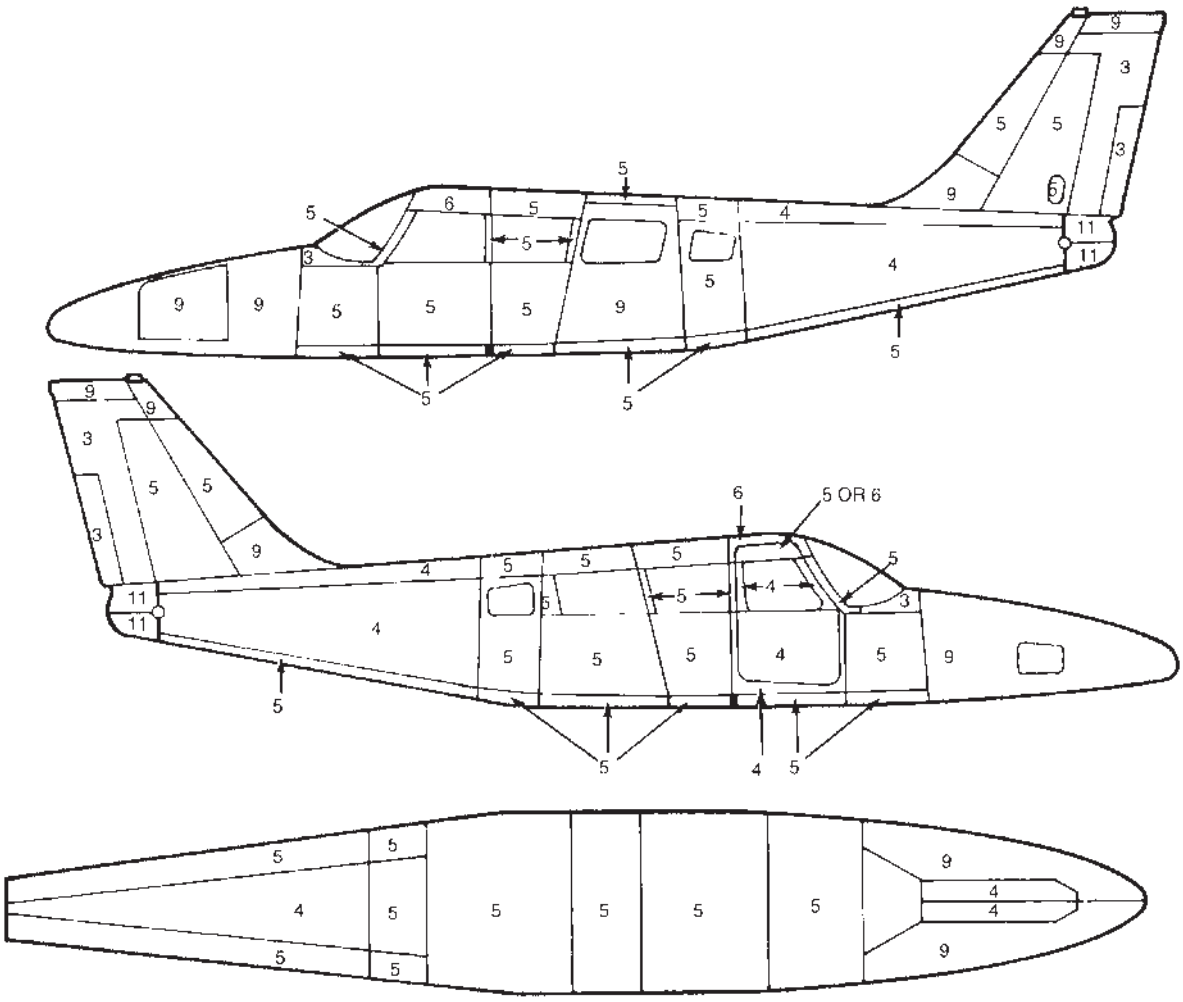
GENERAL

The PA-34-220T is an all metal semi-monocoque structure with a fuselage constructed of bulkheads, stringers and stiffeners, to which all of the outer skin is riveted. Windows include a single pane windshield and eight side windows; all windows are single pane. A storm window is located in the forward lower section of the left window and can be opened inward when the latch is released. There is a cabin entrance door located on the right side of the fuselage above the wing. Also, a door provided for entrance to the aft passenger compartment is located on the left side of the fuselage just aft of the wing trailing edge, with a baggage or cargo door adjacent and to the rear of the entrance door. Additionally, a forward baggage compartment door is located on the left side of the nose at F. S. 20.

Each wing is an all metal, full cantilever, semi-monocoque type construction, with a removable fiberglass or thermoplastic tip. Attached to each wing is an aileron, flap, main landing gear and power plant. The wings are attached to each side of the fuselage by inserting the butt ends of the main spars into a spar box carry-through. The spar box is an integral part of the fuselage structure which provides, in effect, a continuous main spar with splices at each side of the fuselage. There are also fore and aft attachments at the front and rear spars.

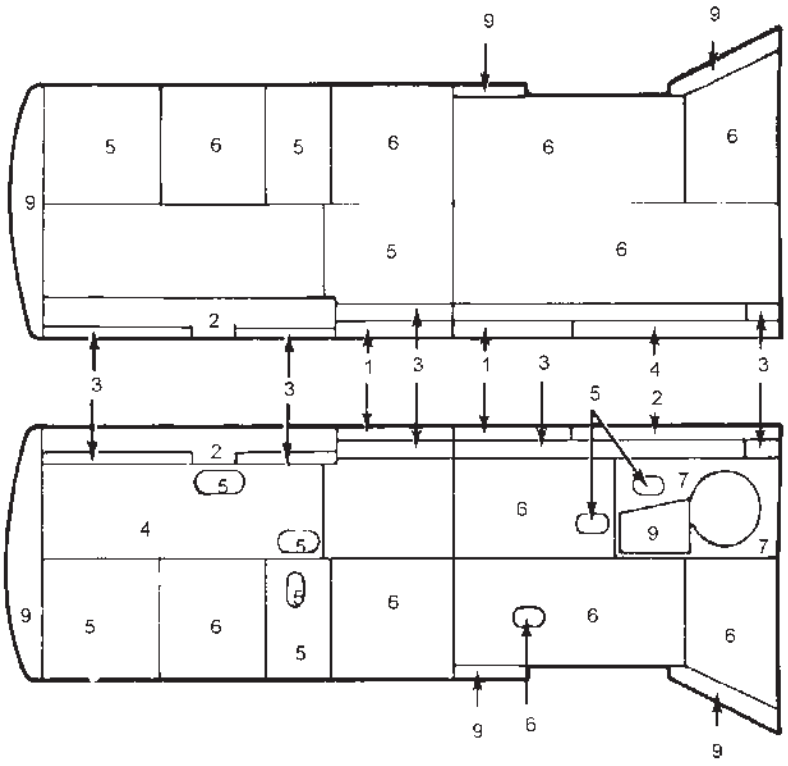
The all metal empennage group is a full cantilever design consisting of a vertical stabilizer (fin), rudder and stabilator, all with removable fiberglass or thermoplastic tips. The rudder and stabilator have trim tabs attached that are controllable from the cockpit. The stabilator also incorporates one channel main spar that runs the full length of the stabilator and hinges to the aft bulkhead assembly of the fuselage.

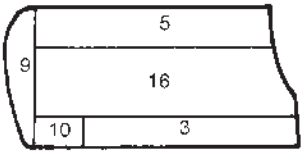
All exterior surfaces are coated with specially formulated primers and polyurethane overcoats. The standard airplane is completely primed with zinc chromate and/or other corrosion resistant paint processes.



NUMBER	MATERIAL	THICKNESS
1	2024-T3	.018
2	2024-0*	.020
3	2024-T3	.020
4	2024-T3	.025
5	2024-T3	.032
6	2024-T3	.040
7	2024-T3	.051
8	2024-T3	.025
9	FIBERGLASS	
10	2024-T3	.020
11	THERMOPLASTIC OR FIBERGLASS	
12	2024-0*	.025
13	5052-H34	.040
14	321A ST. STL.	.016

Skin Materials and Thickness
Figure 1 (Sheet 1 of 2)

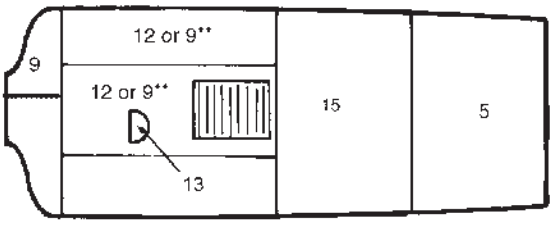


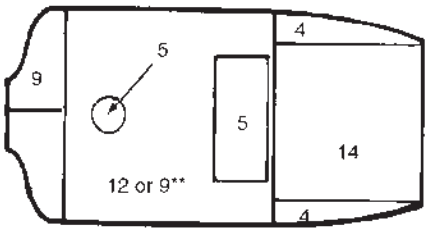


LEFT WING AND STABILATOR SHOWN,
RIGHT WING AND STABILATOR OPPOSITE



NUMBER	MATERIAL	THICKNESS
1	2024-T3	.018
2	2024-0*	.020
3	2024-T3	.020
4	2024-T3	.025
5	2024-T3	.032
6	2024-T3	.040
7	2024-T3	.051
8	2024-T3	.025
9	FIBERGLASS	
10	2024-T3	.020
11	THERMOPLASTIC OR FIBERGLASS	
12	2024-0*	.025
13	5052-H34	.040
14	321A ST. STL.	.016
15	2024- T3*	.025
16	2024-T3*	.018





* HEAT TREAT TO 2024-T42 AFTER FORMING
** FIBERGLAS ON SENECA V

Skin Materials and Thickness
Figure 1 (Sheet 2 of 2)

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INVESTIGATION, CLEANUP AND AERODYNAMIC SMOOTHNESS

Corrosion Control

Corrosion is the deterioration of metal by chemical or electrochemical attack. Water which is allowed to remain on the aircraft and industrial pollution are the major causes of corrosion in aircraft. The two general types of corrosion are:

-- Direct chemical attack. (i.e. spilled battery acid)

-- Electrochemical attack which requires a medium. (usually water)

The latter is the most common and is responsible for most forms of aircraft corrosion.

Since corrosion is a constant threat, the only effective method to control it is a routine of regular inspection, cleaning, and surface refinishing.

A. Forms of Corrosion

The following are the most common forms of corrosion:

(1) Surface corrosion appears as a general roughening or pitting on the surface usually accompanied by a powdery deposit of corrosion products. It may spread under the surface and not be recognized until the paint or plating is lifted off the surface in small blisters.

(2) Dissimilar metal corrosion may occur when two dissimilar metals are contacting each other. This type may be serious because it usually takes place out of sight. The only way to find it before structural failure is by disassembly and inspection. Insulating is necessary between two contacting dissimilar surfaces (2-3 coats of waterborne, chromated, fluid resistant, epoxy primer (i.e., PRC Desoto) or a coat of epoxy polyimide on each surface; plus a 0.003 thick piece of vinyl tape if one of the surfaces is magnesium).

(3) Intergranular corrosion is difficult to detect in its early stages. When severe, it causes the surface of the metal to exfoliate (flake or lift).

(4) Stress Corrosion is the result of sustained tensile stresses and corrosive environment. It usually occurs in assemblies such as aluminum alloy bellcranks with pressed in bushings; landing gear shock struts with pipe thread grease fittings, clevis pin joints and shrink fit parts.

(5) Fretting Corrosion takes place when two parts rub together, constantly exposing fresh active metal to the corrosive effects of the atmosphere

(6) Filiform Corrosion is the appearance of numerous meandering thread-like filaments of corrosion on the surface of various types of metal.

B. Conditions Affecting Corrosion

Some conditions which affect the occurrence of corrosion are:

(1) Heat and humidity increase corrosion.

(2) Different metal and their relative sizes affect resistance or susceptibility to corrosion.

(3) Frequent contributing factors to corrosion are:

(a) Soil and atmosphere dust.

(b) Oil, grease and exhaust residues.

(c) Salt water and salt moisture condensation.

(d) Spilled battery acids and caustic cleaning solutions.

(e) Welding, brazing and soldering flux residue

A clean aircraft will resist corrosion better than a dirty one. Cleaning frequency depends on several factors (such as geographical location, type of operation, etc). Soil should be removed as soon as possible, especially when it is on a high temperature area.

CHART 1
THERMOPLASTIC REPAIR LIST OF MATERIALS

Type of Material	Type of Corrosion	Remedy (2)
Steel.	Rust (1).	Complete removal of corrosion by mechanical means.
Aluminum.	White to grey powdery material.	Mechanical polishing or brushing with material softer than aluminum.
Magnesium (highly susceptible to corrosion).	White powdery snow-like mounds and white spots.	Mechanical polishing or brushing for a smooth finish.
Cadmium (plating).	White to brown to black mottling of surface (plating is still protecting until iron appears).	Mechanical removal of corrosion is limited to metal surfaces from which cadmium has been depleted.
Chromium (plating).	May pit in chloride environment.	Polishing and buffing.

— NOTES —

(1) Red rust generally shows on bolts, nuts, and other aircraft hardware. Rust in these areas is generally not dangerous, however, it shows a need for maintenance and the possibility of corrosive attack in more critical areas. Any surface corrosion on highly stressed steel parts is potentially dangerous. A careful removal of corrosion using mild abrasives (rouge or fine grit aluminum oxide paper) is necessary. Do not overheat metal when removing corrosion.

(2) For abrasion, do not use dissimilar material (for example steel wool on aluminum). Remove only material required to clean affected area.

After cleaning, ensure that no cleaning solution remains in any holes, crevices or joints, as it may lead to increased corrosion. Also, all exposed areas (landing gear, flap tracks, control surface, hinge parts, etc) should be lubricated after cleaning.

C. Corrosion Inspections

Corrosion should be inspected for at every inspection. In trouble areas, the inspection frequency should be increased.

In addition to routine inspections:

(1) Aircraft operating around a marine environment should be given special checks on a weekly basis.

(2) Aircraft operating in a semi-acid condition should be inspected monthly. A semi-acid condition is likely to occur in industrialized areas where sulfur bearing particles in dust, smoke and smog attack painted surfaces.

(3) Inspections for corrosion should be performed by personnel familiar with corrosive problems and their remedies.

(a) Daily and preflight inspections should include the engine frontal areas, all intake vents, engine compartments, gaps, seams, and faying surfaces in the exterior skins, wheel and wheel well areas, battery compartment, fuel cell and all other drains, and any bilge areas not requiring extensive removal of inspection access covers.

(b) Detailed inspection should include the above referenced areas along with areas requiring removal of screw attached inspection plates and panels to thoroughly inspect the internal cavities of the aircraft.

(4) During inspection remember that paint tends to hide corrosion in its initial stages. However, the results of corrosion can sometimes be seen as blisters, flakes, chips and other irregularities in the paint.

D. Corrosion Removal and Control

CAUTION: REMOVAL OF "SEVERE CORROSION" MAY BE CONSIDERED A MAJOR REPAIR. ANY REPAIR OF THIS TYPE MUST BE APPROVED BY THE FAA BEFORE RETURNING THE AIRPLANE TO SERVICE.

Corrosion cannot be prevented or eliminated on aircraft; it can only be reduced to an acceptable level by proper control methods.

All corrosion products must be removed prior to refinishing. If they are not removed, corrosion will begin again, even though the affected area is refinished.

(1) Before beginning any rework:

(a) Position the airplane in a wash rack or provide some type of washing apparatus for rapid rinsing of all surfaces.

(b) Connect a static ground line to the airplane.

(c) Remove the airplane battery if required.

(d) Protect the pitot-static ports, engine openings, airscoops, louvers, wheels, tires and other portions of the airplane from moisture and chemical brightening agents.

(e) Protect the surfaces next to the rework areas from chemical paint strippers, corrosion removal agents and surface treatment materials.

(2) An evaluation of the corrosion damage is necessary to determine the type and extent of repairs required.

The following are general guidelines:

(a) Light Corrosion: Discoloration or pitting; normally removed by light hand sanding or a small amount of chemical treatment.

(b) Moderate Corrosion: Similar to light corrosion except there could be some blistering or evidence of scaling and flaking. Removed by extensive hand sanding or mechanical sanding.

(c) Severe Corrosion: Similar to moderate corrosion with severe blistering, exfoliation, scaling and / or flaking, normally removed by extensive mechanical sanding or grinding.

NOTE: The depth of material removed should not exceed safe limits.

E. Corrosion Prone Areas

Certain areas are more prone to corrosion than others. The following list is intended to be a general guide to areas where corrosion is frequently found.

(1) Areas around steel fasteners are susceptible to corrosion. The paint on these areas cracks which allows moisture to seep in and corrode the underlying metal. Each time the fastener is removed, it should be coated with fluid resistant epoxy primer before reinstallation. The paint should be wet when the fastener is installed.

(2) Fluids tend to seep into faying surfaces, seams and joints due to capillary action. The effect of this type of intrusion is usually detectable by irregularities in the skin's surface.

(3) Spot welded assemblies are particularly prone to corrosion. The only means to prevent this type of corrosion is by keeping potential moisture entry points in the spot weld filled with a sealant or preservative compound. On an aluminum spot welded assembly, a chromate conversion coating before paint is applied will help prevent corrosion.

(4) Areas which are exposed to exhaust gases may have their finish damaged by deposits. These deposits may result in an aggressive attack on the metal by corrosion. Heat from the exhaust may also blister or otherwise damage the paint. Gaps, seams, hinges and fairings are some places where exhaust gas deposits may be trapped and not reached by normal cleaning methods.

(5) The wheels and landing gear are the most exposed parts of the aircraft. Due to the complexity of its shape, assemblies and fittings, maintaining a protective coverage is difficult. The especially troublesome areas are:

(a) Magnesium wheels: around bolt heads, lugs and wheel well areas:

(b) Exposed rigid tubing, B-nuts, ferrules, under clamps and tubing identification tape:

(c) Exposed electrical equipment:

(d) Crevices between stiffeners, ribs and lower skin surfaces.

(6) Flaps, flight control slots and equipment installed in these areas may corrode unnoticed unless a careful surveillance is maintained.

(7) Engine frontal areas, air inlet ducts and the leading edge of wings, because they are constantly exposed to abrasion by dirt, dust, gravel and rain, should be checked frequently for the beginning of corrosion.

(8) Hinges (piano hinges especially) are extremely vulnerable to corrosion due to the wearing away of their protective coating and their being a natural trap for dirt, salt and moisture.

(9) Control cables may have bare spots in their preservative coating which could lead to corrosion. Cables having external corrosion should be checked for internal corrosion. If internal corrosion is present, replace the cable. If only external corrosion is present, remove corrosion with a wire brush and recoat cable with preservative.

(10) Any area where water may be trapped is a trouble spot for corrosion. Drain holes should be checked and cleaned regularly.

(11) Battery compartment and vent openings are particularly prone to corrosion due to spilled electrolyte. Fumes from overheated battery electrolyte will spread to adjacent areas and cause rapid corrosion of unprotected surfaces. Frequent cleaning and neutralization of deposits will minimize corrosion from this cause.

(12) Due to magnesium parts being prone to corrosion, special attention should be given to their surface treatment, proper insulation (due to dissimilar metal corrosion) and paint coatings.

(13) Electrical components and connectors should be checked. Their inspection frequency should be based on their operational environment and past trouble with them.

(14) Skin joints and lap-overs are two areas which may contain moisture. Corrosion in these areas may go unnoticed unless particular attention is paid to them during inspection.

(15) Hoses having an internal wire braid which are located in a position where they are frequently water soaked need a protective treatment.

(16) Drilled holes and the trimmed end of sandwich panels should be protected. An inhibitor solution and/ or sealant application is recommended. Any gaps or cavities which allow dirt or moisture to enter should be filled with a sealant.

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REPAIRS

WARNING: NO ACCESS HOLES ARE PERMITTED IN ANY CONTROL SURFACE.

WARNING: USE OF PATCH PLATES FOR REPAIRS OF ALL MOVABLE CONTROL SURFACES IS PROHIBITED. USE OF ANY FILLER MATERIAL NORMALLY USED FOR REPAIR OF MINOR DENTS AND/OR MATERIALS USED FOR FILLING INSIDE OF SURFACES IS ALSO PROHIBITED ON ALL MOVABLE CONTROL SURFACES.

CAUTION: CONTROL SURFACE SKINS MUST BE REPLACED IF THEY SUSTAIN DAMAGE OR EXHIBIT CRACKS.

Structural repair methods used for minor repairs must be in accordance with FAA Advisory Circular 43.13-1, latest revision. To assist in making repairs, “Figure 1” on page 51002, identifies type and thickness of various skin material used.

Repairs to areas defined in FAR Part 43, Appendix A, must be shown (using approved engineering data) to not diminish strength or the Fatigue Life of the component, if a life limit is stated herein (see 4-00-00) or in the aircraft type certificate data sheet (TCDS).

When a repair is proposed, it is the responsibility of the repairer per AC 43.13-1 to determine that the proposed repair is not contrary to manufacturer’s data. The repairer or aircraft owner or his agent should contact Piper directly to determine that a proposed repair is not in conflict with minimum type design capability.

Temporary repairs, when required, must add Instructions for Continued Airworthiness (ICA) to the maintenance record. Any such ICA must be based on approved data.

1. Fiberglass Repairs

The following procedures describe methods for repair of Fiberglass Reinforced Structures; Fiberglass Touch-Up, and Surface Repairs such as blisters, open seams, delamination, cavities, small holes and minor damages that have not harmed fiberglass cloth material; and, Fiberglass Fracture and Patch Repairs such as puncture, breaks and holes that have penetrated through structure and damaged fiberglass cloth. A repair kit, part number 766-222, furnishes necessary material for such repairs, and is available through Piper Dealers.

NOTE: Very carefully follow resin and catalyst mixing instructions furnished with repair kit.

A. Fiberglass Touch-Up and Surface Repairs

(1) Remove wax, oil and dirt from around damaged area with acetone, Methylethylketone or equivalent and remove paint to gel coat.

(2) Damaged area may be scraped with a fine blade knife or a power drill with a burr attachment to roughen bottom and sides of damaged area. Feather edge surrounding scratch or cavity. Do not undercut edge. (If scratch or cavity is shallow and penetrates only surface coat, continue to paragraph (8), below.)

(3) Pour a small amount of resin into a jar lid or on a piece of cardboard, just enough to fill area being worked on. Mix an equal amount of milled fiberglass with resin, using a putty knife or stick. Add catalyst, according to kit instruction, to resin and mix thoroughly. A hypodermic needle may be used to inject gel into small cavities not requiring fiberglass millings mixed with gel.

(4) Work mixture of resin, fibers and catalyst into damaged area, using sharp point of a putty knife or stick to press it into bottom of hold and to puncture any air bubbles which may be present. Fill scratch or hole above surrounding undamaged area about 1/16 inch.

(5) Lay piece of cellophane or waxed paper over repair to cut off air and start cure of gel mixture.

(6) Allow gel to cure 10 to 15 minutes until it feels rubbery to touch. Remove cellophane and trim flush with surface, using a sharp razor blade or knife. Replace cellophane and allow to cure completely for 30 minutes to an hour. Patch will shrink slightly below structure surface as it cures. (If wax paper is used, ascertain wax is removed from surface.)

(7) Rough up bottom and edges of hole with electric burr attachment or rough sandpaper. Feather hole into surrounding gel coat, do not undercut.

(8) Pour out a small amount of resin, add catalyst and mix thoroughly, using a cutting motion rather than stirring. Use no fibers.

(9) Using tip of putty knife or fingertips, fill hole to about 1/16 inch above surrounding surface with gel coat mixture.

(10) Lay piece of cellophane over patch to start curing process. Repeat paragraph (6), above, trimming patch when partially cured.

(11) After trimming patch, immediately place another small amount of gel coat on cut edge of patch and cover with cellophane. Then, using a squeegee or back of a razor blade, squeegee level with area surrounding patch, leave cellophane on patch for one or two hours or overnight, for complete cure.

(12) After repair has cured for 24 hours, sand patched area using sanding block with fine wet sandpaper. Finish by priming, again sanding and applying color coat.

B. Fiberglass Fracture and Patch Repairs

(1) Remove wax, oil and dirt from around damaged area with acetone, methylethyketone or equivalent.

(2) Using a key hole saw, electric saber saw, or sharp knife cut away ragged edges. Cut back to sound material.

(3) Remove paint three inches back from around damaged area.

(4) Working inside structure, bevel edges to approximately a 30 degree angle and rough-sand hole and area around it, using 80-grit dry paper. Feather back for about two inches all around hole. This roughens surface for strong bond with patch.

(5) Cover a piece of cardboard or metal with cellophane. Tape it to outside of structure covering hole completely. Cellophane should face toward inside of structure. If repair is on a sharp contour or shaped area, a sheet of aluminum formed to a similar contour may be placed over area. Aluminum should also be covered with cellophane.

(6) Prepare a patch of fiberglass mat and cloth to cover an area two inches larger than hole.

(7) Mix small amount of resin and catalyst, enough to be used for one step at a time, according to kit instructions.

(8) Thoroughly wet mat and cloth with catalyzed resin. Daub resin on mat first, and then on cloth. Mat should be applied against structure surface with cloth on top. Both pieces may be wet out on cellophane and applied as a sandwich. Enough fiberglass cloth and mat reinforcements should be used to at least replace amount of reinforcements removed in order to maintain original strength. If damage occurred as a stress crack, an extra layer or two of cloth may be used to strengthen area.

(9) Lay patch over hole on inside of structure, cover with cellophane, and squeegee from center to edges to remove all air bubbles and assure adhesion around edge of hole. Air bubbles will show white in patch and they should all be worked out to edge. Remove excess resin before it gels on part. Allow patch to cure completely.

(10) Remove cardboard or aluminum sheet from outside of hole and rough-sand patch and edge of hole. Feather edge of hole about two inches into undamaged area.

(11) Mask area around hole with tape and paper to protect surface. Cut a piece of fiberglass mat about one inch larger than hole and one or more pieces of fiberglass cloth two inches larger than hole. Brush catalyst resin over hole, lay mat over hole and wet out with catalyzed resin. Use a daubing action with brush. Then apply additional layer or layers of fiberglass cloth to build up patch to surface of structure. Wet out each layer thoroughly with resin.

(12) With a squeegee or broad knife, work out all air bubbles in patch. Work from center to edge, pressing patch firmly against structure. Allow patch to cure for 15 to 20 minutes.

(13) As soon as patch begins to set up, but while still rubbery, take a sharp knife and cut away extra cloth and mat. Cut an outside edge of feathering. Strip cut edges of structure. Do this before cure is complete, to save extra sanding. Allow patch to cure overnight.

(14) Using dry 80-grit sandpaper on a power sander or sanding block, smooth patch and blend with surrounding surface. Should air pockets appear while sanding, puncture and fill with catalyzed resin. A hypodermic needle may be used to fill cavities. Let cure and resand.

(15) Mix catalyzed resin and work into patch with fingers. Smooth carefully and work into any crevices.

(16) Cover with cellophane and squeegee smooth. Allow to cure completely before removing cellophane. Let cure and resand.

(17) Brush or spray a coat of catalyzed resin to seal patch. Sand patch, finish by priming, again sanding and applying color coat.

NOTE: Brush and hands may be cleaned in solvents such as acetone or methylethylketone. If solvents are not available, a strong solution of detergent and water may be used.

2. Thermoplastic Repairs

The following procedures will assist in making field repairs to items made of thermoplastic which are used throughout the airplane. A list of material needed to perform these repairs is given in “Chart 1” on page 51707, along with suggested suppliers of the material. Common safety precautions should be observed when handling some of the materials and tools used while making these repairs.

A. Surface Preparation:

(1) Surface dirt and paint if applied must be removed from item being repaired. Household cleaners have proven most effective in removing surface dirt.

(2) Preliminary cleaning of damaged area with perchlorethylene or VM&P Naphtha will generally insure a good bond between epoxy compounds and thermoplastic.

B. Surface Scratches, Abrasion or Ground-in-Dirt

See “Figure 1” on page 51704.

(1) Shallow scratches and abraded surfaces are usually repaired by following directions on containers of conventional automotive buffing and rubbing compounds.

(2) If large dirt particles are embedded in thermoplastic parts, they can be removed with a hot air gun capable of supplying heat in temperature range of 300°F to 400°F. Use care not to overheat material. Hold nozzle of gun about 1/4 of an inch away from surface and apply heat with a circular motion until area is sufficiently soft to remove dirt particles.

(3) Thermoplastic will return to its original shape upon cooling.

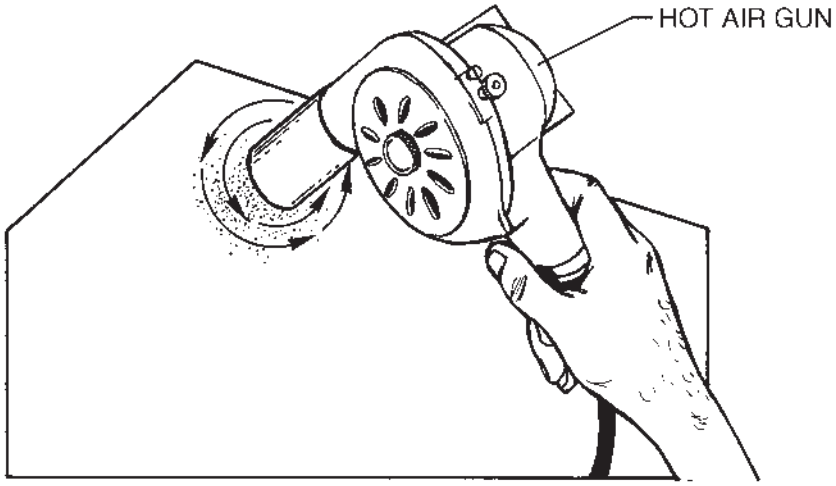
C. Deep scratches, Shallow Nicks and Small Holes: (Less than 1 inch diameter.)

See “Figure 2” on page 51704.

(1) Solvent cements will fit virtually any of these applications. If area to be repaired is very small, it may be quicker to make a satisfactory cement by dissolving thermoplastic material of the same type being repaired in solvent until desired paste-like consistency is achieved.

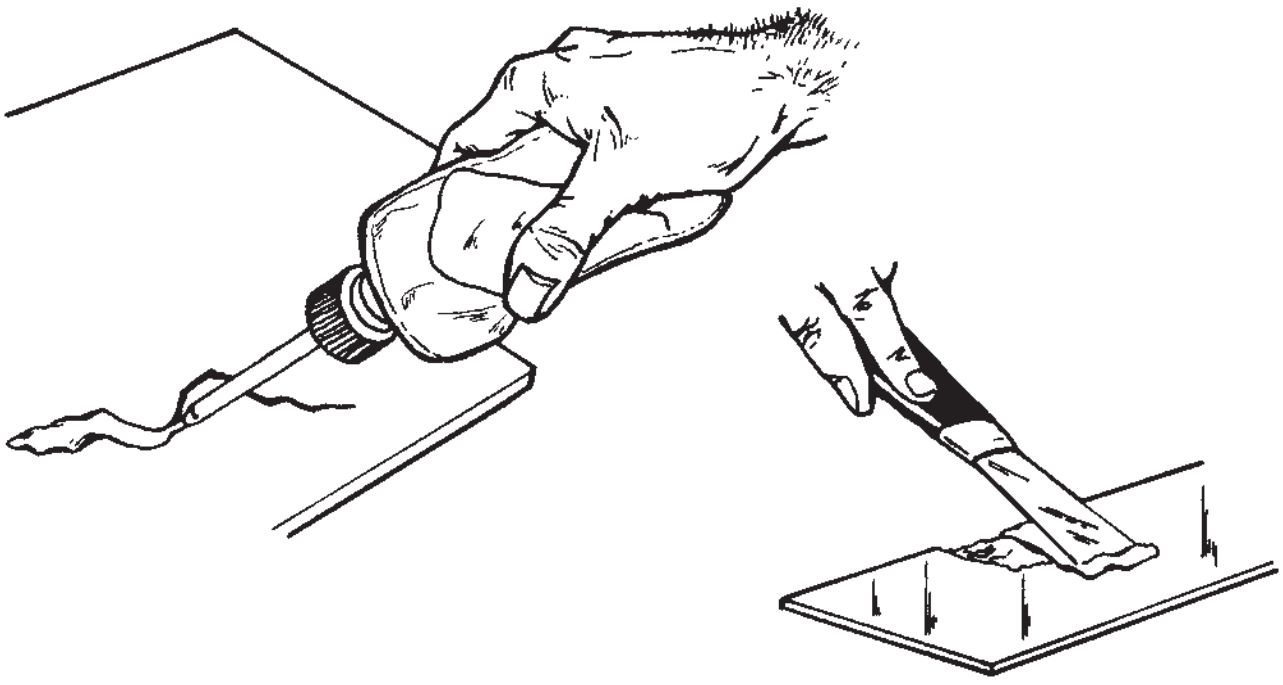
(2) This mixture is then applied to damaged area. Upon solvent evaporation, hard durable solids remaining can easily be shaped to desired contour by filing or sanding.

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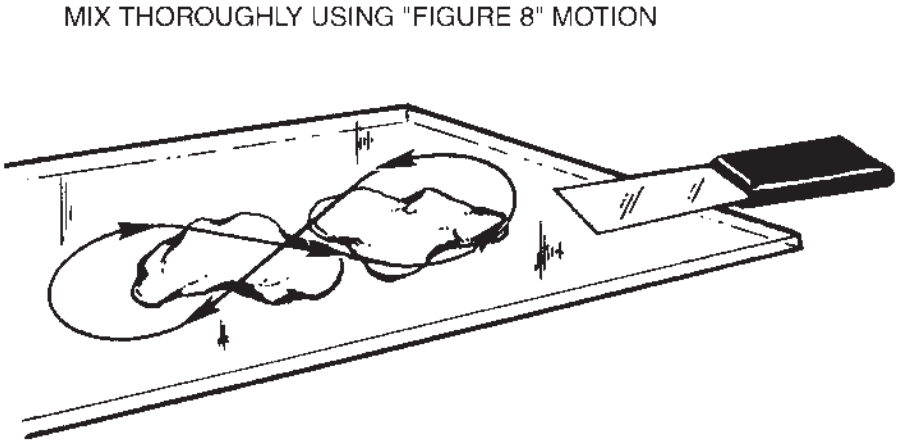
Surface Scratches - Abrasions or Ground in Dirt
Figure 1

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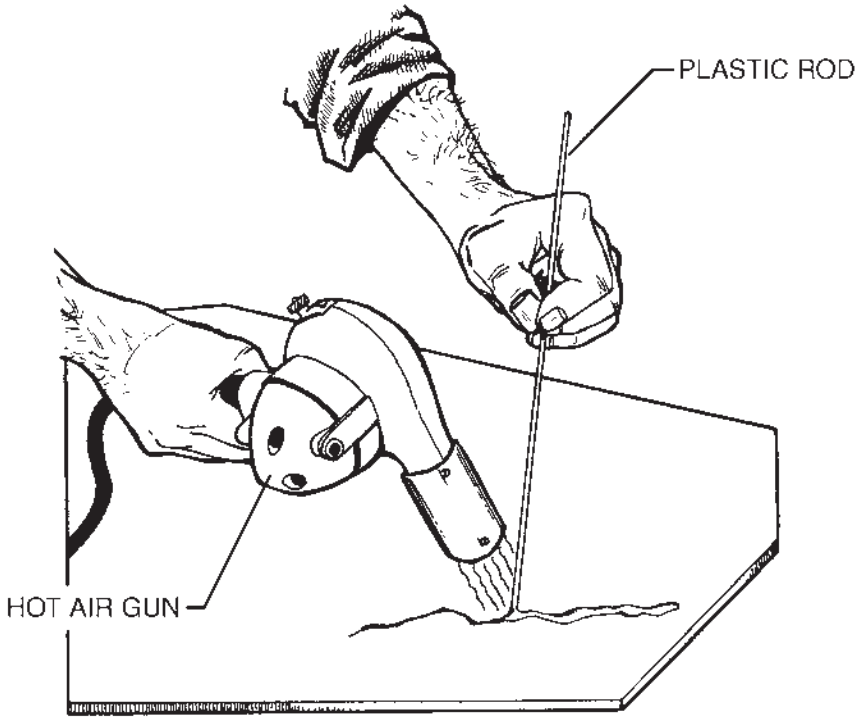
Deep Scratches, Shallow Nicks and Small Holes
Figure 2

825



Mixing Epoxy Patching Compound
Figure 3

870



Welding Repair Method
Figure 4

(3) Solvent adhesives are not recommended for highly stressed areas, on thin walled parts or for patching holes greater than 1/4 inch in diameter.

(4) For larger damages an epoxy patching compound is recommended. This type material is a two part, fast curing, easy sanding commercially available compound.

(5) Adhesion can be increased by roughing bonding surface with sandpaper and by utilizing as much surface area for bond as possible.

(6) Patching compound is mixed in equal portions on a hard flat surface using a figure eight motion. Damaged area is cleaned with perchlorethylene or VM&P Naphtha prior to applying compound. (Refer to “Figure 3” on page 51705.)

(7) A mechanical sander can be used after compound is cured, providing sander is kept in constant motion to prevent heat buildup.

(8) For repairs in areas involving little or no shear stress, hot melt adhesives, polyamids which are supplied in stick form may be used. This type of repair has a low cohesive strength factor.

(9) For repairs in areas involving small holes, indentations or cracks in material where high stress is apparent or thin walled sections are used, welding method is suggested.

(10) Welding method requires a hot air gun and ABS rods. To weld, gun should be held to direct flow of hot air into fusion (repair) zone, heating damaged area and rod simultaneously. Gun should be moved continuously in a fanning motion to prevent discoloration of material. Pressure must be maintained on rod to insure good adhesion. (Refer to “Figure 4” on page 51705.)

(11) After repair is completed, sanding is allowed to obtain surface finish of acceptable appearance.

D. Cracks

See “Figure 5” on page 51707.

(1) Before repairing a crack in thermoplastic part, first determine what caused crack and alleviate that condition to prevent it recurring after repair is made.

(2) Drill small stop holes at each end of crack.

(3) If possible, a double plate should be bonded to reverse side of crack to provide extra strength to part.

(4) Crack should be V-grooved and filled with repair material, such as solvent cement, hot melt adhesive, epoxy patching compound or hot air welded, whichever is preferred.

(5) After repair has cured, it may be sanded to match surrounding finish.

E. Repairing Major Damage: (Larger than 1 inch in diameter.)

See “Figure 6” on page 51708.

(1) If possible a patch should be made of same material and cut slightly larger than section being repaired.

(2) When appearances are important, large holes, cracks, tears, etc, should be repaired by cutting out damaged area and replacing it with a piece of similar material.

(3) When cutting away damaged area, under cut perimeter and maintain a smooth edge. Patch and/or plug should also have a smooth edge to insure a good fit.

(4) Coat patch with solvent adhesive and firmly attach it over damaged area.

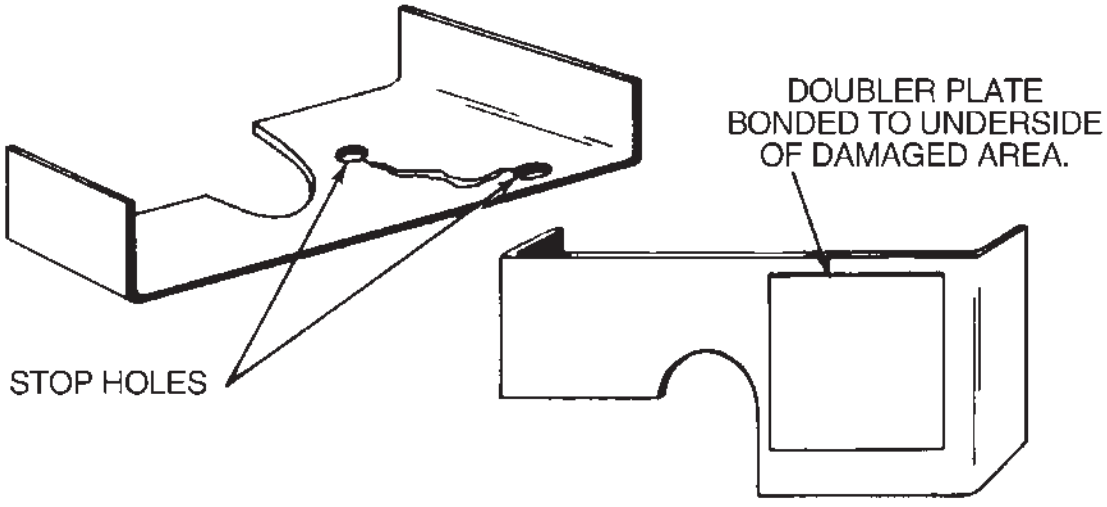
(5) Let patch dry for approximately one hour before any additional work is performed.

(6) Hole, etc, is then filled with repair material. A slight overfilling of repair material is suggested to allow for sanding and finishing after repair has cured. If patching compound is used, repair should be made in layers, not exceeding a 1/2 inch in thickness at a time, thus allowing compound to cure and insuring a good solid buildup of successive layers as required.

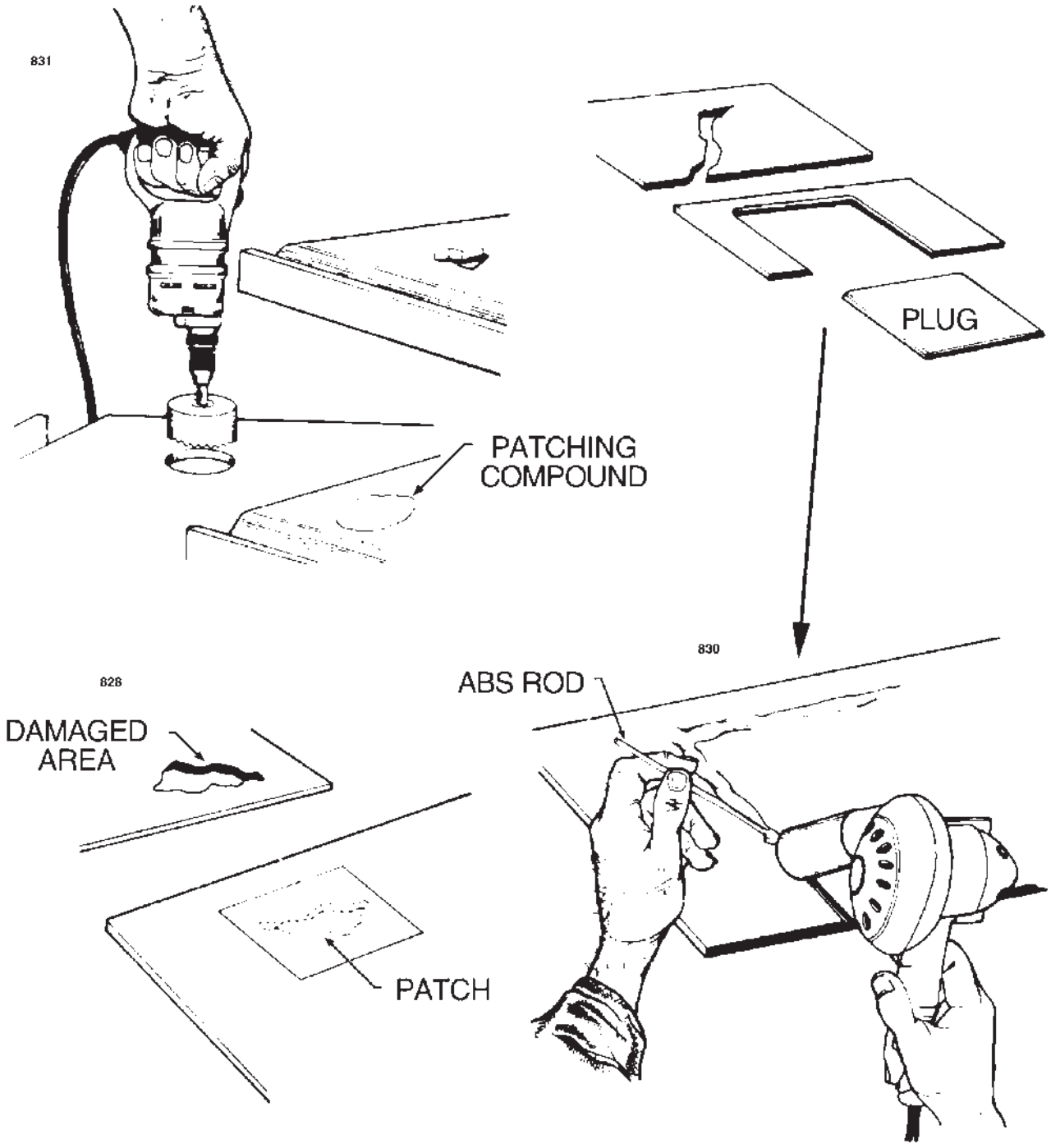
CHART 1
THERMOPLASTIC REPAIR LIST OF MATERIALS

Item	Description	Source
Buffing and Rubbing Compounds	Automotive Type – Rubbing Compound	Local Suppliers
	Ram Chemical #69 x 1	Ram Chemicals Gardenia, CA 90248
	Mirror Glaze #1	Mirror Bright Polish Co.,Inc. Irvine, CA 92713
Cleaners	Fantastic Spray	Local Suppliers
	Perchlorethylene	
	VM&P Naphtha (Lighter Fluid)	
ABS-Solvent Cements	Solarite 11 Series	Solar Compounds Corp. Linden, NJ 07036
Solvents	Methylethyl Ketone	Local Suppliers
	Methylene Chloride	
	Acetone	
Epoxy Patching Compound	Solarite #400	Solar Compounds Corp. Linden, NJ 07
Hot Melt Adhesives Polyamids and Hot Melt Gun	Stick From 1/2 in. dia. 3 in. long	Sears Roebuck & Co., or most hardware stores
Hot Air Gun	Temp. Range 300° to 400°F	Local Suppliers

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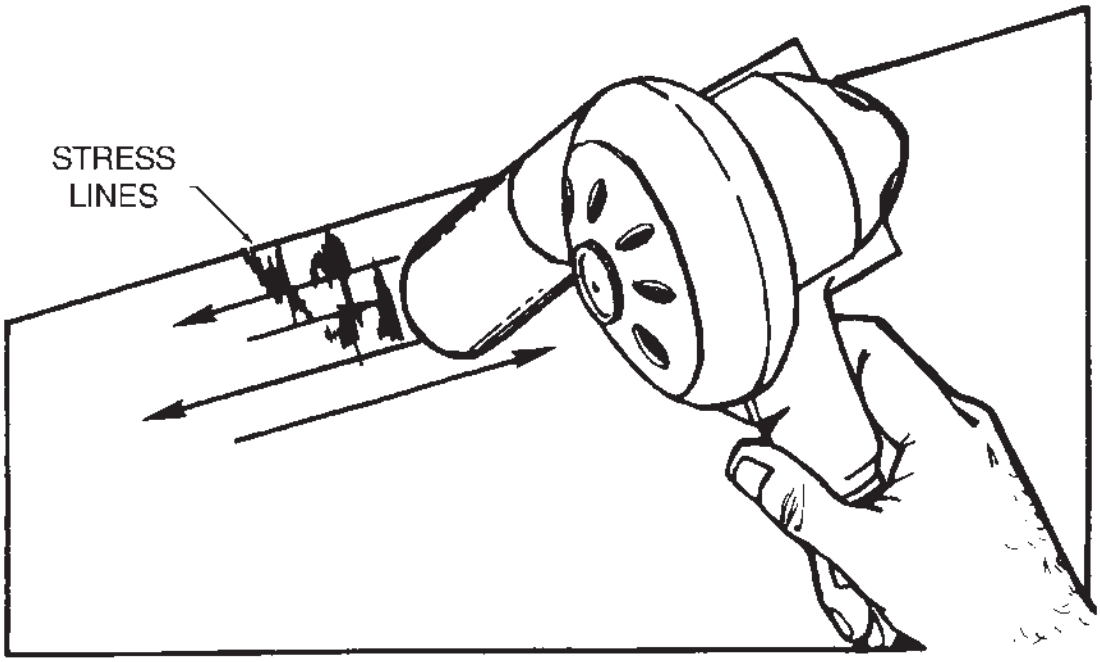


Repairing Cracks
Figure 5



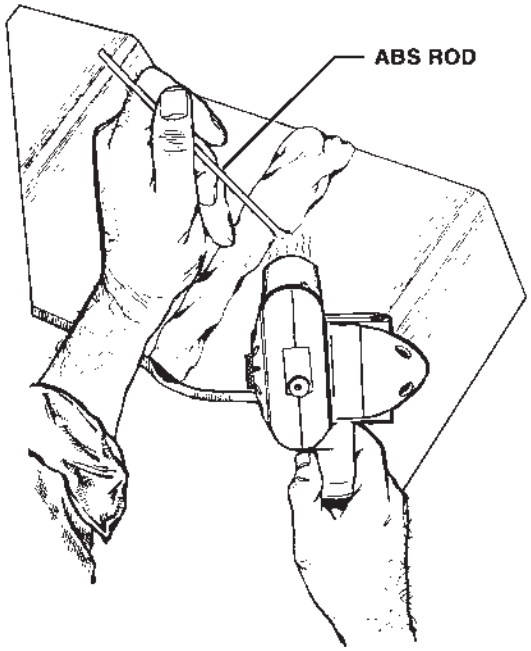
Various Repairs
Figure 6

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Repairing Stress Lines
Figure 7

829



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Repairing Impact Damage
Figure 8

F. Stress Lines

See “Figure 7” on page 51709.

(1) Stress lines produce a whitened appearance in a localized area and generally emanate from severe bending or impacting of material. (Refer to “Figure 8” on page 51709.)

(2) To restore material to its original condition and color, use a hot air gun or similar heating device and carefully apply heat to affected area. Do not overheat material.

G. Painting the Repair

(1) An important factor in obtaining a quality paint finish is proper preparation of repair and surrounding area before applying any paint.

(2) It is recommended that parts be cleaned prior to painting with a commercial cleaner or a solution made from one-fourth cup of detergent mixed with one gallon of water.

(3) Paint used for coating thermoplastic can be either lacquers or enamels depending on which is preferred by repair facility or customer.

NOTE: It is extremely important that solvent formulations be considered when selecting a paint, because not all lacquers or enamels can be used satisfactorily on thermoplastics. Some solvents used in paints can significantly affect and degrade plastic properties.

(4) Another important matter to consider is that hard, brittle coatings that are usually best for abrasion resistance should not be used in areas which incur high stress, flexing or impact. Such coating may crack, thus creating a weak area.

3. Safety Walk Repair

A. Pressure Sensitive Safety Walk

(1) Surface Preparation For Pressure Sensitive Safety Walk

NOTE: Areas to which pressure sensitive safety walk is to be installed must be free from all contaminates and no moisture present.

If liquid safety walk is installed the area must be prepared as follows:

(a) Area must be masked off to protect painted surfaces.

(b) Apply suitable stripper MEK Federal Spec. TT-M-261, U.S. Rubber No. 3339 to windwalk compound. As compound softens remove by using putty knife or other suitable tool.

(c) Area must be clean and dry prior to painting.

(d) Prime and paint area.

(2) Application of Pressure Sensitive Safety Walk

NOTE: Newly painted surfaces shall be allowed to dry for 2.5 hours minimum prior to application of safety walk.

NOTE: Wipe area with a clean dry cloth to insure that no moisture remains on surface. Do not apply when surface temperature is below 50F.

(a) Peel back full width of protective liner approximately 2 inches from leading edge of safety walk.

(b) Apply safety walk to wing area, begin at leading edge, insure proper alignment and position from wing lap.

(c) Remove remaining protective liner as safety walk is being applied from front to back of wing area.

(d) Roll firmly with a long handled cylindrical brush in both lengthwise directions. Make sure all edges adhere to wing skin.

(e) Install and rivet leading edge retainer.

B. Liquid Safety Walk

These airplanes were delivered from the factory with Pressure Sensitive Safety Walk installed. Use of existing shelf stocks of Liquid Safety Walk Compound is an authorized replacement option.

(1) Surface Preparation

(a) Clean all surfaces with a suitable cleaning solvent to remove dirt, grease and oils. Solvents may be applied by dipping, spraying or mopping.

(b) Insure that no moisture remains on surface by wiping with a clean, dry cloth.

(c) Outline area to which liquid safety walk compound is to be applied, and mask adjacent surfaces.

(2) Product Listing For Liquid Safety Walk Compound

(a) Suggested Solvents:

Safety Solvent per MIL-S-18718
Sherwin Williams Lacquer Thinner R7KC120
Glidden Thinner No. 207

(b) Safety Walk Material:

Walkway Compound and Matting Nonslip (included in Piper Part No. 179872)

(3) Application Of Liquid Safety Walk Compound

NOTE: Newly painted surfaces shall be allowed to dry for 2.5 hours minimum prior to application of safety walk.

NOTE: Liquid safety walk compound shall be applied in an area free of moisture for a period of 24 hours minimum after application. Do not apply when surface to be coated is below 50°F. Apply liquid safety walk compound as follows:

(a) Mix and thin liquid safety walk compound in accordance with manufacturer's instructions on container.

(b) Coat specified surfaces with a smooth, unbroken film of liquid safety walk compound. A nap type roller or a stiff bristle brush is recommended, using fore and aft strokes.

(c) Allow coating to dry for 15 minutes to one hour before recoating or touchup, if required after application of initial coating.

(d) After recoating or touchup, if one, allow coating to dry for 15 minutes to one hour before removing masking.

NOTE: Coated surface shall not be walked on for six hours minimum after application of final coating.

4. Metal / Wire Stitching Repair

(PIR-PPS-20024, Rev. A.)

See “Figure 9” on page 517013.

CAUTION: METAL/WIRE STITCHING (AND THE ALTERNATE METHOD OF JOINING DESCRIBED BELOW) SHALL ONLY BE USED FOR NON-STRUCTURAL, NON-LOAD CARRYING APPLICATIONS.

A metal/wire stitching process is used to staple fabric and rubber seal materials to engine baffles and some composite materials. The following alternate method of joining is approved for field use when replacing these fabric and rubber seal materials.

Alternate (Rivet) Method of Joining.

(1) Substitute two rivets in lieu of each staple where stitching was previously used or is specified. Maintain a minimum of .75 inch spacing between rivets.

(2) When materials being joined include Stainless Steel, Galvanized Steel or Steel, use:

(a) MS20615M4 Rivet (Monel) and NAS1149CN432R Washer (See “Figure 9” on page 517013.)

(b) Install with manufactured (factory) head against hardest material. Install washer against opposite side of joint and upset rivet (bucktail) against washer.

(3) When materials being joined include only aluminum and nonmetallic materials use:

(a) MS20470A4 Rivet and NAS1149DN432H Washer (See “Figure 9” on page 517013.)

(b) Install with manufactured (factory) head against hardest material. Install washer against opposite side of joint and upset rivet (bucktail) against washer.

5. Hail Damage

NOTE: The following procedures incorporate the requirements of Piper Service Letter No. 1095C.

Classification and disposition of hail damage can be accomplished as follows.

A. Inspection of Dents

(1) Determine in which of the following locations the dents are present:

- Metallic skin of the fuselage, wings, or empennage.
- Control surfaces.
- Composite components.

(2) Measure the maximum depth and circumscribable diameter of the dents from the aircraft exterior. For depth measurements on curved surfaces, measure the maximum depth of the dent from the general contour. The dent diameter is measured to the point where the skin material rejoins the general contour. See “Figure 10”.

(3) Measure the distance from the above circumscribable diameter to the nearest fastener. Note the diameter of the fastener.

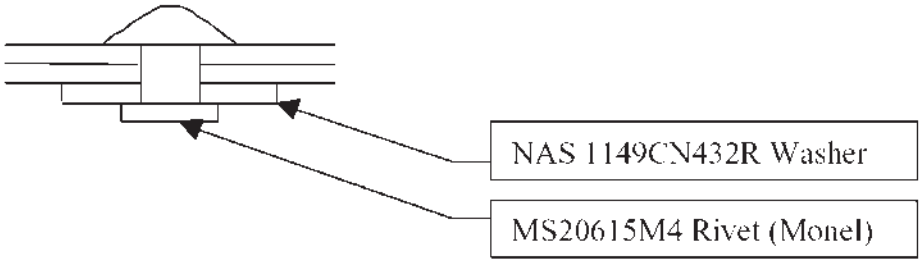
(4) Visually inspect the dent area, inner and outer surfaces, for cracks, abrasions, creases, or abrupt changes in contour within an area encompassing two times the circumscribable diameter. Use a 10-power magnifier or an equivalent inspection method.

(5) Visually inspect interior adjacent structural components for permanent for deformation.

When materials being joined include Stainless Steel, Galvanized Steel or Steel, use:

MS20615M4 Rivet (Monel)
NAS1149CN432R Washer

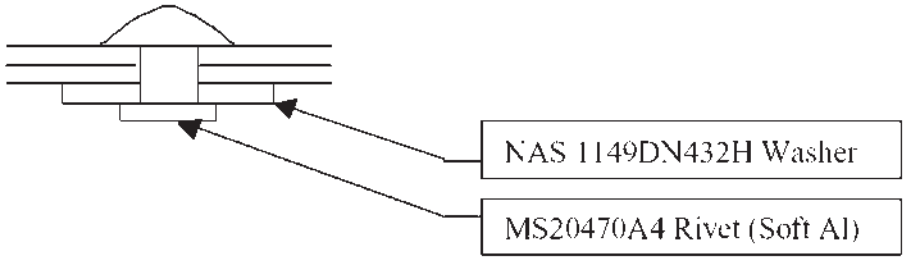
Install with manufactured (factory) head against hardest material. Install washer against opposite side of joint and upset rivet (bucktail) against washer.



When materials being joined include only aluminum and nonmetallic materials use:

MS20470A4 Rivet
NAS1149DN432H Washer

Install with manufactured (factory) head against hardest material. Install washer against opposite side of joint and upset rivet (bucktail) against washer.



Metal / Wire Stitching Repair
Figure 9

B. Classification of Dents

NOTE: Dents classified as minor or major require an additional inspection of the inner surface of the dent, using the criteria defined in “Inspection of Dents”, step (4).

(1) Negligible – Must meet all of the following dent limitations:

(a) Dents only in the metal skin of the fuselage, wings and empennage and their control surfaces (not applicable to substructure such as spars, frames and stringers).

(b) Dents must have a depth of no greater than 0.030 inches. Dents must have a circumscribable diameter no greater than 2.0 inches.

(c) Dents are a minimum distance of two times (2X) the nearest fastener’s diameter from that fastener, when measured from a dent’s circumscribable diameter to the nearest fastener.

(d) There are no cracks, abrasions, or creases within an area encompassing two times the dent’s circumscribable diameter.

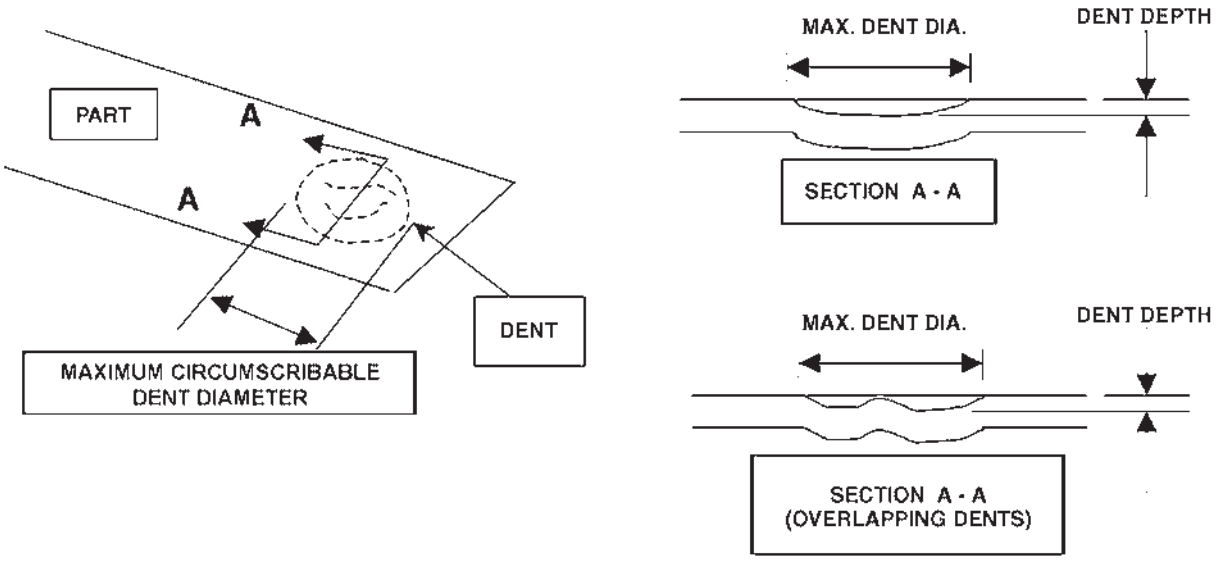
(e) There are no deformations of the interior structure adjacent to the dents.

(2) Minor – Must meet all of the limitations of Negligible, above, except for item (c); the dents may be less than 2X the nearest fastener’s diameter from the fastener.

(3) Major – Exceed any or all of Negligible dent limitations (b), (d), and/or (e).

(4) Composite and Bonded Components

(5) Control Surfaces.



Hail Profile Definitions
Figure 10

C. Disposition of Dents

(1) Negligible – Acceptable as is. Cosmetic repair is not required.

(2) Minor – The affected parts must be replaced, or they may be repaired as follows:

For each fastener that is within a distance of 2X the applicable fastener's diameter from the edge of any dent, install one matching (same size and type) fastener adjacent to the applicable fastener.

New fasteners shall be installed in accordance with the following limitations:

- A minimum of 2X the fastener diameter from the edge of the dent
- A minimum of 3X the fastener diameter from existing fasteners
- A minimum of 2X the fastener diameter from any component's edge

Minor dents may be repaired and restored to within negligible limits using standard suction or glue dent pulling devices.

(3) Major – The affected parts must be replaced, or they may be repaired using Electronic Dent Repair (EDR) devices only or other FAA-approved procedures.

(4) Composite Components – Evaluate and disposition per FAA AC43.13-1B, Chapter 3, "Fiberglass and Plastics," or other FAA-approved procedures, prior to returning to service.

(5) Control Surfaces

- Replace control surface if dents exceed any one of the negligible limits (b), (d), and/or (e).
- Replace surface if dents disturb the trailing edge contour or any formed radius.

NOTE: Movable control surfaces must remain free of body filler or any other modifications/repairs that could affect weight and balance, or aerodynamics.

D. Continued Inspection Requirements

Regardless of classification and disposition, visually inspect all dents for cracks every 250 hours, or annually, whichever comes first. Paint cracking around the area of the dent may indicate cracking of the skin and requires further inspection.

However, dents previously classified as minor that have not been repaired require a recurring inspection at an interval not to exceed 50 hours time in service, until a suitable repair is accomplished.

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ELECTRICAL BONDING

1. General

(PIR-PPS55006, Rev. AI.)

All electrical and electronic equipment and specified components shall be installed in such a manner as to provide a continuous low resistance path (bonds) from the equipment enclosure/component to the airplane structure. Bonds must be installed to ensure that the structure and equipment are electrically stable and free from the hazards of lightning, static discharge, electrical shock, etc.

A. All parts shall be bonded with as short a lead as possible.

B. All bonding surfaces shall be cleaned prior to the installation of the bonded joint.

C. All nuts used in bonding shall be of the self-locking type. (Do Not use fiber-locking type).

D. All electrical bonding shall be accomplished without affecting the structural integrity of the airframe.

2. Inspection

(PIR-AC 43.13-1, Rev. B.)

A. 100 Hour

Each 100 hours, visually inspect shield and shield terminations of each electrical harness for integrity, condition, and security. If electrical arcing is evident, check for intermittent contact between conducting surfaces. Arcing can be prevented by bonding or insulation, as appropriate.

Inspect the components listed in “Chart 1” on page 51802 as follows:

(1) Bond connections shall be secure and free from corrosion.

(2) Bonding jumpers installed so as not to interfere in any way with the operation of moveable components of the aircraft.

(3) No self-tapping screws used for bonding purposes.

(4) Exposed conducting frames or parts of electrical or electronic equipment should have a low resistance bond of less than 2.5 milliohms to structure. If the equipment design includes a ground terminal or pin, which is internally connected to such exposed parts, a ground wire connection to such terminal will satisfy this requirement.

(5) Parts shall be bonded directly to the primary structure rather than to other bonded parts.

(6) Where aluminum or copper is bonded to dissimilar metallic structures, ensure installed hardware (typically washers) is as called out in the parts catalog to minimize electrolytic corrosion and ensure the hardware should corrode first.

B. On Condition

Whenever any electrically bonded component (see “Chart 1” on page 51802) is removed and reinstalled, or visual inspection reveals the electrical bonding to be suspect, measure resistance between component and aircraft structure.

To ensure proper operation and suppression of radio interference from hazards, electrical bonding of equipment must not exceed the maximum allowable resistance values specified in “Chart 1” on page 51802.

(1) Measurements should be performed after the grounding and bonding mechanical connections are complete to determine if the measured resistance values meet the basic requirements.

(2) A high quality test instrument (AN/USM-21A or equivalent) will accurately measure the very low resistance values specified.

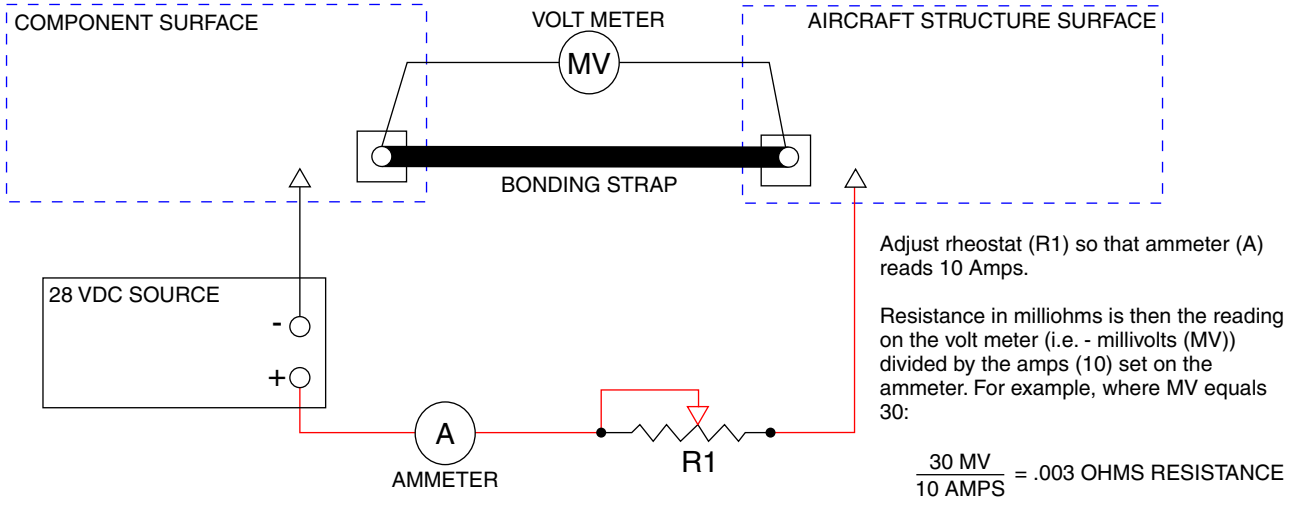
(3) Another method of measurement is the millivolt drop test as shown in “Figure 1” on page 51802.

CHART 1
ELECTRICAL BONDING RESISTANCE INDEX

Component		Maximum Allowable Resistance Value in Ohms
Engine Mount(s)		.003
Generator(s)		.010
Ailerons		.003
Elevator / Stabilator		.003
Rudder		.003
Alternator(s)		.010
Trim Tab(s)		
Conventional Hinge		.003
Piano Hinge		.010
Instrument Panel Inserts		.003
Exterior Lights Mounted on Non-Conductive Material		.003
Avionics 'Black Boxes'		.003
NOTE: Harnesses should be installed and connected for this check, internal chassis wiring through the connector to ground is permissible for this grounding.		
Battery Ground Point		.010
Static wick mounting plates:	P/N 452-094 (TCO Model B-4)	1.00
	P/N 452-124 (R90-550)	.001

NOTE: Where jumper wires or cables are used to accomplish a proper bond, resistance between the jumper terminal and the component or structure shall not exceed .001 ohms. The controlling points for measuring resistance will be within the limits of the cleaned area to be bonded and within 1/4 inch of the exterior limits of the bonding jumper terminal or material called for in the bill of materials of the drawing.

Resistance to ground will be measured from wire terminal to structure for electrical / electronic equipment not internally grounded and from mounting flange to structure for equipment that is internally grounded.



Millivolt Drop Test
Figure 1

3. Garmin G600 and G1000/G1000 NXi Equipment Electrical Bonding Check

The following procedure verifies proper electrical bonding for Garmin G600 and G1000/G1000 NXi system equipment. Perform when equipment is removed and installed, for troubleshooting, and each 1000 hours.

NOTE: Any area that fails shall be re-bonded and retested per AC 43.13-1 latest revision.

A. Required Equipment

A milli-ohmmeter capable of measuring 0.001–1.00 ohms.

B. GRS 7X AHRS (2 ea.)

(1) The units must be installed in racks with connectors removed.
(2) Test each AHRS unit between unit base and airframe.
(3) PASS: = ≤ 10 mohms

C. GDC 7X ADC (2 ea.)

(1) The units must be installed in racks with connectors removed.
(2) Test each ADC unit between unit base and airframe.
(3) PASS: = ≤ 10 mohms

D. GMU 44 Magnetometer (2 ea.)

(1) Make sure units are unplugged and secured in mounts, mounts secured to inspection covers.
(2) Test between the sensor base and inspection cover mating area with wing skin.
(3) PASS: = ≤ 10 mohms

E. GDU 620 Display Unit (2 ea.) or GDU-1040 Display Unit (3 ea.)

(1) The Display units must be secured to Instrument Panel with connectors removed.
(2) Test from a clean metal spot on back side of GDU to back of Instrument Panel.
(3) PASS: = ≤ 20 mohms.

F. GTP 59 OAT Probe (2 ea.)

(1) OAT secured to wing skin with harness disconnected.
(2) Test between the OAT probe and a point on airframe approximately two feet away.
(3) Use the nut on the OAT probes as test points.
(4) PASS: = ≤ 2.5 mohms

G. GNS 430W Radio (2 ea.) - G600 only.

(1) The units must be installed in rack with connectors removed.
(2) Test from a clean metal spot on radio to back side of Instrument Panel.
(3) PASS: = ≤ 20 mohms.

H. SYSTEM 55X Computer

(1) The unit must be installed in rack with connectors removed.
(2) Test from a clean metal spot on AP computer to back side of Instrument Panel.
(3) PASS: = ≤ 20 mohms.

I. KR 87 ADF Receiver

(1) The unit must be installed in rack with connectors removed.
(2) Test from a clean metal spot on ADF to back side of Instrument Panel.
(3) PASS: = ≤ 20 mohms.

J. GTX 330 Transponder - G600 only.

(1) The unit must be installed in rack with connectors removed.

(2) Test from a clean metal spot on Transponder to back side of Instrument Panel.

(3) PASS: = ≤ 20 mohms.

K. GPS Antennas

(1) GPS antenna secured to airframe with coax cable removed.

(2) Test from the screws to a point on the airframe that is two feet away.

(3) PASS: ≤ 2.5 mΩ

L. MFD and PFD Racks

(1) Test between the rack and aircraft floor.

(2) PASS: ≤ 10 mΩ

CHAPTER

52

DOORS

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CHAPTER 52

LIST OF EFFECTIVE PAGES

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	2	Sep 30/18			
52-Table of Contents	1	Sep 30/18			
	2	Sep 30/18			
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	2	Sep 30/18			
	3	Sep 30/18			
	4	Sep 30/18			
	5	Sep 30/18			
	6	Sep 30/18			
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	2	Sep 30/18			

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CHAPTER 52 - DOORS

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Installation		1
PASSENGER / CREW	52-10-00	1
Cabin Doors		1
Removal		1
Installation		1
Adjustment		1
Door Locks		1
Removal		1
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GENERAL

1. Description

This airplane is provided with a crew entrance door located on the forward right side of the fuselage and a passenger compartment door on the left side of the fuselage aft of the wing trailing edge. A rear baggage compartment door adjoins the passenger compartment door. The forward baggage compartment door is located on the left side of the fuselage at station 20.

2. Door Snubber Seals

See “Figure 1” on page 52002.

Door snubber seals are incorporated in the door jambs to improve door sealing in all doors except the forward cabin door. The latching mechanism used in the forward cabin door has improved sealing characteristics sufficient to allow the removal of snubber seals from those doors.

NOTE: Seneca IV S/N’s 3447001 thru 3447012 use an older forward cabin door latch design and, accordingly, do use snubber seals on the forward cabin door.

NOTE: Seneca V S/N’s 3449089 and up use a new type door seal on the aft cabin door that make the snubber seal no longer necessary.

NOTE: If existing seal is torn or badly deteriorated, it should be replaced. If seal is loose or bond is “marginal’’, it should be rebonded. Adhesives listed below are recommended for rebonding:

3M EC 1300L (Preferred)
Proco Adhesive 6205-1
Scotch Grip 2210

A. Removal

(1) Remove windlace retainers, “roll” back windlace (tape to secure) out of way, remove all scuff plates and disconnect door holder.

(2) Remove all striker plates except where shown in “Figure 1” on page 52002, Section A-A.

(3) With a plastic scraper or other appropriate instrument, scrape off snubber while applying mineral spirits as necessary to loosen strip and wipe off excess adhesive with a clean cloth.

B. Installation

(1) If door jamb is flaking or excessively scuffed, rub down with wet and dry emery cloth. Clean surface using Prep-Sol or equivalent cleaner which will not leave an oily residue.

(2) Mask jamb as shown in “Figure 1” on page 52002, view E.

(3) Apply adhesive to door jamb as shown in “Figure 1” on page 52002, view E.

(4) Apply adhesive to inside surface of snubber.

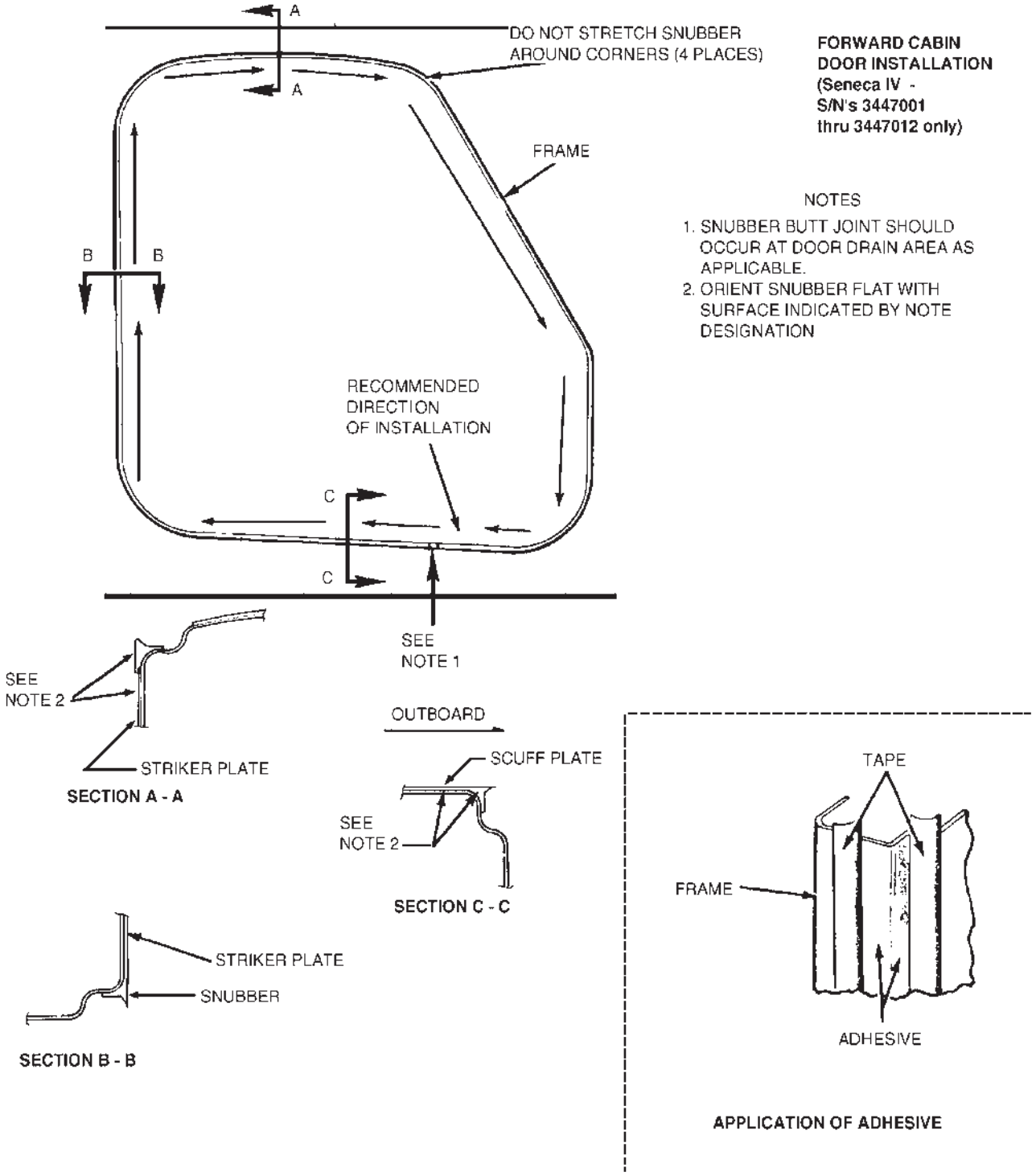
(5) Position snubber with protruding leg facing outboard beginning at lower center of door jamb and work progressively around jamb applying pressure to snubber to remove any trapped air and to ensure a proper bond. Do not prestretch snubber as this can induce cracks.

NOTE: Normal tack time for 3M EC 1300L is 30-45 minutes at 75°F. However, adhesive that has “set” may be reactivated by a clean rag moistened with Toluol or MEK.

(6) It takes approximately 1 day for bond to cure. Do Not allow door to close during this period. It is recommended that door be left open as long as possible to effect curing.

(7) Remove masking tape if used and clean off excessive adhesive smears using Mineral Spirits or Toluol and a clean cloth. Install striker plates and windlacing. Cut snubber for aft cabin door as shown in “Figure 1” on page 52002.

(8) Check that doors close properly and readjust as necessary to achieve a flush fit. Latching effort must not have increased.

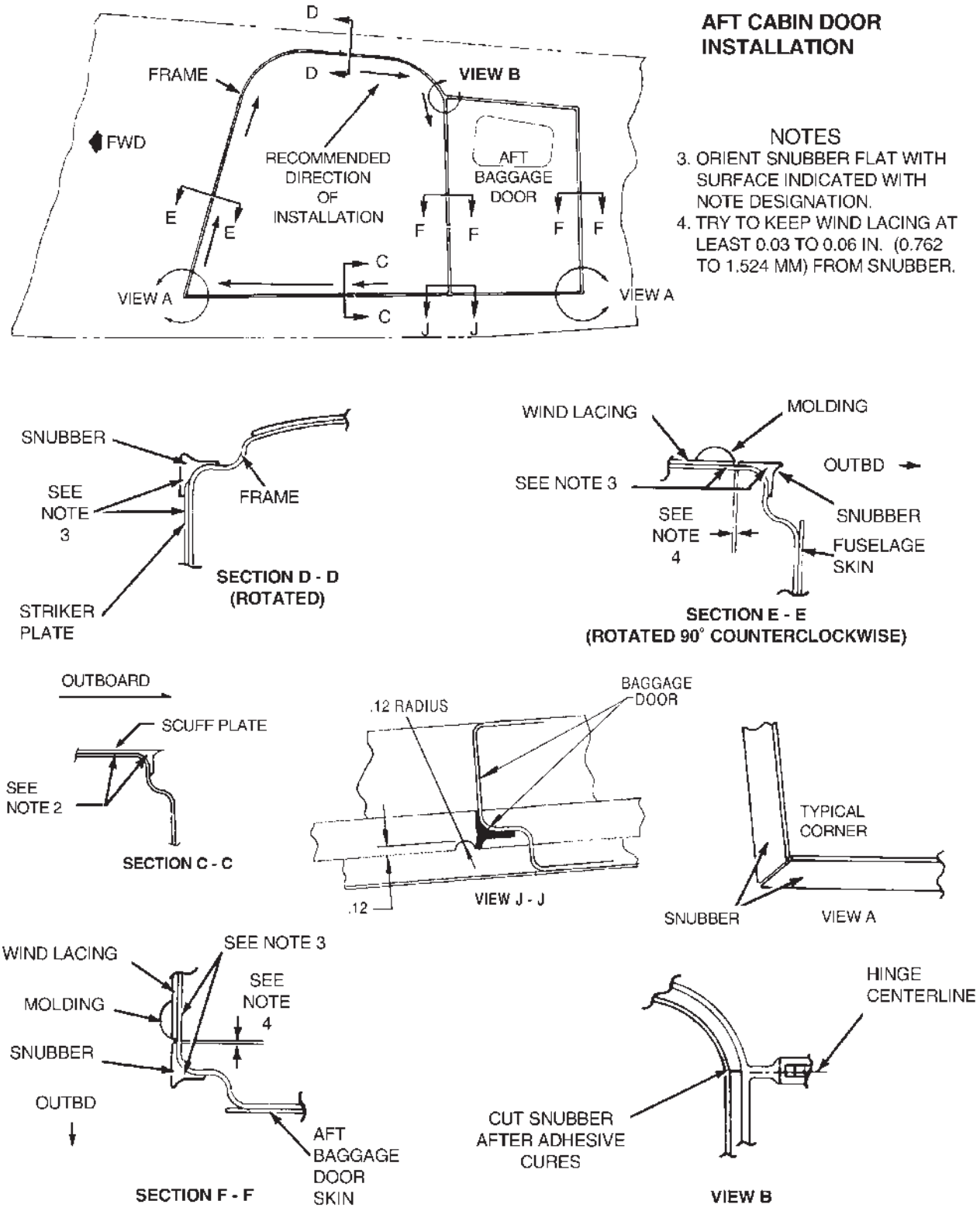


FORWARD CABIN DOOR INSTALLATION (Seneca IV - S/N's 3447001 thru 3447012 only)

NOTES

1. SNUBBER BUTT JOINT SHOULD OCCUR AT DOOR DRAIN AREA AS APPLICABLE.
2. ORIENT SNUBBER FLAT WITH SURFACE INDICATED BY NOTE DESIGNATION

Door Snubber Seal Installation
Figure 1 (Sheet 1 of 2)



Door Snubber Seal Installation
Figure 1 (Sheet 2 of 2)

(9) With all hardware and plates installed, coat snubbers with silicone.

(10) To check for proper cure, try peeling back a small local area of the snubber leg.

(11) With adhesive properly cured, remove the masking tape. Replace scuff plates and windlacing. If the snubber for the aft cabin door has just been installed, cut snubber as shown in "Figure 1" on page 52002.

(12) With both the aft cabin and aft baggage door held open, install the door scuff plate and the baggage door vertical trim strip.

(13) Check that the door closes properly and readjust as necessary to achieve a flush fit. Latching effort must not have increased.

(14) With all hardware and plates reinstalled coat snubbers with silicone.

PASSENGER / CREW

1. Cabin Doors

   A. Removal

      (1) Remove the clevis bolt, washer and bushing from the door holder assembly.
      (2) Remove cotter pins, clevis pins and washers from door hinges eye bolts.
      (3) Remove the door from the airplane.

   B. Installation

      (1) Insert the door into position and install the washers, clevis bolts and cotter pins on the door hinges eye bolts.
      (2) For adjustment of door, refer to Adjustment, below.
      (3) Hook up and install the clevis bolt, bushing and washer into the door holder assembly.

   C. Adjustment

      (1) To acquire the proper vertical adjustment of the door, insert the necessary washer combination between the cabin door hinge and fuselage hinge eyebolt.

         NOTE: When installing a new door, proper horizontal adjustment can be achieved by inserting or removing washers (NAS1149F0463P or NAS1149F0432, as required) to/from under the head of the fuselage hinge eyebolt.

      (2) To ensure long life of door seals and improve sealing characteristics, lubricate with a dry lubricant in a spray can.

2. Door Locks

   A. Removal

      (1) Remove the door trim upholstery by removing the attachment screws.
      (2) Loosen the nut on the lock assembly and remove the lock by turning it sideways.

   B. Installation

      (1) Install the lock in the door by turning it sideways and placing it through the opening provided.
      (2) Replace the nut on the back of the lock assembly and tighten.
      (3) Replace the door trim upholstery and secure with the attachment screws.

3. Door Latch Mechanisms

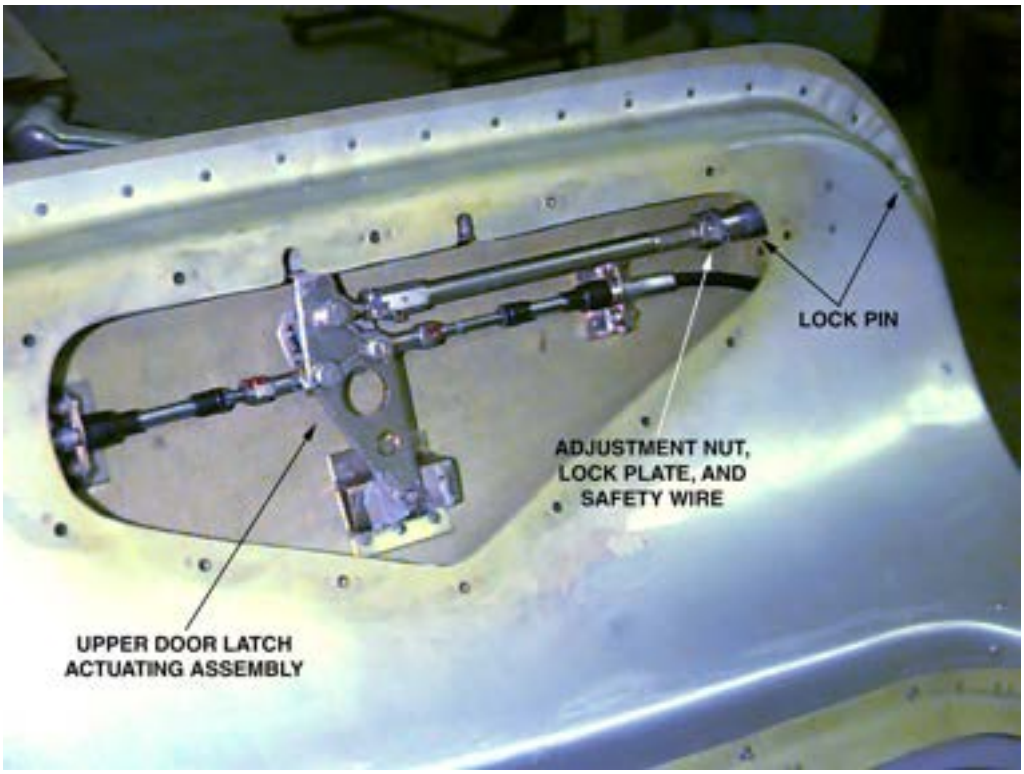
A. Forward Cabin Door - Upper - Adjustment

Refer to “Figure 1”.

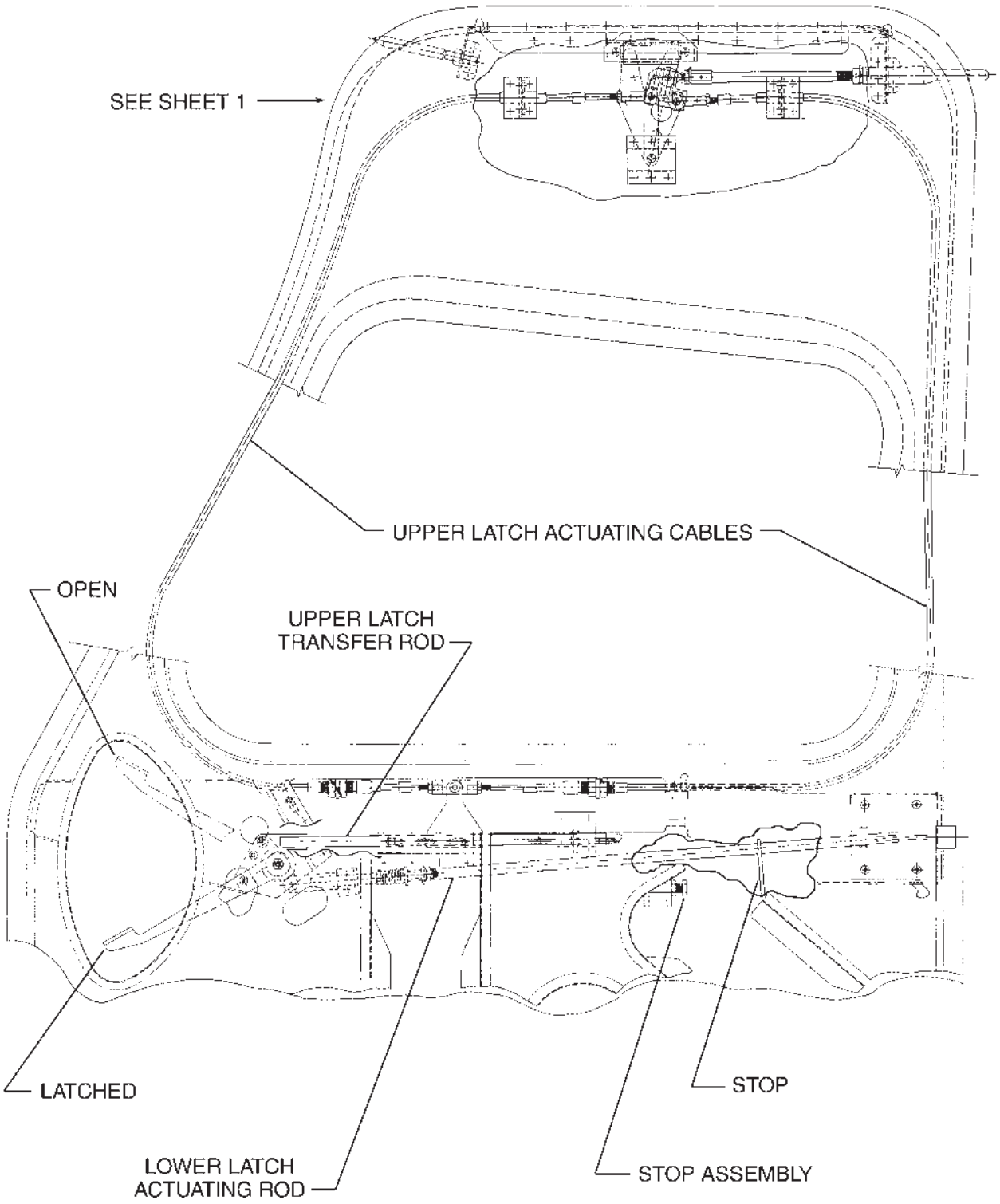
CAUTION: DO NOT LUBRICATE LOCK PIN / LOCK PIN TEFLON GUIDE BEARING.

NOTE: Seneca IV S/N’s 3447001 thru 3447012 are equipped with the same upper latch described under Aft Cabin Door, below.

(1) Remove the door trim upholstery by removing the attachment screws.
(2) Remove upper cabin door access cover on inside of door to gain access to the upper door latch assembly.
(3) Remove lockwire from nut at aft end of pin assembly.
(4) Back nut off from lock plate.
(5) Move lock plate to disengage from aft pin.
(6) Adjust pin so that, in extended position, rigging groove on pin aligns with forward face of pin receptacle on aft door frame.
(7) Engage lock plate, making sure safety wire tab on lock plate is facing inboard.
(8) Tighten nut.
(9) Install safety wire.



Fwd Cabin Door Latch Installation
Figure 1 (Sheet 1 of 2)



Fwd Cabin Door Latch Installation
Figure 1 (Sheet 2 of 2)

B. Aft Cabin Door

(1) Lower Latch

(a) Removal

1) Remove the door latch mechanism by removing the door trim upholstery and the screws that attach the latch plate and latch mechanism to the door.

2) Disconnect the latch pull rod from the inside door handle.

3) Remove the complete latch mechanism.

(b) Installation

1) Place the latch assembly into position on the door.

2) Connect the latch pull rod to the inside door handle.

3) Replace the screws that attach the latch plate and mechanism to the door. Install the door trim upholstery and secure with screws.

(c) Adjustment

To adjust the door latch, loosen the screws on the striker plate, make necessary adjustment, and retighten the screws.

(2) Upper Latch

(a) Removal

1) Remove the inside and outside handles and the screws holding the pan on the inside of the door.

2) Remove the pan and pull the latch assembly through the opening on the door. With the aft door only, pull the pan and latch forward to ensure the locking pin assembly comes free from its receptacle and exits the opening without bending.

(b) Installation

1) Place the latch assembly into position for installation. With the aft door only, insert the locking pin assembly first and guide it into its receptacle as the latch assembly reaches its final position.

2) Replace the pan and install the screws and handles.

3) Check the latch assembly for operation and be certain that it is free of rubbing on the trim panels.

(c) Adjustment

1) To adjust the door safety latch, remove the two screws from latch plate found at the top of the door opening.

2) Remove the plate and turn the loop assembly in or out to make necessary adjustments.

3) Replace the latch plate and secure with the two attachment screws.

4) In the aft door only, the locking pin may be adjusted through the opening near the locking pin receptacle.

CARGO

1. Baggage Doors

   A. Removal

   With door open remove hinge pin from hinge and remove the door.

   B. Installation

   Place door in position so that hinge halves are properly matched and install hinge pin. It will not be necessary to replace hinge pin with a new pin if it is free of bends and wear.

   C. Seal Installation

   (1) Remove any traces of the old seal.
   (2) Clean the surface to be bonded with methylethylketone (MEK) to remove any dirt, oil, and grease.
   (3) Bond the door seal to the retainer with RTV-732 (P/N 279-195) or RTV-103 (P/N 179-759).

2. Forward Baggage Door Lock

   A. Removal

   (1) With door open remove nut from back of lock assembly by use of a special made wrench. (This tool may be fabricated from dimensions given in “Figure 1” on page 52302.)
   (2) Remove lock assembly through front of door.

   B. Installation

   (1) Place lock into position for installation.
   (2) Install nut on lock assembly and tighten with use of a special wrench.

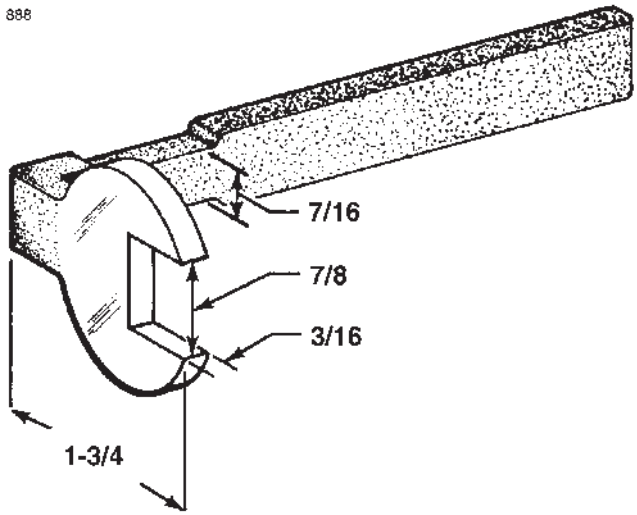
3. Baggage Door Hinges

   A. Removal

   (1) Remove door from airplane as described in Baggage Doors, Removal, above.
   (2) Remove hinge half from airplane or door by drilling out rivets and removing hinge.

   B. Installation

   (1) Place hinge halves together and install hinge pin.
   (2) Install door into closed position and drill two end rivet holes and install rivets.
   (3) Operate door and check for proper fit and installation. Drill remaining holes and install rivets.



Baggage Door Lock Tool
Figure 1

CHAPTER

53

FUSELAGE

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CHAPTER 53

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MAIN

1. Aft Wing Attach Fittings 100 Hour Inspection

A. Background

Should the seals for the windows and doors not be maintained, leaks may develop which, if not corrected, will allow an ingress of water. This water contamination will wet the insulation around the aft wing attach fittings creating a highly corrosive environment.

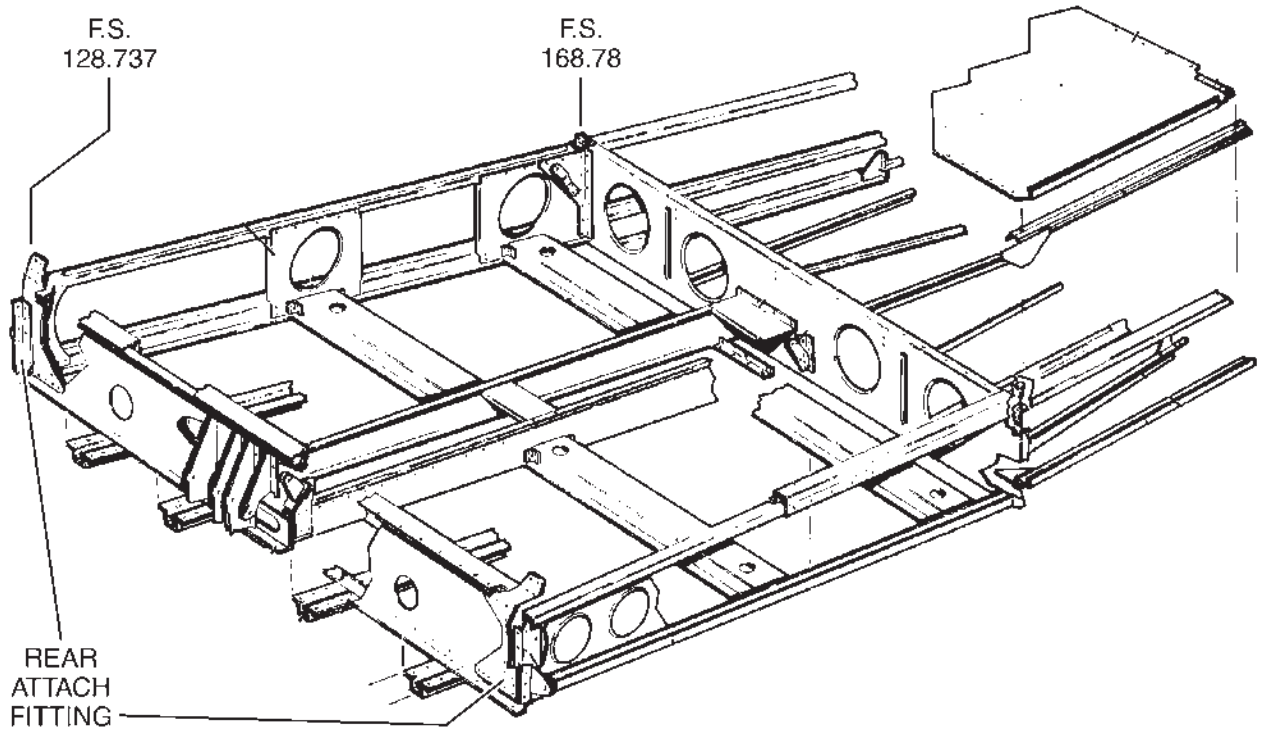
B. Procedure

Each 100 hours, inspect to determine condition of the aircraft window and door seals, the condition of the aft wing attach fittings, the insulation material around the affected area, and the drain holes in the bottom fuselage skin at the aft attach fittings area.

(1) Gain access to the left and right aft wing attach fittings.

(a) Remove center seats and the center floorboard.

(b) Remove interior mouldings and carpet as necessary.



Aft Wing Attach Fittings
Figure 1

(2) Inspect thoroughly the left and right aft wing attach fittings for evidence of flaking paint and/or corrosion. (Flaking paint may be a symptom of hidden corrosion.)
   (a) If no corrosion exists, continue with these instructions.
   (b) If corrosion is found, proceed as follows:
      1) If corrosion is superficial and there is no metal flaking and/or pitting, clean and paint fittings, using a good quality aircraft primer.
      2) If serious corrosion is found, consult the Piper Illustrated Parts Catalog (P/N 761-887) for replacement part numbers and obtain and install new parts before next flight. See Aft Wing Attach Fittings Replacement, below.
      3) Upon completion of the inspection and after replacement or refurbishment of fittings, treat the aft attach fittings area using DINOL AV 8 corrosion compound (P/N 89500-800). The treatment may be brushed or sprayed.

(3) Inspect insulation in and around the rear fittings.
   (a) If insulation is wet or matted down where it has been wet, it will be necessary to replace this insulation and it will be necessary to inspect all windows, doors, and exterior panels leading to the cabin.
      1) Check door seals for deterioration, cracks, and voids in adhesive.
      2) Check window seals for voids, cracks, and deterioration.
      3) Perform a leak check with water to determine where the water is entering. Cure all leak paths before continuing these instructions.
      4) Consult the Piper Illustrated Parts Catalog (P/N 761-887) for replacement part numbers and obtain and install new parts before continued operation.
      5) If sealing windows, use P/N 279-058 Sealant (Bostik 1100 FS) or equivalent.
      6) If using insulation other than Piper original material, be sure that the insulation is flame resistant and conforms to FAR part 23.853.
   (b) If the insulation material has not been wet, or if new material is being installed, ensure a six (6) inch clearance in the insulation has been cut out in all directions around each attach fitting.

(4) Locate the two 0.191 inch drain holes, one beneath each aft attach fitting, in the bottom fuselage skin and ensure each is clean and free of obstruction.

(5) Re-install floorboards, seats, interior panels, and other articles previously removed. Perform a functional test of any system or component that may have been interrupted or removed.

(6) Make an appropriate logbook entry documenting completion of this inspection.

2. Aft Wing Attach Fittings Replacement

NOTE: The following is basic guidance. More extensive disassembly may be required to remove the rear attach fitting(s). Thoroughly access the job before beginning to determine if additional steps or parts may be required. Consult the Piper Illustrated Parts Catalog (P/N 761-897) for additional parts as required.

A. Removal

(1) Remove electrical power from aircraft by disconnecting the battery.

(2) Place jacks under wings and tail - tie down to stabilize aircraft. Provide support for the fuselage in the affected area.

(3) Remove seats,center floorboard and interior panels necessary to gain access to the aft wing attach fittings.

(4) Remove or relocate systems components to gain access to the attach fittings.

(5) Remove bolt from aft wing spar-to-fuselage attach fitting.

NOTE: If replacing both left and right attach fittings, work one side at a time.

(6) Carefully remove rivets, screws, inner panels, channels, and brackets necessary to remove attach fitting.

NOTE: In order to remove some channels, removal of the wings may be required.

(7) Remove wing attach fitting from the aircraft. Discard fitting.

(8) Clean and inspect the areas that were under bracket for any signs of corrosion.

(a) If corrosion is found, repair or replace parts as necessary. Coat the area with primer and allow to dry.

(b) If no corrosion is found, coat the area with primer. Allow time to dry.

B. Installation

(1) Install new wing attach fitting and align rivet holes. It may be necessary to ream open the bolt hole to proper size. The hole is close tolerance and should be .3745 / .3765 (3/8) inch. Replace attach fitting bolt should there be any sign of wear or corrosion.

(2) Re-rivet wing attach fitting into place with appropriate fasteners.

NOTE: For hard to reach areas. it is permissible to replace the existing MS20470AD-5 rivets with Hi Lok fasteners. Use HL30-5 with HL-94 Hi Lok collars. Torque to 15 to 25 inch lbs. Observe standard practices for use of Hi Lok fasteners.

(3) Install wing spar and fuselage attach bolt per 57-40-00, Figure 1.

(4) Seal edges of attach fittings with PRC PR1422 (or equivalent) before installing interior.

(5) If required, repeat this process for the opposite side.

(6) Replace or reinstall any systems components removed or relocated for access.

(7) Reinstall center floorboard, interior panels, and seats.

(8) Connect battery and check for operation.

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CHAPTER

55

STABILIZERS

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CHAPTER 55

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CHAPTER 55 - STABILIZERS

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GENERAL

CAUTION: BEFORE ENTERING THE AFT PORTION OF THE FUSELAGE TO ACCOMPLISH ANY OF THE FOLLOWING PROCEDURES, ATTACH A STAND TO THE TAIL SKID FOR SUPPORT. USE A HEAVY PAD TO PROTECT THE BULKHEADS SO AS NOT TO DAMAGE THE FUSELAGE SKIN OR BULKHEAD.

1. Description

The all metal empennage group is a full cantilever design consisting of a vertical stabilizer (fin), rudder and stabilator, all with removable fiberglass or thermoplastic tips. The rudder and stabilator have trim tabs attached that are controllable from the cockpit. The stabilator also incorporates a one channel main spar that runs the full length of the stabilator and hinges to the aft bulkhead assembly of the fuselage.

Refer to “Figure 1” on page 55002 for overview of empennage assemblies.

2. Repairs

WARNING: NO ACCESS HOLES ARE PERMITTED IN ANY CONTROL SURFACES.

WARNING: USE OF PATCH PLATES FOR REPAIRS OF ALL MOVABLE CONTROL SURACES IS PROHIBITED. USE OF ANY FILLER MATERIAL NORMALLY USED FOR REPAIR OF MINOR DENTS AND/OR MATERIALS USED FOR FILLING INSIDE OF SURFACES IS ALSO PROHIBITED ON ALL MOVABLE CONTROL SURFACES.

CAUTION: CONTROL SURFACE SKINS MUST BE REPLACED IF THEY SUSTAIN DAMAGE OR EXHIBIT CRACKS.

Structural repair methods used must be in accordance with FAA Advisory Circular 43.13-1, latest revision. See also 51-70-00.

3. Balancing Equipment

See “Figure 2” on page 55005.

Balancing must be done using a suitable tool capable of measuring unbalance in inch-pounds from centerline of control surface hinge pin. A suggested tool configuration is shown in “Figure 2” on page 55005. Other tool configurations may be used, provided accuracy is maintained and recalibration capability is provided.

A. Ensure that the control surface is in its final flight configuration. Static wicks, trim tabs, trim tab push-pull rod and control surface tip (as applicable) should be installed. The surface should be painted, decals installed, and trim/servo tabs should be in the neutral position.

NOTE: Because paint is a considerable balance factor, it is recommended that existing paint be removed prior to repainting a control surface.

B. Place hinge bolts through control surfaces and place control surface on a holding fixture.

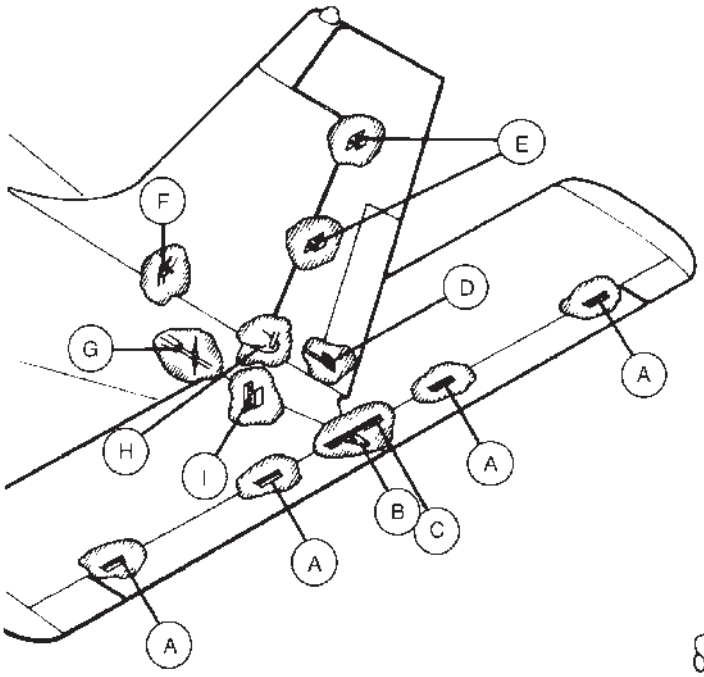
C. Calibrate the tool.

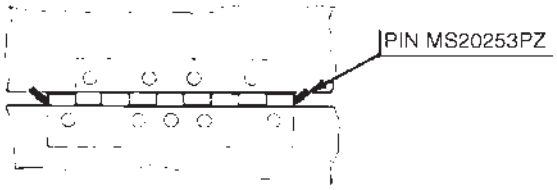
(1) Avoiding rivets, place the balancing tool on the control surface with the tool’s hinge centerline directly over the hinge line of the control surface.

(2) Adjust the movable trailing edge support to fit the width of the control surface. Tighten the set screw on the trailing edge support.

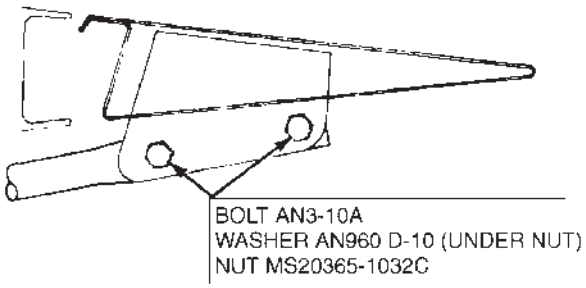
(3) Adjust the trailing edge support vertically until the beam is parallel with the control surface chord line.

(4) Remove the tool from the control surface and balance the tool itself by adding or removing nuts or washers from the beam balancing bolt. When balancing the tool, the movable weight must be at the bar’s hinge centerline.

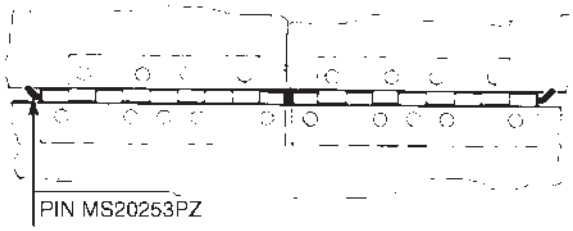




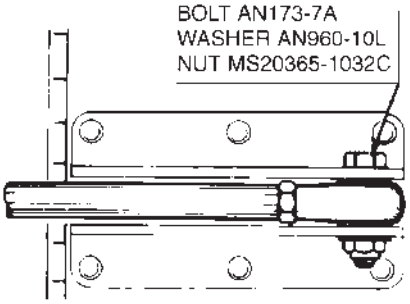
SKETCH A



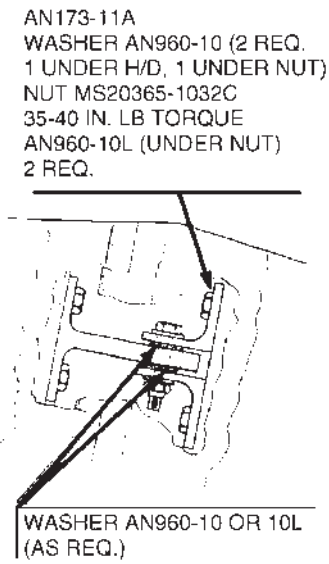
SKETCH B



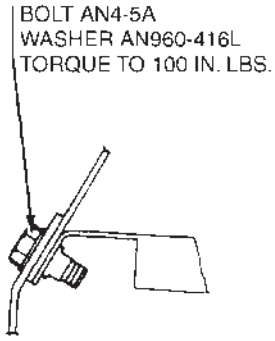
SKETCH C



SKETCH D

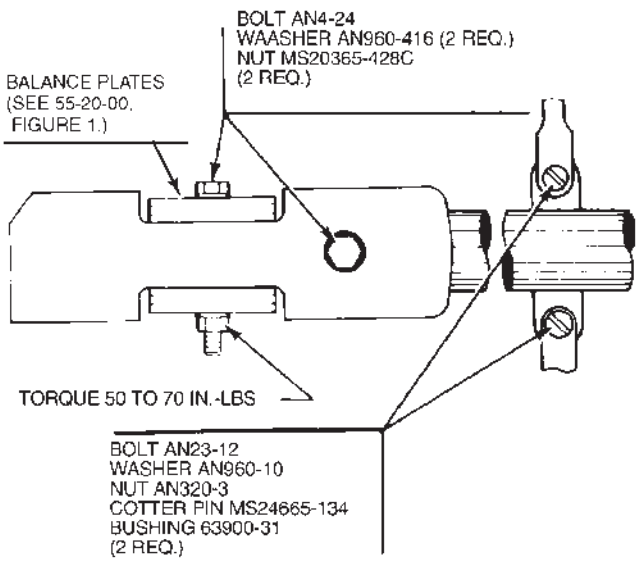


SKETCH E

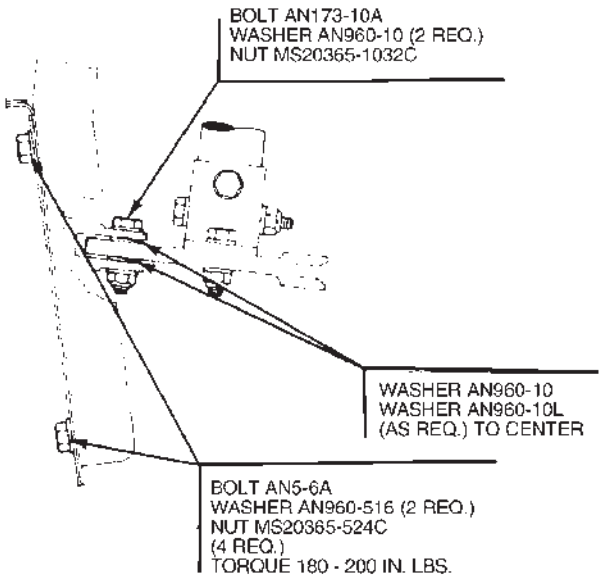


SKETCH F

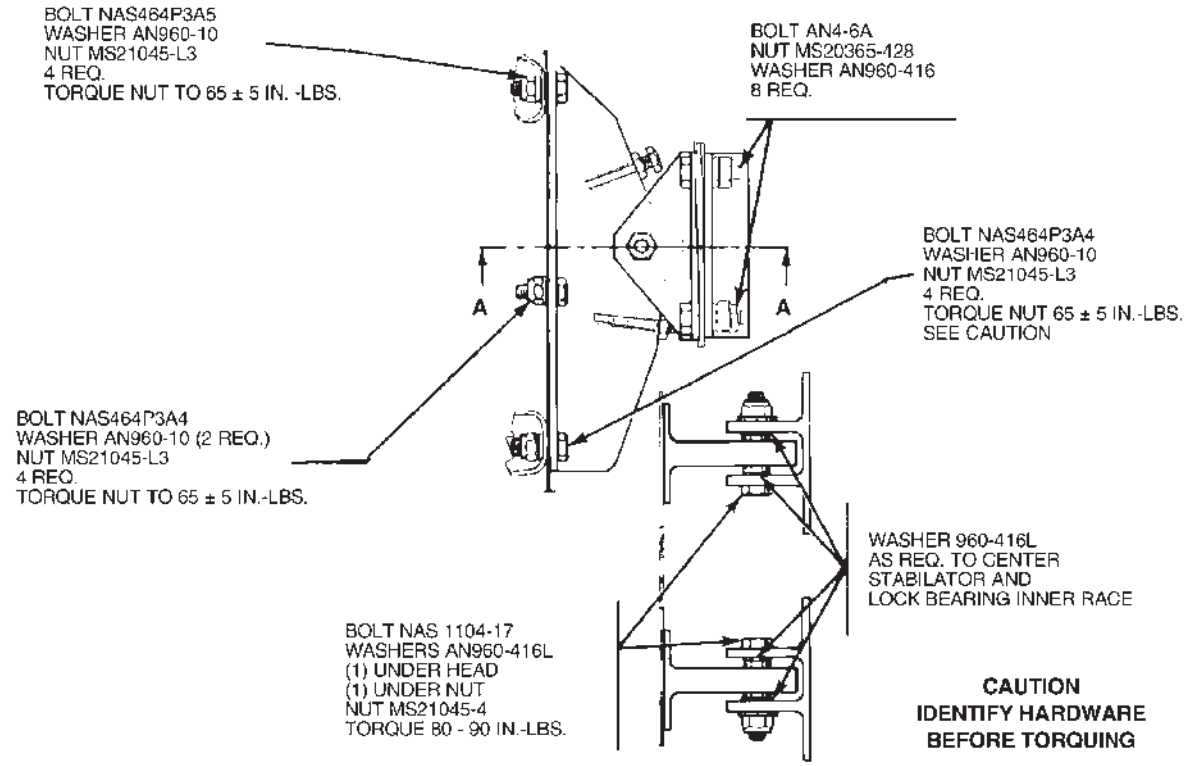
Empennage Installation
Figure 1 (Sheet 1 of 2)



SKETCH G



SKETCH H



SKETCH I

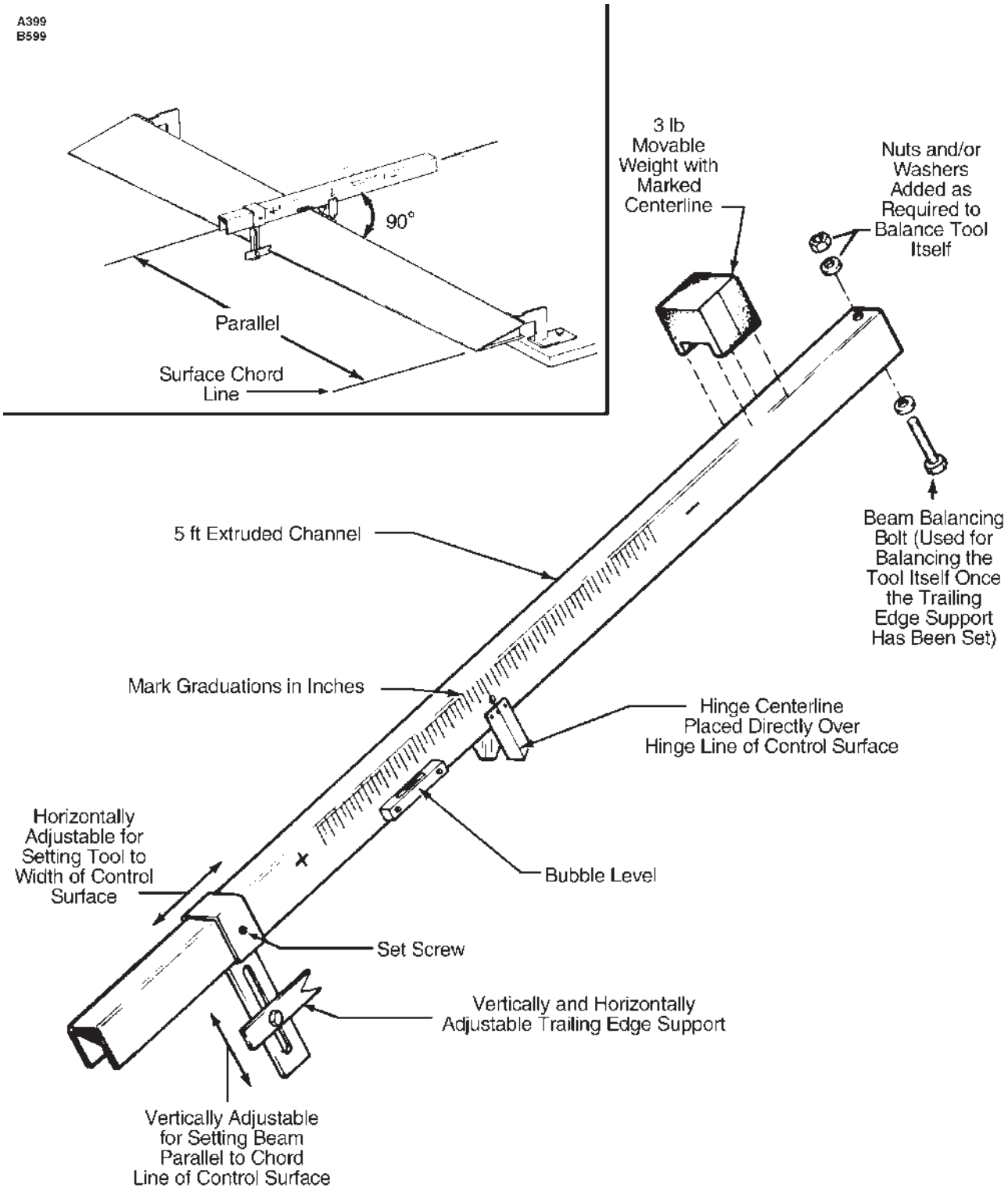
DETAIL A - A

Empennage Installation
Figure 1 (Sheet 2 of 2)

D. After balancing the tool, reattach it to the control surface. Keep the beam positioned 90° from the control surface hinge line.

E. Determine balance of control surface by sliding movable weight along the balance beam.

F. Read the scale when the bubble level has been centered. Multiply by three to determine in-lbs. (I.E., Since the movable weight weighs three pounds, every inch it is moved from the center of the beam equals three inch-lbs of force.)



Control Surface Balancing Tool
Figure 2

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STABILATOR

CAUTION: CONTROL SURFACE SKINS MUST BE REPLACED IF THEY SUSTAIN DAMAGE OR EXHIBIT CRACKS.

1. Stabilator

See “Figure 1” on page 55002.

A. Removal

CAUTION: AT EACH REMOVAL OF THE STABILATOR, CONDUCT ATTACH BRACKETS CORROSION CONTROL INSPECTION, BELOW.

(1) Remove the screws from around the upper and lower tail cone fairings and remove the assemblies separately.

(2) Block the trim cable at the barrel of the trim screw assembly to prevent the cable from unwrapping.

(3) Remove the access panel to the aft section of the fuselage located at the back wall of the baggage compartment.

(4) Install cable blocks (see Figure 2 in 27-30-00) on the stabilator trim control cable at the first set of pulleys forward of the cable turnbuckles to prevent the forward cable from unwrapping.

(5) Disconnect the trim cables at the turnbuckles within the aft section of the fuselage.

(6) Relieve tension from the stabilator control cables by loosening one of the cable turnbuckles in the aft section of the fuselage.

(7) Disconnect the stabilator control cables from the stabilator balance arm by removing the connecting hardware.

(8) Disconnect the trim assembly from the aft bulkhead of the fuselage by removing the attaching hardware of the horizontal and diagonal support brackets.

(9) Move the trim assembly up through the tail cone fairing cutout in the stabilator and remove, with cable, from the airplane.

(10) Remove the two hinge bolts at the pivot points and remove the stabilator as a complete assembly.

B. Installation

WARNING: IF THE STABILATOR HAS BEEN REPLACED, REPAINTED, OR HAS HAD DEICE BOOTS ADDED OR REMOVED, OR THE TRIM TAB HAS BEEN REPLACED OR REPAINTED: THE STABILATOR MUST BE BALANCED BEFORE INSTALLATION. SEE BALANCING, BELOW.

NOTE: A clearance of 0.25 ± .06 of an inch (6.350 ± 1.524 mm) between the stabilator and the side of the fuselage and 0.18 of an inch (4.572 mm) minimum between all parts of the stabilator and the tail cone assembly must be maintained throughout the stabilator travel. Use a proper washer combination on the stabilator hinges to attain the necessary tolerances.

(1) Insert the stabilator in position and install attaching hinge bolts, washers and nuts. Torque as specified in “Figure 1” on page 55002.

(2) Move the trim assembly through the cutout in the stabilator and attach the brackets of the assembly to the aft bulkhead with bolts, washers and nuts. Insert the trim cable ends into the fuselage.

(3) Attach the stabilator control cables to the stabilator balance arm with clevis bolts, bushings, washers, nuts and cotter pins.

(4) Connect the ends of the fore and aft trim cables at the turnbuckles within the aft section of the fuselage.

(5) Remove the cable block from the trim control cable within the fuselage.

(6) Set stabilator control cable tension and check rigging and adjustment according to Rigging and Adjustment, Stabilator Control Cables, 27-30-00.

(7) Remove the cable blocks from the trim cable at the barrel of the trim screw assembly.

(8) Set stabilator trim control cable tension and check rigging and adjustment according to Rigging and Adjustment, Stabilator Trim Controls (Aft), 27-30-00.

(9) Remove the pad from the aft section of the fuselage and replace the access panel.

(10) Install the tail cone fairing and remove tail stand.

2. Stabilator Trim Tab

See “Figure 1” on page 55002.

A. Removal

(1) Disconnect the stabilator trim control rod by removing the bolts that attach the control rod to the stabilator trim tab.

(2) Remove the stabilator trim hinge pins by cutting one end of the wire pins and removing.

(3) The stabilator trim tab can now be removed.

B. Installation

(1) Place the trim tab in position on the aft end of the stabilator.

(2) Replace the old hinge pins with new pins.

(3) Insert the pins and secure by bending the end to a 45 degree angle.

(4) Install the control rod and attach with the four bolts and washers.

(5) The trim tab free play must not exceed 0.125 inches (3.175 mm) maximum.

3. Checking Free Play

A. Stabilator

Check the stabilator for any free play at its attachment points by grasping each half near the tip and gently trying to move it up and down, fore and aft, and in and out. No play is allowed.

B. Stabilator Trim Tab

Set the stabilator trim tab in neutral position. This neutral position is determined with the airplane properly rigged per instructions given in 27-30-00 and the trim indicator at its neutral position. Obtain a straightedge long enough to extend from the ground up to a few inches above the trim tab trailing edge. Place the straightedge next to the trim tab inboard (center) trailing edge, secure the stabilator in neutral and grasping the tab, gently move it up and down, mark the limit of tab free play on the straightedge. The overall travel (free play) must not exceed 0.125 of an inch (3.175 mm). The use of a dial indicator and fixed stand is recommended.

4. Balancing Stabilator

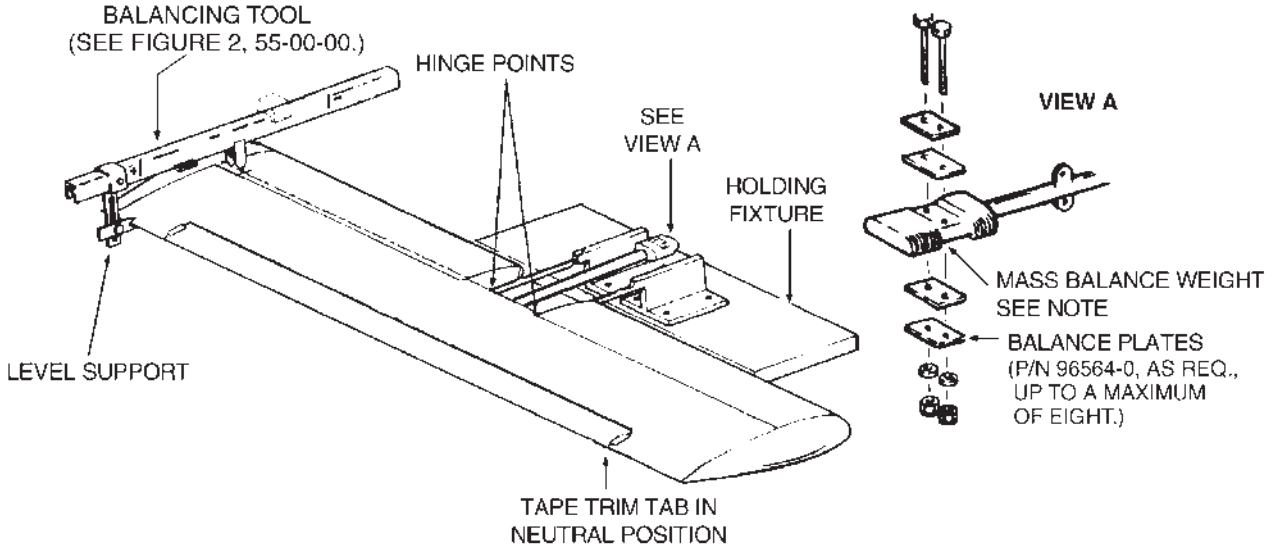
(PIR-PPS50011-1, Rev. Q.)

Refer to “Figure 1”.

To balance stabilator, assembly must be complete including trim tab, tab pushrod and end bearing, stabilator tips, all attaching screws, and in its final flight configuration as described in 55-00-00, Balancing Equipment. Before balancing, tape trim tab in neutral position with a small piece of tape. Place complete assembly horizontally on knife edge support in a draft-free area in a manner that allows unrestricted movement. Do not place tool on trim tab. Calibrate tool as described in “Balancing Equipment” on page 55001. Read scale when bubble level has been centered by adjustment of movable weight and determine static balance limit. If static balance is not within limits given in “Figure 1”, proceed as follows:

A. If the stabilator is out of limits on the leading edge heavy side, remove balance plates from the mass balance weight until the static balance is within limits.

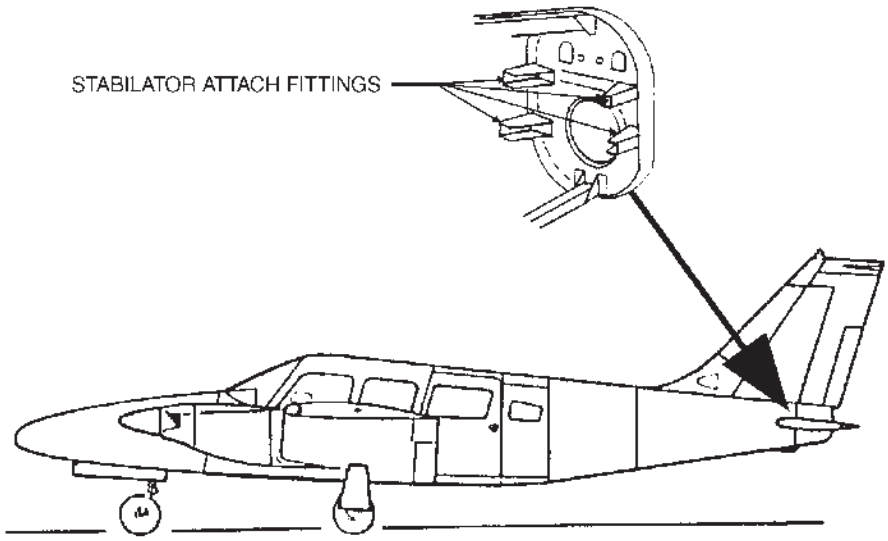
B. If stabilator is out of limits on trailing edge heavy side, add balance plates (as shown in “Figure 1”) to mass balance weight until static balance is within limits.



BALANCE LIMITS (IN-LBS)	
LEADING EDGE HEAVY	TRAILING EDGE HEAVY
0.00	-13.0

NOTE: WEIGHT OF MASS BALANCE WEIGHT ON AIRCRAFT WITH DEICER BOOTS 6.00 LBS. ± 5 OZ. ON AIRCRAFT WITHOUT DEICER BOOTS 6.69 LBS. ± 5 OZ.

Balancing Stabilator
Figure 1



Stabilator Attach Brackets (Typical)
Figure 2

5. Attach Brackets Corrosion Control Inspection

During each annual inspection, use the following method to inspect stabilator attach brackets for rust and corrosion between the steel attach fittings and the adjacent fuselage structure. Take corrective action as required.

A. Remove upper and lower tail cone fairing assembly.

B. Remove the aft fuselage closeout plate assembly on the applicable models.

C. Inspect the steel stabilator attach fittings (4 places) and adjacent fuselage structure for the presence of rust and/or corrosion. (Refer to “Figure 2”.)

NOTE: Refer to F.A.A. Advisory Circular (AC) 43-4A, Corrosion Control for Aircraft.

D. If rust and/or corrosion is present, repair or replace as required. Add corrosion protection per AC43-4A.

E. Install aft fuselage closeout plate assembly. Verify integrity of rubber seals; replace if required

F. Install upper and lower tail cone fairing assembly.

VERTICAL STABILIZER

Vertical Fin

Refer to “Figure 1” on page 55002.

A. Removal

(1) Remove the screws from the upper and lower tail cone fairing; the fin tip cover and the fairing at the forward base of the fin.

(2) Remove the rudder per “Removal” on page 55402.

(3) Disconnect the leads from the antenna terminals (optional) and attach a line to the leads to assist in reinstallation.

(4) Disconnect the wire antenna (optional) that attaches to the leading edge of the fin.

(5) Disconnect the positive lead to the rotating beacon (optional) and attach a line prior to removal. Disconnect the ground lead by removing the attachment screw.

(6) Remove the rudder trim assembly and trim cable in accordance with 27-20-00, Rudder Trim Controls (Aft), Removal.

(7) Remove the bolt and washer that attaches the leading edge of the fin to the fuselage.

(8) Remove the nuts, washers, and bolts that secure the fin spar to the aft bulkhead and remove the vertical fin.

B. Installation

(1) Insert the vertical fin into position and install the bolts, washers, and nuts that secure the fin spar to the aft bulkhead.

(2) Install the bolt and washer that attaches the leading edge of the fin to the fuselage.

(3) Torque fasteners installed in (1) and (2) as specified in “Figure 1” on page 55002.

(4) Install the rudder trim assembly and trim cable per instructions given in 27-20-00, Rudder Trim Controls (Aft), Installation.

(5) Install the rudder per “Installation” on page 55401

(6) Pull the electrical and antenna leads through the vertical fin with the line that was attached.

(7) Connect the antenna leads to the proper terminals and secure with the washers and nuts.

(8) Connect the electrical leads at the disconnects and insulate.

(9) Rig and adjust the rudder and trim control cables as given in 27-20-00: Rigging and Adjustment; and, Rudder Trim Controls (Aft), Rigging and Adjustment.

(10) Check the operation of the radios and electrical lights.

(11) Replace all fairings and access plates and secure with attaching screws.

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RUDDER

CAUTION: CONTROL SURFACE SKINS MUST BE REPLACED IF THEY SUSTAIN DAMAGE OR EXHIBIT CRACKS.

1. Rudder

Refer to “Figure 1” on page 55002.

A. Removal

(1) Remove the screws from around the upper tail cone fairing assembly and remove the fairing.

(2) Remove the rudder tip by removing the attaching screws and disconnect the tail position light wire at the quick disconnect located at the tip of the rudder. Open the access panel in the rear of the baggage compartment to gain access to the aft section of the fuselage.

(3) Relieve the cable tension from the rudder control system by loosening one of the cable turnbuckles in the aft section of the fuselage.

(4) Disconnect the two control cables from the rudder horn by removing the cotter pins, nuts, washers, bushings and bolts.

(5) Disconnect the rudder trim tab push rod from the actuating link by removing cotter pin, nuts, washer and bolt.

(6) Disconnect the jumper lead between the rudder and vertical fin.

(7) Remove the cotter pins, nuts, washers, and bolts from the upper and lower rudder hinge pivot points.

(8) Pull the rudder up and aft from the vertical fin.

B. Installation

WARNING: IF THE RUDDER HAS BEEN REPLACED OR REPAINTED, OR THE TRIM TAB HAS BEEN REPLACED OR REPAINTED; THE RUDDER MUST BE BALANCED BEFORE INSTALLATION. SEE BALANCING, BELOW.

(1) Place the rudder in position and install the hinge bolts, washers, nuts, and cotter pins. Torque as specified in “Figure 1” on page 55002.

NOTE: Use any washer combination of the hinge assembly to suit best centering and operation of the rudder.

(2) Connect the rudder trim tab push rod to the actuating link with bolt, washer, nut and cotter pin.

(3) Connect the tail position light electrical lead at the quick disconnect and cover the connector with an insulating sleeve. Tie both ends of the sleeve with number six electrical lacing twine.

(4) Connect the jumper lead between the rudder and vertical fin.

(5) Connect the control cables to the rudder horn with bolts, washers, nuts and cotter pins.

(6) Check the rudder in accordance with 27-20-00, Rigging and Adjustment.

(7) Install the upper tail cone fairing and rudder tip and secure with the attachment screws. Secure the access panel to the aft section of fuselage.

2. Rudder Trim Tab

Refer to “Figure 1” on page 55002.

A. Removal

(1) Remove the bolt assembly which connects the trim tab actuating arm to the tab assembly.

(2) Remove the trim tab hinge pin and remove the tab assembly from the rudder.

B. Installation

WARNING: IF THE RUDDER HAS BEEN REPLACED OR REPAINTED, OR THE TRIM TAB HAS BEEN REPLACED OR REPAINTED; THE RUDDER MUST BE BALANCED BEFORE INSTALLATION. SEE BALANCING, BELOW.

(1) Position the trim tab assembly into the rudder aligning the two hinge bolts.

(2) Install a new hinge pin. Ensure that at least 0.50 of an inch (12.7 mm) of hinge pin extends out from each end of the hinge.

(3) Bend both ends of the hinge pin to a 30° angle to secure it in place.

(4) Connect the trim tab actuating arm to the bracket and the tab and secure with bolt assembly.

(5) Rig per 27-20-00, Rudder Trim Controls (Aft), Rigging and Adjustment.

3. Balancing Rudder

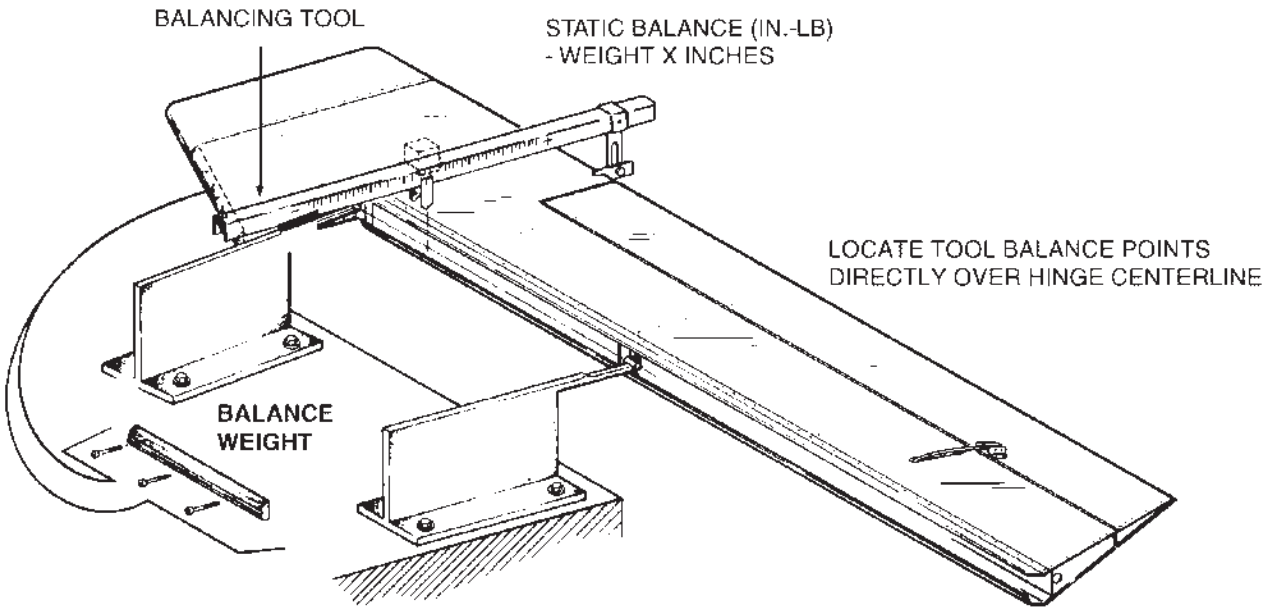
See “Figure 1” on page 55403.

WARNING: ALL CONTROL SURFACES THAT HAVE BEEN REPLACED OR REPAINTED MUST BE BALANCED BEFORE INSTALLATION.

To balance rudder, assembly must be complete including sector assembly and in its final flight configuration as described in “Balancing Equipment” on page 55001. Place complete assembly horizontally on knife edge support in a draft-free area in a manner that allows unrestricted movement. Place tool on rudder with beam perpendicular to hinge centerline. Calibrate tool as described in “Balancing Equipment” on page 55001. Read scale when bubble level has been centered by adjustment of moveable weight and determine static balance limit. If static balance is not within limits given in “Figure 1” on page 55403, proceed as follows:

A. Nose Heavy: This condition is highly improbable; recheck calculations and measurements.

B. Nose Light: In this case, the mass balance weight is too light or the rudder is too heavy because of painting; it will be necessary to strip the paint and repaint. If the rudder is too heavy as a result of repairs, the repair must be removed and the damaged parts replaced.



	LEADING EDGE HEAVY	TRAILING EDGE HEAVY
SENECA IV	-16.00	-35.00
SENECA V	-16.00	-32.00

Balancing Rudder
Figure 1

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CHAPTER

56

WINDOWS

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CHAPTER 56

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CHAPTER 56 - WINDOWS

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FLIGHT COMPARTMENT

1. Windshield

Refer to "Figure 1" on page 56102.

These airplanes have a one-piece windshield. As shown in "Figure 1" on page 56102 , the windshield aft of Fuselage Station 69.8 is fitted in a channel and sealed with vinyl foam tape. Forward of F.S. 69.8, the windshield is held in place by inner and outer collars which are secured to the airplane by two different types of screws. If removing the windshield, mark the screws and collars to ensure the proper screws are replaced in their appropriate positions during installation.

A. Removal

(1) Remove the screws around the forward part of the windshield. (On Seneca V models it may be necessary to also loosen screws on the sideposts.)

NOTE: To facilitate installation, the screws should be cross marked to the collars ensuring the proper screws are replaced in their appropriate positions.

(2) Remove the outer collars. With the tape and sealant between the outer collars and windshield, it may be necessary to carefully pry the collars off the windshield.

NOTE: Save the old or damaged windshield for use as a pattern for trimming the new windshield.

(3) Raise the forward portion of the windshield slightly and slide the assembly forward out of the fuselage channel.

(4) Clean old tape and sealer from collars and channel.

B. Installation

(1) Ensure the outline or outside contour of the new windshield edge properly fits. Grind or cut the new windshield as necessary to acquire proper dimensions.

NOTE: If saved, the old windshield can be used as a pattern for trimming the new windshield.

(2) Apply black vinyl tape to the inside collar surfaces that mate to the windshield and outside collars. Refer to "Figure 1" on page 56102, Section A-A.

(3) Apply vinyl foam tape around the edge of the windshield to be inserted in the fuselage channel. The affected edge of the windshield is that which extends behind F.S. 69.8.

(4) Align the windshield in position and slide it aft into the fuselage channel.

(5) Apply vinyl foam tape (see List of Consumable Materials, 91-10-00) to the inside of the outer collars and loosely connect them to the fuselage around the edge of the windshield.

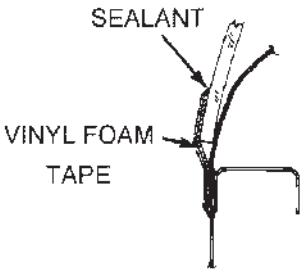
(6) Insert screws into place on collars but do not tighten.

(7) Apply sealant ((PRC) PR 307, as specified in List of Consumable Materials) by forcing it under the flange edges specified in "Figure 1" on page 56102. Mating parts can be separated slightly using a soft wooden wedge or tongue depressor. The sealant must be forced deep into the gap. Avoid bending or scratching aluminum or window surfaces.

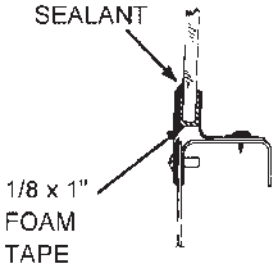
(8) Tighten the screws on the windshield collars.

(9) Clean off excess sealant immediately, using clean rags, plastic scrapers, and solvent. Use only Tripolene or Apperson Solvent No. 12D around windows. Toluol may be used in areas away from windows.

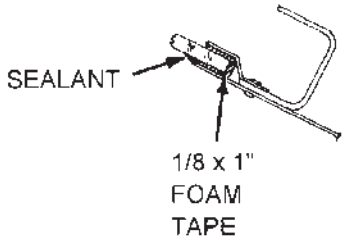
NOTE: Joints may be completely filled. All sealed areas should be smoothly blended after clean up.



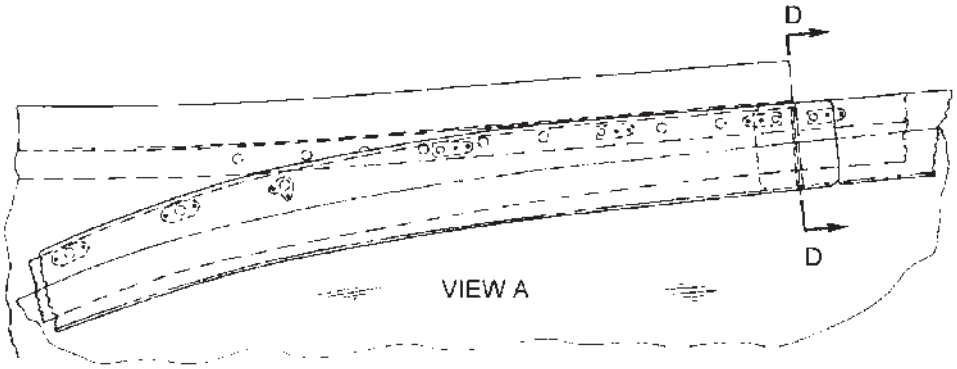
SECTION A-A

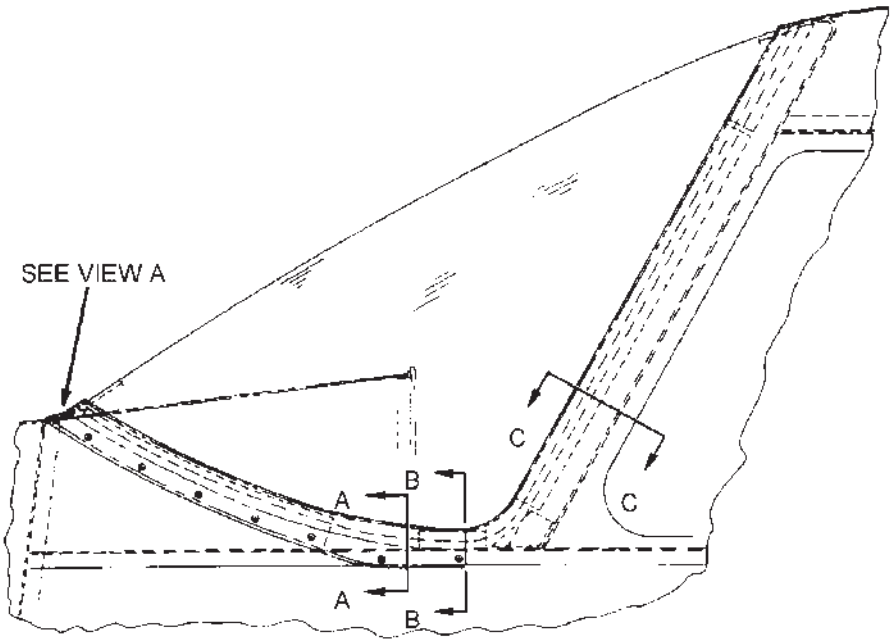


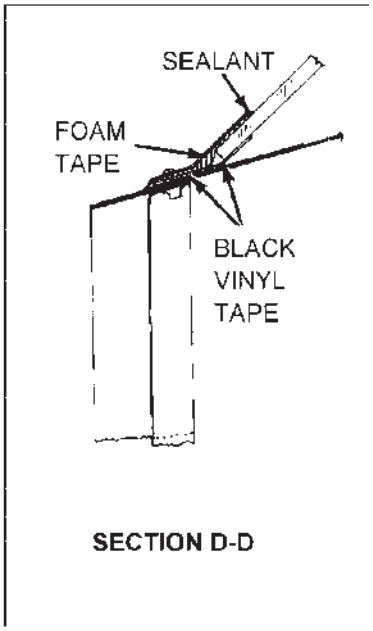
SECTION B-B



SECTION C-C



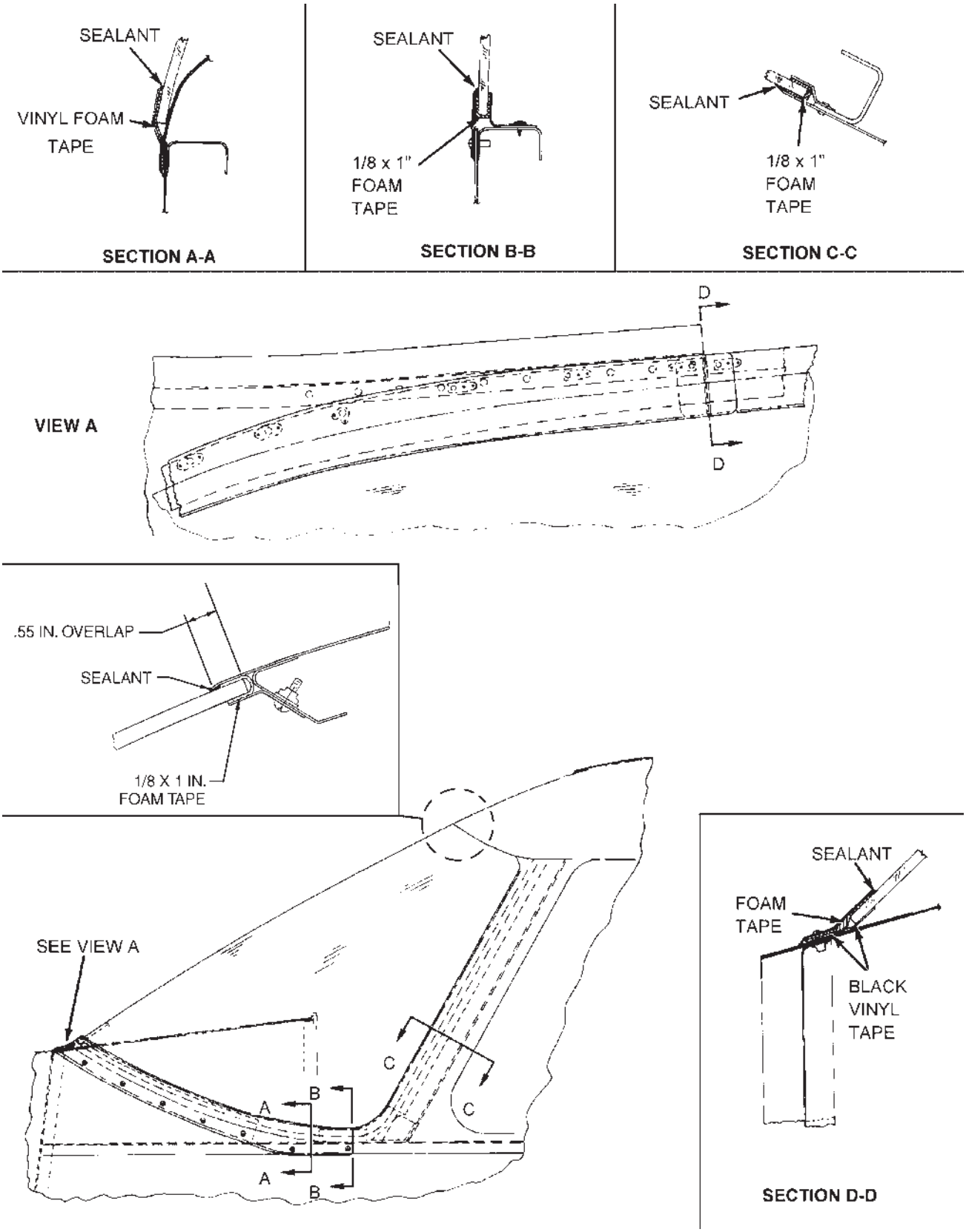




SECTION D-D

Effectivity
Seneca IV

Windshield Installation
Figure 1 (Sheet 1 of 2)



Windshield Installation
Figure 1 (Sheet 2 of 2)

Effectivity
Seneca V

2. Storm Window

Refer to “Figure 2” on page 56105.

The pilot side window assembly includes a storm window, which is a small, openable window. The storm window has a hinge attached with five (5) rivets along its bottom edge. The lower plate of the hinge is attached to the side window assembly with five (5) rivets. The storm window is held closed by a latch that fastens at its top. The latch assembly is attached to the side window assembly with screws, washers and a nut.

A. Removal

(1) Unlatch the storm window.

(2) Hold the storm window in place. Remove the pin which links the two plates of the storm window hinge.

(3) To remove the storm window assembly, lean the top toward you, then grip it along its edges and slide the upper hinge plate away from the lower plate.

B. Installation

(1) Place the storm window over its opening in the side window and hold it in place.

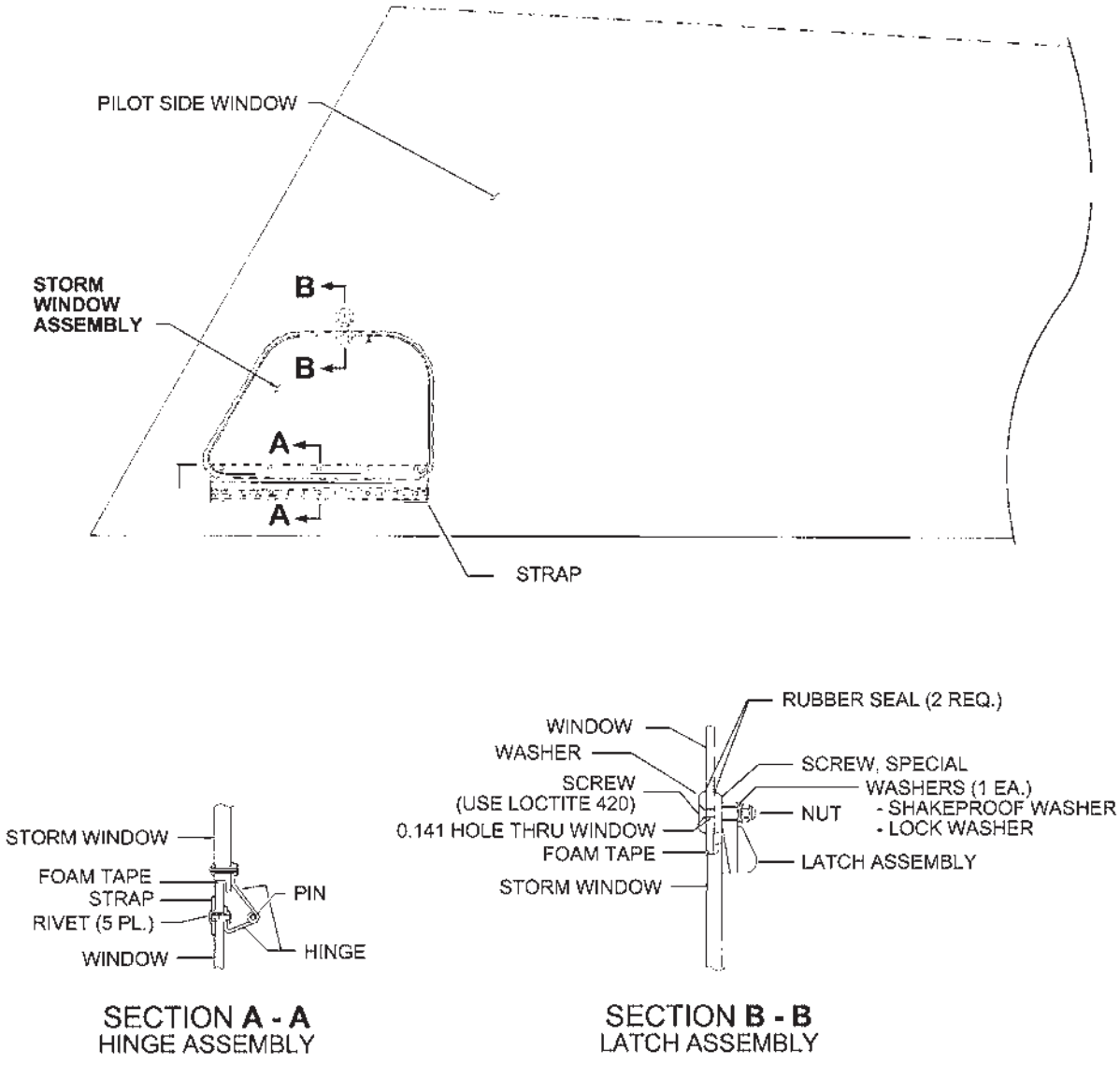
(2) Adjust the storm window and the upper hinge plate so the plate meets the lower hinge plate correctly.

(3) Push the top of the storm window into place and close the latch over it.

(4) Insert the pin which links the two hinge plates.

(5) Crimp hinge end to keep pin from sliding out.

69713 S



Storm Window
Figure 2

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PASSENGER COMPARTMENT

1. Side Windows

Refer to “Figure 1”.

These airplanes are equipped with single pane side windows.

A. Removal

(1) Remove the molding and retainer from around the window by removing attaching screws.

(2) Carefully remove the damaged window from the frame.

(3) Remove old tape and sealer from window frame and molding.

NOTE: A damaged window should be saved to provide a pattern for trimming the new window.

B. Installation

Install all side windows except the cabin door window as follows:

(1) Match new window to old. If necessary, cut or grind the new window to the same dimensions.

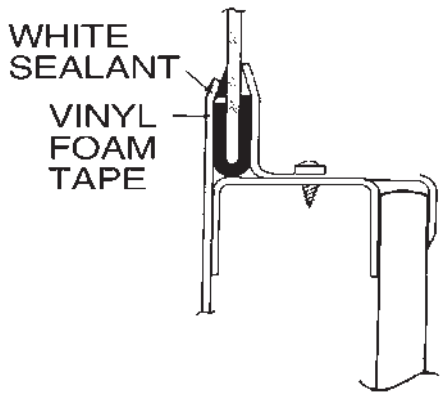
(2) Apply 1/8 in. by 1 in. vinyl foam tape, Norton V510 or equivalent (i.e. - Tape - Vinyl Foam, Type 2; 91-10-00, Consumable Materials), around entire edge of window.

(3) Insert the window into the frame, install the retainer moldings and attachment screws, but do not tighten. Take care not to damage or dislocate the vinyl foam tape.

(4) Apply polyurethane, urethane, acrylic, or polysulfide sealant (i.e. - Sealant - Window and Airframe; 91-10-00, Consumable Materials), completely around the outer surface of the window at all attachment flanges as indicated in “Figure 1”. Force the sealant between the mating parts, which may be separated slightly using a soft wooden wedge or a tongue depressor. Force sealant deep into the gap. Take care to avoid bending or scratching aluminum or window surfaces. Joints should be completely filled, and blended smoothly with adjacent surfaces after clean-up.

(5) Tighten attachment screws until vinyl foam tape is compressed approximately 25 percent.

(6) Remove excess sealant from window areas using rags, disposable wipers or plastic scrapers. A tool made of acrylic sheet with a wedged end (.25 inch thick and 1.5 inch wide) can be fabricated and used. Tirpolene solvent or Apperson solvent No. 120 may be used.to clean polysulfide sealants.



Side Window Installation
Figure 1

2. Cabin Door Windows

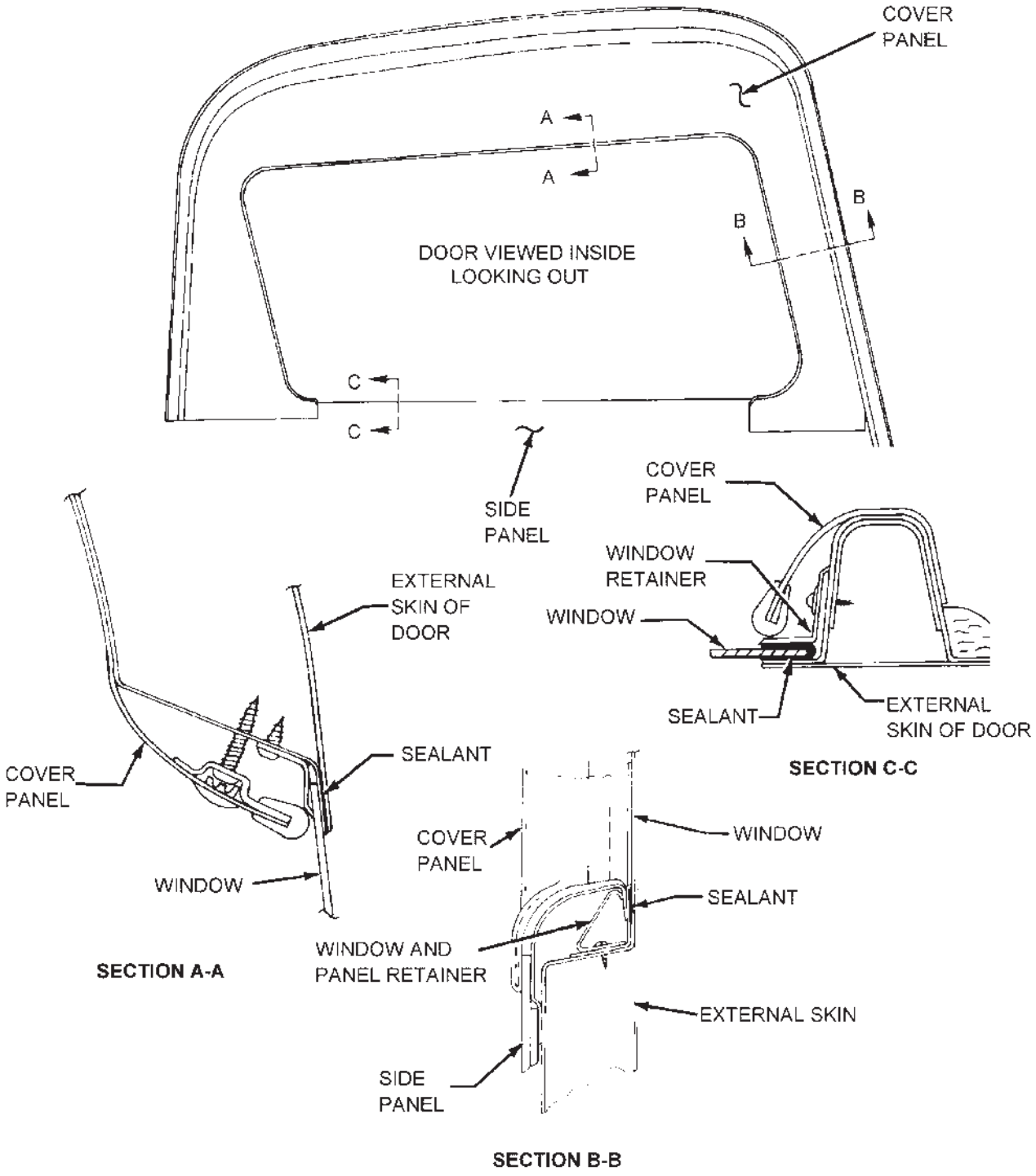
Refer to “Figure 2” on page 56203.

A. Removal

Remove cabin door windows per Side Windows, Removal, above.

B. Installation

(1) Apply sealant, with protective paper in place, to edges of windows.
(2) Remove protective paper just before installing window.
(3) Insert window in frame and apply hand pressure to the perimeter of the window by using a narrow rubber roller.
(4) Install window retention parts and hardware.



Cabin Door Window Installation
Figure 2

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CHAPTER

57

WINGS

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CHAPTER 57

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	5	Oct 10/23			
	6	Oct 10/23			
57-20-00	1	Oct 10/23			
	2	Sep 30/18			
57-40-00	1	Oct 10/23			
	2	Oct 10/23			
	3	Oct 10/23			
	4	Oct 10/23			
	5	Oct 10/23			
	6	Oct 10/23			
	7	Oct 10/23			
	8	Oct 10/23			
	9	Oct 10/23			
	10	Oct 10/23			
	11	Oct 10/23			
	12	Oct 10/23			
57-50-00	1	Sep 30/18			
	2	Sep 30/18			
	3	Sep 30/18			
	4	Sep 30/18			
	5	Sep 30/18			
	6	Sep 30/18			

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CHAPTER 57 - WINGS

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GENERAL

1. Description

Each wing is an all metal, full cantilever, semi-monocoque type structure with removable tips and access panels. Attached to each wing is an aileron, flap, main landing gear and power plant. Installed in each wing ahead of the main spar are two metal fuel tanks with a capacity of 24.5 U.S. gallons each, or 49 U.S. gallons per wing. Also, a 15 U.S. gallon bladder cell is interconnected between the two tanks, providing each wing with a capacity of 64 U. S. gallons, giving a total capacity of 128 U. S. gallons.

The wings are attached to each side of the fuselage by inserting the butt ends of the main spars into a spar box carry through. The spar box is an integral part of the fuselage structure which provides, in effect, a continuous main spar with splices at each side of the fuselage. There are also fore and aft attachments at the front and rear spars.

NOTE: The major subassemblies of the wing may be removed individually or the wing may be removed as a unit. To remove a wing, a fuselage and wing supporting cradle is required.

2. Repairs

WARNING: NO ACCESS HOLES ARE PERMITTED IN ANY CONTROL SURFACES.

WARNING: USE OF PATCH PLATES FOR REPAIRS OF ALL MOVABLE CONTROL SURFACES IS PROHIBITED. USE OF ANY FILLER MATERIAL NORMALLY USED FOR REPAIR OF MINOR DENTS AND/OR MATERIALS USED FOR FILLING INSIDE OF SURFACES IS ALSO PROHIBITED ON ALL MOVABLE CONTROL SURFACES.

CAUTION: CONTROL SURFACE SKINS MUST BE REPLACED IF THEY SUSTAIN DAMAGE OR EXHIBIT CRACKS.

Structural repair methods used must be in accordance with FAA Advisory Circular 43.13-1, latest revision. See also 51-70-00.

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MAIN FRAME

1. Wing Rib Assembly W.S. 49.25

NOTE: The following incorporates the requirements of Piper Service Bulletin No. 1161D.

In S/N's 3447001–3447029 and 3449001–3449508, the factory installed left and right rib assemblies, located forward and aft of the main spar at W.S. 49.25, can crack under certain conditions. The cracking is typically observed vertically along the bend radius of the flange common to the main spar and the main landing gear side brace attach fitting (see "Figure 1"). Accordingly, in S/N's 3447001–447029 and 3449001–3449508, perform the following inspections at the intervals indicated:

NOTE: Installation of Kits No. 767-397 (LH) and 767-398 (RH) (with kit instructions marked as drawing number (Dwg.No.) 88373 revision "J" (or later)) will eliminate the following 100 hour and 500 hour repetitive inspection requirements.

A. 100 Hour Wing Rib Inspection

In S/N's 3447001 thru 3447029 and 3449001 thru 3449508, for airplanes which have not installed Kits No. 767-397 (LH) and 767-398 (RH) (with kit instructions marked as Dwg.No. 88373 revision "J" (or later)) each 100 hours time-in-service:

(1) Place the airplane on jacks per 7-10-00.

(2) In both the right and left wings:

(a) Inspect the Forward and Aft Rib Assemblies at W.S. 49.25 for any evidence of cracks (see "Figure 1" on page 57103), limited to a visual examination.

(b) Inspect the exposed (upper half) portion of the bend radius of the flange common to the Forward and Aft Rib Assemblies and the Main Spar Web.

NOTE: Prior to visual inspection, wipe surfaces clean using a soft cloth dampened with isopropyl alcohol, mineral spirits, naphtha, or other suitable cleaning agent.

NOTE: This visual inspection shall be accomplished using a 10X magnifier, a mirror and a suitable light source or other equipment capable of providing equal or better resolution.

(3) On the Forward Rib Assembly, if any crack is detected visually, then the repair or replacement of the Forward Rib Assembly is required. (Refer to the Parts Catalog, P/N 761-887, for the applicable Rib Assembly.)

(4) On the Aft Rib Assembly, if any crack is detected visually, proceed to this detailed examination.

(a) Remove the Main Landing Gear Side Brace by removing the five (5) bolts that fasten it to the wing structure. Retain hardware for reassembly.

(b) Inspect the cracked Rib Assembly in the bend radius of the flange common to the Main Spar Web (as shown in "Figure 1" on page 57103), using dye penetrant inspection techniques, to determine the path of crack propagation. Use guidance from FAA Advisory Circular AC43.13-1B.

(c) Determine if trimming, as shown in "Figure 1" on page 57103, will remove all the material affected by the crack.

1) If so, install the appropriate kit (see above).

2) If this cannot be accomplished, then replace the cracked Rib Assembly. To relieve the recurring inspection requirements, the appropriate kit must also be installed, with the following exception:

If the replacement Rib Assembly is P/N 78475-010 and/or 78475-011, or higher, then the kit is not required. (Refer to the Parts Catalog, P/N 761-887, for the applicable Rib Assembly.)

(5) If no cracks are detected, then reassemble/reinstall any parts or components previously removed.

(6) Verify the proper functioning of landing gear.

(7) Verify that the gear are down and locked and remove airplane from jacks.

(8) Make an appropriate logbook entry.

B. 500 Hour Wing Aft Rib Inspection

In S/N's 3447001–3447029 and 3449001–3449508, for airplanes which have not installed Kits No. 767-397 (LH) and 767-398 (RH) (with kit instructions marked as Dwg. No. 88373 revision "J" (or later)), each 500 hours time-in-service:

(1) Place the airplane on jacks per 7-10-00.

(2) In both the right and left wings:

(a) Remove the Main Landing Gear Side Brace by removing the five (5) bolts that fasten it to the wing structure. Retain hardware for reassembly.

(b) Inspect the Rib Assembly for evidence of cracks in the bend radius of the flange common to the Main Spar Web (as shown in "Figure 1" on page 57103), using dye penetrant inspection techniques. Use guidance from FAA Advisory Circular AC43.13-1B.

(3) If a crack is detected, identify the path of the propagation. Determine if trimming, as shown in "Figure 1" on page 57103, will remove all the material affected by the crack.

(a) If so, install the appropriate kit.

(b) If this cannot be accomplished, then replace the cracked Rib Assembly. To relieve the recurring inspection requirements, the appropriate kit must also be installed, with the following exception:

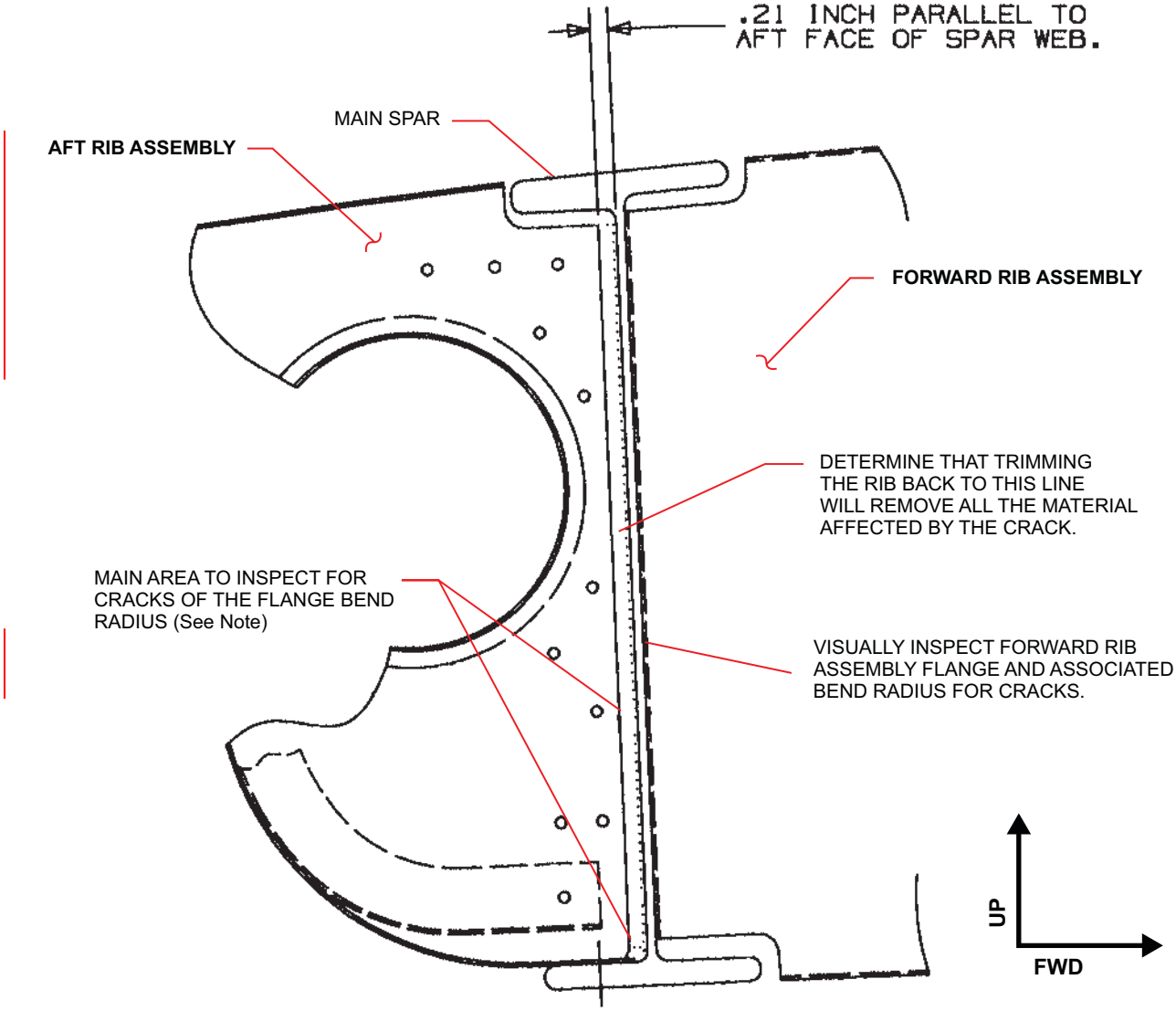
If the replacement Rib Assembly is P/N 78475-010 and/or 78475-011, or higher, then the kit is not required. (Refer to the Parts Catalog, P/N 761-887, for the applicable Rib Assembly.)

(4) If no cracks are detected, reassemble/reinstall any parts or components previously removed.

(5) Verify proper functioning of landing gear.

(6) Verify gear are down and locked and remove airplane from jacks.

(7) Make an appropriate logbook entry.



NOTE: Applicable to the initial 50 hour inspection and 500 hour recurring inspection. Not applicable to the recurring 100 hour inspection.

LOOKING OUTBOARD AT LEFT WING
(RIGHT WING OPPOSITE)

Wing Rib Inspection (W.S. 49.25)
Figure 1

2. Overspeed/Overload Wing Inspection

A. Validate that the external wing skins and internal wing web structure (rib and spar webs) have no permanent buckles and remain flat to within the tolerances as specified below and all line elements (spar flanges, stringers, stiffeners, etc.) remain straight to within tolerances as specified below.

NOTE: Remove all access panels as required.

Suspected contour deviation conditions shall be evaluated span-wise along element lines. Use a straight edge that spans across 2 bay areas (bound by 3 rib lines). With the straight edge resting against the surface, measure the maximum depth (g) and width (b) of each gap to verify none exceed 0.0065 in. depth per inch width. For multiple waves, up to 2 adjacent gaps may be averaged out using the depth of the bigger gap and the width (b) across two bays to meet the 0.0065 in. depth per inch width.

If a gap spans over a rib area, center that gap in between 2 bays to measure the depth and width. If the gap extends beyond the width of 2 bay areas, limit the measurable width to the 2 bay areas. Any out of plane condition that is more than 0.0065 in. deep per inch width, the total deviation shall be cause for rejection.

See below for calculation examples, Chart 1 for maximum gap depths for specific widths, and “Figure 2” on page 57105 for examples of typical contour deviations.

$$b \times 0.0065 = g$$

Example 1: The maximum depth is measured as 0.028 inches and the maximum width (b) of the condition is 5 inches then the condition is acceptable.

Calculation: 5 x 0.0065 = 0.0325 (g - maximum allowable depth over the distance)

Example 2: The maximum depth is measured as 0.028 inches and the maximum width (b) of the condition is 4 inches then the condition is NOT acceptable.

Calculation: 4 x 0.0065 = 0.026 (g - maximum allowable depth over the distance)

NOTE: Out of contour conditions may not be more than 0.060 inches difference between left and right of aircraft centerline (B.L. 0.0) at the same location (to within ± 0.10 inches).

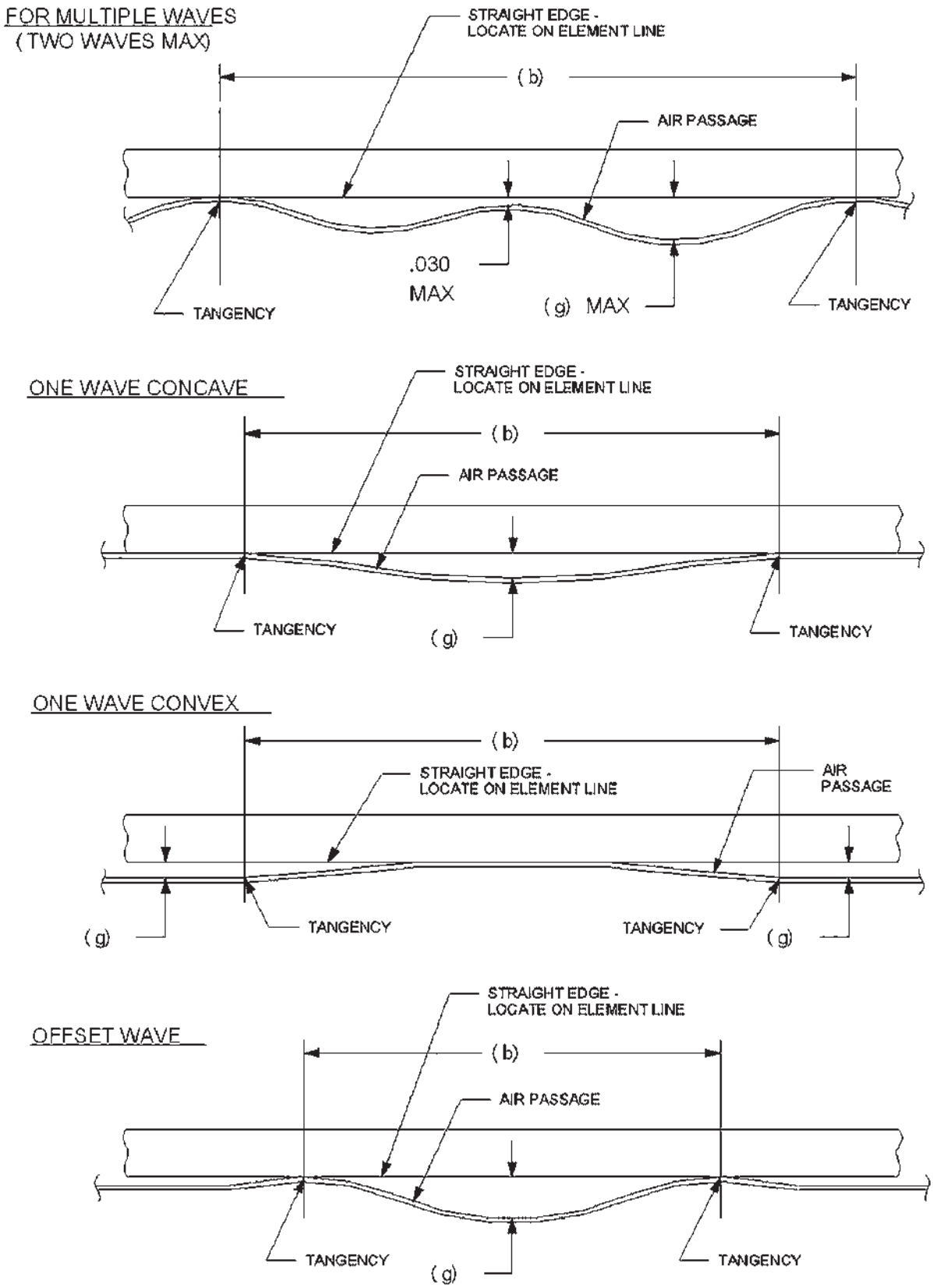
NOTE: Aerodynamic filler is not a repair for unacceptable out of contour conditions.

NOTE: Out of contour conditions are unstable and prone to “oil-can” fatigue and must be repaired.

NOTE: Out of contour conditions shall not be checked in the chord-wise (or hoop) direction without the use of approved contour boards.

CHART 1
MAXIMUM GAP DEPTH FOR SPECIFIC GAP WIDTHS (INCHES)

If the gap width (b) is:	Then the maximum depth (g) allowed is:
3.0	0.020
3.5	0.023
4.0	0.026
4.5	0.029
5.0	0.033
5.5	0.036
6.0	0.039
6.5	0.042
7.0	0.046
7.5	0.049
8.5	0.055
9.0	0.059
9.5	0.062
10.0	0.063



Overspeed/Overload Wing Inspection – Examples of Contour Deviations
Figure 2

B. Visually inspect all bonded members for signs of disbonding along every bonded edge.

C. Visually inspect all wing attachments for signs of dislocation or permanent deformation. Removal of fastener hardware (bolts, nuts, etc.) is required at front and aft spar attachments to validate bolt straightness and integrity. Replace fastener hardware on condition. All torque values are to be verified and torque stripes reapplied as required. (See 57-40-00.)

D. If landing gear and their associated wing attachment fittings are suspected to have endured an atypical “hard” landing, or a wheel tire failure has occurred, they shall be removed and all landing gear components shall be inspected for signs of permanent damage or cracks. Repair or replace on condition. (See 32-10-00, 32-20-00, and 32-40-00.)

E. Symmetry between the left and right wings shall be verified on a leveled aircraft through the use of precise visual inspection techniques and tools such as laser-based distance measuring devices from a fixed reference location. First, level the aircraft using the aircraft leveling procedure outlined in 8-20-00. Using a laser level, record the left and right wing dihedral angles at various wing stations along the main and rear spar centerlines at the upper wing surface. Compare the data between left and right wings for both spars. Angles are to be symmetric to within ± 0.5 degrees.

   If symmetry between the wings has been verified, compare the data between suspected overloaded wings and non-suspect wings from another aircraft of the same type. Angles are to be within ± 0.5 degrees between the suspected and non-suspect wings. If overall wing shape does not pass, then wings should be replaced.

AUXILIARY STRUCTURE

Wing Tip

A. Removal

(1) Remove the screws holding the wing tip to the wing, being careful not to damage the wing or wing tip.

(2) Pull off the wing tip far enough to disconnect the landing light and navigation and strobe light wire assemblies. Be sure to unscrew the ground lead at the wing rib.

(3) Inspect the wing tip to ascertain that it is free of cracks, severe nicks and minor damage. If repair is required, refer to “Repair”.

B. Installation

(1) Place the wing tip in a position that the landing light and navigation and strobe light leads may be connected. Be sure to connect the navigation/strobe ground lead to the wing rib by use of a screw and nut. Ensure that the ground lead is free of dirt and film to insure a good connection.

(2) Insert the wing tip into position and install the screws round the tip. Take care to refrain from damaging the wing tip or wing. Check operation of the lights.

C. Repair

See 51-70-00, Repairs.

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ATTACH FITTINGS

1. Wing

A. Removal

See “Figure 1” on page 57405.

(1) Close the fuel valve and drain the fuel from the wing to be removed. (See 12-10-00, Fuel System, Fuel Tanks, Draining Entirely.)

(2) Drain the brake lines and reservoir. (See 12-10-00, Brake System, Draining Brake System.)

(3) Remove the engine from the wing to be removed. (See Engine, Removal, 71-00-00.)

(4) Drain the hydraulic lines of the landing gear of the wing to be removed by separating the lines and elbows at the actuating cylinder.

(5) Remove the access plate at the wing butt rib and wing inspection panels. (See Access Provisions, 6-40-00.)

(6) Main Wing Spar Hardware Inspection

NOTE: The following inspection incorporates the requirements of Piper Service Bulletin No. 1366.

NOTE: Temporary removal of adjacent access plates and inspection panels will facilitate this inspection. See 6-40-00.

Each time the fasteners common to the main wing spar and the spar box are to be removed, visually examine the fasteners prior to wing removal, as follows:

(a) Identify the location and usage of washer P/N 96352-003 (use the Bolt Legend in “Figure 1 (Sheet 3 of 4)” for reference). They are to include a radius feature along the circumference of the outside diameter, which provides additional positive clearance from the adjacent structure. When properly installed, this radius is oriented to match the radius of the adjacent structure, as shown in “Figure 1” on page 57406, Detail F.

NOTE: Some washers, in inventory or purchased, may be identified as P/N 96352-003, but were manufactured without the radius (that is, they are flat washers). Check all replacement washers for the radius prior to installation.

(b) Examine all washers which share a common mating surface with the aircraft structure, including existing washers and washers to be used as replacements. For P/N 96352-003 washers, verify the presence of the radius feature, as shown in “Figure 1” on page 57406, Detail G. A conforming washer will have an external radius measuring between 0.040 and 0.070 inches.

(c) Verify that the outer edge of all washers, both flat and radiused, have positive clearance from (i.e., aren’t touching or pressing into) the radius of adjacent structure.

- If positive clearance exists at all locations, proceed to step (d).
- If any washer touches or presses into the radius of the adjacent structure, examine for damage to the structure (upon wing removal) and disposition with an FAA-approved repair as needed. Where applicable, replace any existing flat washers with P/N 96352-003 washers (see the Bolt Legend in “Figure 1 (Sheet 3 of 4)”).

(d) Make an appropriate logbook entry documenting compliance with this inspection and/or any necessary repairs.

(7) Remove the front and back seats from the airplane.

(8) Expose the spar box and remove the side trim cockpit panel assembly that corresponds with the wing being removed.

(9) Place the airplane on jacks. (See Jacking, 7-10-00.)

NOTE: To facilitate reinstallation of control cables, power plant controls, and fuel and hydraulic lines, mark cable and line ends in some identifying manner. Attach a line, where applicable, to cables before drawing them through the fuselage or wing.

(10) Disconnect the aileron balance and control cables at the turnbuckles that are located within the fuselage aft of the spar.

(11) If the left wing is being removed, remove the cotter pin from the pulley bracket assembly to allow the left aileron balance cable end to pass between the pulley and bracket.

(12) Disconnect the flap from the torque tube by extending the flap to its fullest degree and removing the bolt and bushing from the bearing at the aft end of the control rod.

CAUTION: PLACE A PROTECTIVE COVER OVER THE ENDS OF FUEL, HYDRAULIC, AND MISCELLANEOUS LINES TO PREVENT CONTAMINATION OR DAMAGE TO LINE FITTINGS AND ENDS .

(13) Disconnect the fuel line at the fitting located inside of the wing by removing the access panel on the forward inboard portion of the wheel well and reaching through to the fuel line coupling.

(14) Remove the clamps that are necessary to release the electrical harness assembly. Disconnect the leads from the terminal strip by removing the cover and appropriate nuts and washers.

(15) With the appropriate trim panel removed, disconnect the hydraulic brake line at the fitting located within the cockpit at the leading edge of the wing.

(16) Disconnect the landing gear hydraulic lines at the fittings aft of the spar and within the fuselage.

(17) If the left wing is being removed, it will be necessary to disconnect pitot and static tubes at the elbows located within the cockpit at the wing butt line.

(18) Arrange a suitable fuselage cradle and supports for both wings.

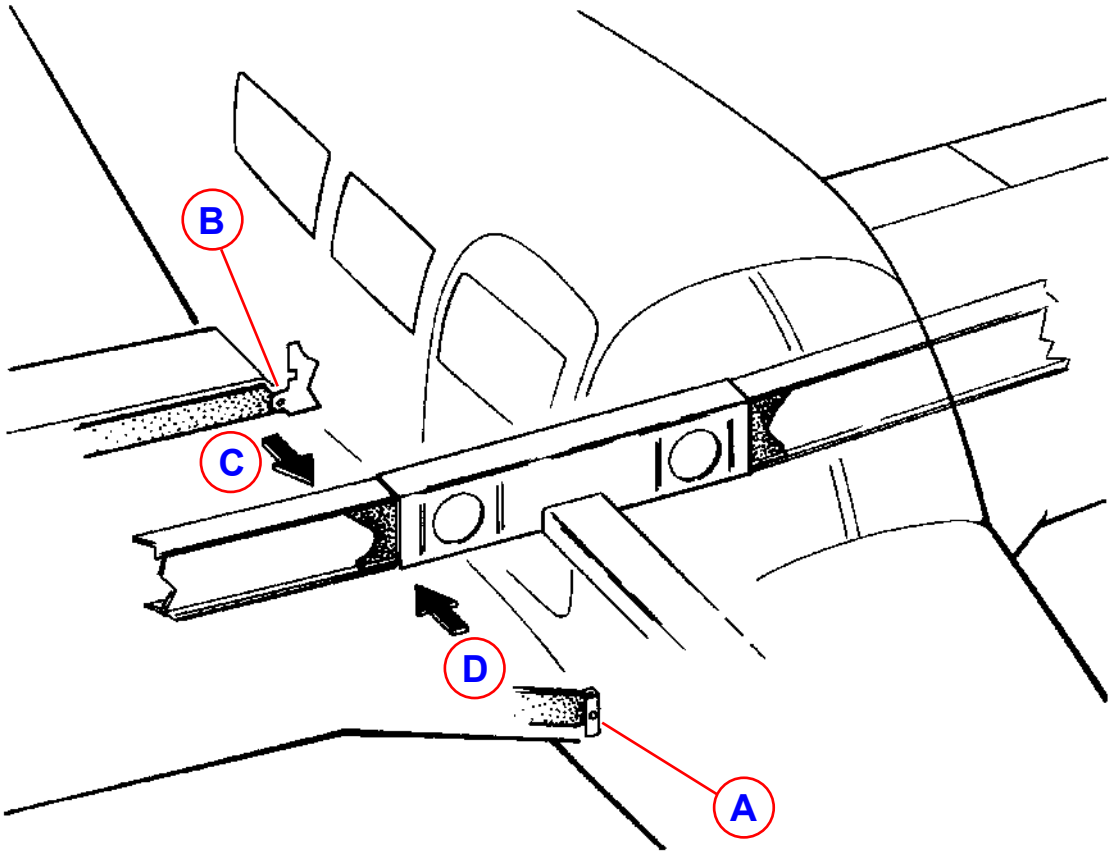
(19) Remove the wing jacks.

WARNING: DO NOT DRIVE BOLTS INTO OR OUT OF THE MAIN SPAR ATTACHING BOLT HOLES. CAUTIOUSLY PREVENT DAMAGE TO THE BOLT HOLES WHILE REMOVING OR INSTALLING THESE BOLTS.

(20) Remove the front and aft spar bolts. Discard the bolts and nuts; retain washers for reinstallation if they are in serviceable condition.

(21) Remove the 18 main spar bolts. Discard the bolts and nuts; retain washers for reinstallation if they are in serviceable condition.

(22) Slowly remove the wing being certain that all electrical leads, control cables, power plant controls, and fuel lines are disconnected.

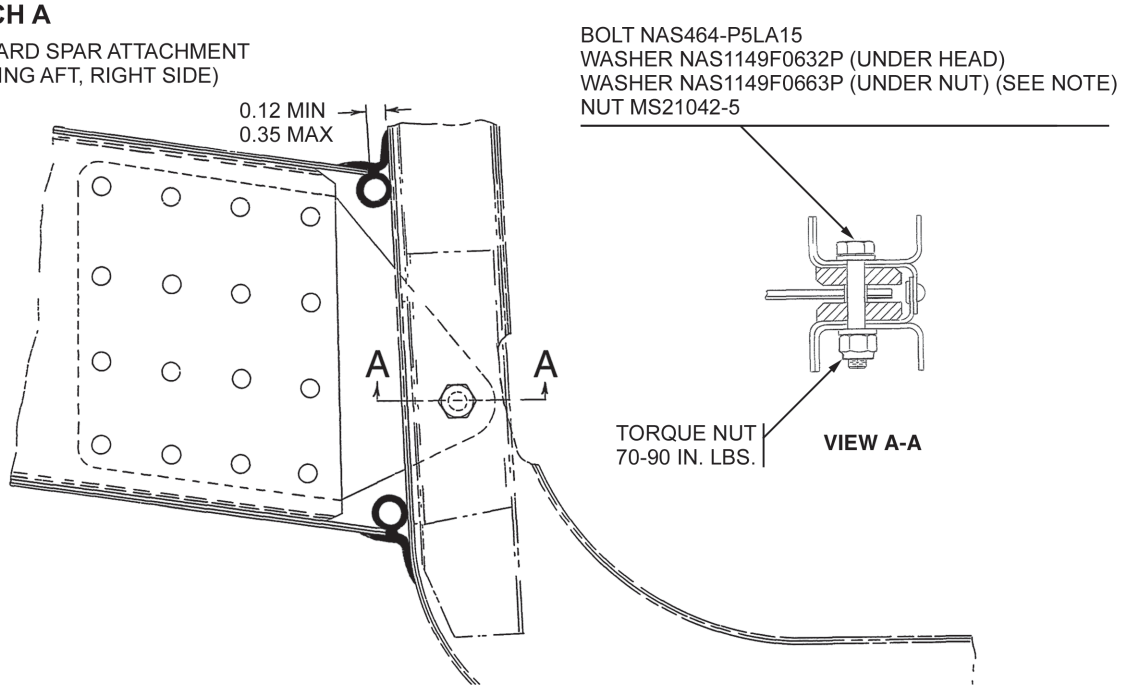


Wing Installation
Figure 1 (Sheet 1 of 4)

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SKETCH A

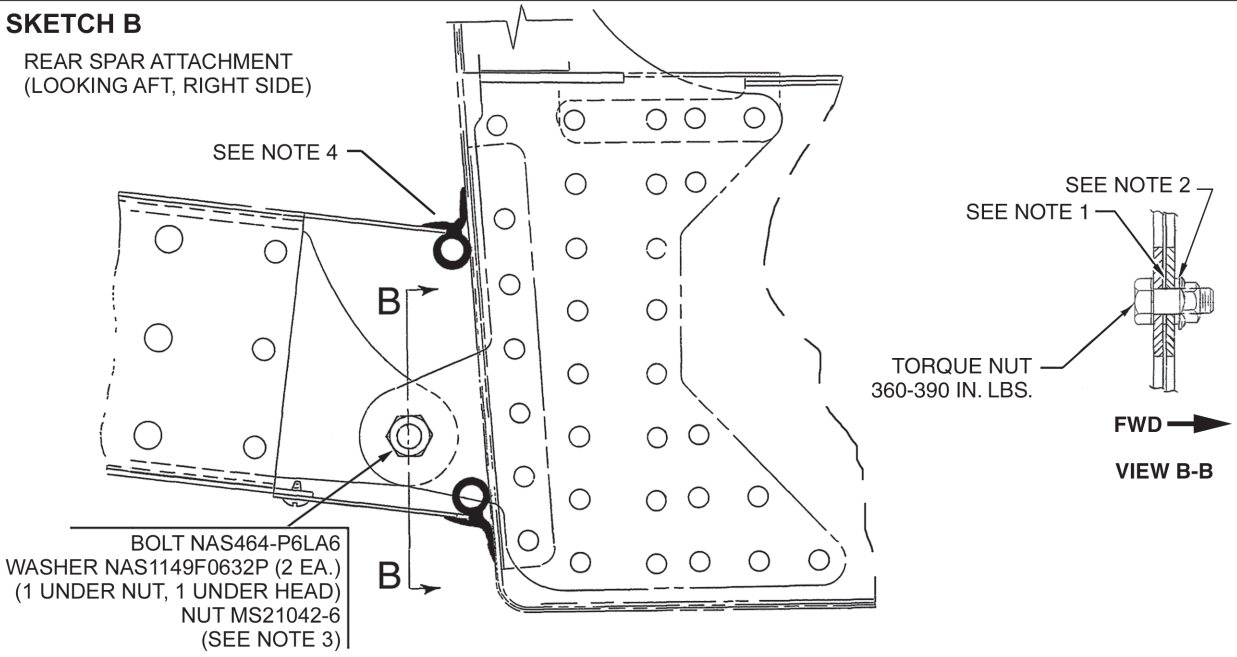
FORWARD SPAR ATTACHMENT
(LOOKING AFT, RIGHT SIDE)



NOTE: Use one (1) NAS1149F0532P or one (1) NAS1149F0563P washer under nut, as required, to leave a minimum of 1½ usable threads exposed on bolt.

SKETCH B

REAR SPAR ATTACHMENT
(LOOKING AFT, RIGHT SIDE)



NOTES: (1) Maximum gap between forward face of wing fitting and aft face of fuselage fitting is 0.15 inch. If required to shim gap, use any combination of NAS1149F0663P and NAS1149F0632P washers.
(2) Use NAS1149F0663P and/or NAS1149F0632P washer(s) under nut, as required, to leave a minimum of 1½ usable threads exposed on bolt.
(3) New service replacement wings are not drilled for the rear spar attachment bolt. Accordingly, using suitable tooling to ensure proper hole alignment, drill a 0.3745/0.3765 Inch hole in the rear attach fitting.
(4) Trim bulb section from molding at top and bottom spar caps and at flap bellcrank arm to prevent flap binding, if required.

Wing Installation
Figure 1 (Sheet 2 of 4)

BOLT LEGEND			WASHER	
POS.	BOLT (1)	NUT (1)	UNDER HEAD	UNDER NUT
A-1	NAS464P6LA17	MS21042-6	NAS1149F0663P	NAS1149F0663P (2 max)
A-2	NAS464P6LA16	MS21042-6	NAS1149F0663P	NAS1149F0663P (3)
A-3	NAS464P6LA16	MS21042-6	NAS1149F0663P	NAS1149F0663P (3)
A-4	NAS464P6LA16	MS21042-6	NAS1149F0663P	NAS1149F0663P (3)
A-5	NAS464P6LA16	MS21042-6	NAS1149F0663P	NAS1149F0663P (3)
B-1	NAS464P6LA16	MS21042-6	NAS1149F0632P	96352-003 (4)
B-2	NAS464P6LA15	MS21042-6	NAS1149F0663P	96352-003 (4)
B-3	NAS464P6LA15	MS21042-6	NAS1149F0663P	96352-003 (4)
B-4	NAS464P6LA15	MS21042-6	NAS1149F0663P	96352-003 (4)
B-5	NAS464P6LA15	MS21042-6	NAS1149F0663P	96352-003 (4)
C-1	NAS464P5LA20	MS21042-5	NAS1149F0532P	NAS1149F0563P (2 max)
C-2	NAS464P6LA20	MS21042-6	NAS1149F0632P	NAS1149F0663P (3)
C-3	NAS464P6LA20	MS21042-6	NAS1149F0632P	NAS1149F0663P (3)
C-4	NAS464P6LA20	MS21042-6	NAS1149F0632P	NAS1149F0663P (3)
C-5	NAS464P6LA21	MS21042-6	NAS1149F0632P	96352-003 (2 & 4)
C-6	NAS464P6LA23	MS21042-6	NAS1149F0632P	96352-003 (2 & 4)
D-1	NAS464P5LA20	MS21042-5	NAS1149F0532P	NAS1149F0563P (2 max)
D-2	NAS464P6LA20	MS21042-6	NAS1149F0632P	NAS1149F0663P (3)
D-3	NAS464P6LA20	MS21042-6	NAS1149F0632P	NAS1149F0663P (3)
D-4	NAS464P6LA20	MS21042-6	NAS1149F0632P	NAS1149F0663P (3)
D-5	NAS464P6LA21	MS21042-6	NAS1149F0632P	96352-003 (2 & 4)
D-6	NAS464P6LA23	MS21042-6	NAS1149F0632P	96352-003 (2 & 4)

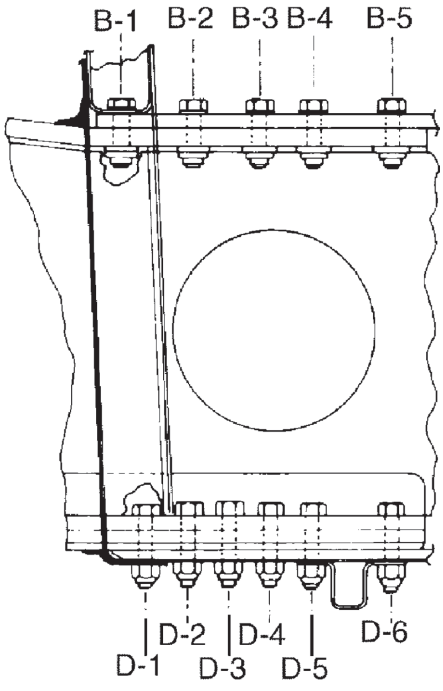
(1) Torque the bolt head on upper spar cap and nut on lower spar cap as follows:
5/16 Bolt to 205–225 in-lb
3/8 Bolt to 360–390 in-lb

(2) A maximum of one (1) NAS1149F0632P and/or one (1) NAS1149F0663P on the appropriate bolt is allowed between the 96352-003 washer and nut.

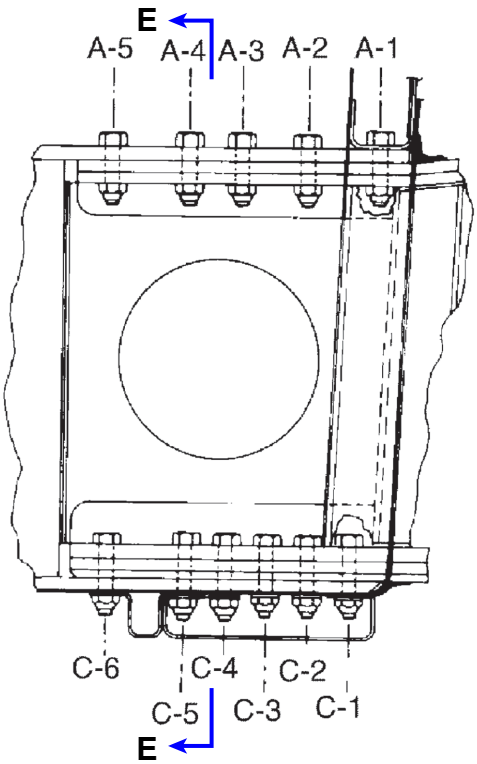
(3) In addition to the one (1) NAS1149F0663P washer specified under nut, an additional one (1) NAS1149F0632P washer or one (1) NAS1149F0663P washer may be installed to accommodate manufacturing tolerances. If the NAS1149F0632P washer is used, place it between the NAS1149F0663P washer and the nut. Total stackup under the nut must not exceed two (2) washers.

(4) The 96352-003 washers feature a rounded edge (i.e., radius) on one side. Ensure that the flat side of each 96352-003 washer is installed against the bolt head, nut, or adjacent washer. The rounded edge is designed to face and abut the aircraft structure.

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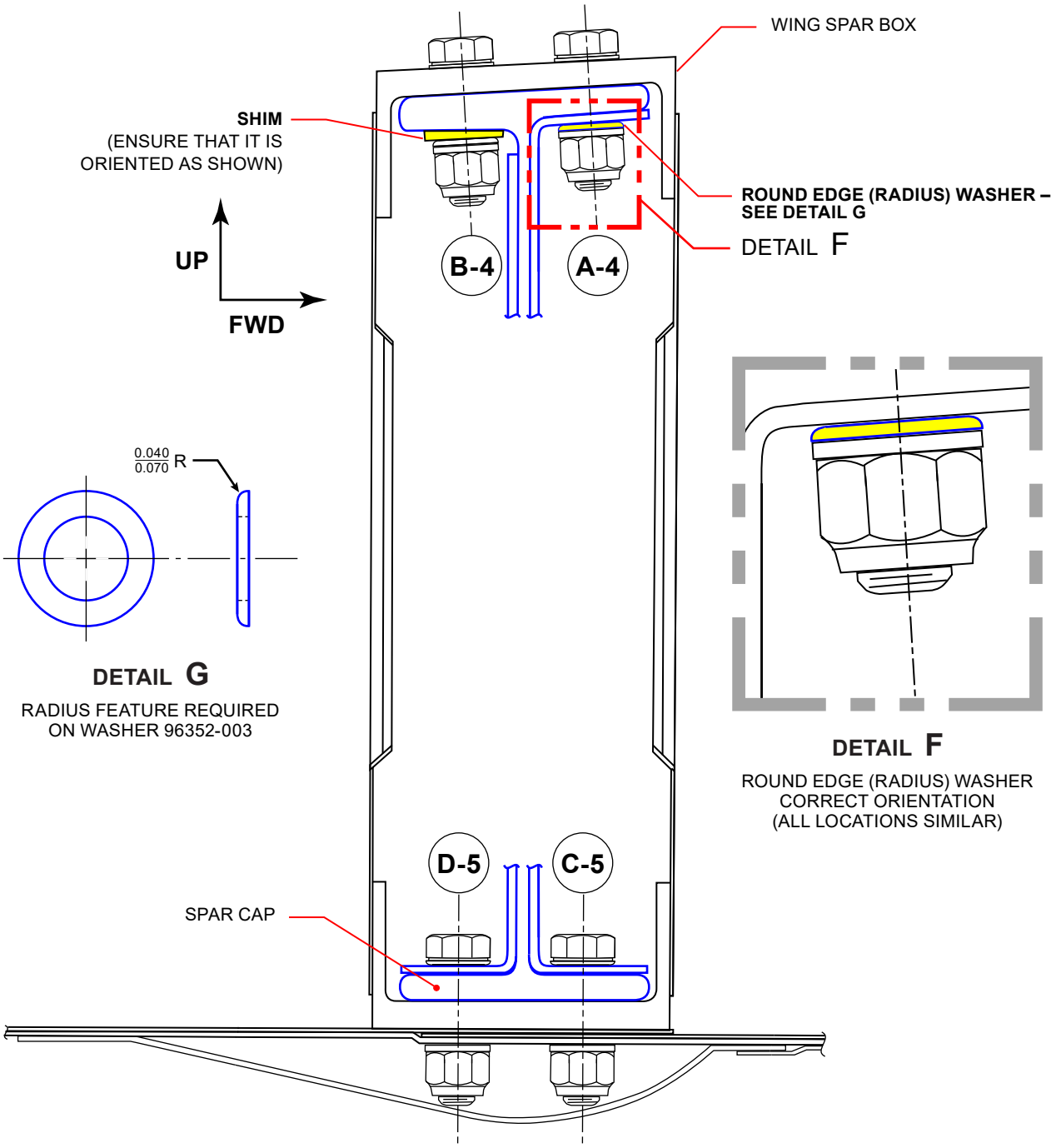


SKETCH C



SKETCH D

Wing Installation
Figure 1 (Sheet 3 of 4)



SECTION E–E
TYPICAL WING SPAR CROSS SECTION

Wing Installation
Figure 1 (Sheet 4 of 4)

B. Installation

See “Figure 1” on page 57403.

NOTE: If a factory replacement wing is being installed, the wing aft spar fitting will need a .3745/.3765 in. diameter hole drilled in it to match the existing hole in the fuselage fitting.

NOTE: If installing a “replacement” wing, perform the Flight Test Procedure provided in 27-30-00, Stall Warning System, upon completion of wing installation.

(1) Ascertain that the fuselage is positioned solidly on a support cradle.

(2) Place the wing in position for installation, with the spar end a few inches from the side of the fuselage and set on trestles.

(3) Prepare the various electrical leads, fuel lines, control cables, and power plant controls for insertion into the wing or fuselage when the wing is eased into place.

(4) Slide the wing into position on the fuselage.

WARNING: DO NOT DRIVE BOLTS INTO OR OUT OF THE MAIN SPAR ATTACHING BOLT HOLES. CAUTIOUSLY PREVENT DAMAGE TO THE BOLT HOLES WHILE REMOVING OR INSTALLING THESE BOLTS.

(5) Install the main spar bolts in accordance with the information given in “Figure 1” on page 57405, Sketches C and D, and “(6) Main Wing Spar Hardware Inspection” on page 57401.

(6) Install the bolt, washers, and nut that attach the front spar and fuselage fitting. A minimum of one washer is required under the bolt head; then add washers as needed to leave a maximum of one and one-half threads visible or a minimum of bolt chamfer exposed. (See “Figure 1” on page 57404, Sketch A, View A–A.)

(7) Install the bolt, washers, and nut that attaches the rear spar and fuselage fitting. It is acceptable to have the faces of the fittings against each other in which case the NAS1149F0632P washer should be used under the bolt head. The NAS1149F0663P washer may be added under the nut when not used as a shim. (See “Figure 1” on page 57404, Sketch B, View B–B.) Check to ensure that no threads are bearing on the forward plate prior to installing the nut.

(8) Torque the main spar bolts in accordance with specifications given in the Bolt Legend in “Figure 1 (Sheet 3 of 4)”. Torque the forward and rear spar to fuselage attach bolts as specified in “Figure 1” on page 57404, Sketches A and B.

(9) Install the wing jacks and tail support to the tail skid with approximately 250 pounds of ballast on the base of the tail support. Remove the fuselage cradle and wing supports.

(10) If the left wing was removed, reconnect the pitot and static tubes at the elbows located within the cockpit at the wing butt line. Replace or install clamps where found necessary.

(11) Connect the hydraulic brake line onto the fitting located within the cockpit at the landing edge of the wing and the landing gear hydraulic lines at the fittings within the fuselage aft of the spar.

(12) Connect the leads to the appropriate posts on the terminal strip and install the washers and nuts. (Refer to the electrical schematics in Chapter 91.) Place the clamps along the electrical harness to secure it in position and install the terminal strip dust cover.

(13) Connect the fuel line at the fitting located inside the wing by reaching through the access panel on the forward inboard portion of the wheel well.

(14) Connect the aileron balance and control cables at the turnbuckles that are located within the fuselage aft of the spar. After the left balance cable has been inserted through the bracket assembly and connected, install a cotter pin cable guard into the hole that is provided in the bracket assembly.

(15) Connect the flap by placing the flap handle in the full flap position; place the bushing on the outside of the rod end bearing and insert and tighten bolt.

(16) Install the engine. (See Engine, Installation, in 71-00-00.)

(17) Check the rigging and control cable tension of the ailerons and flaps. (See Rigging and Adjustment, 27-30-00; and also 27-50-00.)

(18) Service and refill the brake system with hydraulic fluid in accordance with Brake System, 12-10-00. Bleed the system as outlined in 32-40-00 and check for fluid leaks.

(19) Check the fluid level of the landing gear hydraulic system and fill in accordance with 12-10-00, Hydraulic System, Hydraulic Pump/Reservoir. With the airplane sitting on jacks, operate the gear through several retraction and extension cycles to be certain that there are no hydraulic leaks. Test the hydraulic system in accordance with 29-10-00. Ascertain that the landing gear is down and locked.

(20) Service and fill the fuel system in accordance with 12-10-00, Fuel System, Fuel Tanks, Filling. Open the fuel valve and check for leaks and fuel flow.

(21) Check the operation of all electrical equipment, pitot and static systems.

(22) Remove the airplane from jacks.

(23) Install the cockpit trim panel assembly, spar box carpet, the front and back seats and wing root rubber. Replace all the access plates and panels.

2. Wing Aft Spar-to-Fuselage Attachment Hardware 100 Hour Inspection

NOTE: The following incorporates the requirements of Piper Service Letter No. 1087.

Each 100 hours time-in-service, inspect the aft wing spar-to-fuselage attachment hardware for both the right and left wings as follows:

NOTE: Perform the following with weight on wheels (not on jacks).

A. Remove the lower wing-to-fuselage fairing in the area of the wing aft spar.

B. Lower the flaps to gain access to the head of the wing-to-fuselage attachment bolt.

C. Inspect the bolt for tightness in the fittings.

(1) If bolt is tight:

(a) Ensure that the washer(s) are in place on the bolt and ensure the bolts are torqued as specified in “Figure 1” on page 57405, Sketch B.

(b) Reinstall the fairings and make a log book entry indicating completion of this inspection.

(2) If the bolt is loose:

(a) Remove the bolt and inspect the holes in the fittings for size and make sure no out-of-round condition exists outside of allowable limits. The specified hole size for each fitting is 0.3745–0.3765 inch diameter.

(b) If the holes meet the requirements specified above, inspect gap between the fittings.

1) If gap is below 0.032 inches, it is acceptable to install a replacement nut P/N 404-532 (MS21042-6), bolt P/N 402-915 (NAS464P6LA6), and washers P/N 690-629 (NAS1149F0663P) and P/N 690-620 (NAS1149F0632P) and torque as specified in Figure 1, Sketch B. Ensure that 1½ threads of the bolt protrude from the nut.

2) If the gap is above 0.032 inches, install a thick or thin washer (or any combination) as necessary to fill the gap, as described in “Figure 1” on page 57405, Sketch B, prior to installation of the nut/bolt combination above.

NOTE: If the gap (before shimming with washers) exceeds 0.15 inches, contact Piper Technical Support through your Piper Dealer for further instructions.

(c) If the holes do not meet the requirements specified above, contact Piper Technical Support through your Piper Dealer for further instructions.

(d) Upon successful disposition, reinstall the fairings and make a log book entry indicating completion of this inspection.

3. Wing Aft Attach Fitting Inspection

NOTE: This inspection incorporates the requirements of Piper Service Bulletin No. 1244C.

NOTE: See also FAA Advisory Circular (AC) 43-4A Corrosion Control for Aircraft, which is available from https://www.faa.gov.

A. If Piper Kit P/N 764-998 has been installed, determine the revision level of the kit instructions used.

(1) If one of the following are true, then proceed to “Inspection”:

(a) The kit instructions used were DWG 87584 Rev. E or later.

(b) The kit instructions used were DWG 87584 Rev. New, A or B.

(c) The kit was installed on or before July 16, 2013.

(2) If the kit instructions used were DWG 87584 Rev. C or D, or the revision cannot be determined, then:

(a) Remove the aft spar attach bolt. Discard the nut. Examine all of the hardware and the adjacent structure for any damage.

(b) Replace the nut and any damaged hardware; repair or replace any damaged mating surfaces.

(c) Reinstall the hardware (existing and/or replacements) using the correct torque value as shown in “Figure 1” on page 57404, Sketch B:

(d) Proceed to “Inspection”.

B. Inspection

Each 2,000 hours or seven (7) calendar years time-in-service, whichever occurs first, inspect the aft wing attach fittings and adjacent structure for both the right and left wings as follows:

(1) Remove wing inspection plates and fairings, as required to gain visual access to the aft wing attach fittings. See Access Provisions in 6-40-00.

(2) Identify the aft wing attach fittings which are riveted to the aft wing spars. See “Figure 2” on page 574011. Using a solvent-based degreaser spray that conforms to SAE AMS 1525 (such as LPS Presolve Orange Degreaser) and/or an alkaline cleaner that complies with SAE AMS 1526 (such as Chemetall Ardrox 6333A), thoroughly clean the aft wing attach fittings and the adjacent wing spar structure (fore and aft sides of each wing spar), removing surface oil, grease, loose paint, and soil, followed by a clean water rinse and dry.

(3) Carefully inspect for evidence of corrosion. The initial stages of corrosion are often masked by paint coatings and hidden under faying surfaces such as riveted lap joints. Since corrosion products occupy more volume than the original metal, carefully inspect these areas for irregularities such as blisters, flakes, chips, lumps, bulging skins and missing rivets.

(a) If no corrosion is present proceed to “Corrosion Protection”, below.

(b) If superficial corrosion is detected, remove per FAA Advisory Circular AC 43.13-1B, Chapter 6, Section 7. After removal of the corrosion, verify that all the affected areas meet or exceed the minimum thicknesses in “Chart 1”. If the affected areas meet the minimums, then treat the corrosion with guidance from AC 43.13-1B, Chapter 6, Section 8 or 10. Paint affected areas using a good quality aircraft primer that complies with MIL-PRF-23377 (such as Akzo-Nobel 10P30-5). Proceed to “Corrosion Protection”, below.

(c) If corrosion is detected and removal of corrosion in the affected areas will result in a part thickness in any location that is less than the minimum values shown in “Chart 1”, structural repair is required. Proceed to “Structural Repair”, below.

CHART 1
CORROSION LIMITS

Part Description	Minimum Thickness
Aft wing spar (aluminum)	0.060 inches
Aft wing attach fitting (steel)	0.117 inches

NOTE: Measure thickness using a nondestructive inspection method, such as ultrasound, eddy current, or equivalent, with a measurement accuracy of ± 0.005 inches or better.

C. Structural Repair

Order and install Kit, Corrosion - Aft Spar Rework, Piper P/N 764-998. This kit provides parts and instructions to replace (via a splicing operation) the aft spar fitting, as well as the inboard 10.80 inches of the aft wing spar on both left and right hand sides. Corrosion damage beyond these areas may require additional and/or different repairs. If that is the case, contact Piper Aircraft at (772) 299-2141. Piper normal business hours are 7:30 a.m. to 4:30 p.m. ET, Monday through Friday.

NOTE: When ordering Kit 764-998, specify the kit must be revision F. The kit instructions will be marked drawing number 87584, revision F or higher.

Upon completion of structural repair, make a logbook entry documenting installation of Piper Kit No. 764-998 and proceed to Corrosion Protection, below.

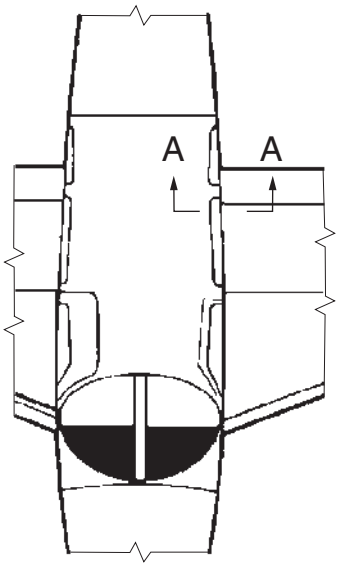
D. Corrosion Protection

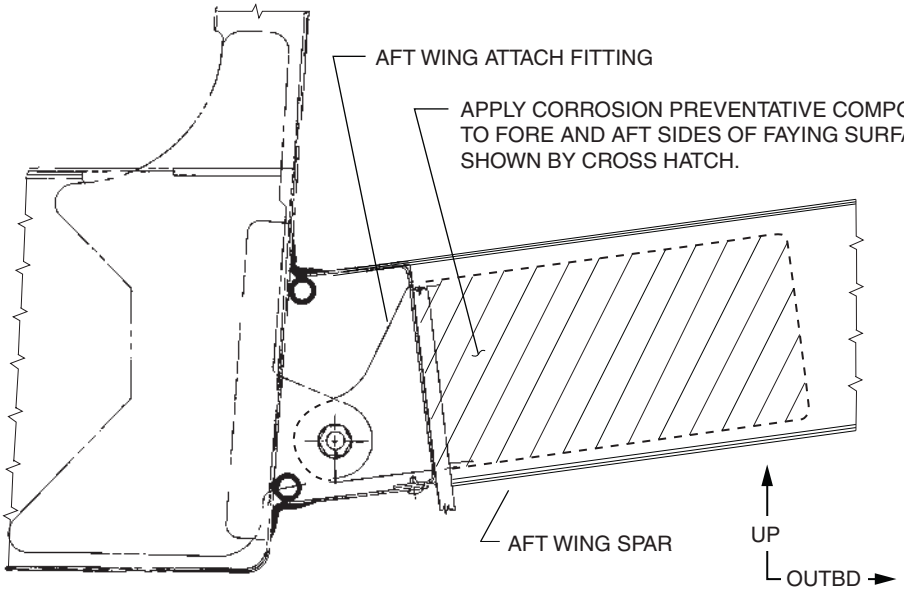
Treat the area identified in “Figure 2” on page 574011, View A-A, with Ardrox (Dinitrol) AV8, or other MIL-PRF-16173 Class I or II, Grade 1 or 4 compliant Corrosion Preventive Compound (procure locally, or order Piper P/N 89500-800). The treatment may be brushed or sprayed.

NOTE: After applying Corrosion Preventive Compound verify that all drain holes and drain passages are clear before proceeding.

E. Reinstall inspection plates and fairings. Perform a functional test of any system or component that may have been disconnected or removed.

F. Make a logbook entry documenting compliance with this inspection.





VIEW A-A
LOOKING AFT AT LEFT HAND SIDE

Wing Aft Attach Fitting
Figure 2

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FLIGHT SURFACES

CAUTION: CONTROL SURFACE SKINS MUST BE REPLACED IF THEY SUSTAIN DAMAGE OR EXHIBIT CRACKS.

1. Aileron

See “Figure 1” on page 57502.

A. Removal

(1) Disconnect the aileron control rod at the aileron attachment point by removing the nut, washers and bolt from the rod end bearing. To simplify installation note location of washers removed.

(2) Remove the attaching nuts, bolts and washers from the hinges at the leading edge of the aileron, and remove the aileron.

B. Installation

WARNING: AILERONS THAT HAVE BEEN REPLACED OR REPAINTED MUST BE BALANCED BEFORE INSTALLATION. SEE BALANCING, BELOW.

(1) Move the aileron into place and install attaching bolts, washers and nuts. Ascertain that the aileron is free to move with no interference.

(2) Attach the aileron control rod with bolts, washers and nut, dividing the washers so that the aileron is free to rotate from stop to stop without the control rod binding or rubbing on the opening in the aft spar. Be certain that the rod end bearing has no side play when tightening the bolt and that the rod does not contact the side of the bracket.

(3) Actuate the aileron controls to insure freedom of movement.

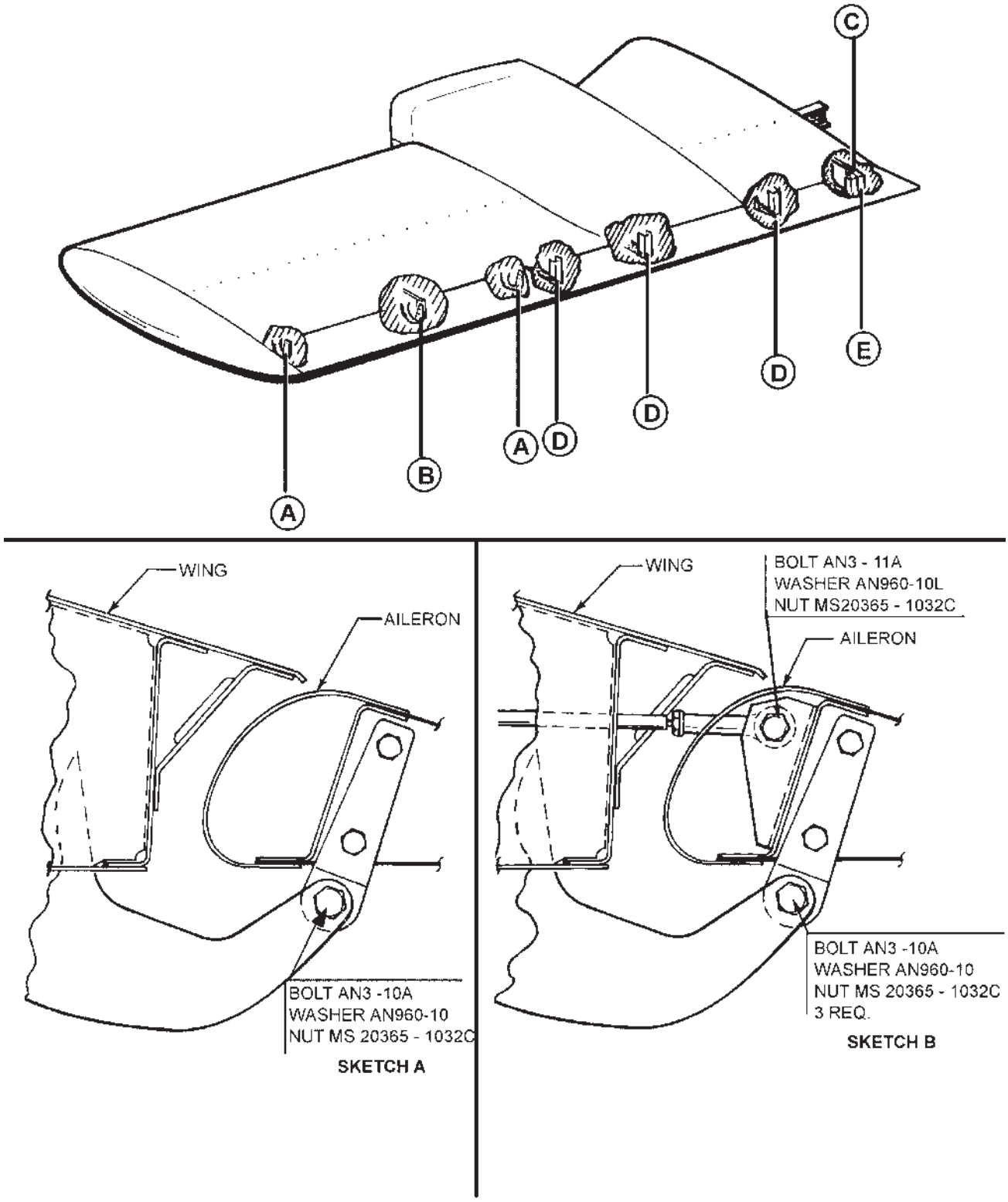
C. Checking Free Play

(1) Set the aileron in its neutral position and secure.

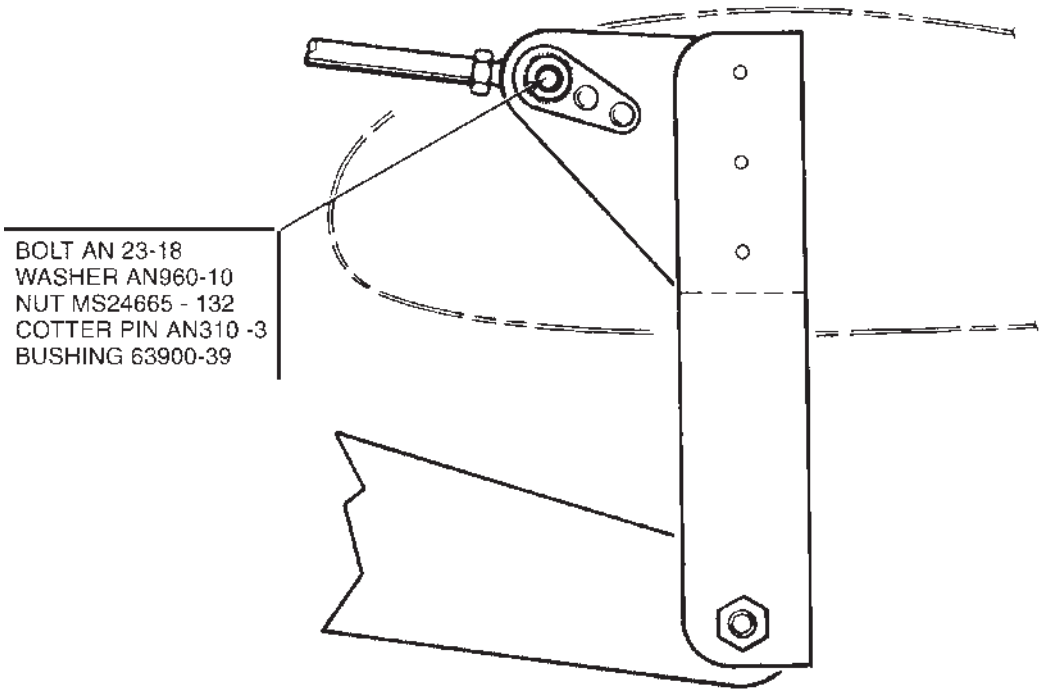
(2) Obtain a straightedge long enough to extend from the ground up to a few inches above the aileron trailing edge. Place the straightedge next to the aileron trailing edge and gently move the aileron up and down, mark the limit of travel (free play) on the straightedge.

(3) The overall travel (free play) must not exceed 0.24 of an inch (6.096 mm). Should free play exceed the limit stated, make necessary repairs as required to eliminate free play.

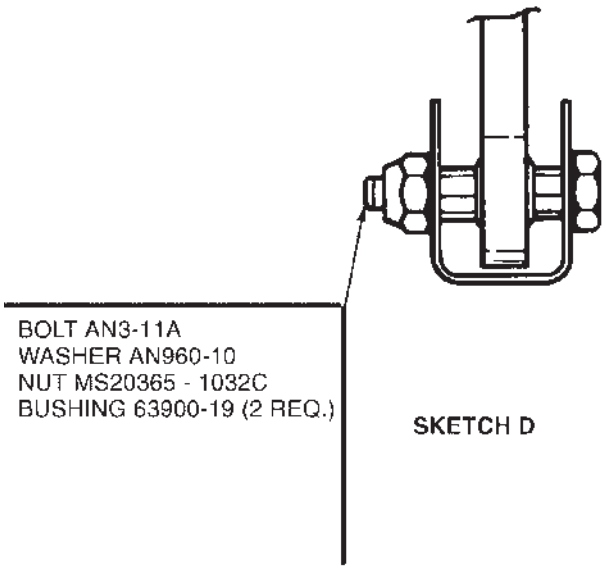
(4) Grasp the aileron and move it spanwise (inboard/outboard) to insure maximum end play of 0.035 of an inch (0.889 mm) is not exceeded.



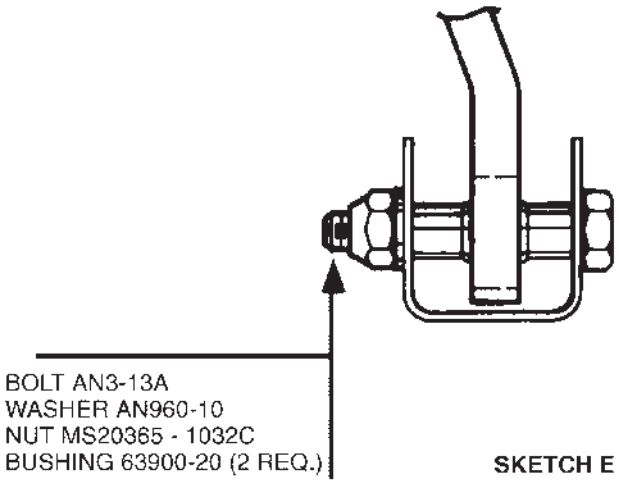
Aileron and Flap Installation
Figure 1 (Sheet 1 of 2)



SKETCH C



SKETCH D



SKETCH E

Aileron and Flap Installation
Figure 1 (Sheet 2 of 2)

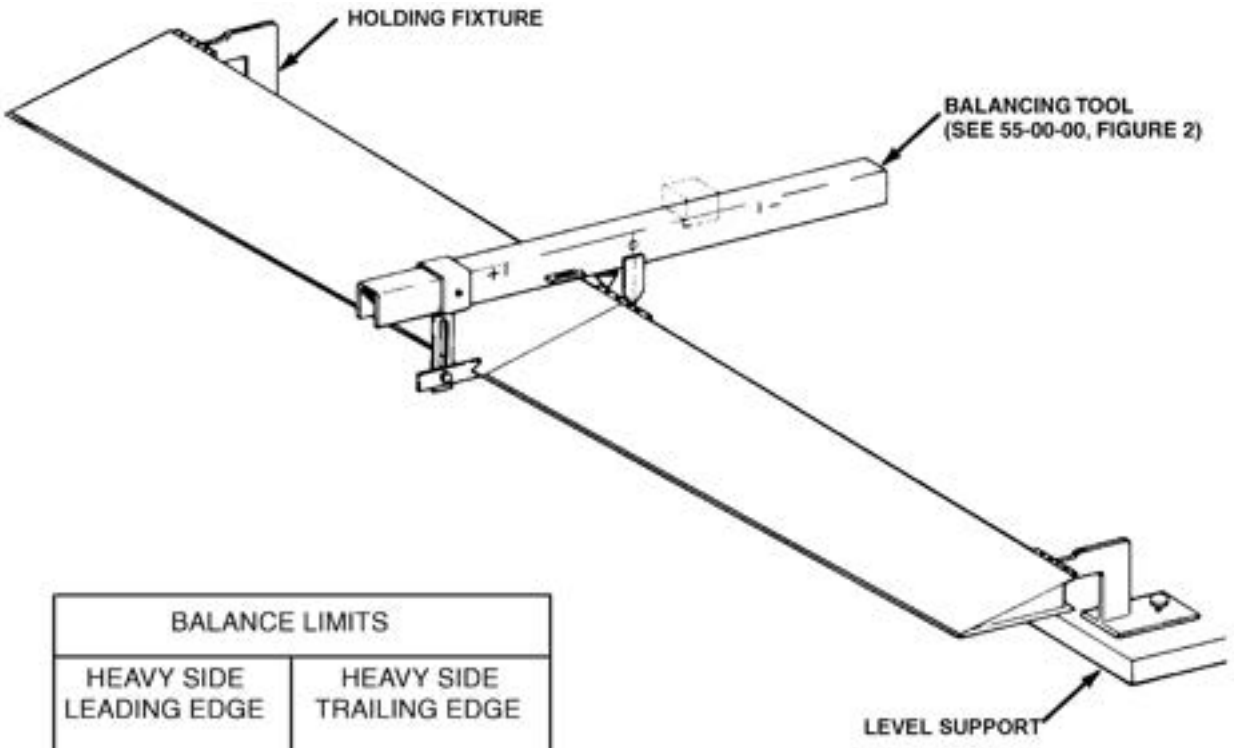
2. Balancing

Balancing Aileron

To balance the aileron, it must be complete and in its final flight configuration as described in 55-00-00, Balancing Equipment. Position the aileron on the balancing fixture in a draft free area and in a manner which allows unrestricted movement of the aileron. Place the tool on the aileron, avoiding rivets, and keep the beam perpendicular to the hinge centerline. Calibrate the tool as described in Balancing Equipment, 55-00-00. Read the scale when the bubble level has been centered by adjustment of the movable weight and determine the static balance. If the static balance is not within the limits specified in “Figure 2” on page 57505, proceed as follows:

(1) Leading edge heavy: This condition is highly improbable; recheck measurements and calculations.

(2) Trailing edge heavy: There are no provisions for adding weight to balance weight to counteract a trailing edge heavy condition. Therefore, it will be necessary to determine the exact cause of the unbalance. If the aileron is too heavy because of painting over old paint, it will be necessary to strip all paint from the aileron and repaint. If the aileron is too heavy resulting from repair to the skin or ribs, it will be necessary to replace all damaged parts and recheck the balance.



BALANCE LIMITS	
HEAVY SIDE LEADING EDGE	HEAVY SIDE TRAILING EDGE
+5 LBS.	-2.75 LBS.

Balancing Aileron
Figure 2

3. Wing Flap

See “Figure 1” on page 57502.

CAUTION: CONTROL SURFACE SKINS MUST BE REPLACED IF THEY SUSTAIN DAMAGE OR EXHIBIT CRACKS.

A. Removal

(1) Extend the flaps to their fullest degree and remove the bolt and bushing from the rod end bearing.
(2) Remove the nuts, washers, bushing and hinge bolts that hold the flap to the wing assembly.
(3) Pull the flap straight back off the wing.

B. Installation

(1) Replace the wing flap by placing the flap onto its proper position and inserting the hinge bolts, bushings, washers and nuts.
(2) With the flap control in the full flap position, place the bushing on the outboard side of the rod end bearing and insert and tighten the bolt.
(3) Operate the flap several times to be certain it is operating freely. (See 27-50-00 for rigging information.)

CHAPTER

61

PROPELLERS

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CHAPTER 61

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CHAPTER 61 - PROPELLERS

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GENERAL

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

Description

On Seneca IV models, the McCauley three bladed propeller is the standard installation, and the Hartzell two bladed propeller is available as an option. On Seneca V models, the Hartzell two bladed propeller is the standard installation, and the McCauley three bladed propeller is available as an option.

Both the Hartzell two bladed or the McCauley three bladed installations are constant speed, controllable pitch, feathering propellers.

Pitch is controlled by oil and nitrogen pressure. Oil pressure sends the propeller toward high rpm or unfeather position; nitrogen pressure sends the propeller toward low rpm or feather position, which also prevents propeller overspeed. Feathering is accomplished by releasing the governor oil pressure, allowing the counterweights, nitrogen pressure and internal feathering spring to feather the blades.

Each engine is equipped with a governor that supplies engine oil at varying pressure through the propeller shaft to maintain constant rpm, which controls engine speed by varying propeller blade angle (pitch) to match load torque to engine torque in response to changing flight conditions.

Feathering is accomplished by moving the desired propeller control lever fully aft through the low rpm detent into the FEATHER position. Unfeathering in flight is accomplished by moving the propeller control forward past the low rpm detent and engaging the starter until the engine begins to windmill.

Unfeathering on the ground can be accomplished by moving the mixture to full rich position, engaging the starter until the engine fires, and then moving the propeller control full forward to the high rpm position. Unfeathering on the ground may also be accomplished by moving the propeller control forward past the low rpm detent and using blade paddles to mechanically pull the propeller out of feather.

An unfeathering accumulator system is available as an option. It uses stored air/oil pressure to aid unfeathering in flight.

DO NOT unfeather a propeller if the engine was stopped due to mechanical failure.

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PROPELLER ASSEMBLY

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

1. Hartzell

A. Spinner

WARNING: ENSURE THE MAGNETO AND MASTER SWITCHES ARE OFF AND THE MIXTURE CONTROL IS IN THE IDLE CUT-OFF POSITION.

(1) Removal

See “Figure 1” on page 61102.

(a) Disassemble the spinner nose cap from the spinner.

(b) Remove locknut B from the valve boss.

(c) Remove the screws securing the spinner to its aft bulkhead.

(d) If it is necessary to remove the spinner bulkhead, remove the bolts securing the bulkhead to the propeller hub. Do not lose the spacers, if any, between the valve boss locknut and front spinner bulkhead.

(2) Installation

(a) Install the spacer(s) (A169-7) over the valve boss. The spacer(s) provide proper alignment of the spinner.

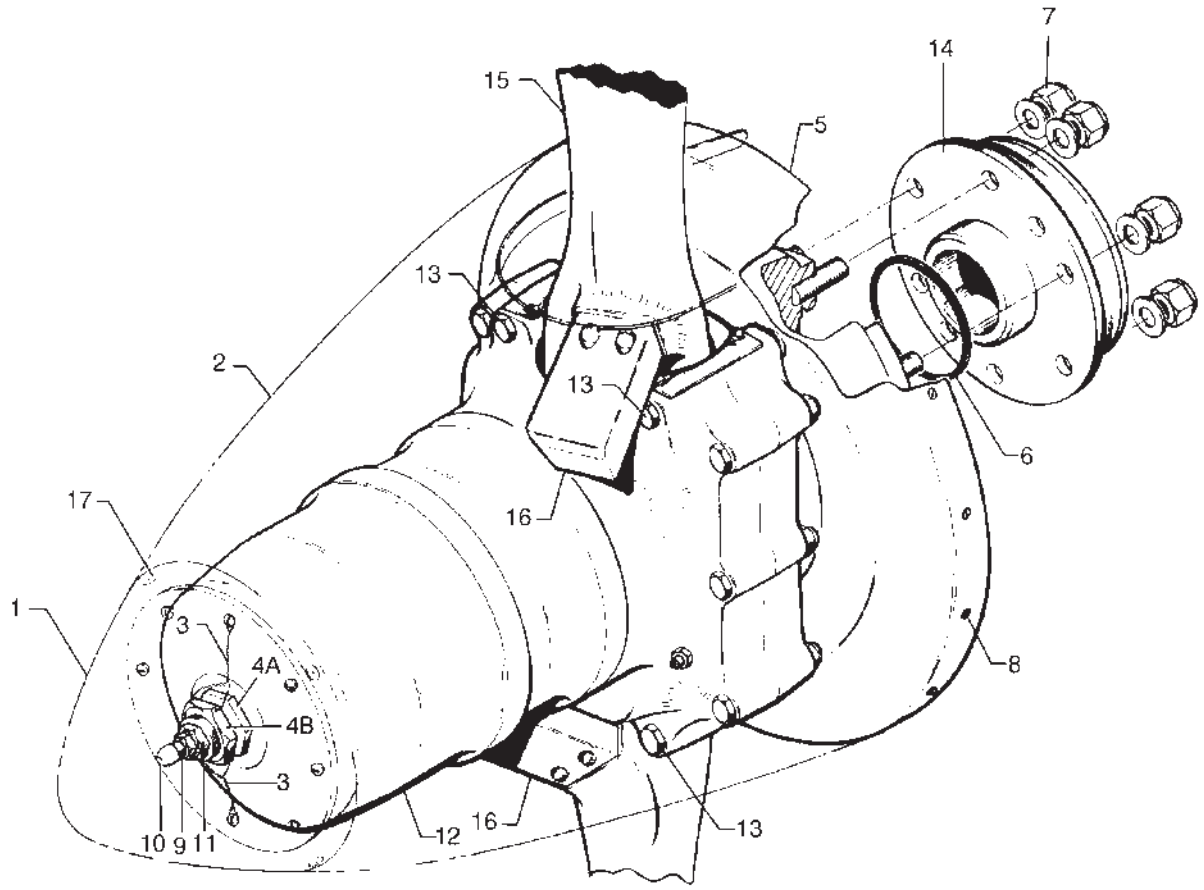
(b) Slide the spinner onto the propeller. Align the holes in the spinner dome with those in the aft bulkhead and push the dome onto the bulkhead until the holes line up. Install attaching screws.

(c) Install locknut B on the valve boss and torque as specified in “Chart 1” on page 61109, or as specified in the appropriate Hartzell Propeller Owner’s Manual.

(d) Safety wire the locknut and screws on the forward spinner bulkhead.

(e) Check air pressure per “Chart 2” on page 611010.

(f) Install spinner nose cap.



CAUTION: THE FOLLOWING PROPELLER ASSEMBLIES MUST BE MOUNTED IN PAIRS AND NOT MIXED.

SENECA IV

LEFT
BHC-C2YF-2CKUF/FC8459-8R

RIGHT
BHC-C2YF-2CLKUF/FJC8459-8R

SENECA V

LEFT
BHC-J2YF-2CUF/FC8459(B)-8R

RIGHT
BHC-J2YF-2CLUF/FJC8459(B)-8R

1. SPINNER CAP
2. SPINNER
3. SAFETY WIRE
4A. LOCKNUT A (LOW PITCH STOP JAM NUT)
4B. LOCKNUT B (SPINNER)
5. AFT BULKHEAD
6. 0-RING
7. PROPELLER MOUNTING NUT
8. SPINNER ATTACHMENT SCREW
9. AIR VALVE
10. AIR VALVE CAP
11. LOW PITCH ADJUSTMENT
12. PROPELLER - R DOME
13. BULKHEAD BOLT
14. ENGINE FLANGE
15. PROPELLER BLADE
16. COUNTERWEIGHT
17. CAP ATTACHMENT SCREW

Hartzell Two Bladed Propeller Installation
Figure 1

B. Propeller

WARNING: ENSURE THE MAGNETO AND MASTER SWITCHES ARE OFF AND THE MIXTURE CONTROL IS IN THE IDLE CUT-OFF POSITION.

(1) Removal

CAUTION: WHEN REMOVING A PROPELLER EQUIPPED WITH ENGINE SYNCHROPHASER MAGNETIC PICKUPS, REMOVE PICKUPS PRIOR TO REMOVING PROPELLER TO PREVENT DAMAGE TO THE PICKUP.S

(a) Remove the spinner per “Removal” on page 61101.
(b) Remove the cowling.
(c) Place a drip pan under the propeller and engine to catch oil spillage.
(d) Remove safety wire and nuts from the propeller mounting studs and withdraw propeller.
(e) Make sure to cover or plug the crankshaft port as well as the propeller hub port.

(2) Installation

Refer to “Figure 1” on page 61102.

(a) Remove any coverings from the propeller and engine crankshaft and clean the mounting flanges. Make sure that dirt, lint, or other foreign material does not enter any of the propeller or crankshaft passages.
(b) Remove the appropriate bolts and install the spinner aft bulkhead. Torque the nuts as specified in “Chart 1” on page 61109.
(c) Lubricate and install O-ring in the propeller hub.
(d) Position the propeller on the shaft mounting flange.
(e) Install the retaining nuts and torque as specified in “Chart 1” on page 61109, or as specified in the appropriate Hartzell Propeller Owner’s Manual.
(f) Install spinner per “Installation” on page 61101.

2. McCauley

A. Spinner

WARNING: ENSURE THE MAGNETO AND MASTER SWITCHES ARE OFF AND THE MIXTURE CONTROL IS IN THE IDLE CUT-OFF POSITION.

(1) Removal

See “Figure 2” on page 61105.

The propeller is held in position by a support on the hub dome inside the spinner and is not attached to the spinner.

(a) Remove the cowling.

(b) Remove the spinner by unscrewing the screws attaching the spinner to its bulkhead at the rear of the propeller and slide off the assemblies.

(c) Inspect bulkhead for cracks, if cracks found remove and replace bulkhead using the Improved bulkhead kit (P/N 688-023).

NOTE: S/N’S 3449001–3449505 have six (6) mounting screws securing the bulkhead Assy. The improved bulkhead kit (P/N 688-023) adds a row of twelve mounting (12) screws to the six (6) screws in the earlier bulkhead assembly. If replacing the bulkhead assembly with the kit torque screws per “Chart 1” on page 61109 Propeller Specifications. This procedure incorporates the requirements of Piper Service Letter 1197.

S/N’s 3449508 and up are manufactured with the improved bulkhead assembly, See “Chart 1” on page 61109 Propeller Specifications For Torque Values.

(d) The support can be removed from the dome by pulling it off the assembly.

Do not lose the spacers inside the support assembly.

(2) Installation

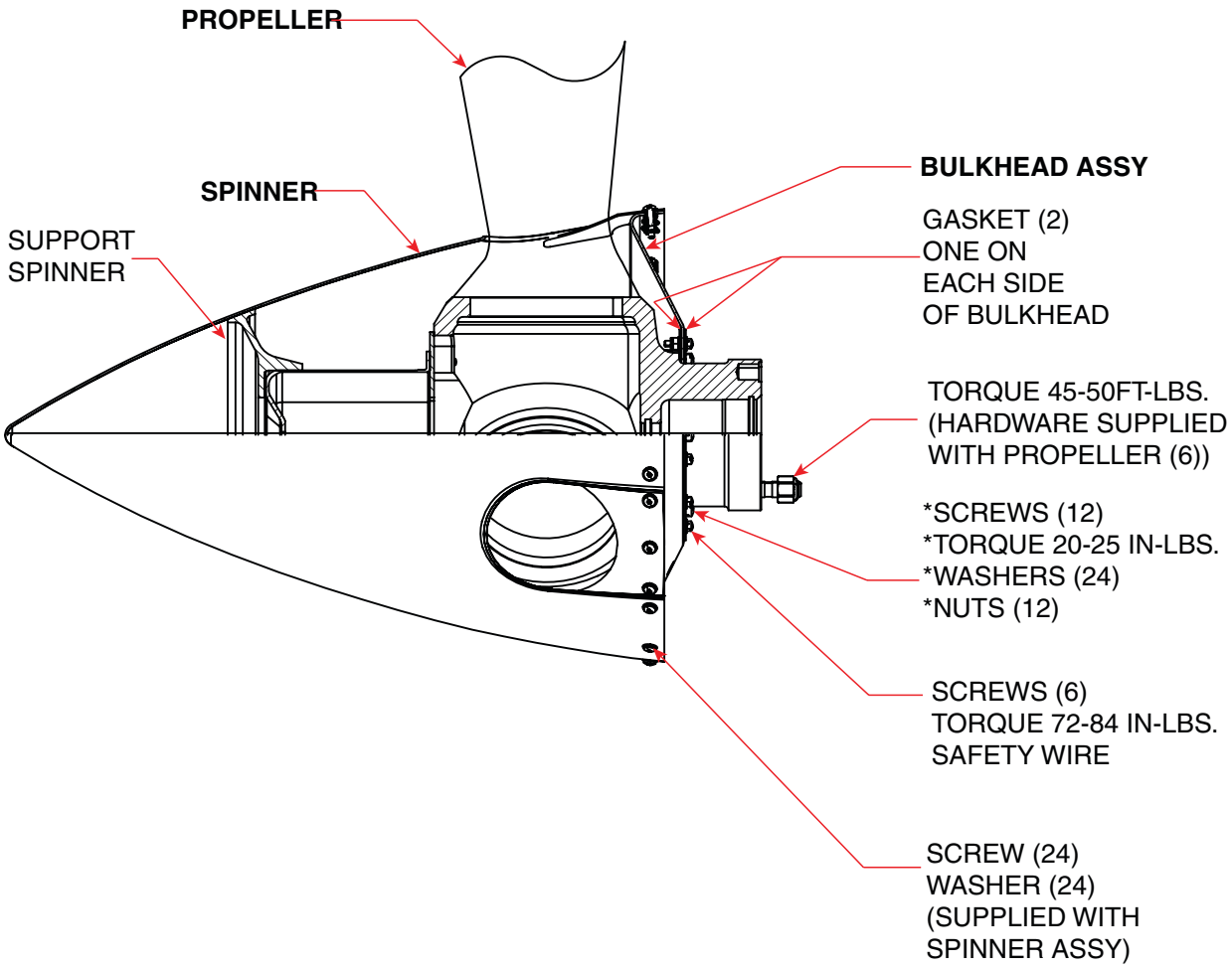
See “Figure 2” on page 61105.

With the same amount of spacers in the spinner dome support as when removed.

(a) install the spinner support on the propeller hub dome.

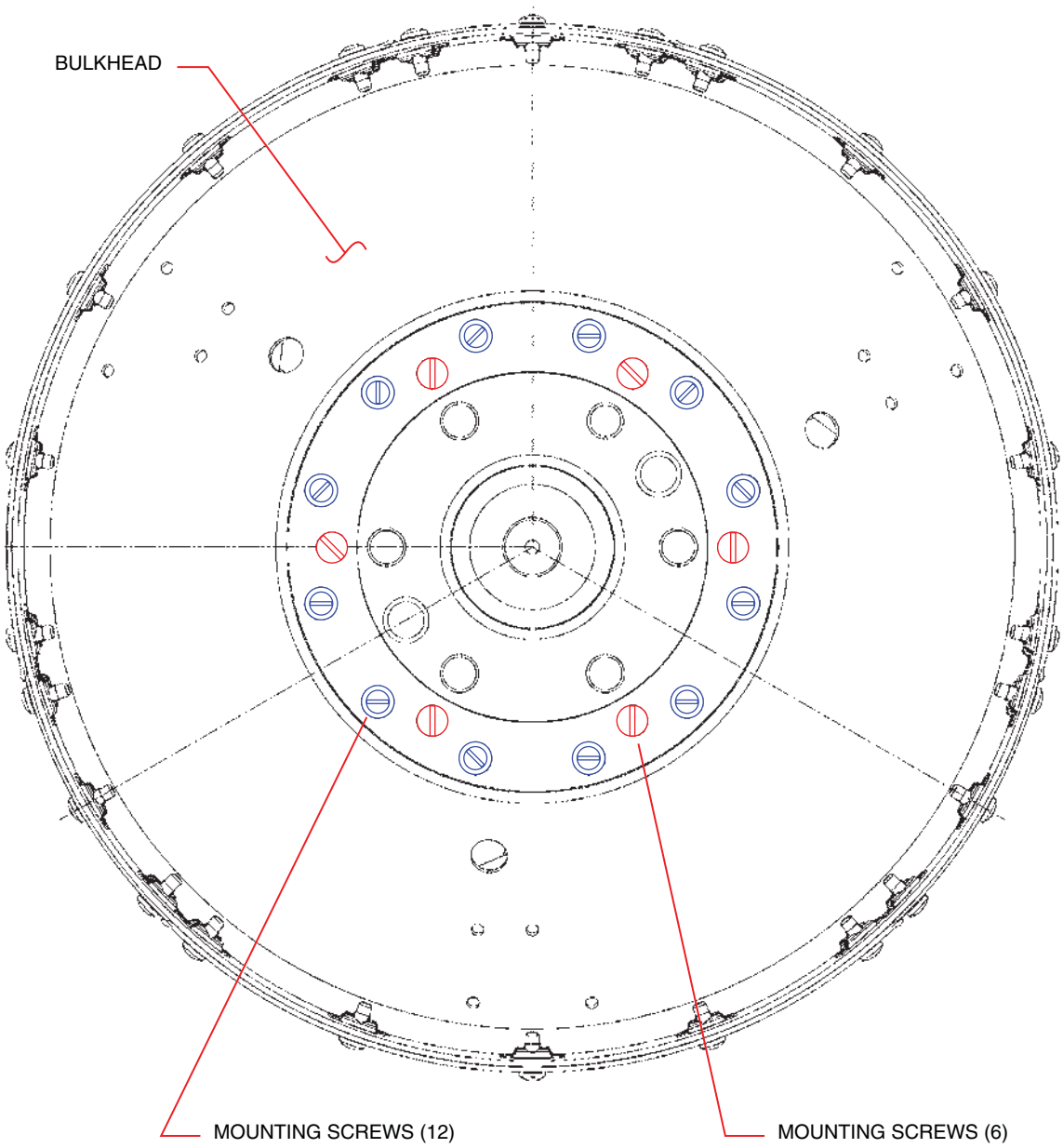
(b) Align the spinner and slide it over the propeller. The spinner holes must be misaligned forward of those in the bulkhead by half a hole, so that the spinner need be pressed rearward to install the hardware. If the holes align perfectly, or misalign to the rear of the holes in the bulkhead, remove spinner and support and add more spacers inside the spinner support.

100710 REV AK 5.2



* NOTE: S/N'S 3449001 THRU 3449505 HAVE SIX (6) MOUNTING SCREWS SECURING THE BULKHEAD ASSY. THE IMPROVED BULKHEAD KIT (P/N 688-023) ADDS A ROW OF TWELVE (12) MOUNTING SCREWS TO THE SIX (6) MOUNTING SCREWS IN THE EARLIER BULKHEAD ASSEMBLY. IF REPLACING THE BULKHEAD ASSEMBLY WITH THE KIT TORQUE SCREWS PER CHART 2 PROPELLER SPECIFICATIONS.

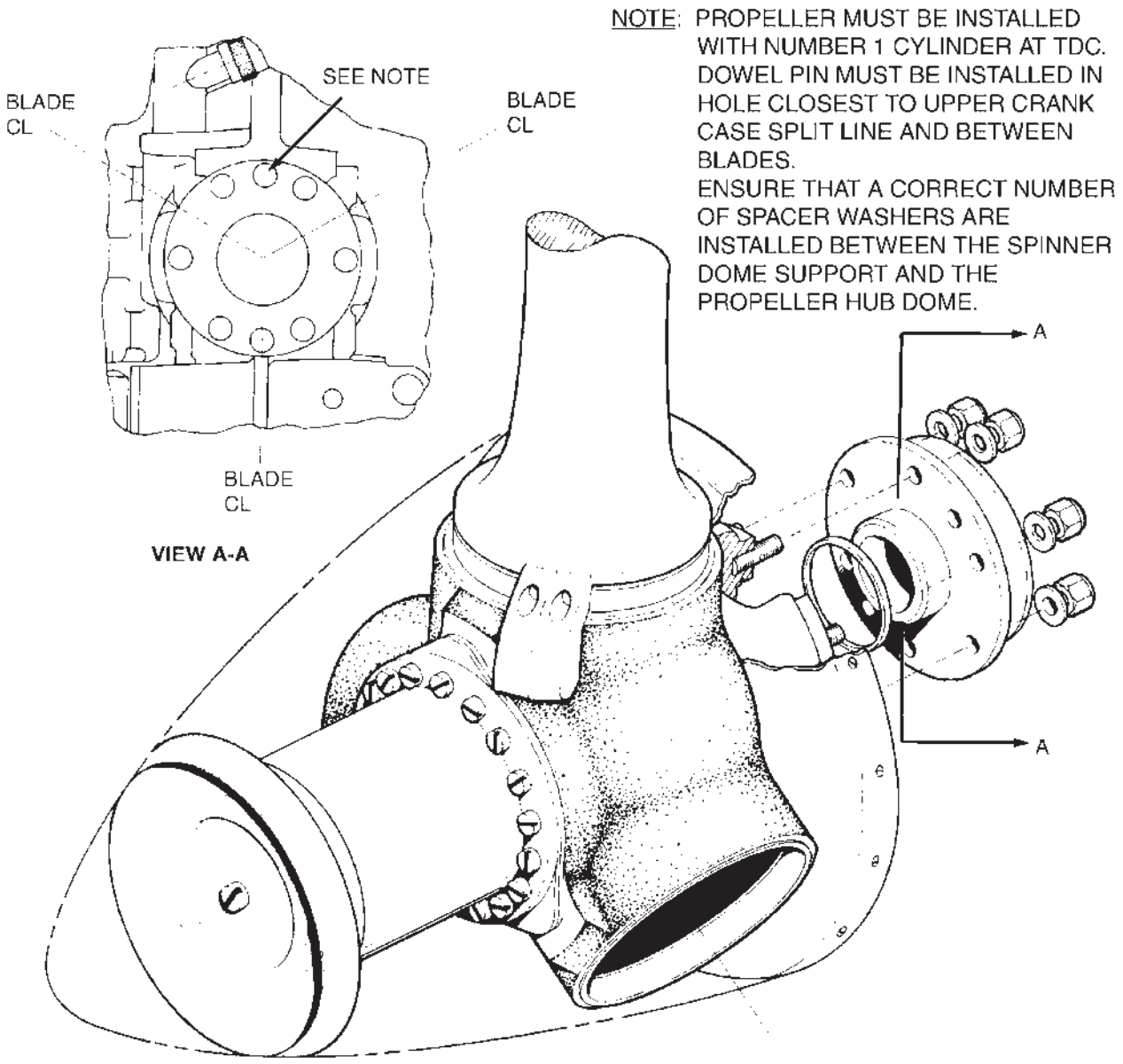
McCauley Three Bladed Propeller Installation
Figure 2 (Sheet 1 of 3)



REPLACEMENT BULKHEAD ASSEMBLY
(Piper Kit P/N 688-023, McCauley Kit P/N E-7258-5)

LOOKING FORWARD

McCauley Three Bladed Propeller Installation
Figure 2 (Sheet 2 of 3)



SENECA IV
LEFT
3AF32C508/82NFA-6

RIGHT
3AF32C509/L82NFA-6

SENECA V
LEFT
3AF32C522/82NJA-6

RIGHT
3AF32C523/L82NJA-6

McCauley Three Bladed Propeller Installation
Figure 2 (Sheet 3 of 3)

B. Propeller

WARNING: ENSURE THE MAGNETO AND MASTER SWITCHES ARE OFF AND THE MIXTURE CONTROL IS IN THE IDLE CUT-OFF POSITION.

(1) Removal

CAUTION: WHEN REMOVING A PROPELLER EQUIPPED WITH ENGINE SYNCHROPHASER MAGNETIC PICKUPS, REMOVE PICKUPS PRIOR TO REMOVING PROPELLER TO PREVENT DAMAGE TO THE PICKUPS.

(a) Remove the spinner per “Removal” on page 61104.

(b) Remove the cowling.

(c) Place a drip pan under the propeller and engine to catch oil spillage.

If desired, the spinner on the McCauley installation can now be removed by removing the screws at the spinner bulkhead and withdrawing spinner. See 2A(1) Spinner Removal.

(d) Remove safety wire and nuts from the propeller mounting studs and withdraw propeller.

(e) Make sure to cover or plug the crankshaft port as well as the propeller hub port.

(2) Installation

Refer to “Figure 2” on page 61105.

(a) Remove any coverings from the propeller and engine crankshaft and clean the mounting flanges. Make sure that dirt, lint, or other foreign material does not enter any of the propeller or crankshaft passages.

(b) Lubricate and install O-ring in the propeller hub.

(c) On Seneca IV models, position the spinner back plate over the studs on the propeller hub. On Seneca V models, attach spinner bulkhead to back of aft end of propeller hub. Tighten six (6) mounting screws to a torque as specified in “Chart 1” on page 61109. Safety wire screws in pairs.

NOTE: S/N’S 3449001–3449505 have six (6) mounting screws securing the bulkhead Assy. The improved bulkhead kit (P/N 688-023) adds a row of twelve mounting (12) screws to the six (6) screws in the earlier bulkhead assembly. If replacing the bulkhead assembly with the kit torque screws per “Chart 1” on page 61109 Propeller Specifications. This procedure incorporates the requirements of Piper Service Letter 1197.

S/N’s 3449508 and up are manufactured with the improved bulkhead assembly, see “Chart 2” on page 611010 Propeller Specifications for torque values.

(d) Rotate the engine until the number one cylinder is at its top dead center position.

(e) Mount the propeller on the crankshaft mounting flange such that the hub dowel pin, located between two of the blades, is inserted in the flange hole closest to the split line of the upper crankcase. Install mounting nuts.

(f) Torque the mounting hardware as specified in “Chart 2” on page 611010.

(g) Install spinner per “Installation” on page 61104.

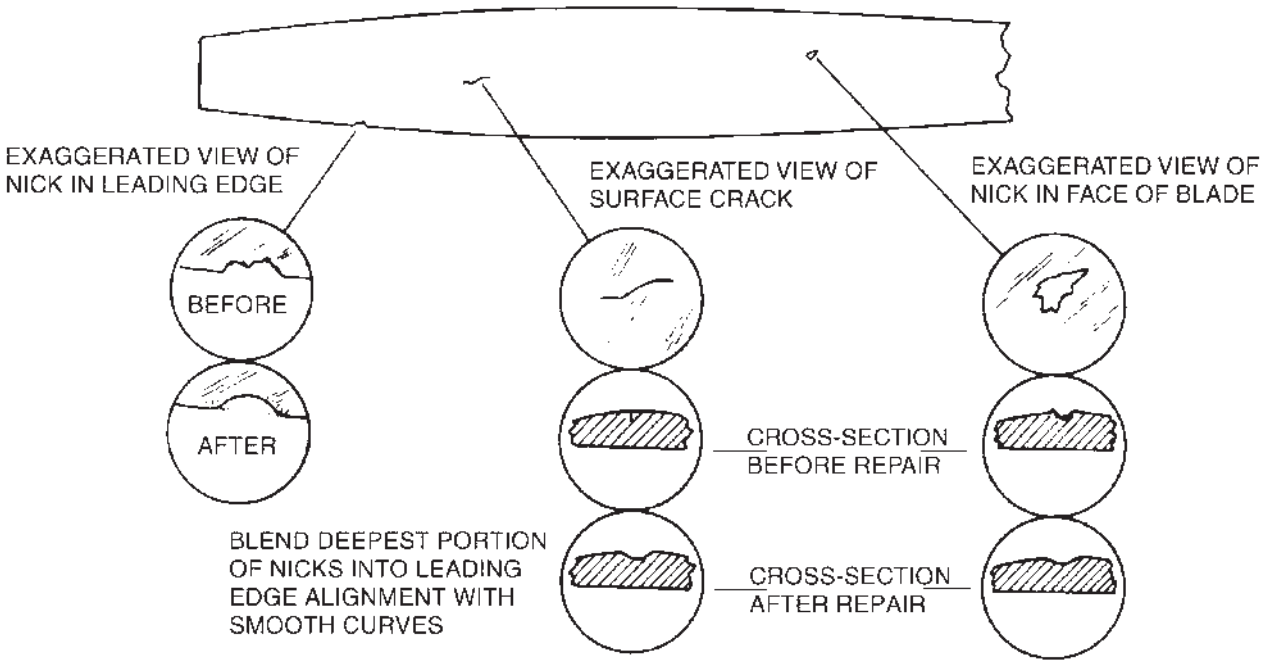
CHART 1
PROPELLER SPECIFICATIONS

	Hartzell (2-Blade)	McCauley (3-Blade)
Blade Angle		
Low Pitch (High RPM) (1)		
(Seneca IV)	12.6° ± 0.2°	11.0° ± 0.2°
(Seneca V)	14.6° ± 0.2°	12.6° ± 0.2°
High Pitch (Low RPM) (1)		
(Seneca IV)	80° to 81.5°	81° to 83.5°
(Seneca V)	80° to 81.5°	81.6° to 82.6°
Propeller RPM Setting		
Engine Static High RPM		
(Seneca IV)	2800 rpm max.	2800 rpm max.
(Seneca V)	2600 rpm max.	2600 rpm max.
Propeller Torque Limits (Dry (2))		
Spinner Bulkhead (Aft)		
(Seneca IV)	20–22 foot-pounds	N/A
(Seneca V)	20–22 foot-pounds	72–84 in·lb (6) (A-1635-78 Screws) 20–25 in·lb (12) (A-1635-121 Screws)
Propeller Mounting		
(Seneca IV)	70–80 foot-pounds	Per Placard on Hub
(Seneca V)	70–80 foot-pounds	45–50 foot-pounds (Lubricated with McCauley P/N A-1637-16 (MIL-T-83483) only.)
Locknut A (Low Pitch Stop Jam Nut)	15–20 foot-pounds	N/A
Locknut B (Spinner)	28–30 foot-pounds	N/A
Spinner Attachment Screws	35–40 inch-pounds	35–40 inch-pounds

(1) Measured at 30 inch station.
(2) Unless otherwise noted.

CHART 2
HARTZELL PROPELLER CHAMBER PRESSURE VS. TEMPERATURE

	Temp °F (C)	Pressure (psi)
Seneca IV	**Propeller Hubs BHC-C2YF-2CKUF and BHC-C2YF-2CLKUF**	
	70 to 100 (21.11 to 37.78)	22 ± 2
	40 to 70 (4.44 to 21.11)	17 ± 2
	0 to 40 (-17.8 to 4.44)	14 ± 2
	-30 to 0 (-34 to to -17.8)	9 ± 2
Seneca V	**Propeller Hubs BHC-J2YF-2CUF and BHC-J2YF-2CLUF**	
	70 to 100 (21.11 to 37.78)	41 ± 1
	40 to 70 (4.44 to 21.11)	38 ± 1
	0 to 40 (-17.8 to 4.44)	36 ± 1
	-30 to 0 (-34 to to -17.8)	33 ± 1
NOTE: Do not check pressure or charge with propeller in feather position.		



Typical Nicks and Removal Methods
Figure 3

3. Propeller Cleaning, Inspection, and Repair

CAUTION: DO NOT ATTEMPT TO DISASSEMBLE THE PROPELLER ANY FURTHER THAN STATED IN THIS MANUAL. PROPELLER SHOULD BE REFERRED TO THE HARTZELL OR MCCAULEY FACTORY, OR A CERTIFIED REPAIR STATION, FOR INTERNAL REPAIRS AND REPLACEMENT OF PARTS.

A. Check for oil and grease leaks.

B. Clean the spinner, propeller hub, and blades with a non-corrosive solvent.

C. Inspect the hub parts for cracks.

D. Steel hub parts should not be permitted to rust. Use aluminum paint to touch up, if necessary, or replate them during overhaul.

E. Check all visible parts for wear and safety.

F. Check blades to determine whether they turn freely on the hub pilot tube. This can be done by rocking the counterweights or blades back and forth through the slight freedom allowed by the pitch change mechanism. If they appear tight and are properly lubricated, the propeller should be disassembled by an authorized Service Center.

G. Inspect the blades for damage or cracks. Nicks in the leading edges of blades should be filed out and all edges rounded, as cracks sometimes start from such places. Use fine emery cloth for finishing. (Refer to “Figure 3” on page 611010 for propeller blade care.)

H. Check the condition of the propeller mounting nuts and studs.

I. Each blade face should be sanded lightly with fine sandpaper and painted, when necessary, with a flat black paint to retard glare. A light application of oil or wax may be applied to the surfaces to prevent corrosion.

J. Grease the blade hub through the zerk fittings. Remove one of the two fittings for each propeller blade; alternate the next time. Apply grease through the zerk fitting until fresh grease appears at the fitting hole of the removed fitting. Care should be taken to avoid blowing out the hub gaskets.

K. On Hartzell propellers, check for air leaks by applying a soap solution around the air valve and stop adjustment nut. Internal leakage will show up as air flows through the piston rod.

L. Hartzell propellers usually require periodic lubrication (grease), McCauley propellers are not lubed. Verify propeller lubrication requirements are correct and correctly caveated for “Hartzell only”, if McCauley propellers are an option.

4. Checking Propeller Blade Track

Blade track is the ability of one blade tip to follow the other, while rotating, in almost the same plane. Excessive difference in blade track - more than 0.0625 inch (1.5867 mm) – may be an indication of bent blades or improper propeller installation. Check blade track as follows:

A. With the engine shut down and blades vertical, secure to the aircraft a smooth board just under the tip of the lower blade. Move the tip fore and aft through its full “blade-shake” travel, making small marks with a pencil at each position. Then center the tip between these marks and scribe a line on the board for the full width of the tip.

B. Carefully rotate propeller by hand to bring the opposite blade down. Center the tip and scribe a pencil line as before and check that lines are not separated more than 0.125 inch (3.175 mm).

C. Propellers having excess blade track should be removed and inspected for bent blades, or for parts of sheared O-ring, or foreign particles, which have lodged between hub and crankshaft mounting faces. Bent blades will require repair and overhaul of assembly.

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CONTROLLING

1. Propeller Governor

A. Removal

The propeller governor is mounted on the lower left forward portion of the engine crankcase. Remove the governor as follows:

(1) Remove the left side of the nose cowl to gain access to the governor.

(2) Disconnect the governor control cable end from the governor control arm.

(3) Remove the governor mounting nuts and withdraw the governor from the mounting pad. Cover the mounting pad to prevent foreign material from entering the engine.

B. Installation

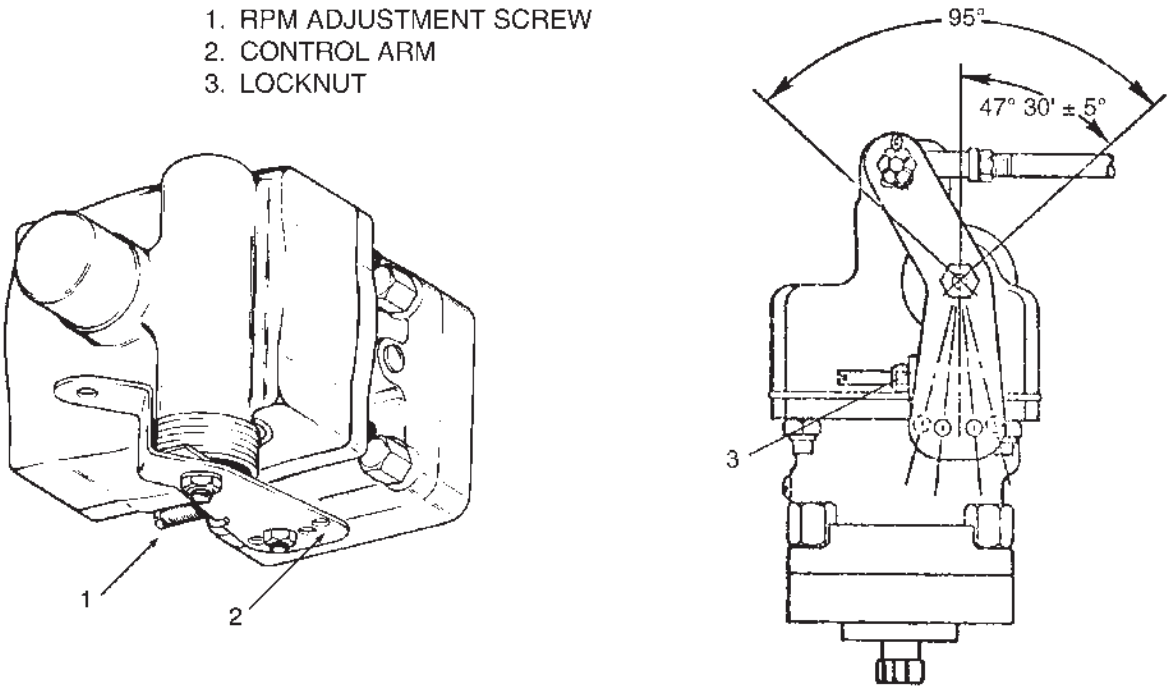
(1) Clean the mounting pad and the governor drive shaft thoroughly.

(2) Coat the mounting gasket with Dow Corning release agent or equivalent.

(3) Lubricate governor drive shaft with engine oil and install governor on the mounting pad.

(4) Tighten the mounting nuts evenly and tighten to a final torque of 140 to 160 inch-pounds.

(5) Connect the control cable to the control arm. Check to be sure the attachment bolt does not contact the governor body while moving the control arm through its full travel. Clearance should be 0.03 inch (0.762 mm) minimum.



Propeller Governor
Figure 1

C. Rigging and Adjustment

(PIR-FTP2001-12, Rev. F.

Refer to “Figure 1” on page 61201.

(1) Start engine; park 90° to wind direction and warm in normal manner.

(2) (Seneca IV models) To check high rpm, low pitch setting, move the propeller control all the way forward. At this position the governor speed control arm should be against the high rpm fine adjusting screw. With the throttle full forward, observe engine rpm, which should stabilize between 2700 and 2800 rpm. A takeoff must be conducted during which the engine maintains 2800 rpm.

(Seneca V models) To check high rpm, low pitch setting, move the propeller control all the way forward. At this position the governor speed control arm should be against the high rpm fine adjusting screw. With the throttle full forward, observe engine rpm, which should stabilize at 2575-2600 rpm. A takeoff must be conducted during which the engine maintains 2575-2600 rpm.

(3) (Seneca IV models) If the engine rpm does not read 2800 rpm in flight, the high rpm setting must be adjusted as follows.

(Seneca V models) If the engine rpm does not read 2575-2600 rpm in flight, the high rpm setting must be adjusted as follows:

(a) Land, shut down the engine and open the cowl door(s).

(b) (Seneca IV models) Adjust the governor by means of the fine adjustment screw to 2800 rpm. To do this, loosen the high rpm fine adjustment screw locknut and turn the screw in a clockwise direction to decrease engine or speed or in a counterclockwise direction to increase engine speed.

(c) (Seneca V models) Adjust the governor by means of the fine adjustment screw to 2575-2600 rpm. To do this, loosen the high rpm fine adjustment screw locknut and turn the screw in a clockwise direction to decrease engine or speed or in a counterclockwise direction to increase engine speed.

NOTE: One revolution of the fine adjustment screw will increase or decrease the engine speed approximately 20 rpm.

(d) Secure the cowl door(s) and repeat step (2) to ascertain proper rpm setting.

(e) After setting the proper high rpm adjustment, run the self-locking nut on the fine adjustment screws against the base projection to lock.

(4) With the high rpm adjustment complete, the control system should be adjusted so that the governor control arm will contact the high rpm stop when the cockpit control knob is .032 to .047 of an inch from its full forward stop. To adjust the control knob travel, disconnect the control cable end from the control arm; loosen the cable end jam nut and rotate the end to obtain the desired level clearance. Reconnect the cable end and tighten jam nut.

(5) It is usually only necessary to adjust the high rpm (low pitch) setting of the governor control system, as the action automatically takes care of the positive low rpm (high pitch) setting.

2. Propeller Unfeathering Accumulator System

See”Figure 2” on page 61204.

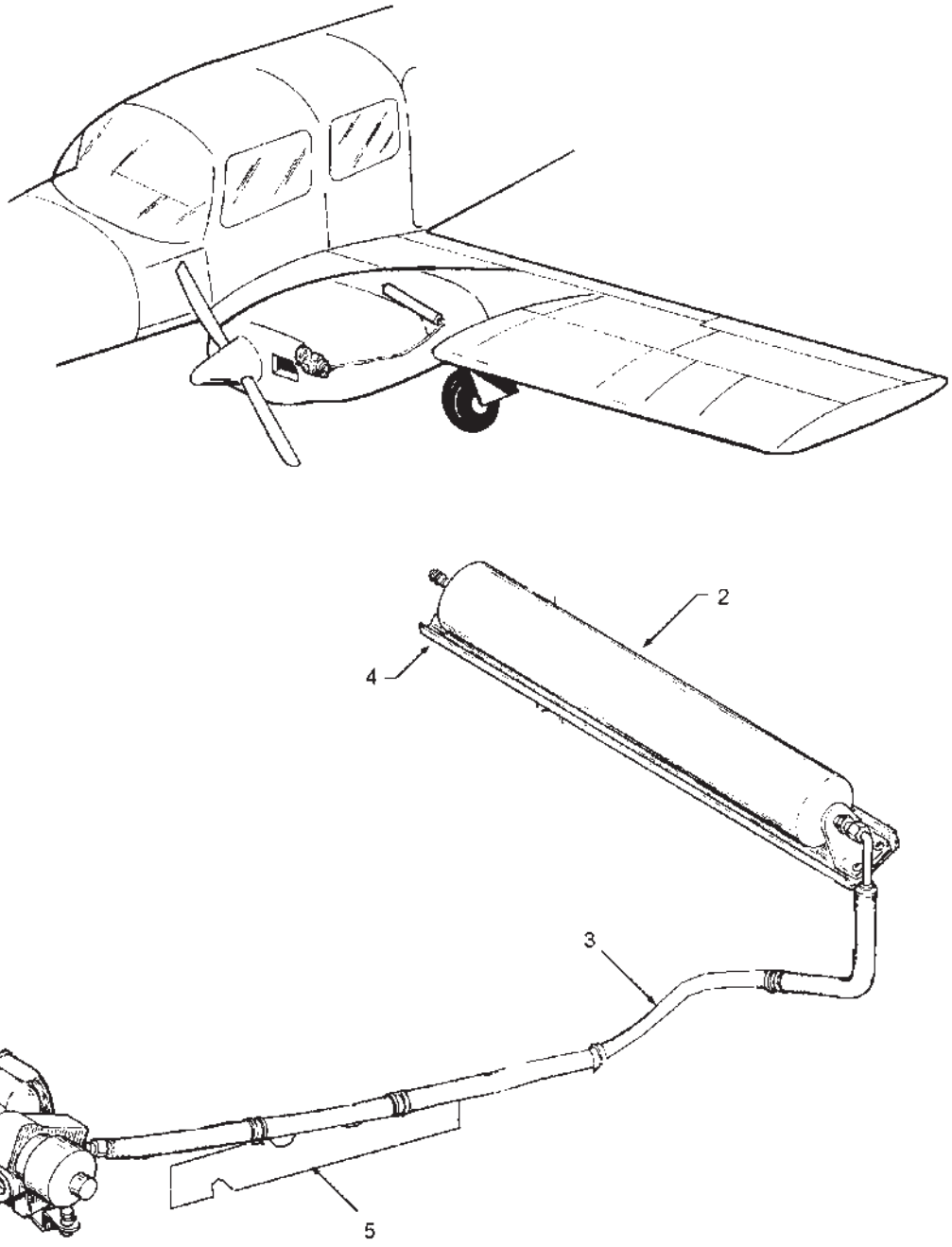
The propeller unfeathering system provides a means for storing air and oil pressure in an accumulator so that the propeller may be moved out of the feathered position when so desired. Refer to the Pilot’s Operating Handbook for proper operating procedure.

Accumulator

WARNING: COMPLETELY DISCHARGE ALL NITROGEN PRESSURE BEFORE DISCONNECTING THE OIL LINE, PRIOR TO REMOVAL OF THE UNIT FROM THE AIRPLANE.

This is a free piston type accumulator which is charged with nitrogen to a working pressure of 90 to 100 psig at normal room temperature. Accumulator overhaul should coincide with governor overhaul. Refer to McCauley Service Manual No. 780401 for detailed instructions.

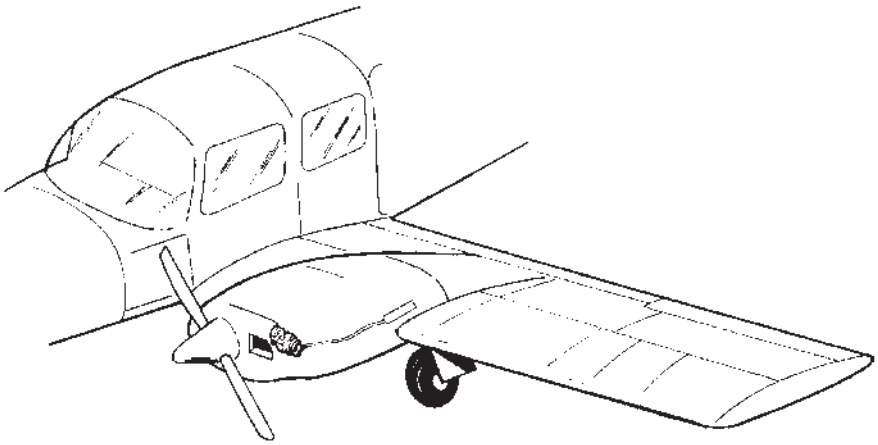
General servicing of the accumulator during its use between overhauls consists of periodically checking the nitrogen charge and visually inspecting the unit for any oil leaks. To perform an operational check, refer to the Pilot’s Operating Handbook for proper operating procedure.

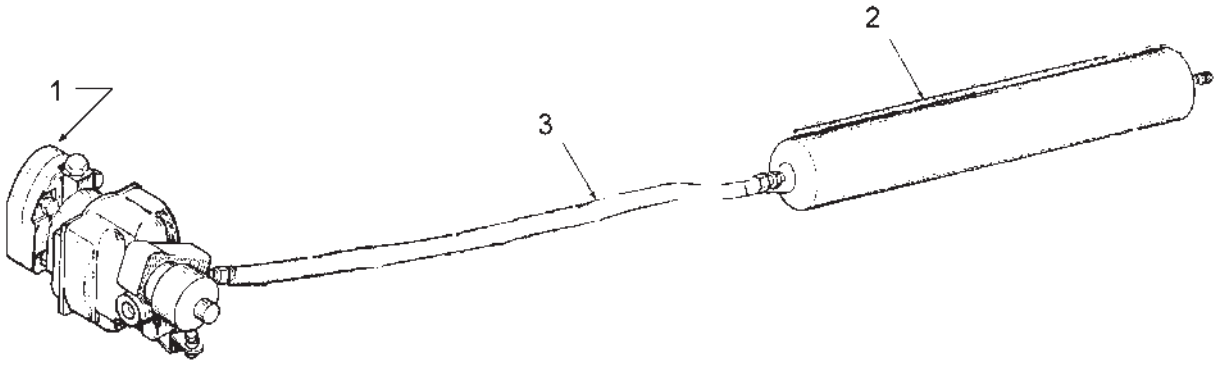


1. GOVERNOR ASSY.
2. ACCUMULATOR
3. HOSE ASSY.
4. BRACKET
5. HEAT SHIELD

Effectivity
Seneca IV

Propeller Unfeathering Accumulator Installation
Figure 2 (Sheet 1 of 2)





1. GOVERNOR ASSY.
2. ACCUMULATOR
3. HOSE ASSY.

Propeller Unfeathering Accumulator Installation
Figure 2 (Sheet 2 of 2)

Effectivity
Seneca V

3. Engine Synchrophaser

See “Figure 3” on page 61207.

The synchrophaser installation is a Hartzell system which utilizes a computer, and an electrically slaved and mechanically operated propeller governor.

The function of the synchrophaser is to maintain both propellers at the same rpm and at a selected phase angle. This eliminates propeller “beat” effect and minimizes vibration. The left engine is utilized as the master engine. The right engine is equipped with a slave governor, which automatically maintains, or synchronizes, its rpm with the left engine rpm. The synchrophaser is turned ON with a three position switch located on the throttle quadrant below the propeller controls. It is labeled OFF for manual control and 1 or 2 for propeller synchrophaser. A blue press-to-test light, which illuminates when the propellers are out of synchronization, is located below the switch.

NOTE: Be certain Magneto Switches are OFF.

A. System Operation

During taxi, takeoff, landing or single engine operations, the propeller synchrophaser switch should be in the OFF position. The blue press-to-test light below the switch will illuminate while the propellers are out of synchronization, whether or not the switch is in the OFF, 1, or 2 position. When the switch is in the OFF position, the propellers can be synchronized manually. The light will go out when propeller synchronization is complete. To utilize automatic synchronization, the propellers should first be synchronized manually to within approximately 10 rpm of each other; then the switch placed in position 1. The blue light will go out when synchronization is complete. For a given rpm and power setting, switch position 2 may provide smoother operation by means of providing a different phase angle. Set the switch to position 1 or 2, whichever provides the smoothest operation. Normally, propeller synchrophasing will take place within a few seconds, but occasionally it may take up to a full minute. Position the synchrophaser switch OFF for 30 seconds before making power setting adjustments. The synchrophaser switch may then be returned to position 1 or 2, whichever provides the smoothest operation. Should propeller rpm differential exceed 50 rpm, the switch should be selected OFF for 30 to 40 seconds; then the propellers can be synchronized again and the synchrophaser switch returned to position 1 or 2. Pulling the circuit breakers completely deactivates the propeller synchrophaser system. If the master switch is turned OFF, or if there is an electrical system failure, the slaved engine will return to the controlled selected rpm plus approximately 25 rpm “out of synchronization” regardless of the position of the synchrophaser switch.

B. Test Procedure

(PIR-PPS50041-1, Rev. New.

The following procedure ensures all circuits and the propeller governor solenoid coil are functioning properly. Use the Hartzell B-4657 Test Set to perform the following tests.

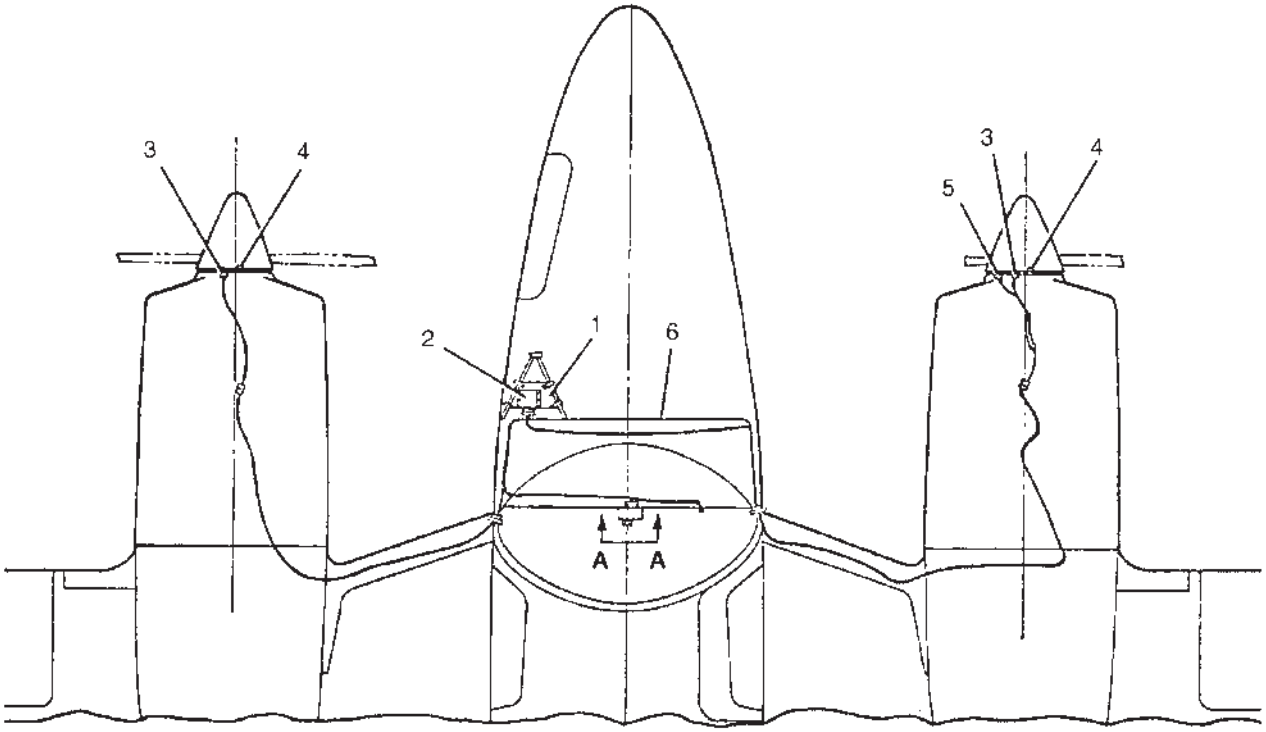
NOTE: If the B-4657 indicator light does not come ON in any position, a check for proper battery voltage and polarity at system plug must be accomplished (positive at pin 14, negative at pin 1, 2 or 3). If voltage is correct, check indicator bulb. If bulb is burned out, replace with Sylvania #330 or equivalent. If the bulb is good, check the internal fuse. If fuse is damaged, replace with Buss AGC1/2 or equivalent. The 1/2 amp fast-blow fuse may not be replaced with anything heavier. After replacing the fuse, recheck the wiring harness for improper connections and/or shorts. Do this prior to reconnecting the test box.

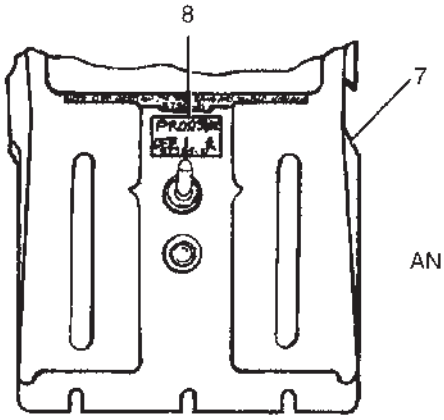
(1) Master switch OFF; “BATT” and “PROP SYNC” circuit breakers pulled.

(2) Remove forward baggage compartment trim panels to gain access to the computer.

(3) Visually check installation for inadequate connections, incorrect connections, shorts, etc.

(4) With the aircraft master switch OFF and the test box function switch in Position 1 (OFF), connect the test box into the synchrophaser system in place of the computer.





SECTION A - A

SWITCH INSTALLATION
AND SYSTEM IDENTIFICATION

"AUTO-SYNC"
(FIXED PHASE)
INSTALLATION

1. MOUNTING BRACKET
2. COMPUTER
3. PICKUP ASSEMBLY
4. MAGNET ASSEMBLY
5. SLAVE GOVERNOR ASSEMBLY
6. WIRING HARNESS
7. CONTROL PEDESTAL
8. SYNCHROPHASER CONTROL SWITCH

Propeller Synchrophaser Installation
Figure 3

(5) Close (push-in) the “BATT” and “PROP SYNC” circuit breakers, turn the Battery Master switch ON, and turn the B-4657 ten-position switch to Position 2 (Power) and observe the B-4657 indicator light. If the B-4657 indicator light is “ON”, this indicates the battery voltage and polarity to the check out box is correct.

(6) Turn the ten-position switch to Position 3 (Right Engine) and using the Hartzell supplied magnet (Hartzell P/N A-647), touch it to the magnetic pickup on the right engine. The B-4657 indicator light should “Flash” as the magnet touches the pickup.

(7) Turn the ten-position switch to Position 4 (Left Engine) and using the Hartzell supplied magnet, touch it to the magnetic pickup on the left engine. The B-4657 indicator light should “Flash” as the magnet touches the pickup.

(8) Turn the ten-position switch to Position 5 (manual). Turn the synchrophaser phase / manual switch to “MANUAL” (Switch towards keyway). The B-4657 indicator light should be “ON” with the ten-position in Position 5, and “OFF” in Positions 6 and 7.

(9) Turn the synchrophaser phase / manual switch to “PHASE 1” (Switch Centered). The B-4657 indicator light should be “OFF” with the ten-position switch in Position 5 and 7, and “ON” in Position 6.

(10) Turn the synchrophaser phase / manual switch to “PHASE 2” (switch opposite keyway). The B-4657 indicator light should be “ON” with the ten-position switch in Positions 6 and 7 and “OFF” in Position 5.

(11) Turn the ten-position switch to Position 8 (Phase Light Check). The B-4657 indicator light should be “ON”. Un-screw the lamp from the synchrophaser system Cockpit Indicator light. The B-4657 indicator light should go “OFF”.

(12) Turn the ten-position switch to Position 9 (Coil). The B-4657 indicator light should be “ON”. This indicates the leads to the coil are correct.

(13) Turn the ten-position switch to Position 10 (Coil Short) and depress the “Coil Short Test Switch”. The B-4657 indicator light should remain “OFF”. If “ON”, the coil is either electrically shorted or out of tolerance.

(14) This completes the system test. Open (Pull) the “PROP SYNC” and “BATT” circuit breakers, remove the B-4657 Wiring Test Box, and re-install the Hartzell Prop Synchrophaser Computer.

C. Prop Sync Switch

Removal and Installation

(1) Remove the knobs from the control levers and remove the upper control quadrant cover.

(2) Remove the retaining nut from the toggle side of the switch and remove the switch from the cover.

(3) Slide the shaded wire protective covering back off the soldered connections.

(4) Make note of where each specific wire is soldered and remove the wires from their terminals.

(5) Install the wires on the new switch as noted and reinstall the switch in the opposite manner of removal.

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D. Magnetic Pickup

Refer to “Figure 4”, below, and “Figure 5” on page 612011.

The magnetic pickup consists of a permanent magnet with a coil placed near the counterweight assembly. Low output may be the result of a defective or incorrectly adjusted magnetic pickup and will result in improper synchrophaser system operation.

CAUTION: MAGNETIC PICKUPS SHOULD NOT BE HAMMERED OR JARRED AS THIS MAY DECREASE THE STRENGTH OF THE MAGNET.

(1) Removal

(a) Remove the top cowl of the desired engine.

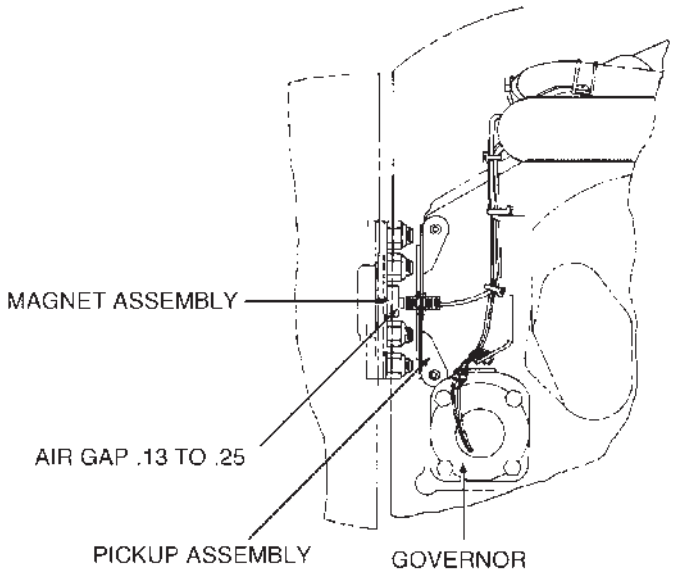
(b) Remove the nut attaching the pickup to the bracket.

(c) The unit can now be removed by cutting the tie wraps and disconnecting the wires.

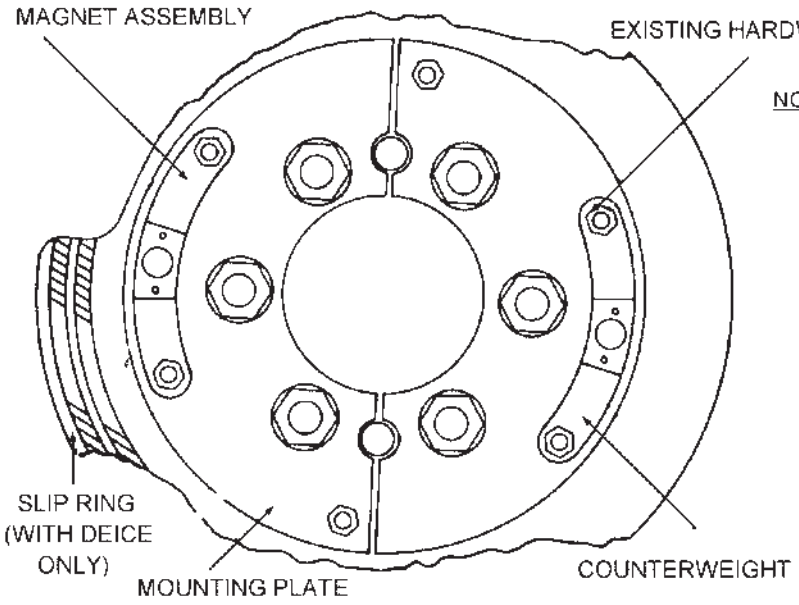
(2) Installation

(a) The clearance between the dowel pin and pickup should be 0.13–0.25 inch with the crankshaft pushed aft. Using a feeler gauge, adjust the nuts securing the pickup to the bracket to give the proper gap.

(b) Rotate propeller, making sure the crankshaft is pushed aft and ensure the proper clearance is maintained.

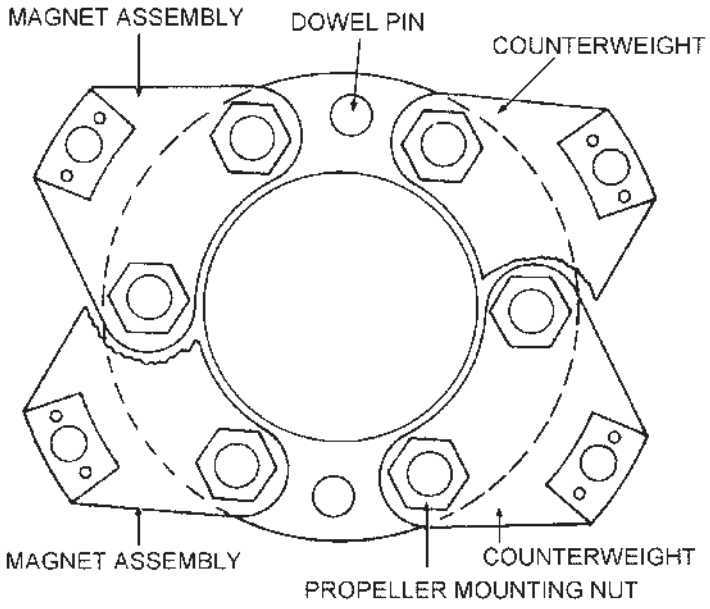


Magnetic Pickup and Governor Installation
Figure 4



SENECA IV WITH DEICE AND SENECA V WITH AND WITHOUT DEICE

NOTE: ORIENT THE RIGHT ENGINE TO PLACE DOWEL PIN AT TOP ALIGNED WITH THE CRANKCASE SPLIT LINE. REMOVE NO. 5 SPARK PLUG (RIGHT FRONT) AND ROTATE TO TDC ON COMPRESSION STROKE FOR APPROXIMATE LOCATION. ROTATE MINIMUM NO. DEGREES TO PLACE DOWEL AT CRANKCASE SPLIT LINE. ACCOMPLISH THE SAME ON THE LEFT ENGINE USING NO. 6 CYLINDER (LEFT FRONT). REMOVE THE SLIP RING MOUNTING NUTS WHICH WILL BE USED TO MOUNT THE MAGNET AND COUNTERWEIGHT ASSEMBLIES. INSTALL AS SHOWN, IN THE SAME LOCATION ON RIGHT AND LEFT ENGINES. MAGNET ASSEMBLIES WILL BE ON THE LEFT SIDE OF EACH ENGINE WITH COUNTERWEIGHTS OPPOSITE.



SENECA IV WITHOUT DEICE

NOTE: ORIENT THE RIGHT ENGINE TO PLACE DOWEL PIN AT TOP ALIGNED WITH THE CRANKCASE SPLIT LINE. REMOVE NO. 5 CYLINDER SPARK PLUG AND ROTATE AS NECESSARY TO FIND TDC ON COMPRESSION STROKE. ROTATE THE MINIMUM NO. DEGREES TO PLACE DOWEL AT CRANKCASE SPLIT LINE. ACCOMPLISH THE SAME ON THE LEFT ENGINE USING NO. 6 CYLINDER (LEFT FRONT). REMOVE PROPELLER MOUNTING NUTS AND WASHERS FROM THE 4 LUGS TO BE USED FOR MOUNTING. INSTALL MAGNET ASSEMBLY, COUNTER BALANCE AND NUTS AS SHOWN. DO NOT USE WASHERS UNDER NUTS SECURING MAGNET AND COUNTER BALANCE ASSEMBLIES. (NOTE: DIFFERENT LUGS ARE USED ON LEFT AND RIGHT ENGINES.)

Magnet Assemblies
Figure 5

4. Control Lever Feather Stop Plate

"Figure 6" on page 612013.

NOTE: If installed, remove and discard feather stop plate P/N 78713-000 and replace with P/N 78713-003. You may retain mounting hardware, if it is in good condition, for reuse with replacement plate instalaltion.

A. Removal

(1) Set the propeller control levers in the full forward position, clear of the feather stop plate assembly.

(2) Remove the screws and washers that fasten the plate assembly to an angle support.

B. Installation

Install the feather stop plate assembly (P/N 78713-003) as follows:

NOTE: By using this procedure, the final position of the feather stop plate assembly will allow stress to be balanced across all of the plate fingers. This will assure optimum service life of the plate and the smooth operation of the control levers.

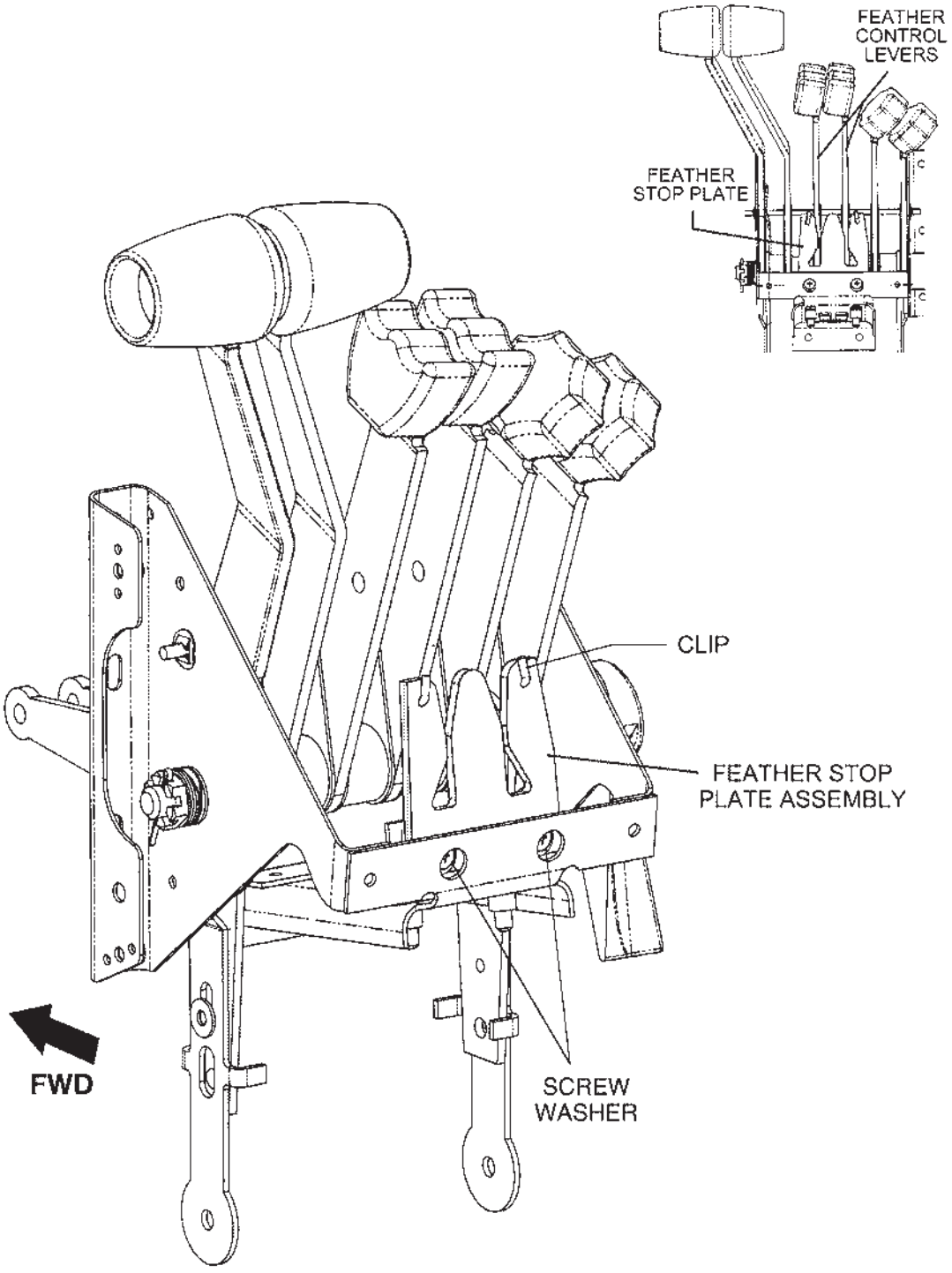
(1) Tighten the plate screws until snug, then loosen one full turn in order to allow the plates to float.

(2) Fully engage the propeller levers into the slots of the plate assembly.

(3) Retighten the plate screws to 4-5 in.-lbs. of torque.

(4) Cycle the propeller control levers until operation is smooth.

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Control Lever Feather Stop
Figure 6

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CHAPTER

70

STANDARD PRACTICES – ENGINE

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CHAPTER 70

LIST OF EFFECTIVE PAGES

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70-List of Effective Pages	1	Sep 30/18			
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70-Table of Contents	1	Sep 30/18			
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	2	Sep 30/18			

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CHAPTER 70 - STANDARD PRACTICES - ENGINE

TABLE OF CONTENTS

SUBJECT	SECTION	PAGE
GENERAL	**70-00-00**	1

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GENERAL

WARNING: ENSURE MAGNETOS AND MASTER SWITCH ARE OFF AND MIXTURE IS IN IDLE CUT-OFF PRIOR TO WORKING AROUND ENGINE.

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

Review the following suggestions before working on the power plant.

A. To insure proper reinstallation and/or assembly, tag and mark all parts, clips, and brackets as to their location prior to their removal and/or disassembly.

NOTE: Use tags or temporary marking methods which are durable enough to ensure identification during ordinary handling, storage and final assembly of parts.

B. During removal of various tubes or engine parts, inspect them for indications of scoring, burning or other undesirable conditions. To facilitate reinstallation, observe the location of each part during removal. Tag any unserviceable part and/or units for investigation and possible repair.

CAUTION: DUST CAPS USED TO PROTECT OPEN LINES MUST ALWAYS BE INSTALLED OVER THE TUBE ENDS AND NOT IN THE TUBE ENDS. FLOW THROUGH THE LINES MAY BE BLOCKED OFF IF LINES ARE INADVERTENTLY INSTALLED WITH DUST CAPS IN THE TUBE ENDS.

C. Extreme care must be taken to prevent foreign matter from entering the engine, such as lockwire, washers, nuts, dirt, dust, etc. This precaution applies whenever work is done on the engine, either on or off the aircraft. Suitable protective caps, plugs, and covers must be used to protect all openings as they are exposed.

D. Should any items be dropped into the engine, the assembly process must stop and the item removed, even though this may require considerable time and labor.

E. Insure that all parts are thoroughly clean before assembling.

F. Never reuse any lockwire, lockwashers, tablocks, tabwashers or cotter pins. All lockwire and cotter pins must fit snugly in holes drilled in studs and bolts for locking purposes. Cotter pins should be installed so the head fits into the castellation of the nut, and unless otherwise specified, bend one end of the pin back over the stud or bolt and the other end down flat against the nut. Use only corrosion resistant steel lockwire and/or cotter pins. Bushing plugs shall be lockwired to the assembly base or case. Do not lockwire the plug to the bushing.

G. All gaskets, packings and rubber parts must be replaced with new items of the same type at reassembly. Insure the new nonmetallic parts being installed show no sign of having deteriorated in storage.

H. When installing engine parts which require the use of a hammer to facilitate assembly or installation, use only a plastic or rawhide hammer.

CAUTION: ENSURE THAT ANTI-SEIZE COMPOUNDS ARE APPLIED IN THIN EVEN COATS, AND THAT EXCESS COMPOUND IS COMPLETELY REMOVED TO AVOID CONTAMINATION OF ADJACENT PARTS.

I. Anti-seize lubrication should be applied to all loose-fit spline drives which are external to the engine and have no other means of lubrication. For certain assembly procedures, molybdenum disulfide in either paste or powdered form mixed with engine oil or grease may be used.

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CHAPTER

71

POWER PLANT

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CHAPTER 71

LIST OF EFFECTIVE PAGES

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CHAPTER 71 - POWER PLANT

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Seneca IV		1
Seneca V		1
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Seneca IV		1
Seneca V		1
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Removal		1
Cleaning		1
Installation		1
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Seneca IV		3
Seneca V		3

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GENERAL

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

The purpose of this chapter is to provide instructions for the removal, minor repair, service and installation of the engine and components. For instructions on major repairs and overhauls, consult the appropriate publication of the component manufacturer.

1. Description

Refer to “Chart 1” on page 71002.

A. Seneca IV Models

Seneca IV models are powered by two Continental engines; a TSIO-360-KB (left) and LTSIO-360-KB (right). Each engine is turbocharged with a Rayjay turbocharger and controlled through a ground adjustable exhaust bypass valve. An overboost relief valve is also used in each of the intake systems to protect the engine from an overboost condition. The engines are rated at 200 hp at 2600rpm and 40 in. Hg manifold pressure with a five minute takeoff power rating of 220 hp at 2800 rpm and 40 in. Hg manifold pressure.

Each engine cowl is made up of two nose sections and side panels, an upper support section and a lower cowl assembly. The nose and side panels are interconnected to the upper support, lower cowl and nacelle by camlock fasteners. Each lower cowl assembly also houses a cowl flap assembly which is operated manually through a cable from a lever in the cockpit.

B. Seneca V Models

Seneca V models are powered by two Continental engines: a TSIO-360-RB (left) and LTSIO-360-RB (right). Each engine is turbocharged with a Garrett turbocharger and controlled with a variable absolute controller and automatic exhaust bypass valve (wastegate). An overboost relief valve is also used in each of the intake systems to protect the engine from an overboost condition. The engines are rated at 220 hp at 2600 rpm and 38 in. Hg manifold pressure and are approved for maximum continuous operation at this setting.

Each engine cowl is made up of two nose sections, an upper cowl assembly and a lower cowl assembly. The nose section, lower cowl and upper cowl are connected by camlock fasteners. Each lower cowl assembly also houses a cowl flap assembly which is operated manually through a cable from a lever in the cockpit.

2. Troubleshooting

Troubles peculiar to the powerplant are listed in “Chart 2” on page 71003, along with the probable causes and suggested remedies. When troubleshooting engine, propeller or fuel system, always ground the magneto primary circuit before performing any checks.

NOTE: Seneca V models incorporate the RSA-5 Fuel Injection System. See 73-10-00 to troubleshoot this system.

CHART 1
ENGINE DATA

Manufacturer	Continental	
Model	**TSIO & LTSIO-360-KB**	**TSIO & LTSIO-360-RB**
Number of Cylinders	6 Horizontally Opposed	6 Horizontally Opposed
Bore (Inches)	4.44	4.44
Stroke (Inches)	3.88	3.88
Displacement (Cubic Inches)	360	360
Compression Ratio	7.5:1	7.5:1
Type of Propeller Drive, Flanged	Direct	Direct
Fuel, Minimum Octane	100 or 100 LL	100 or 100 LL
Oil Sump Capacity	8 quarts	8 quarts
Oil Pressure (PSI):		
Minimum	10 (Idle)	10 (Idle)
Normal	30-80	30-80
Maximum	100 (Cold - Ground)	100 (Cold - Ground)
Oil Temperature (°F):		
Minimum	100	100
Normal	100-200	100-200
Maximum	240	240
Probe Location	Above Oil Cooler	Above Oil Cooler
Cylinder Head Temperature (°F) No. 2 cyl.:		
Minimum	240°F	240°F
Normal	240-420°F	240-420°F
Maximum	460°F	460°F
Magnetos	Bendix 25 Series	Slick 6324 (left) 6320 (right)
Left Bank	Fires 20° BTC Lower Right, Upper Left	Fires 22° BTC Lower Right, Upper Left
Right Bank	Fires 20° BTC Lower Left, Upper Right	Fires 22° BTC Lower Left, Upper Right
Firing Order:		
LTSIO-360-KB	1-4-5-2-3-6	1-4-5-2-3-6
TSIO-360-KB	1-6-3-2-5-4	1-6-3-2-5-4
Spark Plugs (Shielded):	AC, SR86, S86R, HSR86 Auto Lt. PH24, PH260 Champ. REM38W, RHM38W, RHM38E Red Seal. SE270, SE270P, SS270, SU270P	AC, SR86, S86R, HSR86 Auto Lt. PH24, PH260 Champ. REM38W, RHM38W, RHM38E Red Seal. SE270, SE270P, SS270, SU270P
Spark Plug Torque	360 to 420 inch-pounds	360 to 420 inch-pounds
Alternator	12-volt, 60 ampere	24-volt, 85 ampere
Starter	12-volt, Prestolite	24-volt, Continental
Engine Dry Weight With Accessories	392 pounds	420 pounds

CHART 2 (Sheet 1 of 6)
TROUBLESHOOTING ENGINE

Trouble	Cause	Remedy
Engine will not start.	No fuel gauge pressure - fuel to engine.	Check fuel control for proper position, auxiliary pump “ON” and operating, feed valves open. Fuel filters open and tank fuel level.
	Have gauge pressure - engine flooded.	Turn off auxiliary pump and ignition switch; set throttle to “FULL OPEN” and fuel control to “IDLE CUT-OFF”, and crank engine to clear cylinder of excess fuel. Repeat starting procedure.
	Have gauge pressure - no fuel to engine.	Check for vent or loose fuel lines. Loosen line at fuel nozzle. If no fuel shows, replace fuel manifold valve.
Engine starts but fails to keep running.	Inadequate fuel to fuel manifold valve.	Set fuel control in “FULL RICH” position; turn auxiliary pump “ON,” check to be sure feed lines and filters are not restricted. Clean or replace defective components.
	Defective ignition system.	Check accessible ignition cables and connections. Tighten loose connections. Replace defective spark plugs.
Engine runs rough at idle.	Improper idle mixture adjustment.	(Seneca IV) Readjust idle setting. Turn adjustment screw clockwise to lean mixture and counter-clockwise to richen mixture.
	Fouled spark plugs. idle.	Remove and clean plugs, adjust gaps. Replace defective plugs.
	Discharge nozzle (injector) air vent manifold restricted or defective.	Check for bent or loose connections. Tighten loose connections. Check for restrictions and replace defective components.

CHART 2 (Sheet 2 of 6)
TROUBLESHOOTING ENGINE

Trouble	Cause	Remedy
Engine has poor acceleration.	Idle mixture too lean.	Readjust idle mixture.
	Incorrect fuel-air mixture, work control linkage or restricted air cleaner.	Tighten loose connections. Service air cleaner.
	Defective ignition system.	Check accessible cables and connections. Replace defective spark plugs.
	Malfunctioning turbocharger.	(Seneca IV) Check operation; listen for unusual noise. Check exhaust bypass screw and for exhaust system defects. Tighten loose connections.
Engine runs rough at speeds above idle.	Improper fuel-air mixture.	(Seneca IV) Check manifold connections for leaks. Tighten loose connections. Check fuel control for setting and adjustment. Check fuel filters and screens for dirt. Check for proper pump pressure and readjust as necessary.
	Restricted fuel nozzle.	Remove and clean all nozzles.
	Ignition system and spark plugs defective.	Clean and re-gap spark plugs. Check ignition cables for defects. Replace defective components.
Engine lacks power, reduction in maximum manifold pressure or critical altitude.	Incorrectly adjusted throttle control, “sticky” linkage or dirty air cleaner.	Check movement of linkage by moving control from idle to full throttle. Make proper adjustments and replace worn components. Service air cleaner.
	(Seneca IV) Improperly adjusted waste gate valve.	Check exhaust bypass screw adjustment.

CHART 2 (Sheet 3 of 6)
TROUBLESHOOTING ENGINE

Trouble	Cause	Remedy
Engine lacks power, reduction in maximum manifold pressure or critical altitude. (cont.)	Defective ignition system.	Inspect spark plugs for fouled electrodes, heavy carbon deposits, erosion or electrodes, improperly adjusted electrode gaps and cracked porcelains.
		Test plugs for regular firing under pressure. Replace damaged or misfiring plugs. Spark plug gap to be 0.015 to 0.019 inches (0.381 to . 0.482 mm).
	Loose or damaged exhaust system.	Inspect entire exhaust system to turbocharger for cracks and leaking connections. Tighten connections and replace damaged parts.
	Loose or damaged intake manifolding.	Inspect entire manifold system for possible leakage at connections. Replace damaged components; tighten all connections and clamps.
	Fuel nozzles defective.	Inspect fuel nozzle vent manifold for leaking connection. Tighten and repair as required. Check for restricted nozzles and lines and clean or replace as necessary.
	Malfunctioning turbocharger.	Check for unusual noise in turbocharger. If malfunction is suspected, remove exhaust and/or air inlet connections and check rotor assembly for possible rubbing in housing, damaged rotor or defective bearings. Replace turbocharger if damage is noted.
	Exhaust system gas leakage.	Inspect exhaust system for gas leakage, gaskets at turbine inlet flanges, etc, and correct.

CHART 2 (Sheet 4 of 6)
TROUBLESHOOTING ENGINE

Trouble	Cause	Remedy
Low fuel flow.	Restricted flow to fuel metering valve.	Check mixture control for full travel. Check for restrictions in fuel filters and lines; adjust control and clean filters. Replace damaged parts.
	Fuel nozzle vent system defective causing improper pressure regulation.	Check venting system for leaks at connections and other defects. Tighten connections and replace defective parts.
	Fuel control lever interference.	Check operation of throttle control and for possible contact with cooling shroud. Adjust as required to obtain correct operation.
	Incorrect fuel injector pump adjustment and operation.	Check and adjust using appropriate equipment. Replace defective pump.
	Air leakage in fuel pump pressurization line.	Locate cause of leakage and correct.
High fuel flow.	Restricted flow beyond fuel control assembly.	Check for restricted fuel nozzles or fuel manifold valve. Clean or replace nozzles. Replace defective fuel manifold valve.
	(Seneca IV) Defective relief valve operation in fuel injector.	Check fuel injector pump control line from turbocharger for loose connections and defects. Tighten connections, replace damaged line.
	Restricted recirculation passage in fuel injector.	Replace pump.
	(Seneca IV) Air leakage in fuel gauge vent pressurization line.	Locate cause of leakage and eliminate.
Fluctuating fuel flow.	Vapor in fuel system.	Normally operating the auxiliary pump will clear system. Operate auxiliary pump and purge system.
	(Seneca IV) Fuel gauge line leak or improperly purged lines.	Purge gauge line and tighten connections.

CHART 2 (Sheet 5 of 6)
TROUBLESHOOTING ENGINE

Trouble	Cause	Remedy
Low oil pressure on engine gauge.	Insufficient oil in oil sump, oil dilution or using improper grade of oil for prevailing ambient temperature.	Add oil or change oil to proper viscosity.
	High oil temperature.	Defective oil temperature control valve in oil cooler; restriction in oil cooler. Replace valve or overhaul oil cooler.
	Leaking, damaged or loose oil line connections - restricted screens and filter.	Check for restricted lines and loose connections, and for partial plugged oil filter and screens. Clean parts, tighten connections, and replace defective parts.
	Leaking oil seal in turbocharger.	Check for oil in turbocharger exhaust outlet. Replace turbocharger.
	Defective check valve in turbocharger oil supply line.	Disassemble and clean valve or replace.
High oil temperature.	Low oil supply.	Fill to proper level with specified oil.
	Oil cooler air passages clogged.	Clean thoroughly.
	Clogged oil lines, screens, or oil cooler.	Remove and clean oil screens. Replace or overhaul oil cooler.
	Defective oil temperature control valve, or valve held open by debris.	Remove valve and clean valve and seat. If condition persists, replace Oil Temperature Control Valve.
	Oil cooler core plugged.	Replace or overhaul oil cooler.
	Oil viscosity too high.	Drain and refill with correct oil see Continental Maintenance Manual, Servicing Oil.
	Prolonged ground operation.	Limit ground operation to a minimum. See aircraft POH.
	Malfunctioning gauge.	Replace gauge.

CHART 2 (Sheet 6 of 6)
TROUBLESHOOTING ENGINE

Trouble	Cause	Remedy
Poor engine idle cutoff.	Engine getting fuel.	Check fuel control for being in full “IDLE CUTOFF” position. Check auxiliary pump for being “OFF.” Check for leaking fuel manifold valve. Replace defective components.
White smoke exhaust.	Turbo choking oil forced through seal in turbine housing.	Clean or change turbocharger. Refer to Chapter 81.

3. Engine

See “Figure 1” on page 710010.

A. Removal

(1) Turn off all electrical switches in the cockpit and disconnect the battery ground wire at the battery.
(2) Move the fuel selector valve in the cockpit to the OFF position.
(3) Remove the engine cowling.
(4) Remove the propeller. (See 61-10-00.)
(5) Disconnect the starter positive lead and ground lead at the starter.
(6) (Seneca IV models) Disconnect the tachometer cable to the engine.
(7) Disconnect the governor control cable at the governor and cable attachment clamps.
(8) Disconnect the throttle and mixture cables from the fuel-air control unit (Seneca IV), or from the fuel servo (Seneca V).
(9) On the left engine, disconnect the air conditioning compressor lines, if compressor is installed.
(10) Disconnect the cylinder temperature sender wire at No. 2 cylinder.
(11) Disconnect the fuel pump supply line and, on Seneca IV models, disconnect vent line from the engine.

NOTE: In some manner, identify all hoses, wires and lines to facilitate installation. Open fuel, oil, vacuum lines and fittings should be covered to prevent contamination.

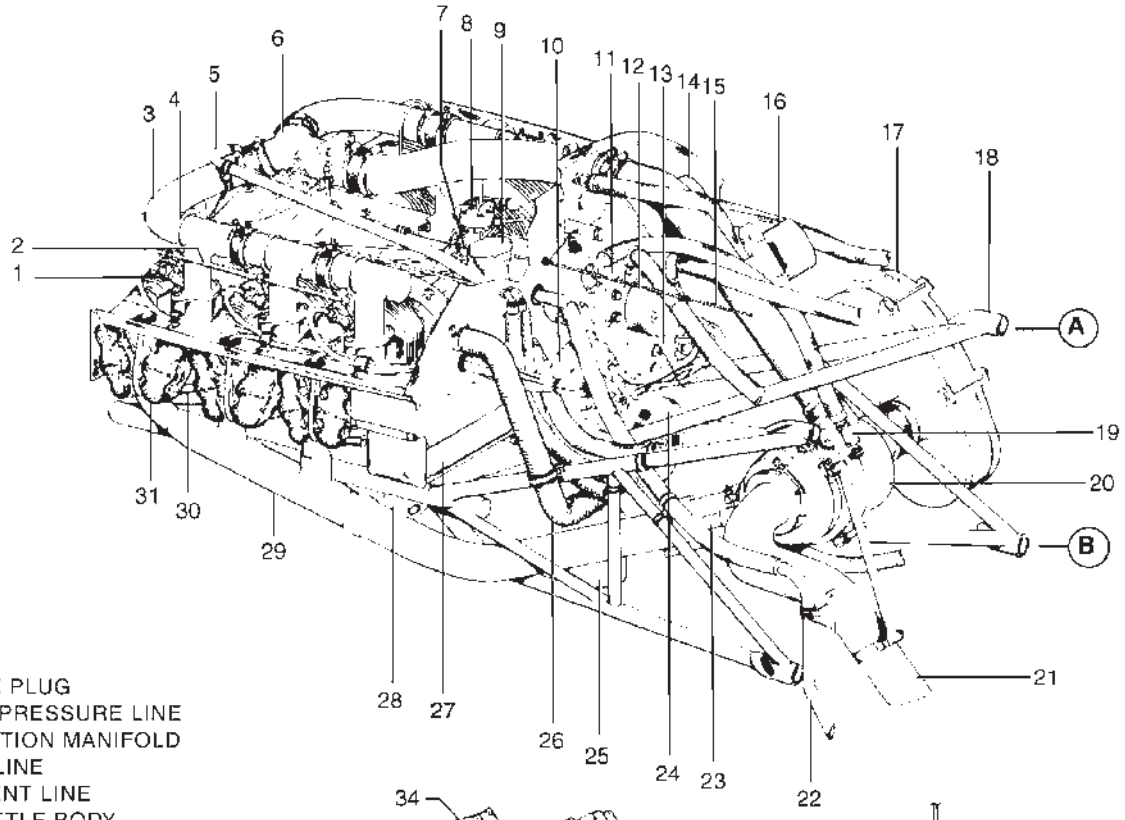
(12) Disconnect the magneto “P” leads at the magnetos.
(13) Disconnect the engine vent tube at the engine.
(14) Disconnect the engine oil temperature lead at the aft end of the engine.
(15) Untie the ignition harness, hoses and lines at the aft end of the engine.
(16) Disconnect the pneumatic pump lines at pump and remove fittings from pump.
(17) Disconnect the oil pressure line at the engine.
(18) Disconnect the fuel flow line at the left rear engine baffle.
(19) Disconnect the manifold pressure line at the left rear side of the engine.
(20) Disconnect the alternator leads and the cable attachment clamps.

CAUTION: PLACE A TAIL STAND UNDER THE TAIL OF THE AIRPLANE BEFORE REMOVING AN ENGINE.

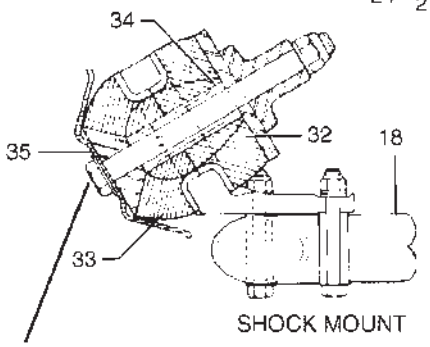
(21) Attach a one-half ton (minimum) hoist to the hoisting straps and relieve the tension from the engine mounts.
(22) Check the engine for any attachments remaining to obstruct its removal.

NOTE: Remove exhaust system components where they pass through engine mount.

(23) Drain the engine oil.
(24) Remove the engine mounting bolts and lower mount assembly.
(25) Carefully raise the engine and pull forward to clear the mount. Ensure there are no connections remaining to obstruct removal of the engine. Remove engine from airplane and place on a suitable stand.

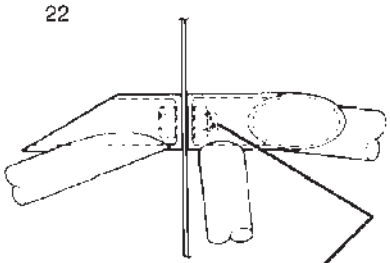


1. SPARK PLUG
2. DECK PRESSURE LINE
3. INDUCTION MANIFOLD
4. FUEL LINE
5. OIL VENT LINE
6. THROTTLE BODY
7. OIL DIPSTICK
8. FUEL MANIFOLD
9. OIL FILLER
10. LEFT MAGNETO
11. OIL PUMP
12. PRESSURE PUMP
13. RIGHT MAGNETO
14. STARTER
15. THROTTLE CABLE
16. PRESSURE RELIEF VALVE
17. INDUCTION AIR FILTER
18. ENGINE MOUNT
19. CHECK VALVE
20. TURBOCHARGER
21. TAIL PIPE
22. OIL VENT
23. EXHAUST BYPASS
24. ALTERNATOR
25. OIL FILTER
26. ALTERNATOR COOLING
27. OIL COOLER
28. SHOCK MOUNT
29. EXHAUST STACKS
30. PUSH RODS
31. ROCKER COVERS
32. UPPER MOUNT
33. LOWER MOUNT
34. SPACER
35. WASHER



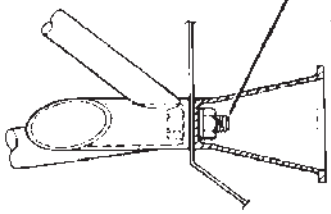
SHOCK MOUNT

BOLT - (AN7-46A)
WASHER - (AN960-716)
NUT - (MS20365-720C)
TORQUE TO 450-500 IN. LBS



VIEW A

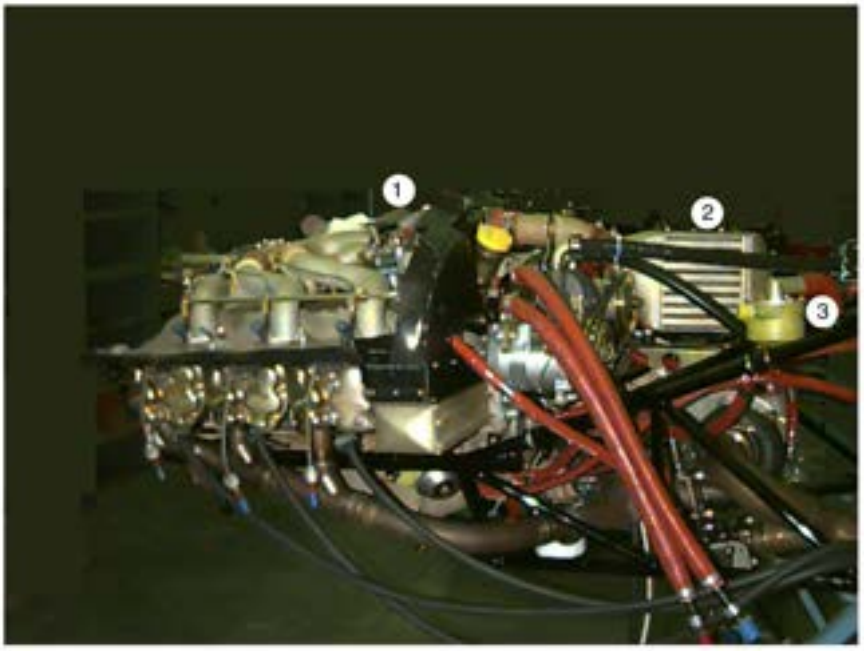
BOLT - (AN6-10A)
WASHER - (AN960-616)
NUT - (PS10062-8-624)
TORQUE TO 240-270 IN. LBS.



VIEW B

Power Plant Installation
Figure 1 (Sheet 1 of 2)

Effectivity
Seneca IV



1. RSA FUEL INJECTOR
2. INTERCOOLER
3. AIR/OIL SEPARATOR

Power Plant Installation
Figure 1 (Sheet 2 of 2)

B. Installation

CAUTION: PRIOR TO INSTALLING THE ENGINE, BE SURE TO INSTALL ALL ITEMS THAT WERE REMOVED AFTER THE ENGINE WAS REMOVED FROM THE AIRCRAFT.

CAUTION: REMOVE ALL PROTECTIVE CAPS AND IDENTIFICATION TAGS AS EACH ITEM IS INSTALLED.

(1) Install the shock mount in the engine mount and hoist the engine into position on the mount.

(2) Install the lower shock mount assemblies and mounting bolts. Torque the bolts as specified in “Figure 1” on page 710010, Sheet 1, for both models.

(3) Route and connect the throttle and mixture control cables and adjust.

(4) Route and connect the propeller governor control cable and adjust.

(5) Connect the alternate air cable and adjust.

CAUTION: APPLY LOCTITE NO. 567 PST SEALANT TO ALL MALE PIPE THREAD FITTINGS. DO NOT ALLOW TO ENTER SYSTEM.

CAUTION: SECURE ALL CABLES, HOSES AND WIRES WITH CLAMPS AND TY-STRAP IN THE SAME LOCATION AS BEFORE REMOVAL.

(6) Reconnect all lines and hoses previously disconnected from the engine.

(7) Route and connect the electrical leads to the appropriate connections on the engine.

(8) Connect the tachometer drive cable (Seneca IV) or electrical connections (Seneca V).

(9) Connect the air conditioning compressor lines, if compressor is installed.

CAUTION: ON SENECA V MODELS, CHECK AND ADJUST ALTERNATOR AND COMPRESSOR BELT TENSIONS. ALTERNATOR BELT TENSION SHOULD BE BETWEEN 80 AND 90 LB. FOR COMPRESSOR BELT TENSION, SEE 21-50-00.

(10) Install the propeller and spinner per 61-10-00.

(11) Service the engine with the proper grade and quantity of oil. (See 12-20-00).

(12) Be certain all switches are in the OFF position and connect the battery cables.

(13) Install the engine cowling.

(14) Make a final check of the security, location and installation of all lines, wires and cables.

(15) Perform an operational check of the engine; inspect for leaks and make final adjustments to engine controls as required.

CAUTION: CHECK EXHAUST PIPE CLEARANCE. MINIMUM CLEARANCE BETWEEN STRUCTURE AND COWL FLAP DOOR OPENING SHOULD BE 0.62 OF AN INCH.

COWLING

1. Engine Cowling

See “Figure 1” on page 71102.

A. Removal

(1) Seneca IV

(a) Release the fasteners securing the two side access panels.

(b) Remove the fasteners securing the top cowl and then remove the top cowl.

(c) Disconnect the cowl flap control.

(d) Support the bottom cowl and remove the screws that attach the bottom cowl to the nose cowl, engine mount and nacelle.

(e) Remove nose cowl by removing the attaching screws and separating the two cowl halves.

(2) Seneca V

(a) Remove the fasteners securing the top cowl and then remove the top cowl.

(b) Disconnect the cowl flap control, alternate air control linkage, induction air hose, and cylinder drain valve.

(c) Support the bottom cowl and remove the screws that attach the bottom cowl to the nose cowl, engine mount and nacelle.

(d) Remove nose cowl by removing the attaching screws and separating the two cowl halves.

B. Installation

(1) Seneca IV

(a) Position the two nose cowl halves on the front of the engine and secure with screw fasteners.

(b) Position the bottom cowl and secure with screw fasteners to the nose cowl, engine mount and nacelle.

(c) Connect the cowl flap control.

(d) Position the top cowl and secure with attaching screw fasteners.

(e) Secure the side cowls to the upper and lower cowling.

(2) Seneca V

(a) Position two nose cowl halves on front of engine and secure with screw fasteners.

(b) Position bottom cowl and secure with screw fasteners to nose cowl, engine mount and nacelle.

(c) Connect cowl flap control, alternate air control linkage, induction air hose, and cylinder drain valve.

(d) Position the top cowl and secure with attaching stud assemblies.

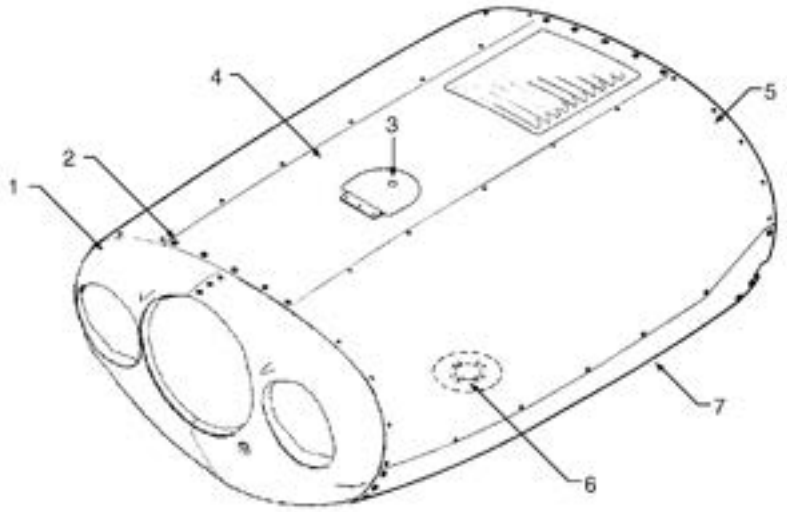
C. Cleaning, Inspection, and Repair

(1) The cowl should be cleaned with a suitable solvent and then wiped with a clean cloth.

(2) Inspect the cowling for dents, cracks, loose rivets, damaged or missing fasteners and damaged fiberglass areas.

(3) Repair all defects to prevent further damage. Fiberglass repair procedures may be accomplished according to Fiberglass Repairs, 51-70-00.

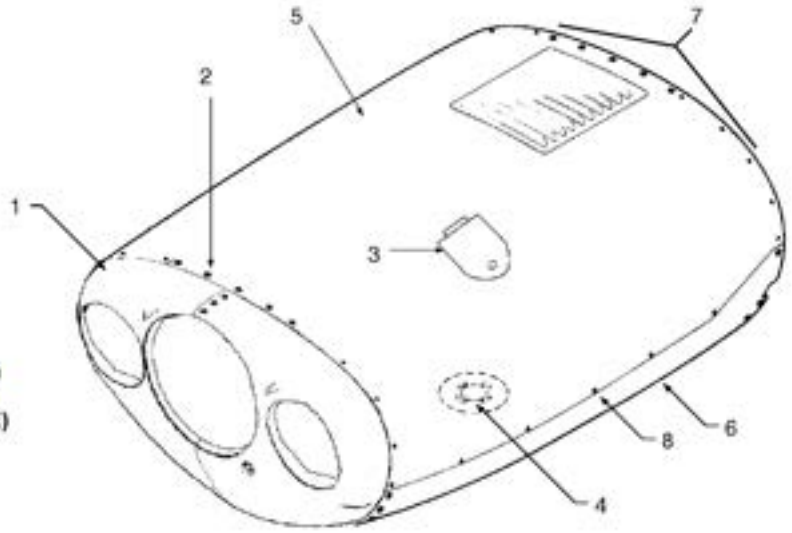
SENECA IV



1. NOSE COWL ASSEMBLY
2. SCREW FASTENER
3. OIL FILLER DOOR
4. TOP PANEL
5. SIDE PANEL
6. ACCESS HOLE, ENGINE OIL DRAIN
7. BOTTOM COWL

37207 AT

SENECA V



1. NOSE COWL ASSEMBLY
2. STUD ASSEMBLIES (11 PLACES)
3. OIL FILLER DOOR
4. ACCESS HOLE, ENGINE OIL DRAIN
5. TOP COWL
6. BOTTOM COWL
7. STUD ASSEMBLIES
   (10 PLACES, S/N'S 3449001 THRU 3449505)
   (19 PLACES, S/N'S 3449506 AND UP)
8. STUD ASSEMBLIES (7 PLACES EACH SIDE)

100754 N

Engine Cowling Installation
Figure 1

2. Engine Cowl Flaps

See “Figure 2” on page 71104.

The cowl flaps are all metal flaps located on the rear of the bottom cowls. The flaps are manually operated through a push-pull control from the cockpit. The cowl flaps are connected to the engine cowls with full length piano type hinges.

Operation and Adjustment

The cowl flaps operate through three positions; closed, intermediate and open by control levers located on the console. When the control levers are in the up position, the flaps are closed. To operate the cowl flaps, depress the lock and move the lever down, releasing the lock after the initial down movement will allow the lock to stop the flap travel at the intermediate position. For full open position, depress the lock and move the control down; release the lock after the initial movement and continue to move the control down until the lock stops the travel of the control. To raise the cowl flaps reverse the procedure. The cowl flaps should be adjusted as follows:

A. Place the control in the up position.

B. Ascertain that the control lock is engaged.

C. Check the cowl flap to visually determine that the flap is flush with the bottom of the engine cowl.

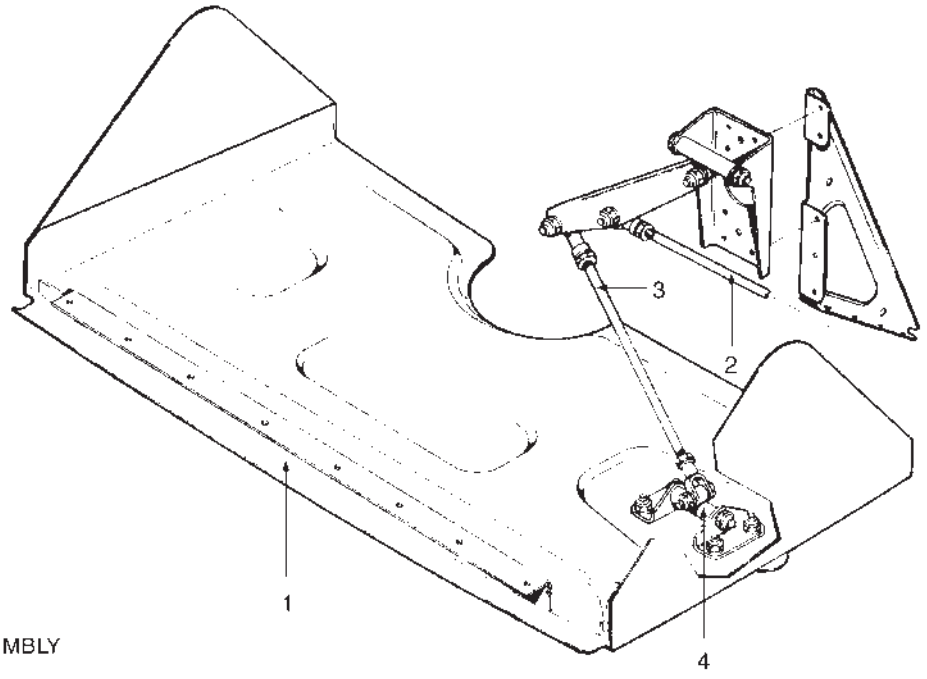
D. If the flap is not flush, disconnect the push-pull control from the arm on the inboard side of the flap.

E. Loosen the jam nut on the clevis end and adjust the clevis to get a flush fit between the cowl flap and engine cowl.

F. Reconnect the control to the flap and operate the cowl flap through its full range a few times; then place the control in the closed position and visually check the flap to determine if it is flush with the engine cowl.

G. If the cowl flap is not flush, repeat steps (D) through (F).

H. When the adjustment is completed, tighten the clevis jam nut and secure the push-pull control to the cowl flap.



1. COWL FLAP ASSEMBLY
2. CONTROL CABLE
3. ROD ASSEMBLY
4. TRUNNION ASSEMBLY

Cowl Flap Installation
Figure 2

MOUNTS

1. Engine Shock Mounts

   Replacement of Engine Shock Mounts (See “Figure 1” on page 710010.)

   (1) Remove the engine cowling.
   (2) Relieve the engine weight on the mounts using a one-half ton hoist attached to the engine lifting points.
   (3) Remove the four engine mounting bolts and the lower half of the mount assemblies.
   (4) Carefully raise the engine just enough to remove the shock mounts. Check all lines, wires and cables for interference. Disconnect any lines and cables if necessary.
   (5) Check all components for wear, damage or cracks and install new mounting kit.
   (6) Lower the engine slowly and use mounting bolts to keep the components aligned.
   (7) When the engine is supported by the mount, check the mounts for proper seating.
   (8) Install the mounting bolt, nut, washer and torque 450 to 500 inch-pounds and safety.
   (9) Reconnect any lines, wires or cable that were disconnected and install engine cowling.

2. Engine Mount Corrosion Inspection, Immersion in Water

   The following guidance is general in nature and should be applied or varied to fit the individual situation based on water level during immersion, length of time immersed, length of time since exposure, etc. Proceed as follows:

   A. Inspection

      (1) Level the aircraft in accordance with 8-20-00 in this maintenance manual.
      (2) In two of the larger, lower, engine mount tubes, drill a 3/16 inch hole in each tube, at the approximate mid-point.
      (3) Visually inspect the interior surface of each tube through the 3/16 inch hole for evidence of internal corrosion. Pay particular attention to the lower end of each tube as this is where corrosion is most likely to appear first.
      (4) Should evidence of corrosion be detected in step (3), above, replace the engine mount. If no corrosion is detected, proceed with corrosion prevention, paragraph B, below.

   B. Corrosion Prevention

      If no evidence of corrosion is detected in step (3), above, proceed as follows:

      (1) Place a drip pan below the inspection holes in each engine mount tube.
      (2) Insert a plastic tube thru each inspection hole and feed it up to the high point of the engine mount tube.
      (3) Using a syringe inserted into the end of the plastic tube, pump linseed oil into the upper end of the engine mount tube while rotating the syringe / plastic tube assembly to assure maximum coverage. Continue pumping until the lower end of the engine mount tube is filled with linseed oil to the level of the inspection hole.
      (4) Now, draw the plastic tube out of the upper end of the engine mount tube and reinsert it in the opposite direction, feeding it to the lower end of the engine mount tube.
      (5) Suck excess linseed oil out of the engine mount tube with the syringe / plastic tube assembly.
      (6) When linseed oil can no longer be picked up by the syringe / plastic tube assembly, remove it and allow the engine mount tube to drain into drip pans for approximately two hours.
      (7) Purge excess oil from tubes by applying air pressure to each 3/16 inch inspection hole, one at a time.

(8) Insure that roughly the same amount of linseed oil that was pumped in is retrieved in the drip pans.

(9) Apply a liberal coating of an approved fuel tank sealant (see Consumable Materials, 91-10-00) to each inspection hole and seal the hole with an appropriate blind rivet. After installing the rivet, apply a liberal coating of the approved fuel tank sealant over the head of the rivet.

AIR INTAKES

1. Induction System Air Filter

See “Figure 1” on page 71602 and “Figure 2” on page 71602.

A. Removal

(1) Seneca IV

(a) Remove the side panel cowl on the right side of the engine.

(b) Release the stud fasteners; remove the filter cover and withdraw the filter element.

(2) Seneca V

(a) Remove the upper cowling.

(b) Disconnect alternate air linkage/cable.

(c) Remove induction air hose.

(d) Remove lower cowl.

(e) Remove six screws securing cover.

(f) Remove filter.

B. Cleaning

CAUTION: NEVER WASH THE FILTER ELEMENT IN ANY LIQUID OR SOAK IT IN OIL. NEVER ATTEMPT TO BLOW OFF DIRT WITH COMPRESSED AIR.

The air filter element should be cleaned as often as it becomes dirty, everyday under severe dust conditions. The filter element should be replaced if any holes or tears exist. When cleaning the filter, it is good practice to remove the filter box assembly and clean with a solvent. Blow the assembly dry and wipe with a clean cloth to remove embedded debris. Be careful not to damage the sealing ends.

C. Installation

CAUTION: CHECK THE INDUCTION SYSTEM TO BE SURE THAT NO AIR LEAKS EXIST AT ANY POINT THAT WOULD ALLOW UNFILTERED AIR TO ENTER THE ENGINE.

(1) Seneca IV

(a) Install the filter box assembly if removed.

(b) Position the filter element in the box assembly and secure the cover assembly with the stud fasteners.

(2) Seneca V

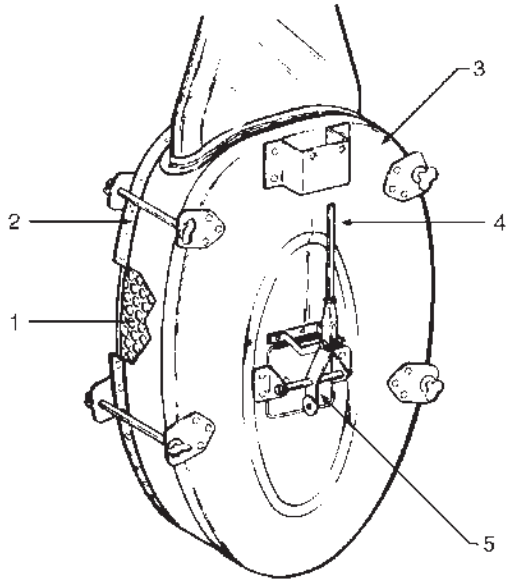
(a) Install the filter box assembly if removed.

(b) Position the filter element in the box assembly and secure the cover assembly with the six screws.

(c) Install lower cowl.

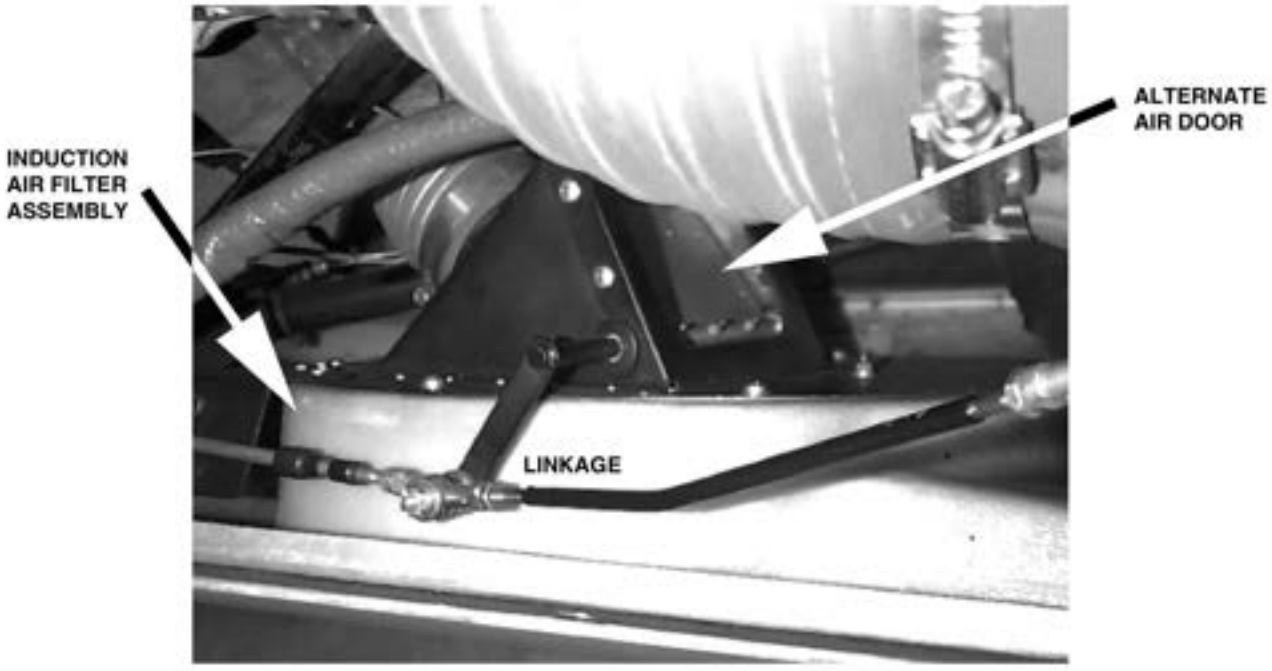
(d) Connect alternate air linkage/cable and induction air hose.

(e) Install upper cowl.



1. FILTER ELEMENT CA-161-PL
2. BOX ASSEMBLY
3. COVER ASSEMBLY
4. ALTERNATE AIR CONTROL CABLE
5. ALTERNATE AIR DOOR

Seneca IV - Induction System Installation
Figure 1



Seneca V - Induction System Installation
Figure 2

2. Alternate Air Door

See “Figure 1” on page 71602 and “Figure 2” on page 71602.

The alternate air door is located in the alternate air box to provide a source of air to the engine should there be an air stoppage through the filter system. The following should be checked during inspection:

A. Seneca IV

(1) Check that air door seals are tight and the hinge and torsion spring are secure.

(2) Adjust the control cable to position the roller on the arm assembly clear of the door in the closed position. Check that when the cockpit control is in the closed position the door is properly seated in the closed position.

(3) Actuate the door by operating the control lever in the cockpit to determine that it is not sticking or binding.

(4) Check the cockpit control cable for free travel.

B. Seneca V

(1) Check that air door seals are tight and the hinge is secure.

(2) Adjust the control cable to position the alternate air door fully shut (against stops) when alternate air control is positioned in the off position.

(3) Adjust linkage to intercooler shutoff valve to position valve against stops (fully open) when alternate air cockpit control is positioned in the off position.

(4) Actuate the door by operating the control lever in the cockpit to determine that it is not sticking or binding.

(5) Check the cockpit control cable for free travel.

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CHAPTER

73

ENGINE FUEL AND CONTROL

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CHAPTER 73

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CHAPTER 73 - ENGINE FUEL AND CONTROL

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DISTRIBUTION

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

1. Description

See “Figure 1” on page 73102.

A. Seneca IV

The Continental Fuel Injection System is used on each of the engines of the Seneca IV. This system uses individual injectors for each cylinder and, while compensating for altitude and engine operating conditions, provides continuous flow to the cylinders.

Each system makes use of a combination fuel pump/mixture control unit to supply fuel pressure to the fuel metering unit on the air throttle body. The metering unit determines and controls fuel flow to the manifold valve and injectors by interconnection with the throttle valve.

The engine driven fuel pump is mounted to the right crankcase and located ahead of the engine mount. Both pumps are of a positive displacement, rotary vane type, each having an integral vapor separator and altitude compensating android valve.

The throttle/metering unit is on the top forward part of the engine connected to the entrance of the intake manifold. A rotary valve makes up the metering unit and is attached to the throttle valve. As the throttle is moved, the cam shaped edge of the rotary valve moves across the fuel delivery port controlling the flow of fuel to the manifold valve and nozzles.

The fuel manifold valve is the central point for dividing fuel to the individual cylinders. A diaphragm and plunger valve within the manifold valve raises or lowers by fuel pressure to open or close the individual fuel supply ports simultaneously.

The fuel discharge nozzles are an air bleed type nozzle with a calibrated orifice. A nozzle is installed in the cylinder head outside each intake valve for each cylinder.

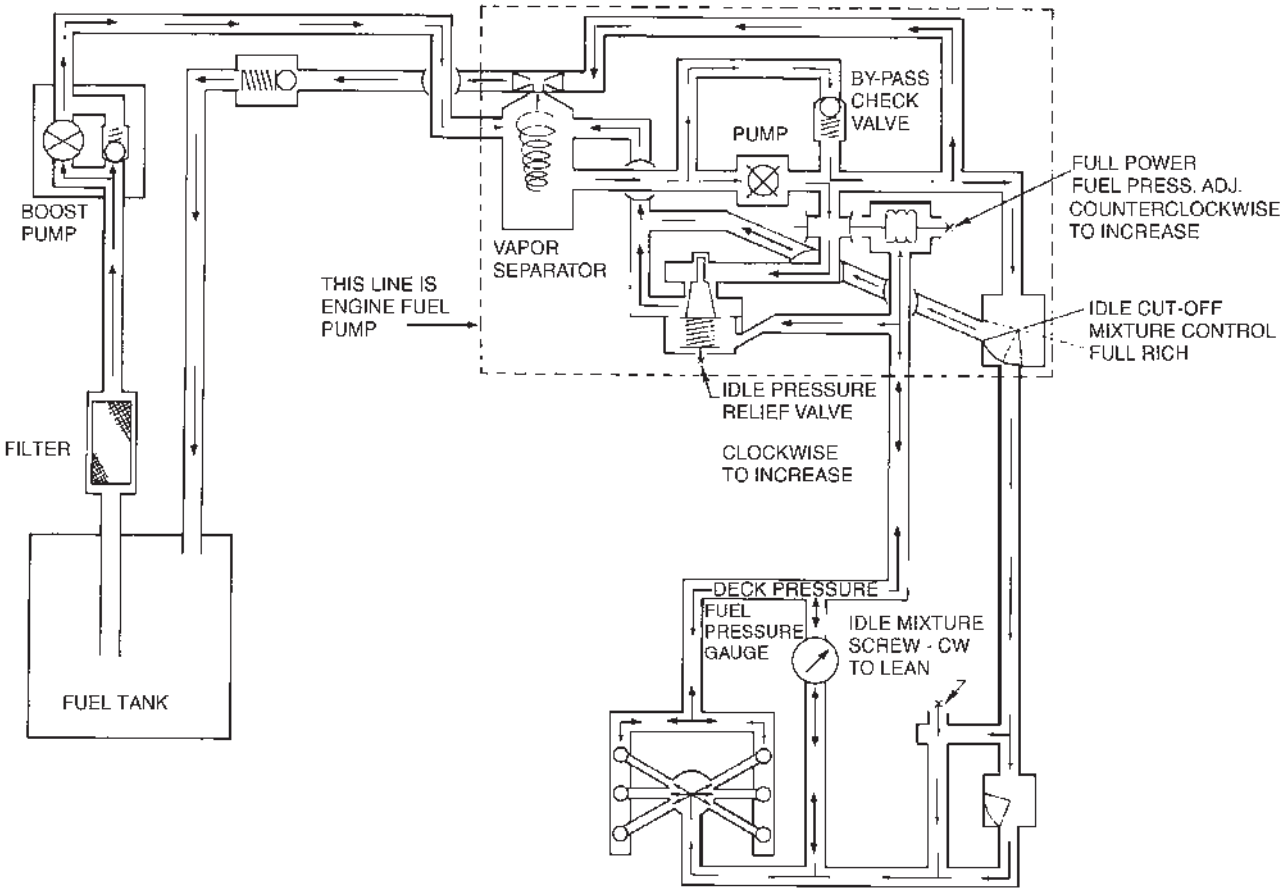
B. Seneca V

The Precision Airmotive Fuel Injection System is used on each of the engines of the Seneca V. The injector is mounted on top of the engine. This system uses individual injectors for each cylinder and, while compensating for engine operating conditions, provides continuous flow to the cylinders.

Precision Airmotive Fuel Injection Systems are designed to meter fuel in direct ratio to the volume of air being consumed by the engine at any given time. This is accomplished by sensing venturi suction and impact air pressures in the throttle body. Opening or closing the throttle valve results in a change in the volume of air being drawn into the engine. This results in a change in the velocity of air passing across the impact tubes and through the venturi. When air velocity increases, the pressure at the impact tubes remains relatively constant depending upon the inlet duct configuration, air filter location, etc. The pressure at the venturi throat decreases, creating a differential (impact minus suction) which is used over the entire range of operation of the fuel injection system as a measurement of the volume of air consumption.

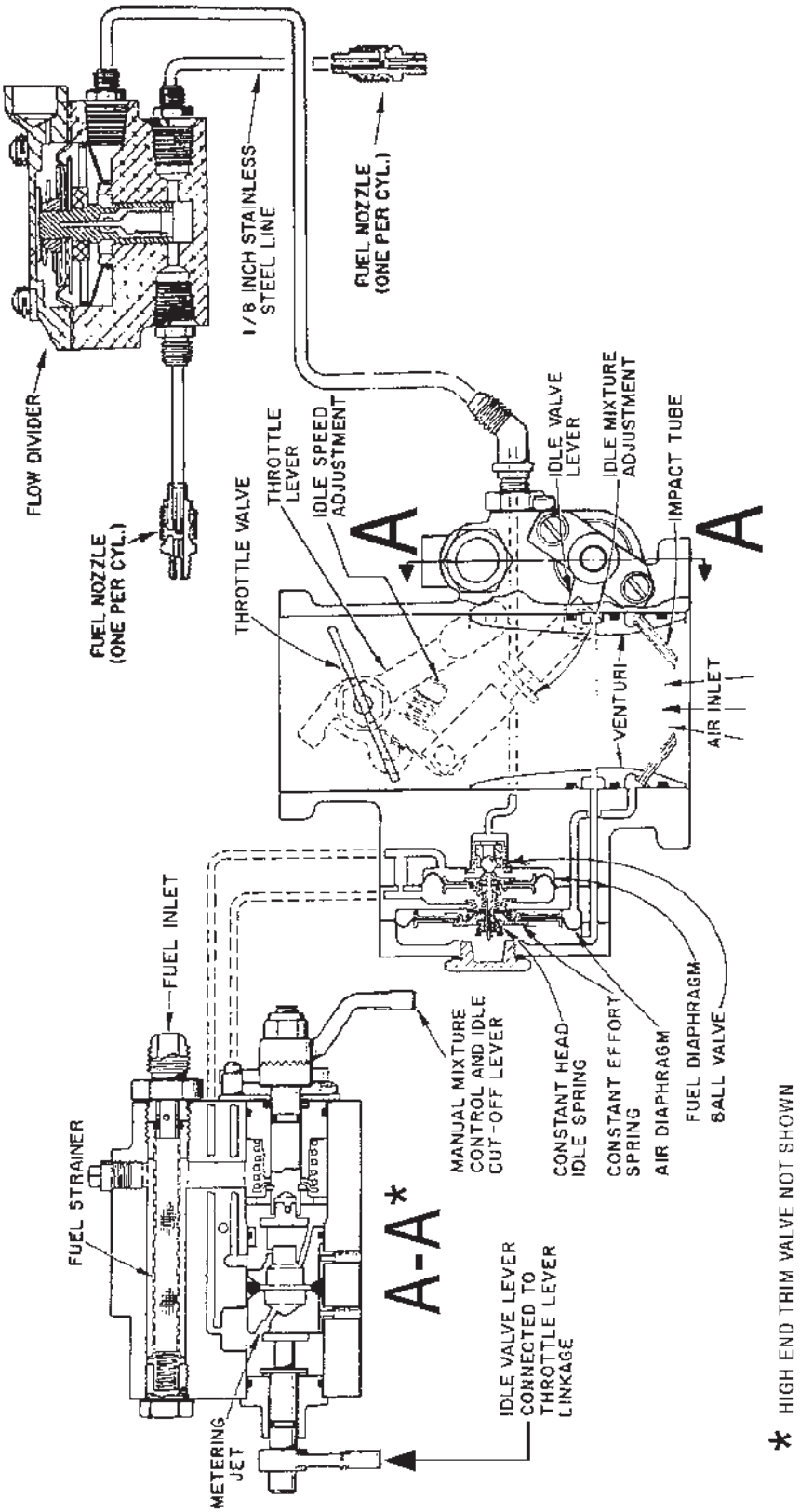
2. Troubleshooting

See “Chart 1” on page 73104.



Effectivity
Seneca IV

Fuel Injection System Functional Schematic
Figure 1 (Sheet 1 of 2)



Fuel Injection System Functional Schematic
Figure 1 (Sheet 2 of 2)

Effectivity
Seneca V

CHART 1 (Sheet 1 of 3)
TROUBLESHOOTING RSA-5 FUEL INJECTION SYSTEM (SENECA V)

Trouble	Cause	Remedy
Hard starting.	Technique.	Refer to aircraft manufacturer's recommended starting procedure.
	Flooded.	Clear engine by cranking with throttle open and mixture control in ICO.
	Throttle valve opened too far.	Open throttle to position approximating 800 rpm.
	Insufficient prime (Usually accompanied by a backfire).	Increase amount of priming.
Rough idle.	Mixture too rich or too lean.	Confirm with mixture control. A too rich mixture will be corrected and roughness decreased during lean-out while a too lean mixture will be aggravated and roughness increased. Adjust idle to give a 25-50 rpm rise @700 rpm.
	Plugged nozzle(s). (Usually accompanied by high take-off fuel flow readings.)	Clean nozzles in Methyl-Ethyl-Ketone, acetone, hydrocarbon cleaning solvent or a chlorinated solvent equivalent to chlorothene. Check system for source of contamination.
	Slight air leak into induction system through manifold drain check valve. (Usually able to adjust initial idle but rough in 1,000 - 1,500 rpm range.)	Confirm by temporarily plugging drain line. Replace check valves as necessary.
	Air leak in fuel line from tank to servo unit.	Confirm by connecting clear tubing between servo and flow divider and watch for air bubbles. Locate and correct source of leakage. May include boost pump or main pump seal leakage.

CHART 1 (Sheet 2 of 3)
TROUBLESHOOTING RSA-5 FUEL INJECTION SYSTEM (SENECA V)

Trouble	Cause	Remedy
Rough idle. (continued)	Slight air leak into induction system through loose intake pipes or damaged “O” rings. (Usually able to adjust initial idle but rough in 1,000-1,500 rpm range.)	Repair as necessary.
	Large air leaks into induction system, such as missing pipe plugs, etc. (Usually unable to throttle engine down below 800-900 rpm.)	Repair as necessary.
	Internal leak in injector. (Usually unable to lean-out idle range.)	Replace injector.
	Unable to set and maintain idle.	Replace injector.
	Fuel vaporizing in fuel lines or distributor. (Encountered only under high ambient temperature conditions or following prolonged operation at low idle rpm’s.)	Refer to the suggestions in Section Four.
Low take off fuel flow.	Strainer plugged.	Remove strainer and clean a suitable solvent. Acetone, MEK, hydrocarbon cleaning solvent, or a chlorinated solvent equal to chlorothene is recommended.
	Injector out of adjustment.	Replace injector.
	Faulty gage.	In a twin engine installation, criss-cross gauges. Replace as necessary. Single engine, change gauge.
	Sticky flow divider valve.	Clean flow divider valves.

CHART 1 (Sheet 3 of 3)
TROUBLESHOOTING RSA-5 FUEL INJECTION SYSTEM (SENECA V)

Trouble	Cause	Remedy
High fuel flow reading.	Plugged nozzle if high fuel flow is accompanied by loss of power and roughness.	Remove and clean nozzles in Acetone, MEK, hydrocarbon cleaning solvent, or a chlorinated solvent equivalent to chlorothene is recommended. Check system for source of contamination.
	Faulty gage.	Criss-cross gages and replace if necessary.
	Injector out of adjustment.	Replace injector.
Staggered mixture control levers.	If take-off is satisfactory, do not be too concerned about staggered mixture control levers because some misalignment is normal with twin engine installation.	Check rigging.
Poor cut-off.	Improper rigging of aircraft linkage to mixture control.	Adjust.
	Mixture control valve scored or not seating properly.	Eliminate cause of scoring (usually burr or dirt) and lap mixture control valve and plug on surface plate.
	Vapor in lines.	See Section Four.
Rough engine (turbo charged) and poor cutoff.	Air bleed hole(s) clogged.	Clean or replace nozzles.
Engine will not accelerate past a given rpm.	Plugged nozzles if accompanied by high fuel flow.	Clean or replace nozzles. Check system for source of contamination.
	Improper internal engine timing or magneto problem.	Correct timing problem.
	Plugged or restricted exhaust manifold.	Refer to engine manual for corrective action.

3. Fuel Injection System Maintenance

CAUTION: DO NOT USE ANY FORM OF THREAD COMPOUND ON FUEL LINE FITTINGS. USE ONLY A FUEL SOLUBLE LUBRICANT SUCH AS ENGINE OIL.

A. Check tightness and lockwiring of all nuts and screws which fasten the injector to the engine.

B. Check all fuel lines for tightness, leaks, evidence of damage, or chafing by metal to metal contact.

C. Check control connections, levers, and linkages for tightness, travel, and lockwiring.

D. Inspect nozzles for cleanliness with particular attention to orifices. Use a standard 1/2 inch spark plug type deep socket to remove nozzles. Do not use wire or other object to clean orifices. To clean nozzles, remove from engine and immerse in fresh cleaning solvent. Use compressed air to dry. O-rings must be removed from nozzles prior to cleaning in certain solvents.

E. (Seneca V only) Remove and clean the injector fuel inlet strainer at the first 25 hour inspection and each 50 hour inspection thereafter. Damaged strainer o-rings should be replaced.

4. Fuel Injector Nozzle Assembly

See “Figure 2” on page 73108.

A. Removal

(1) Remove the cowling side access panels (Seneca IV) or top cowl (Seneca V).
(2) Disconnect the fuel line and remove the reference air line from the nozzle.
(3) Used a standard 1/2 inch spark plug type deep socket to remove the nozzle.

B. Cleaning and Inspection

WARNING: COMPRESSED AIR USED FOR CLEANING PURPOSES MUST NOT EXCEED 30 PSI. USE ONLY WITH EFFECTIVE CHIP- GUARDING AND PERSONAL PROTECTIVE EQUIPMENT (GOGGLES, SHIELDS, GLOVES, ETC.).

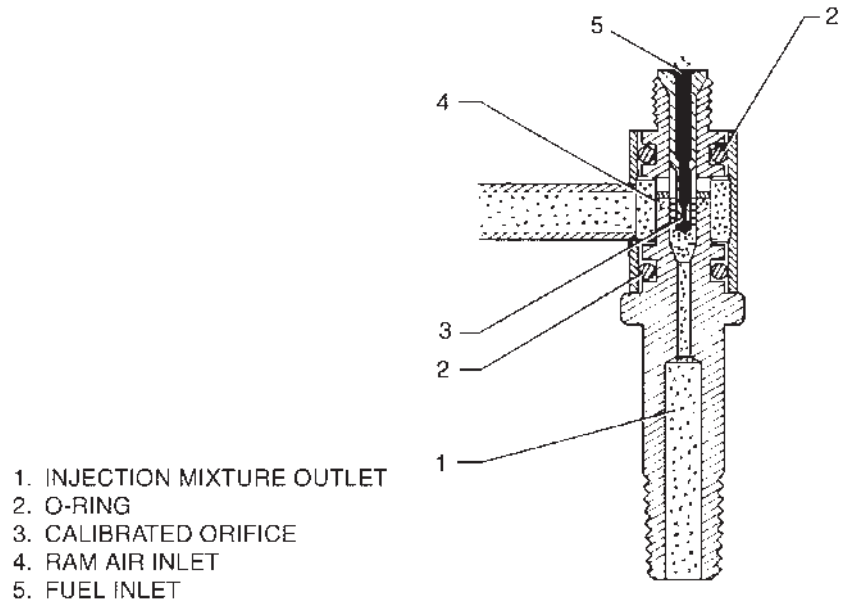
WARNING: METHYL ETHYL KETONE (MEK) AND ACETONE ARE FLAMMABLE AND HARMFUL TO EYES, SKIN, AND BREATHING PASSAGES. KEEP IGNITION SOURCES AWAY. PROVIDE ADEQUATE VENTILATION AND PROTECTIVE CLOTHING.

CAUTION: DO NOT USE WIRE OR OTHER OBJECT TO CLEAN ORIFICES. O-RINGS MUST BE REMOVED PRIOR TO CLEANING IN CERTAIN SOLVENTS.

(1) Clean nozzles by immersing in methyl ethyl ketone or acetone. Use compressed air to dry.
(2) Use compressed air to clean nozzle body in cylinder.
(3) Inspect the nozzles for cleanliness; pay particular attention to the orifices. Check the condition of the nozzle and cylinder threads.

C. Installation

(1) Carefully start the nozzles by hand to prevent cross-threading.
(2) Using a clean, deep well, 1/2 inch socket, tighten and torque nozzle to 60 inch-pounds.
(3) Install reference air lines on nozzles.
(4) Connect the fuel line to the nozzle.
(5) Install cowling side panels (Seneca IV) or top cowl (Seneca V).



Fuel Injector Nozzle Assembly
Figure 2

INDICATING

NOTE: For information on the fuel flow gauge installed in Seneca V models, see 77-40-00.

1. Dual Fuel Flow Gauge (Seneca IV)

   The dual fuel flow gauge is a non-electric differential pressure gauge mounted in the bottom of the instrument panel in the column of engine instruments.

   This instrument measures flow by reading the pressure drop across a fixed orifice. With fuel pressure being supplied by the engine driven pump and a fixed orifice in the fuel divider head, then measuring the pressure drop upstream of the orifice against deck pressure at nozzles, the resultant pressure can be equated to fuel flow in gallons per hour flow.

2. Troubleshooting

   See “Chart 1”.

CHART 1
TROUBLESHOOTING FUEL FLOW GAUGE

Trouble	Cause	Remedy
Pointer oscillates.	Air in fuel line.	Purge line.
Gauge read low at altitude.	Vent line restricted.	Check line and fittings.
Pointer does not return to zero.	Fuel in diaphragm of gauge.	Replace gauge.

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CHAPTER

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IGNITION

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CHAPTER 74

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CHAPTER 74 - IGNITION

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ELECTRICAL POWER SUPPLY

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

CAUTION: ENSURE THAT THE PRIMARY CIRCUITS OF BOTH MAGNETOS ARE GROUNDED BEFORE WORKING ON THE ENGINE.

This section provides only basic timing magnetos to the engine procedures and, if not provided in the appropriate vendor engine maintenance manual, magneto removal and installation instructions. Refer to the appropriate vendor's engine and magneto maintenance manuals for all other magneto maintenance procedures. See Introduction – Supplementary Publications.

1. Ignition System

A. Description

Ignition of the fuel charge in each cylinder is accomplished by two spark plugs independently excited by one of two magnetos, one mounted on each engine. Each magneto separately generates, times and distributes high tension (voltage) through leads to each cylinder. Both magnetos are pressurized to improve magneto efficiency at altitude.

The magnetos are controlled by two switches on the pilot's instrument panel. With the switch OFF, the magneto is grounded and will not produce spark.

B. Replacement Magnetos

NOTE: Check the magneto dataplate to verify the specific model number and series of the magneto being worked on.

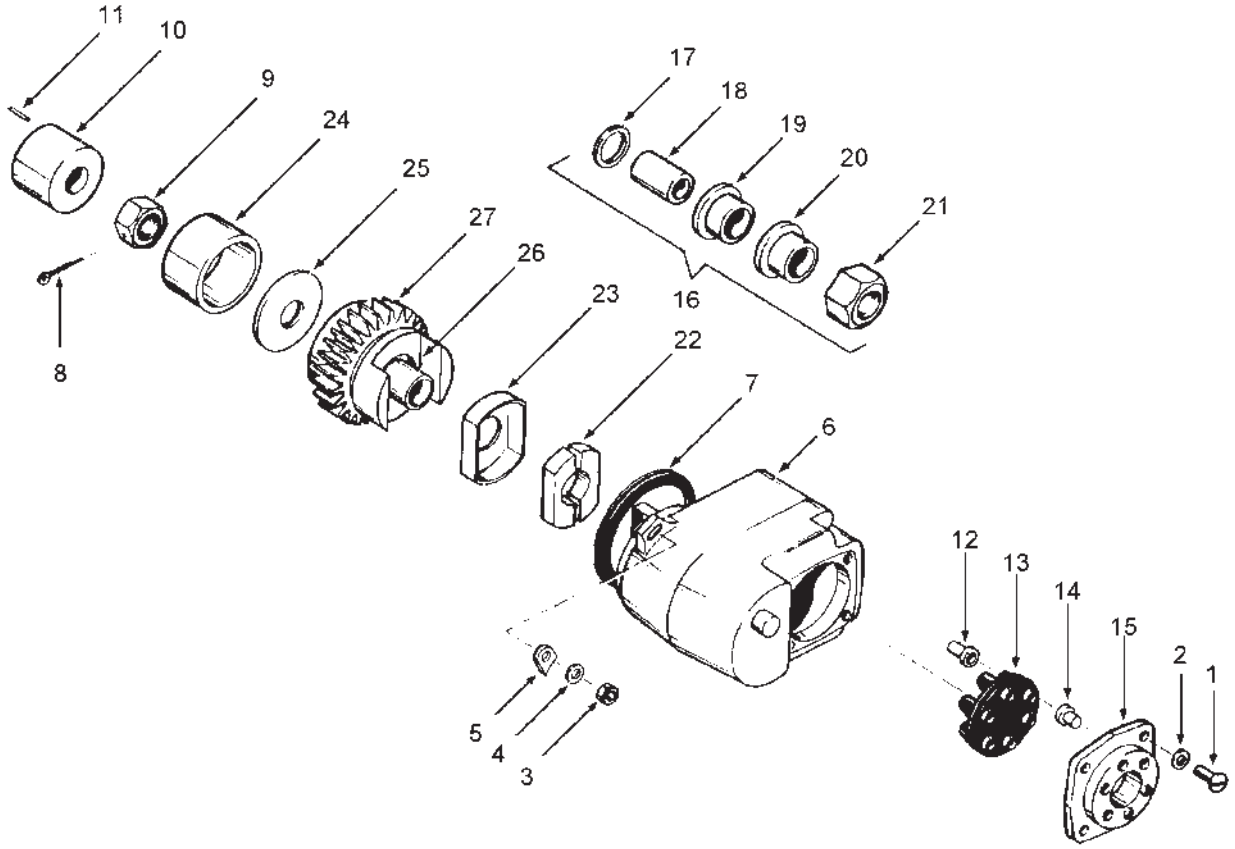
NOTE: Time magneto to engine per the following instructions or as described in Continental Motors MSB94-8D or later.

C. Overhaul

Overhaul is required as conditions indicate, but in no case may Slick 6300 series magnetos time-in-service exceed the TBO for the engine. Magnetos must also be overhauled after a lightning strike or following a sudden engine stoppage.

NOTE: An alternative to overhaul is complete magneto replacement with a new magneto. New magnetos incorporate all the latest design features and may be a cost effective alternative to overhaul.

This chapter deals with general maintenance and timing of the magnetos to the engine. Refer to the appropriate vendor magneto overhaul manual for detailed procedures. See Introduction – Supplementary Publications.



1. SCREW
2. LOCK WASHER
3. NUT
4. LOCK WASHER
5. HOLD DOWN WASHER
6. MAGNETO
7. GASKET
8. COTTER PIN
9. NUT
10. GEAR SUPPORT SHAFT
11. PIN
12. IGNITION CABLE EYELET
13. DISTRIBUTOR CABLE GROMMET
14. IGNITION CABLE FERRULE
15. DISTRIBUTOR CABLE PLATE
16. GROUND TERMINAL KIT
17. WASHER
18. INSULATING SLEEVE
19. INNER FERRULE
20. OUTER FERRULE
21. COUPLING NUT
22. COUPLING BUSHINGS
23. RETAINER
24. NEEDLE BEARING
25. WASHER
26. PILOT SLEEVE BUSHING
27. MAGNETO DRIVE GEAR

CAUTION

WHEN MAINTAINING OR INSPECTING MAGNETOS, BE SURE TO OBTAIN THE MANUFACTURER'S MAINTENANCE DATA FOR THE PARTICULAR MAGNETO INSTALLED IN YOUR AIRPLANE.

Magneto Assembly (Typical)
Figure 1

2. Magnetos

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

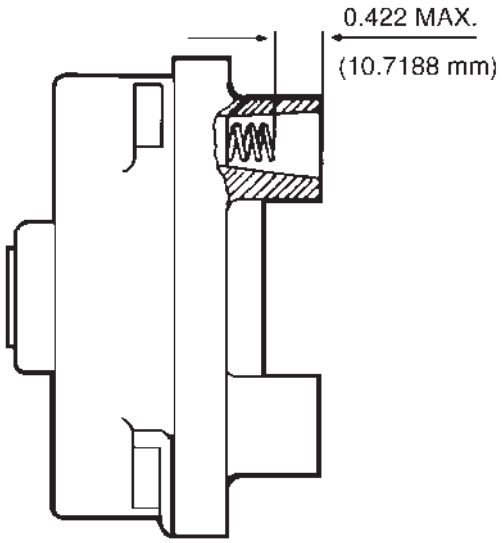
A. Inspection

(1) After the first 25 hour and 50 hour periods, and periodically thereafter, the contact assemblies should be checked. Examine the points for excessive wear or burning. Points which have deep pits or excessively burned areas should be discarded. Examine the cam follower felt for proper lubrication. If necessary, points can be cleaned by using any hard finished paper. Clean breaker compartment with dry cloth.

(2) If engine operating troubles develop which appear to be caused by the ignition system, it is advisable to check the spark plugs and wiring first before working on the magnetos.

(3) Should the trouble appear to be definitely associated with the magneto, the most effective measure is to install a replacement magneto which is known to be in satisfactory condition. Send the suspected unit to the overhaul shop for test and repair.

(4) Should this not be possible, a visual inspection may disclose the source of trouble. Remove the harness outlet plate from the magneto. Inspect for the presence of moisture and foreign matter on the rubber grommet and high tension outlet side of the distributor block. Check height of block contact springs. The top of the spring must not be more than 0.422 of an inch (10.718 mm) below the top of the tower as shown in “Figure 2”. If the springs are broken or corroded, replace them.



Contact Spring inspection
Figure 2

(5) Inspect the distributor block for cracks or burned areas. The wax coating on the block should not be removed. Do not use solvents.

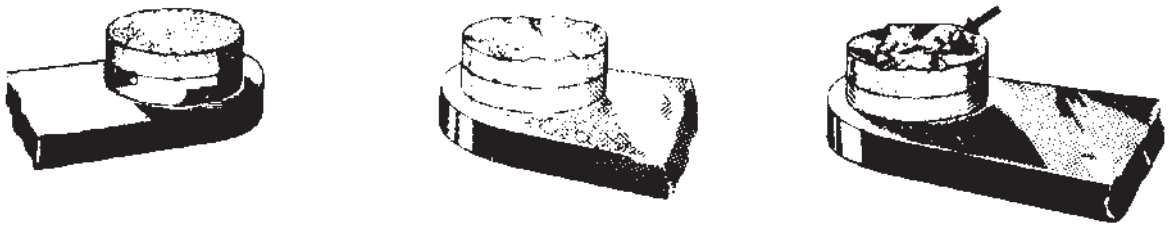
(6) Check for excess oil in the breaker compartment. If present, it may mean a bad oil seal or oil seal bushing at the drive end. Check the magneto manufacturer’s overhaul procedure.

(7) Remove the breaker cover and harness securing screws and nuts and separate cover from magneto housing. Check contact assemblies to see that the cam follower is securely riveted to its spring. Examine the contact points for excessive wear or burning. “Figure 3” on page 74104 shows how the average contact point will look when surfaces are separated for inspection. Desired contact surfaces have a dull gray, sandblasted (almost rough) or frosted appearance over the area where electrical contact is made. This means that points are worn in and mated to each other, thereby providing the best possible electrical contact and highest efficiency of performance.

NORMAL POINT IS SMOOTH AND FLAT. SURFACE HAS DULL GRAY "SANDBLASTED" APPEARANCE.

MINOR IRREGULARITIES - SMOOTH ROLLING HILLS AND DALES WITHOUT ANY DEEP PITS OR HIGH PEAKS. THIS IS A NORMAL CONDITION OF POINT WEAR.

WELL DEFINED MOUND EXTENDING NOTICEABLY ABOVE SURROUNDING SURFACE. REJECT POINTS.



Contact Points
Figure 3

(8) Minor irregularities or roughness of point surfaces are not harmful (refer to “Figure 3”, center), neither are small pits or mounds, if not too pronounced. If there is a possibility of pit becoming deep enough to penetrate pad, “Figure 3”, right, reject contact assembly.

NOTE: No attempt should be made to stone or dress contact points. Should contact assembly have bad points, or show excessive wear, replace complete assembly.

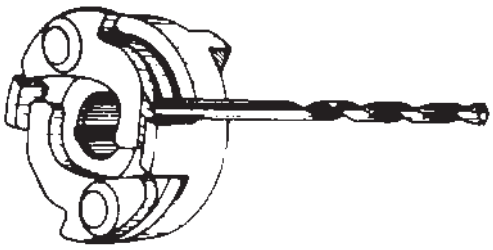
(9) Check the condition of the cam follower felt. Squeeze felt tightly between thumb and forefinger. If fingers are not moistened with oil, re-oil using 2 or 3 drops of Bendix 10-86527 lubricant. Allow approximately 30 minutes for felt to absorb the oil. Blot off the excess with a clean cloth. Too much oil may foul contact points and cause excessive burning.

(10) Inspect the felt washer in the distributor block for oil content. If the felt is dry, inspect the bronze bushing for wear. (Refer to manufacturer’s overhaul instruction.) Oil felt washer with Bendix Distributor Block Lubricant Part No. 10-391200. Blot excess oil from washer until flat surfaces take on a “frosted” appearance and seat washer in its recess in block.

(11) Check the capacitor mounting bracket for cracks or looseness. Using the Bendix 11-1767-1, -2 or -3 Condenser Tester or equivalent, check capacitor for capacitance, series resistance and leakage. Capacitance shall be at least 0.30 microfarads. Series resistance should not be over 1 ohm at 500 kHz.

(12) Inspect coil leads for damaged insulation and terminals for tightness and soldered connection.

(13) Inspect impulse coupling parts for excessive wear. Particularly check clearance between cam and flyweight of the cam assembly. Measure the clearance between the cam flyweight using the shank of a new No. 18 drill (0.1695 inch [4.3053 mm] diameter). If the drill will fit between cam and flyweight as shown in “Figure 4”, the cam assembly must be replaced. Check clearance between both flyweight and the cam of each cam assembly.



Impulse Coupling
Figure 4

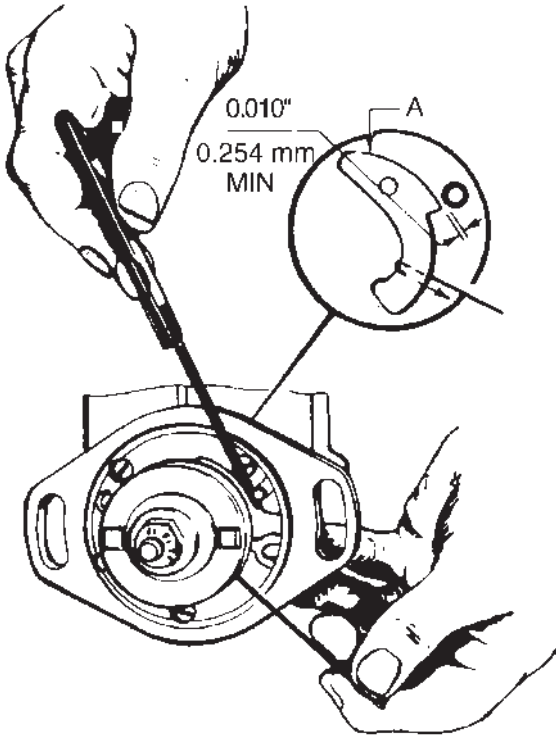
(14) Check the clearance between each flyweight and each stop in as follows:

(a) Bend the end of a stiff piece of wire into a right angle 0.125 inch (3.175 mm) long (maximum).

(b) Hold magneto as shown in Figure 5. Pull heel of flyweight outward with the hooked wire and make certain that feeler gauge of 0.010 inch (0.254 mm) minimum thickness will pass between stop in and the highest point of the flyweight.

NOTE: A true and accurate check of the clearance between flyweight and stop pin can be obtained only by pulling the flyweight outward as described above. Do not attempt the check by pushing in on flyweight as point “A”.

(15) Check internal timing and reinstall and time magneto to engine.



Flyweight Clearance of Impulse Coupling
Figure 5

B. Removal

(1) Remove the side access panel from the engine nacelle.

(2) Disconnect the “P” lead from the magneto.

(3) Remove the harness outlet plate from the magneto by removing the four attaching screws.

(4) Remove the two nuts and washers securing the magneto to the engine accessory housing.

(5) Pull the magneto from the engine. Caution must be used to ensure that magneto drive rubbers do not fall into accessory sections.

C. Installation and Timing Procedure (Magneto to Engine)

LTSIO is Counter Rotating.

NOTE: Time magneto to engine per the following instructions or as described in Continental Motors MSB94-8D or later.

NOTE: Verify correct engine timing for the airplane being worked on by checking the engine dataplate.

(1) Equipment Required

(a) Eastern Technology Corporation Model E25 Timing Indicator (or equivalent).

(b) Top Dead Center (TDC) Locator.

(c) Eastern Technology Corporation Model E50 Timing Light (or equivalent).

(2) Procedure

WARNING: TURN THE IGNITION SWITCH OFF AND DISCONNECT ENGINE ELECTRICAL POWERBEFORE COMMENCING MAINTENANCE OR INSPECTIONS. CONFIRM CONTINUITY BETWEEN THE MAGNETO CAPACITOR AND AIRCRAFT GROUND TO PREVENTACCIDENTAL ENGINE START DURING MAINTENANCE. DO NOT STAND OR PLACEEQUIPMENT WITHIN THE ARC OF THE PROPELLER. ATTACH A “HOT MAGNETO - DO NOT TURN” SIGN TO THE PROPELLER.

CAUTION: INCORRECT TIMING, IN ADDITION TO A ROUGH RUNNING ENGINE, CAN LEAD TO DETONATION, PRE-IGNITION, POSSIBLE INTERNAL ENGINE DAMAGE, OR ENGINE FAILURE. GRAVITY AFFECTS THE TIMING INDICATOR PENDULUM POSITION. PRIOR TO CHECKING OR ADJUSTING ENGINE TIMING, LEVEL THE AIRCRAFT (ENGINE).

NOTE: L/TSIO-360 engines are equipped with a right angle starter adapter that restricts reverse engine rotation. Remove the starter motor to complete magneto to engine timing.

(a) Place the Fuel Selector Valve in the OFF position.

(b) Turn the ignition switch to the OFF position.

(c) See 24-30-00 to gain access to the aircraft main battery.

(d) In airplanes equipped with Garmin G1000/G1000 NXi, disconnect the emergency battery P615 connector.

(e) Disconnect negative (ground) cable from the main battery.

(f) Remove top spark plugs from each cylinder.

(g) Find the No.1 cylinder compression stroke by placing a finger over the spark plug hole and rotating the crankshaft. When the cylinder is on the compression stroke, the valves will be closed and pressure will build as the piston moves to the top of the cylinder. Pressure buildup in the cylinder can be felt at your fingertip.

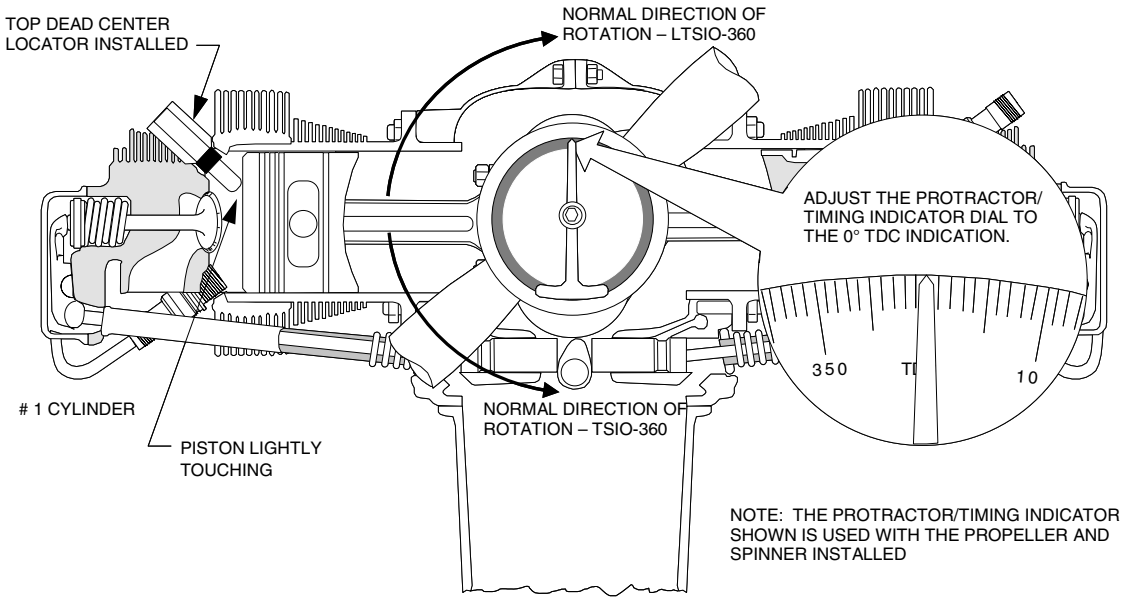
(h) Install the TDC locator (“Figure 6”) in the No. 1 cylinder spark plug hole.

NOTE: Ensure the timing disk is securely fastened to the propeller to prevent movement during engine timing.

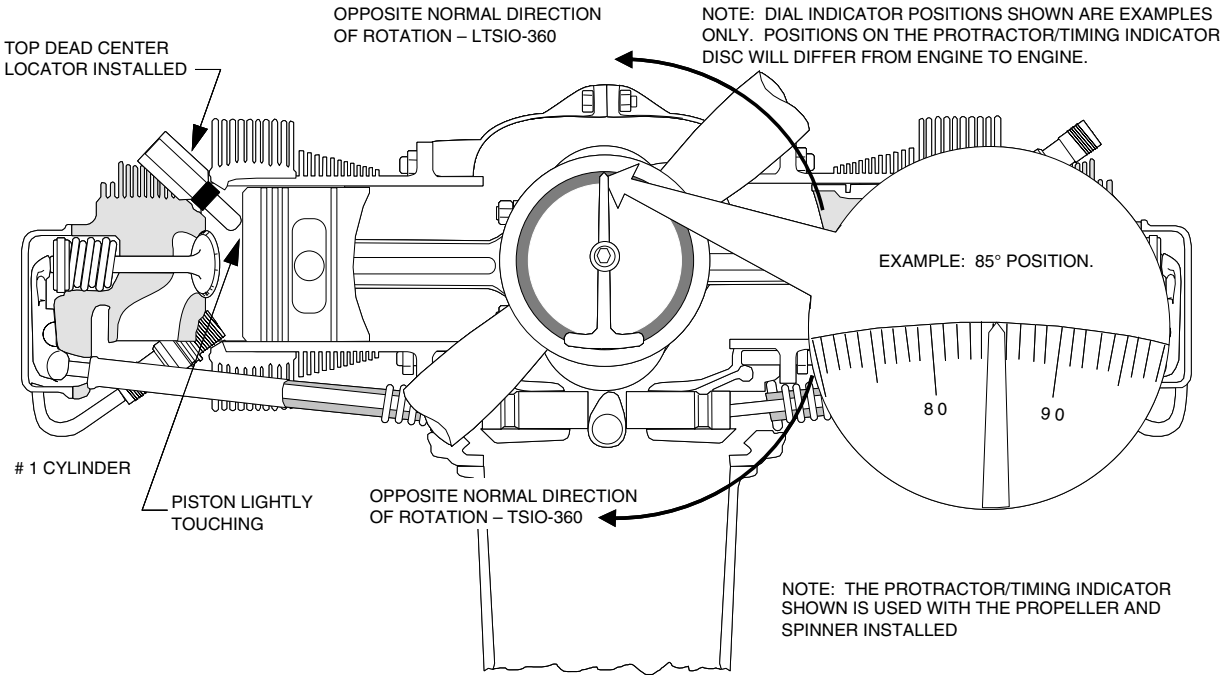
(i) Securely install the timing disk indicator on the crankshaft flange, propeller spinner or propeller hub using the supplied elastic bands.

(j) Turn propeller slowly in the direction of normal rotation until the piston lightly, but positively, touches the TDC locator.

(k) Rotate the timing disc to align the Top Dead Center (TDC) mark on the protractor aligns with the weighted pendulum pointer.



Timing Disk
Figure 6



No. 1 Cylinder Positioned at Top of Intake Stroke
Figure 7

(l) Slowly rotate the crankshaft in the opposite direction of normal rotation until the piston again lightly touches the top dead center locator (“Figure 7” on page 74107).

(m) Determine the difference between TDC and the current pointer indication (“Figure 7” on page 74107) and divide by two; turn the timing disk to align the pointer with the resulting number (“Figure 8”), without turning the timing indicator base.

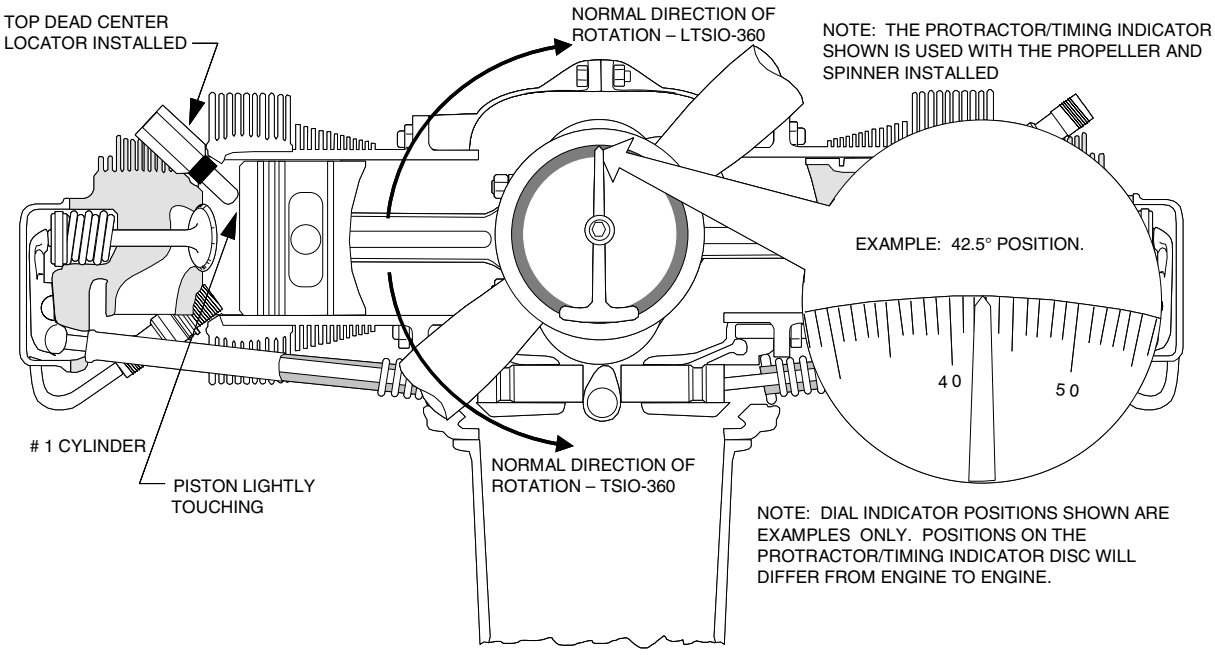
(n) Rotate the crankshaft in the direction of normal rotation on the compression stroke until the pointer aligns with the 0° mark. On magnetos equipped with impulse couplings, continue turning the crankshaft in the direction of normal rotation until each impulse coupling trips. Couplings may trip a few degrees on either side of TDC. If one or both couplings trip after TDC, rotate the crankshaft opposite of the direction of normal rotation a few degrees before TDC, then back in the direction of normal rotation toward TDC.

(o) The crankshaft is now positioned at the top dead center of the No. 1 cylinder compression stroke.

(p) Remove the Top Dead Center locator from the No. 1 Cylinder (“Figure 9”).

(q) Turn the crankshaft in the opposite direction of normal rotation past the specified magneto timing setting to remove gear backlash. Turn the crankshaft in the direction of normal rotation until the pointer aligns with the 20° (Seneca IV) / 22° (Seneca V) BTC mark on the timing disk. The engine is now prepared for magneto removal, installation or magneto to engine timing.

NOTE: Be familiar with the test equipment function prior to use. Some timing lights indicate proper ignition timing by illuminating a lamp; other models illuminate a lamp until the magneto cam lobe opens the circuit and turns the lamp off while other timing indicators use an audible signal to indicate timing status.



Crankshaft Positioned at TDC
Figure 8

(r) Disconnect the P-leads from the magnetos. Connect the timing light right lead to the right magneto ground terminal and the left lead to the left magneto ground terminal. Connect the timing light ground lead to a suitable ground on the engine such as an unpainted bolt or stud.

NOTE: Engines equipped with impulse coupling magnetos must be turned in the direction of normal rotation past the impulse coupling trip point prior to centering the pointer at engine timing mark.

(s) With the crankshaft in the correct timing position, turn the crankshaft a few degrees opposite of the direction of normal rotation before 20° (Seneca IV) / 22° (Seneca V) BTC to clear valve backlash. Turn the crankshaft in the direction of normal rotation toward the point of ignition and observe the timing light as the needle approaches 20° (Seneca IV) / 22° (Seneca V) BTC. As the pointer aligns with the correct timing position, the timing light should indicate proper timing. Lightly tap the end of the pointer with a finger to verify the pointer position.

CAUTION: DO NOT EXCEED THE SPECIFIED MAGNETO TO FLANGE TORQUE. EXCESSIVE TORQUE WILL CRACK THE MAGNETO MOUNTING FLANGE.

(t) If the timing light fails to indicate proper engine timing, loosen the nuts securing the magneto to the mounting flange; rotate the magneto to the left or right in the mount until the lamp illuminates. With the timing light illuminated, torque the magneto fasteners to 100–120 in. lbs. If magneto timing adjustment is not possible, remove the suspect magneto and repair according to the magneto manufacturer's service instructions.

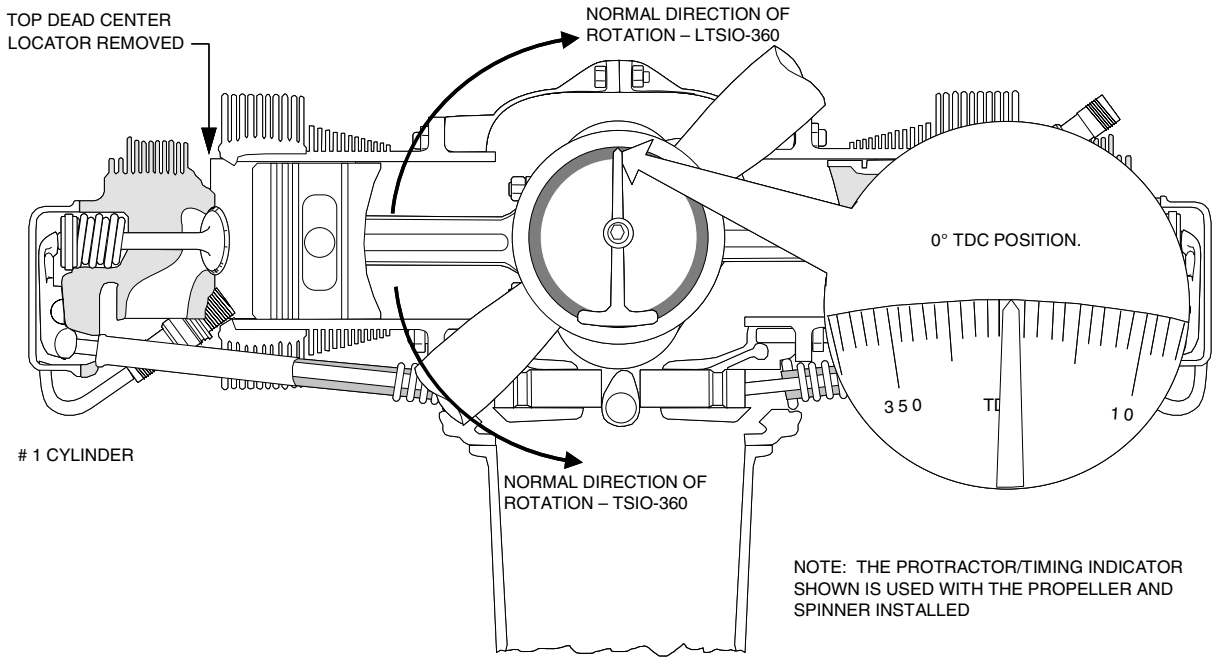
(u) Remove the protractor and timing disk from the propeller flange, spinner or propeller hub. Reconnect the ground switch leads to the magneto.

(v) Inspect the spark plugs according to 74-20-00.

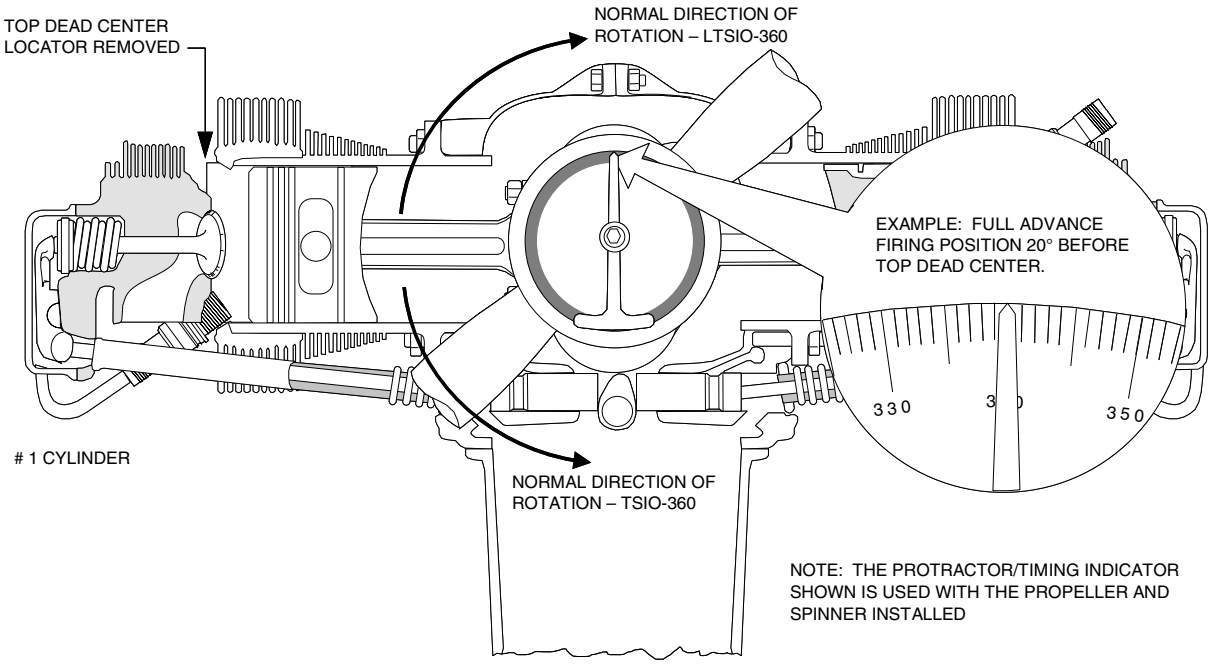
(w) Inspect the ignition harness according to 74-20-00.

(x) In airplanes equipped with Garmin G1000/G1000 NXi, reconnect the emergency battery P615 connector.

(y) Connect the aircraft battery according to the aircraft manufacturer's instructions.



Top of No. 1 Cylinder Compression Stroke
Figure 9



No. 1 Cylinder Full Advance Firing Position
Figure 10

DISTRIBUTION

1. Harness Assembly

A. Inspection

(1) Check the lead assemblies for nicks, cuts, mutilated braiding, badly worn section or any other evidence of physical damage. Inspect the spark plug sleeves for chafing or tears and damaged or stripped threads on coupling nuts. Check the compression spring to see if it is broken or distorted. Inspect the grommet for tears. Check all the mounting brackets and clamps to see that they are secure and not cracked.

(2) Should a harness problem be suspected, integrity of the harness wiring may be checked using an ohmmeter, buzzer, or other suitable device such as the Bendix/ECD High Tension Lead Tester Kits, P/N 11-8950 or 11-8950-1; check each lead for continuity. If continuity does not exist, harness wire is broken and must be replaced.

(3) If an insulation failure is suspected, the condition of the insulation may be determined using the Bendix 11-8950 and the 11-8950-1 High Tension Lead Tester Kits manufactured by the Electrical Components Division, The Bendix Corporation, Sidney, New York.

(4) Test Unit Preparation

(a) Install two “C” cells in the battery holder in accordance with correct position.

(b) Check that red and black leads are open-circuited.

(c) Depress PRESS-TO-TEST push-button switch.

(d) Ensure INDICATOR lamp flashes and GAP fires intermittently as long as PRESS-TO-TEST switch is depressed.

(e) Interconnect both red and black high voltage leads and again depress PRESS-TO-TEST switch. INDICATOR lamp only should flash. GAP does not fire.

(f) Disconnect black and red leads.

(5) Insulation Test

(a) Attach clip of red high voltage test lead to ignition harness lead terminal.

(b) Attach black test lead clip to lead ferrule.

(c) Depress PRESS-TO-TEST push-button switch.

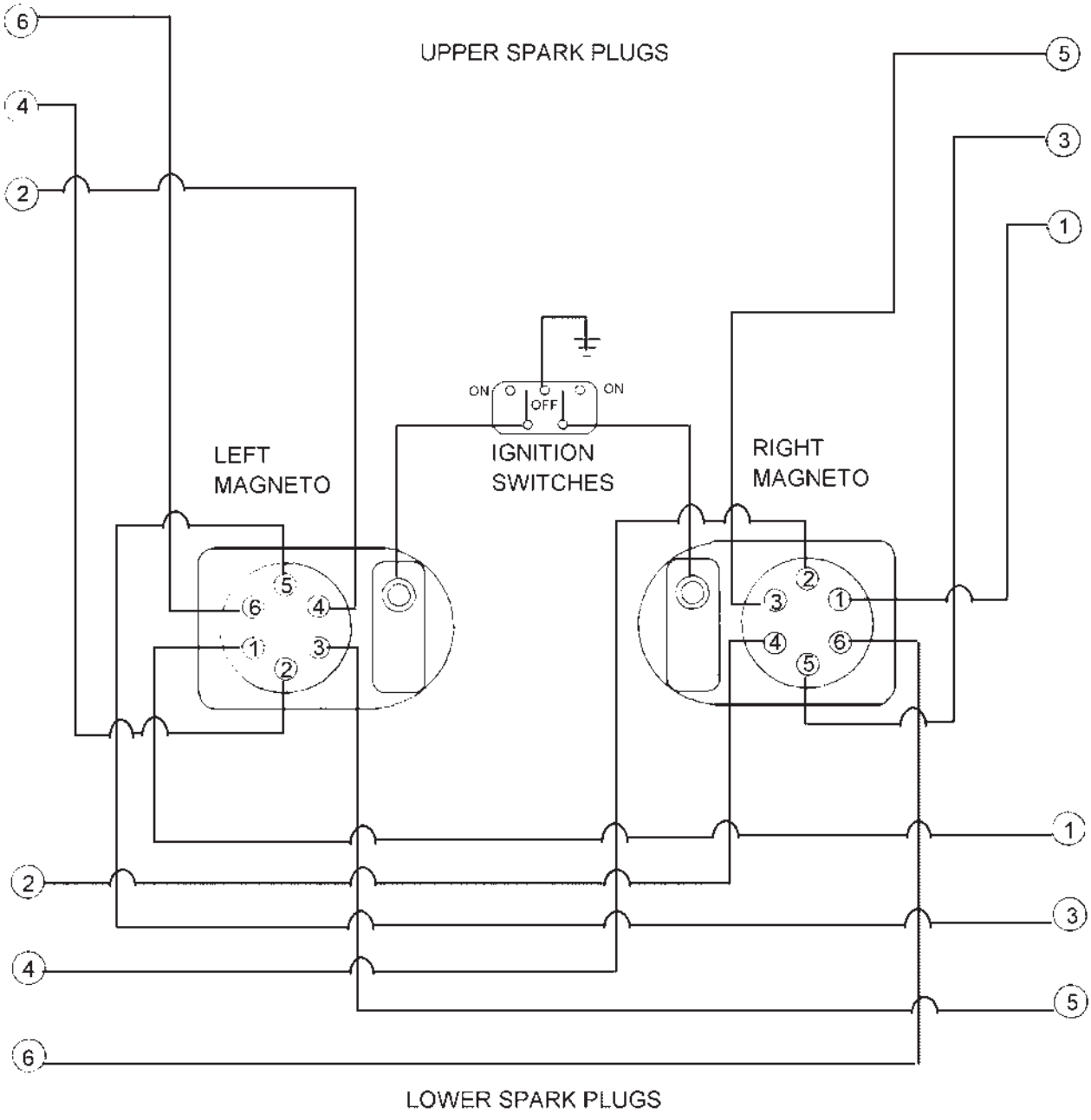
(d) Observe that INDICATOR lamp flashes and GAP fires intermittently as long as PRESS-TO-TEST switch is held depressed.

(e) Whenever INDICATOR lamp flashes and gap fails to fire, lead under test is defective.

(f) When testing leads which are installed on an engine, it may be found that distributed capacitance causes the tester to reject good leads if the tester and red test lead are allowed to lay in close physical contact with the engine parts. For best results, keep the tester and the red high voltage lead well clear of the grounded metal parts of the engine.

(g) On some engines, leakage through the magneto distributor to the magneto coil may occur if the distributor finger electrode is lined up with the lead under test. If this occurs, the tester will indicate a rejection. Before final rejection of a lead which has one end connected to the magneto, turn the engine slightly and repeat test to confirm the reading.

(6) A second acceptable method for performing an insulation check is with a high voltage, direct current tester such as the TAKK Model 86 or 86A or an equivalent direct current tester capable of delivering a test potential of 10,000 volts. Connect ground lead of high voltage tester to outer shielding braid of a single lead. Connect plug terminal. Turn tester ON and apply 10,000 volts. The insulation resistance should be 100 megohms minimum. Proceed to check other leads of harness in the same manner.



ENGINE FIRING ORDER:
LTSIO - 360 - KB - 1-4-5-2-3-6
TSIO - 360 - KB - 1-6-3-2-5-4
LTSIO - 360 - RB - 1-4-5-2-3-6
TSIO - 360 - RB - 1-6-3-2-5-4

MAGNETO FIRING ORDER
1-2-3-4-5-6

Ignition Schematic
Figure 1

B. Removal

(1) Disconnect the clamps that secure the wires to the engine and accessories.

(2) Loosen the coupling nuts at the spark plugs and remove the insulators from the spark plug barrel well. Use caution when withdrawing the insulator so that the insulator spring will not be damaged.

(3) Place a guard over the harness insulators.

(4) Remove the harness assembly terminal plate from the magneto.

(5) Remove the harness from the airplane.

C. Installation

Before installing the harness plate on the magneto, check the mating surfaces for cleanliness. Spray the entire face of the grommet with a light coat of Plastic Mold Spray, SMOOTH Silicone Spray or equivalent. This will prevent the harness grommet from sticking to the magneto distributor block.

(1) Place the harness terminal plate on the magneto and tighten the nuts around the plate alternately to seat the cover squarely on the magneto.Torque the nuts to 18 to 22 inch-pounds.

(2) Route the ignition wires to their respective cylinders as shown in “Figure 1” on page 74202.

(3) Clamp the harness assembly in position.

(4) Connect the leads to the spark plugs.

D. Maintenance

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT,MAYRENDERTHEAIRCRAFTUNAIRWORTHY.(SEEINTRODUCTION-SUPPLEMENTARY PUBLICATIONS.)

2. Spark Plugs

Spark plugs must be removed, cleaned, inspected, gapped and position rotated every 100 hours.

A. Removal

(1) Loosen the coupling nut on the harness lead and remove the terminal insulator from the spark plug barrel well. (A crows foot adapter is needed to remove the lower spark plugs.)

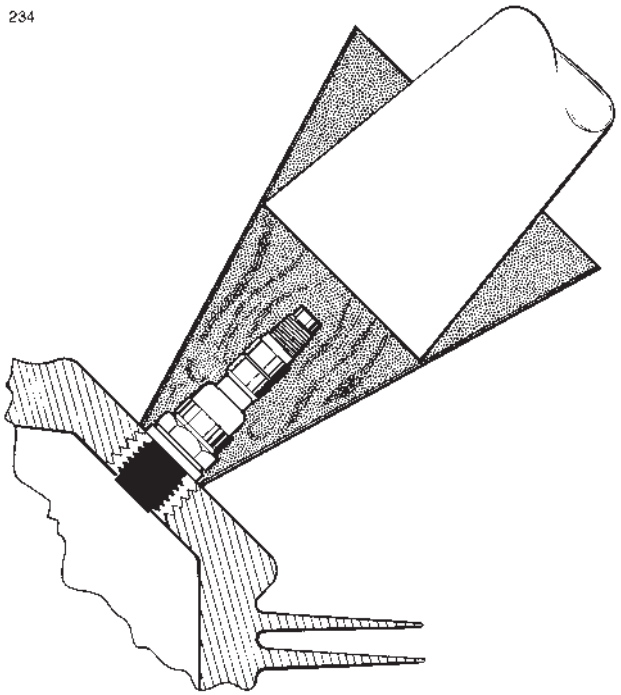
NOTE: When withdrawing the ignition cable lead connection from the plug, care must be taken to pull the lead straight out and in line with the centerline of the plug barrel; otherwise, a side load will be applied which frequently results in damage to the barrel insulator and connector. If the lead cannot be removed easily in this manner, the resisting contact between the neoprene collar and the barrel insulator will be broken by a rotary twisting of the collar. Avoid undue distortion of the collar and possible side loading of the barrel insulator.

(2) Remove the spark plug from the engine. In the course of engine operation, carbon and other combustion products will be deposited on the end of the spark plug and will penetrate the lower threads to some degree. As a result, greater torque is frequently required for removing a plug than for its installation. Accordingly, the torque limitations given do not apply to plug removal, and sufficient torque must be used to unscrew the plug. The higher torque in removal is not as detrimental as in installation, since it cannot stretch the threaded section. It does, however, impose a shearing load on this section and may, if sufficiently severe, produce a failure in this location.

(3) Place spark plugs in a tray that will identify their position in the engine as soon as they are removed.

(4) Removal of seized spark plugs in the cylinder may be accomplished by application of liquid carbon dioxide by a conical metal funnel adapter with a hole at the apex just large enough to accommodate the funnel of a CO_2 bottle (“Figure 2”). When a seized spark plug cannot be removed by normal means, the funnel adapter is placed over and around the spark plug. Place the funnel of the CO_2 bottle inside the funnel adapter and release the carbon dioxide to chill and contract the spark plug. Break the spark plug loose with a wrench. A warm cylinder head at the time the carbon dioxide is applied will aid in the removal of an excessively seized plug.

(5) Do not allow foreign objects to enter the spark plug hole.



Removing Frozen Spark Plug
Figure 2

B. Inspection and Cleaning

Good spark plug maintenance is necessary for engines to operate efficiently. Plugs should be as clean as possible; mechanically sound; exhibit enough electrode for additional use; exhibit properly gapped and contoured electrodes; and, pass necessary tests for electrical soundness. For further information not included herein, contact the spark plug manufacturer for further information.

(1) Visually inspect each spark plug for the following non-repairable defects:
   (a) Terminal barrel sleeve cracked.
   (b) Badly battered or rounded shell hexagons.
   (c) Threads at top of shielding badly nicked or corroded.
   (d) Connector seat badly nicked or corroded.
   (e) Chipped, cracked, or broken ceramic insulators in the firing end or shielding barrel.
   (f) Badly eroded or disfigured electrodes. See “Figure 3” on page 74206.

(2) Clean the spark plug as follows:

NOTE: Do not use carbon tetrachloride to degrease spark plugs except if using vapor degrease method. Do not soak plugs in solvent and ensure solvent does not enter shielding barrel.

   (a) Degrease the spark plugs as required. Refer to manufacturer’s information for specific requirements.
   (b) Dry plugs using clean, dry air. Heating in a small oven is often recommended to thoroughly remove all traces of solvent.

NOTE: If any solvent or oil remains in the firing end, or connector well of the spark plug, abrasive will pack between the shell and insulator, if abrasive blasting is used.

   (c) Clean the firing end terminal well. There are different machines (vibrators, abraisive blasters, etc.) and methods capable of accomplishing this task. Contact the manufacturer if necessary to determine the appropriate procedures and materials. To prevent flashover, all foreign matter, moisture, and stains should be removed from the terminal well insulator..
   (d) Set electrode gaps as specified by the spark plug manufacturer’s specification.

C. Installation

WARNING: DO NOT INSTALL A SPARK PLUG THAT HAS BEEN DROPPED. OR EXHIBITS MECHANICAL DAMAGE

CAUTION: FAILURE TO INSTALL A NEW SPARK PLUG GASKET ON EACH SPARK PLUG MAY RESULT IN INCOMPLETE SEALING OF THE COMBUSTION CHAMBER, LOSS OF SPARK PLUG HEAT TRANSFER, SPARK PLUG OVERHEATING, POSSIBLE PRE-IGNITION / DETONATION AND INTERNAL ENGINE DAMAGE.

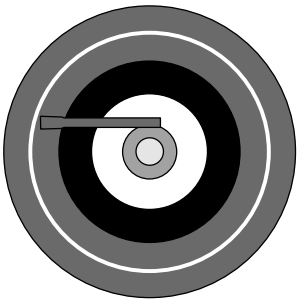
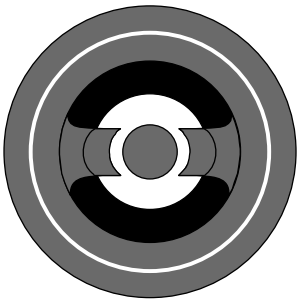
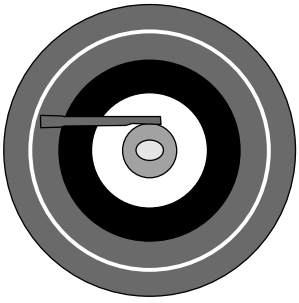
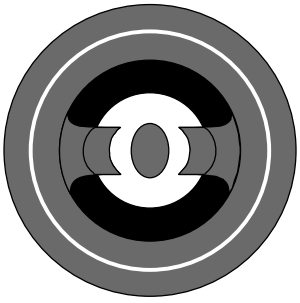
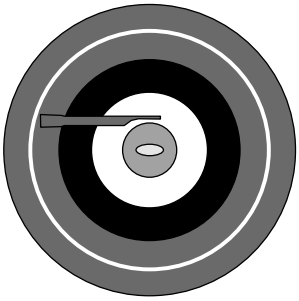
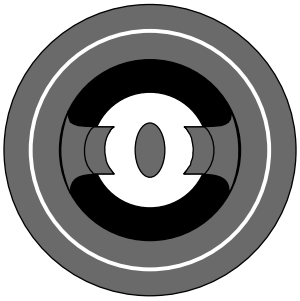
Spark plugs must be rotated every 100 hours. See “Figure 4” on page 74207 for spark plug rotation sequence. Always install a new spark plug gasket when servicing spark plugs or installing new spark plugs.

Install spark plugs as follows, in addition to instructions provided by spark plug manufacturer. Before installing spark plugs, ascertain that the threads within the cylinder are clean and not damaged.

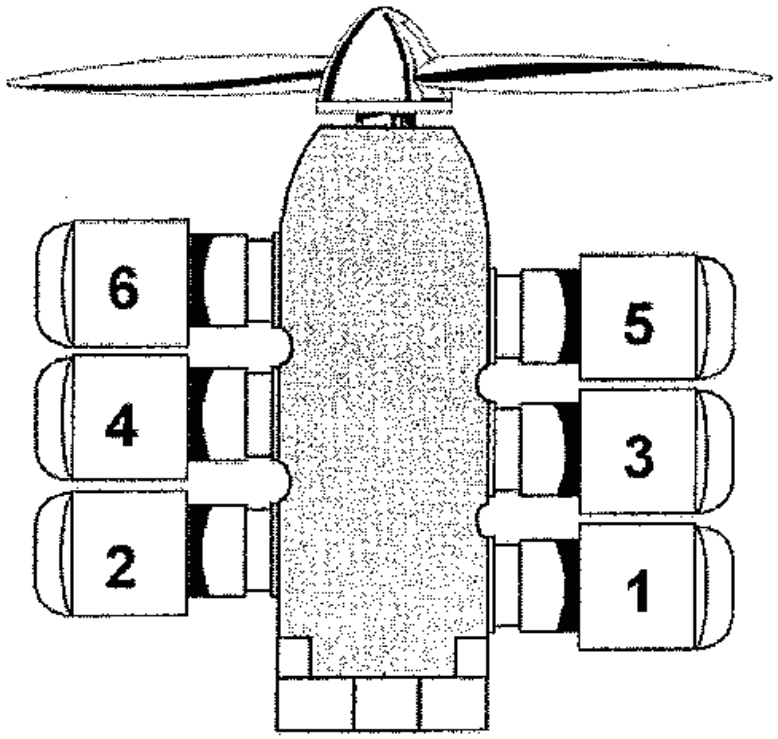
(1) Handle spark plug leads with clean dry hands.
(2) Adjust spark plug electrode to gap recommended by the manufacturer.
(3) Always install spark plugs with new copper gaskets.

CAUTION: DO NOT APPLY ANTI-SEIZE TO FIRST ROW OF THREADS.

(4) Sparingly apply anti-seize to spark plug threads.

	FINE WIRE ELECTRODE	MASSIVE ELECTRODE
NORMAL ELECTRODE CONDITION •INSULATOR TIP GRAY, TAN OR LIGHT BROWN •FEW COMBUSTION DEPOSITS •ELECTRODES NOT BURNED OR ERODED •PROPER TYPE AND HEAT RANGE PLUG FOR ENGINE AND SERVICE. •SPARK PLUG SHOULD BE CLEANED, REGAPPED AND TESTED BEFORE REINSTALLTION		
NORMAL WORN-OUT CONDITION •ELECTRODES ERODED BY HIGH VOLTAGE SPARKING AND BY CORROSIVE GASES FORMED DURING COMBUSTION TO LESS THAN 1/2 ORIGINAL THICKNESS •MORE VOLTAGE NEEDED TO FIRE SPARK PLUGS - OFTEN MORE THAN IGNITION SYSTEM CAN PRODUCE. •REPLACE WITH NEW APPROVED AVIATION SPARK PLUGS.		
SEVERE WORN-OUT CONDITION •EXCESSIVELY ERODED CENTER AND GROUND ELECTRODES PLUS EXTENSIVE NECKING OF FINE WIRE GROUND ELECTRODES INDICATE ABNORMAL ENGINE POWER OR PLUGS LONG OVERDUE FOR REPLACEMENT. •DISCARD SPARK PLUGS AND CHECK HEAT RANGE BEFORE INSTALLING NEW ONES. •REPLACE WITH NEW APPROVED AVIATION SPARK PLUGS IN APPROPRIATE HEAT RANGE.		

Spark Plug Inspection Criteria
Figure 3



FROM CYLINDER NUMBER	TO CYLINDER NUMBER
1 TOP	6 BOTTOM
1 BOTTOM	6 TOP
2 TOP	5 BOTTOM
2 BOTTOM	5 TOP
3 TOP	4 BOTTOM
3 BOTTOM	4 TOP
4 TOP	3 BOTTOM
4 BOTTOM	3 TOP
5 TOP	2 BOTTOM
5 BOTTOM	2 TOP
6 TOP	1 BOTTOM
6 BOTTOM	1 TOP

(VIEW LOOKING DOWN)

Spark Plug Rotation Sequence
Figure 4

CHART 1
SPARK PLUG B-NUT TORQUE

Thread (In.)	Wrench (In.)	Torque
5/8 - 24	3/4	90–95 in·lbs
3/4 - 20	7/8	110–120 in·lbs

(5) Always use a six point deep well socket to install spark plugs.

CAUTION: MAKE CERTAIN THE DEEP SOCKET IS PROPERLY SEATED ON THE SPARK PLUG HEXAGON. THE PLUG COULD BE DAMAGED IF THE WRENCH IS OFF CENTER WHEN PRESSURE IS APPLIED.

(6) Thread spark plug by hand into cylinder to within one or two threads of the gasket. if this is not possible, cylinder head threads or spark plug threads need to be cleaned.

(7) Torque spark plugs to 300–360 in·lbs using calibrated torque wrench.

(8) Before installing the spark plug lead terminal, wipe it clean using a lint free cloth moistened with isopropyl alcohol. Once dry apply MIller-Stephenson MS-122AD dry lubricant to the terminal insulator.

(9) Ensure inside of spark plug barrel is clean and dry.

(10) Insert spark plug lead connector into spark plug barrel with fingers in a straight line with the spark plug.

(11) Screw the B-nut into place until finger tight.

(12) Tighten the B-nut to the torque specified in “Chart 1” while holding the spark plug lead ferrule stationary.

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CHAPTER

76

ENGINE CONTROLS

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CHAPTER 76

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CHAPTER 76 - ENGINE CONTROLS

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POWER CONTROL

This chapter contains information on rigging the power plant controls; throttle, propeller, and mixture.

CAUTION: ALIGNMENT OF THE CORRESPONDING ENGINE CONTROLS MUST BE WITHIN 0.25 IN. (6.350 MM) WHEN THEY ARE IN THEIR AFT POSITIONS AND HALF A KNOB WIDTH IN THEIR FULL FORWARD POSITIONS.

CAUTION: ENSURE ALL COTTER PINS AND/OR SAFETY WIRE ARE REPLACED.

1. Seneca IV

A. Rigging

Refer to “Figure 1” on page 76103.

(1) Throttle Controls

(a) Move the affected throttle lever to its full throttle position (full forward).

(b) At the appropriate engine, remove the upper left cowl panel and disconnect the cable from the throttle valve arm.

(c) Move the throttle valve arm to its full open (throttle) position. Adjust the rod ends of the cable to provide the throttle lever with a 0.03 to 0.06 inch (0.762 to 1.524 mm) clearance or cushion between the quadrant stop and throttle lever, while leaving a clearance of at least 0.03 inch (0.762 mm) at the idle or aft throttle position stop. Check alignment of the corresponding control.

(d) Refer to Chapter 32 and check the rigging of the “Gear Warning Throttle Switch.”

(e) Install cowl panel.

(2) Propeller Controls

(a) If not already done so, move the particular propeller control to full increase (full forward).

(b) At the affected engine, remove the upper left cowl panel and disconnect the control cable from the governor control arm.

(c) Refer to “Figure 1” on page 76103 and check the travel of the control arm. The Hartzell governor must meet the 95° travel requirement.

(d) Move the governor control arm to max speed or full increase rpm and rig cable to provide 0.03 to 0.06 inch (0.762 to 1.524 mm) clearance or cushion between the appropriate prop lever and quadrant stop.

(e) Install cowl panel and ensure alignment of the corresponding control for the other engine.

(3) Mixture Controls

(a) Move the affected mixture control to its full rich (full forward) position.

(b) Remove the upper right cowl panel on the appropriate engine and disconnect the control cable from the mixture control arm on the engine fuel pump.

(c) Move or ensure the arm is to its full rich (full travel) position. Refer to “Figure 1” on page 76103 for arm travel specifications.

(d) Adjust the cable ends to provide the mixture lever with a clearance or cushion of 0.03 to 0.06 inch (0.762 to 1.524 mm) between the lever and quadrant stop when at the full rich position. When the lever is at idle cutoff, there must be a clearance of at least 0.03 (0.762 mm) inch between the lever and aft stop.

(e) Check the alignment of the corresponding mixture lever and install removed panels and cowl panel.

B. Engine Setup Procedures

The following procedures should be used to check and adjust the power plants to maintain the required operating limits and ensure obtaining good setup results. It is important that the following checks be made to both engines before proceeding with any actual system adjustments:

(1) Preparations

(a) Remove the cowl to make access to the fuel injection pump, manifold valve, metering unit, injection nozzles, turbocharger unit, exhaust bypass (waste gate) valve, and overboost relief valve.

(b) Ensure all lines are tight and that there are no indications of leaks. Fuel dye stains around a fuel fitting indicates a leak.

(c) Ensure the overboost relief valve is seated.

(d) Check exhaust and induction systems for tightness and no damage.

(e) Ensure the intake inlets are clear and that the intake filter is clean.

(f) Check the turbocharger for evidence of damage or overheating. If overheat damage is suspect, check that the compressor/turbine are free to turn.

(g) Inspect the ignition harness, cooling baffles, oil lines and fittings (for loss of oil), and other items that could affect engine performance.

(h) Ensure the auxiliary fuel pump system operates properly, refer to Chapter 28.

(2) Leak Check - Gauge Lines

(a) Disconnect the manifold pressure line at the forward part of the throttle assembly; fuel flow vent line from the air throttle adapter fitting; and, the fuel flow pressure line at the manifold valve.

(b) Connect surgical tubing to the fuel flow vent line and evacuate the line until a 10 gallon per hour (maximum) positive indication on the fuel flow gauge is obtained. Clamp off the tubing and observe the gauge for a steady reading. Any change of this reading would indicate a leak in the system, which must be repaired prior to continuing with the setup procedures.

NOTE: A static system test unit can be used to leak check these lines.

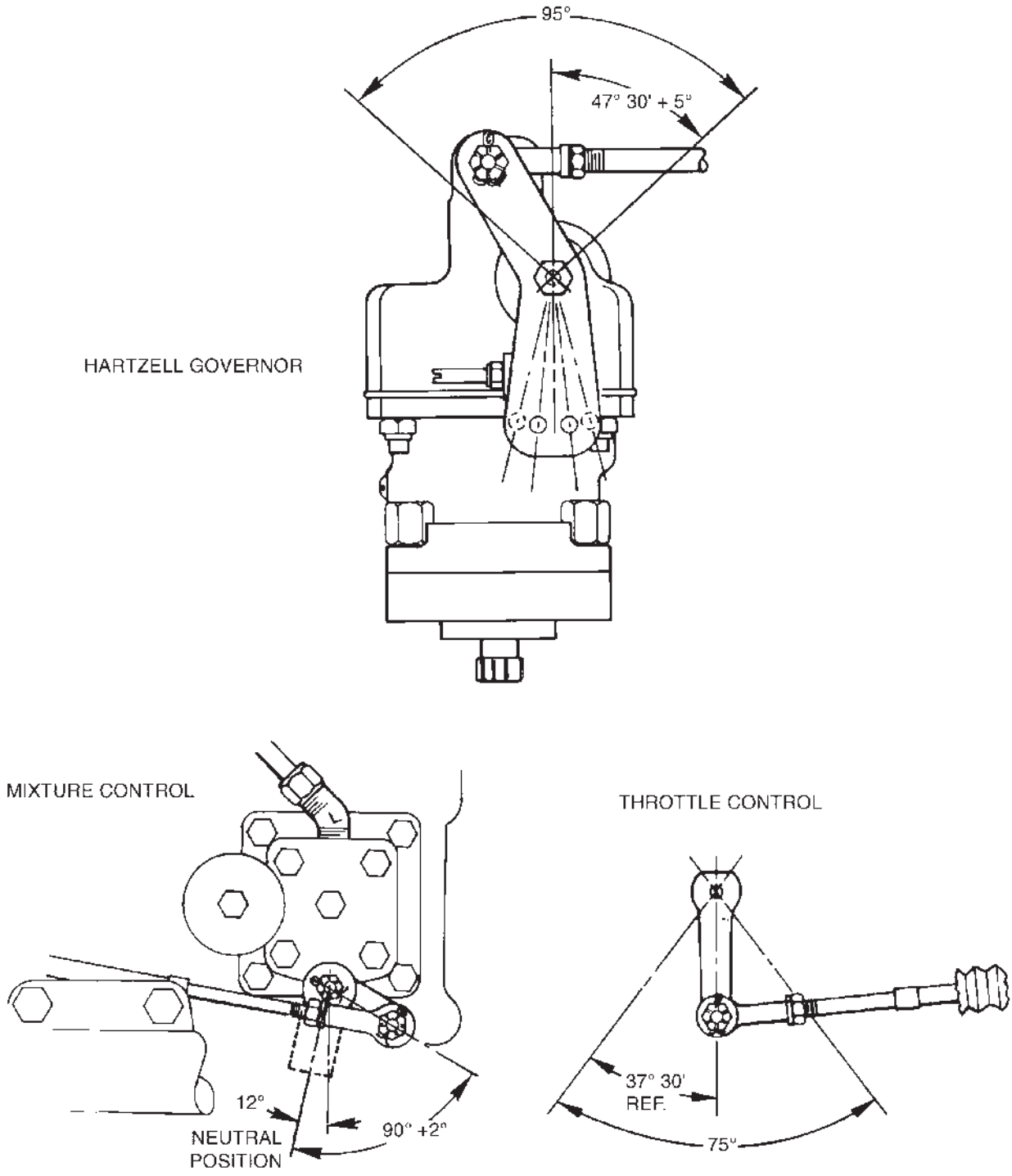
(c) Check the fuel flow pressure and manifold pressure lines in the same manner as given in Step (2) except apply positive pressure to the lines. Do not exceed 4 pounds per square inch (psi) on the fuel pressure gauge.or 4 inches of mercury (in. Hg) increase on the manifold pressure gauge.

(d) Reconnect and tighten the manifold pressure, compressor discharge pressure and fuel pressure lines.

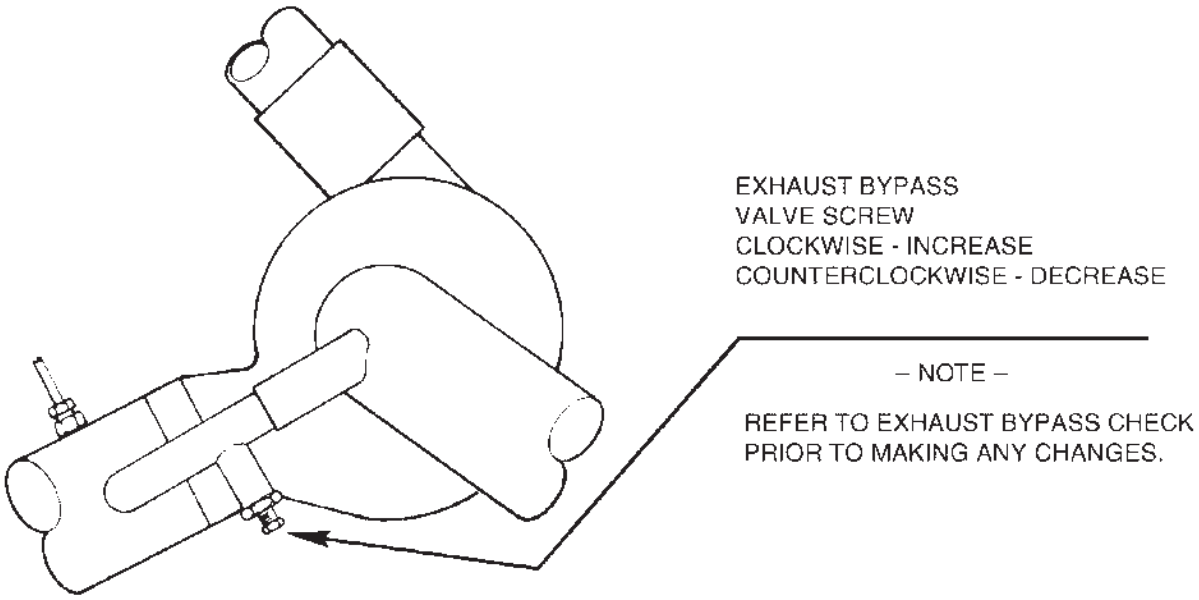
(e) The difference in the static reading on the manifold pressure gauges should not exceed 1/2 in. Hg.

CAUTION: DO NOT RUN PUMPS EXCESSIVELY AS FUEL BEING PUMPED INTO THE CYLINDER WILL WASH THE CYLINDER WALLS.

(f) To reduce the possibility of trapped air in the fuel pressure lines following the test, disconnect the fuel pressure line at the rear of the fuel flow gauge and activate the auxiliary fuel pump long enough to purge the lines; then reconnect the lines.



Engine Controls (Seneca IV)
Figure 1



Exhaust Bypass Valve Screw (Seneca IV)
Figure 2

(3) Exhaust Bypass Check

Refer to “Figure 2”.

CAUTION: DURING ALL ENGINE OPERATIONS OUTLINED IN THESE INSTRUCTIONS, EXERCISE CAUTION TO AVOID HARM OR DAMAGE TO PERSONNEL AND EQUIPMENT BY PROPELLER BLAST AND ROTATING PROPELLER BLADES. WHEN REQUIRED TO MAKE ADJUSTMENTS TO THE ENGINE IN CLOSE PROXIMITY TO THE PROPELLER ARC, SHUT THE ENGINE DOWN BEFORE MAKING ADJUSTMENTS.

The adjusting screw on turbochargers installed on Seneca IV models has course threads. Check that the exhaust bypass adjusting screw consisting of fine threads has from eight to nine threads showing below the jam nut. Bypass adjustment screws having course threads should have three to four threads showing below the jam nut. This screw is preset at the factory and should not require any adjustment, unless it is known that critical altitude is not correct. If adjustment is necessary, use procedure given in Flight Testing.

NOTE: It is extremely important that both engines are thoroughly warmed up, operated and adjusted together to keep them matched. However, excessive engine temperatures must be avoided since setup temperature must closely parallel temperatures in flight.

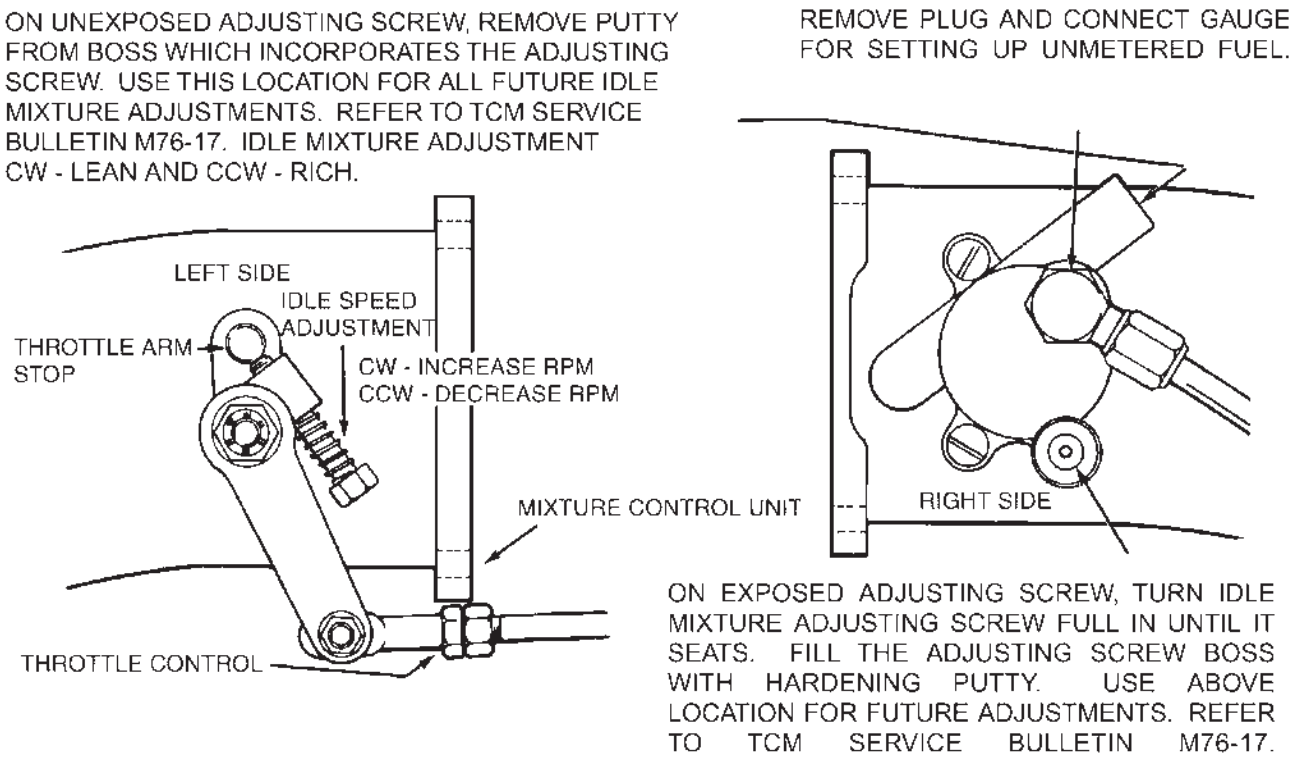
(4) Idle Performance Check

Refer to “Figure 3” on page 76105.

NOTE: Before performing any of these procedures, make sure the auxiliary fuel pump system is operating properly. Refer to Chapter 28.

(a) Remove the cap from the tee fitting on the right side of the throttle body.

(b) Install a 0-60 psig calibrated pressure gauge (vented to the atmosphere) to the tee, using a suitable length of flexible tube. The gauge should always be at the same level as the fuel manifold valve when checking fuel pressure.



Idle Speed and Mixture Adjustment Points (Seneca IV)
Figure 3

(c) Purge the air from the tube.

(d) Perform the following checks and adjustments:

1) Idle Fuel Pressure

NOTE: The following setup procedure is accomplished with the boost pumps OFF. Both engines should be thoroughly warmed up and adjusted together to keep them matched

a) Back off the idle speed adjusting screw two turns. (Refer to “Figure 3”.)

b) Start both engines and warm them up at 1500 to 1800 rpm until the oil pressures are in the green arc, cylinder head temperatures are in the lower one-quarter of the green arc, and the oil temperatures are 160° to 180°F.

c) While maintaining 700 ± 25 rpm, set the idle fuel pressure at 6.5 ± 0.5 psi by adjusting the idle pump adjustment screw (refer to Figure 4, item 6); clockwise adjustment increases pressure; counterclockwise adjustment decreases pressure.

2) Idle Mixture Check and Adjustment (Refer to “Figure 3”.)

a) Operate the engine at 1500 to 1800 rpm until cylinder head temperatures are in the lower one-quarter of the green arc, and the oil temperatures are 160° to 180°F (71° to 82.22°C).

b) Reduce the engine speed and stabilize it at 700 ± 25 rpm.

c) Slowly, but positively, move the mixture control from the full rich position to idle cut-off. The engine speed should increase 75 to 100 rpm before beginning to drop toward zero. Move the mixture control back to full rich before the engine stops.

d) If the engine speed increase is less than 75 rpm, adjust the idle mixture adjustment to enrich the mixture (counterclockwise). If the engine speed increase is more than 100 rpm, adjust the idle mixture to lead the mixture (clockwise).

e) After each adjustment, increase rpm to 1500-1700 for 10 seconds to “clean out” the engine.

f) Double check idle fuel pressure after adjusting idle mixture.

NOTE: Any adjustment of the idle fuel pressure or idle mixture will probably change the reading of each other. Continue to adjust and crosscheck until both are correct.

3) Idle Speed Check and Adjustment (Refer to “Figure 3” on page 76105.)

a) With the idle fuel pressure and idle mixture set in accordance with instructions given previously, cylinder head temperatures in the lower one-quarter of the green arc, and oil temperatures at 160° to 180°F (71° to 82.22°C), set engine speed at 700 ± 25 rpm.

b) Adjust the idle speed adjusting screw until contact is made with the throttle arm stop.

NOTE: After final adjustment, recheck the idle fuel pressure, idle mixture and idle speed to ascertain that all are within specifications given in previous steps.

(5) Full Power Performance Check and Adjustment

Refer to “Figure 4” on page 76107.

CAUTION: BEFORE ATTEMPTING FULL POWER CHECKS, BE SURE THAT THE BRAKES ARE PROPERLY MAINTAINED AND SET, AND THAT GROUND CONDITIONS WILL NOT PERMIT WHEELS TO SLIP DURING FULL POWER CHECK.

NOTE: Fuel flows are given for sea level density altitude. Use Chart 1 to interpolate correct fuel flow for the actual engine rpm. Refer to engine operating manual for further information. Charts 2 and 3 are included for reference only.

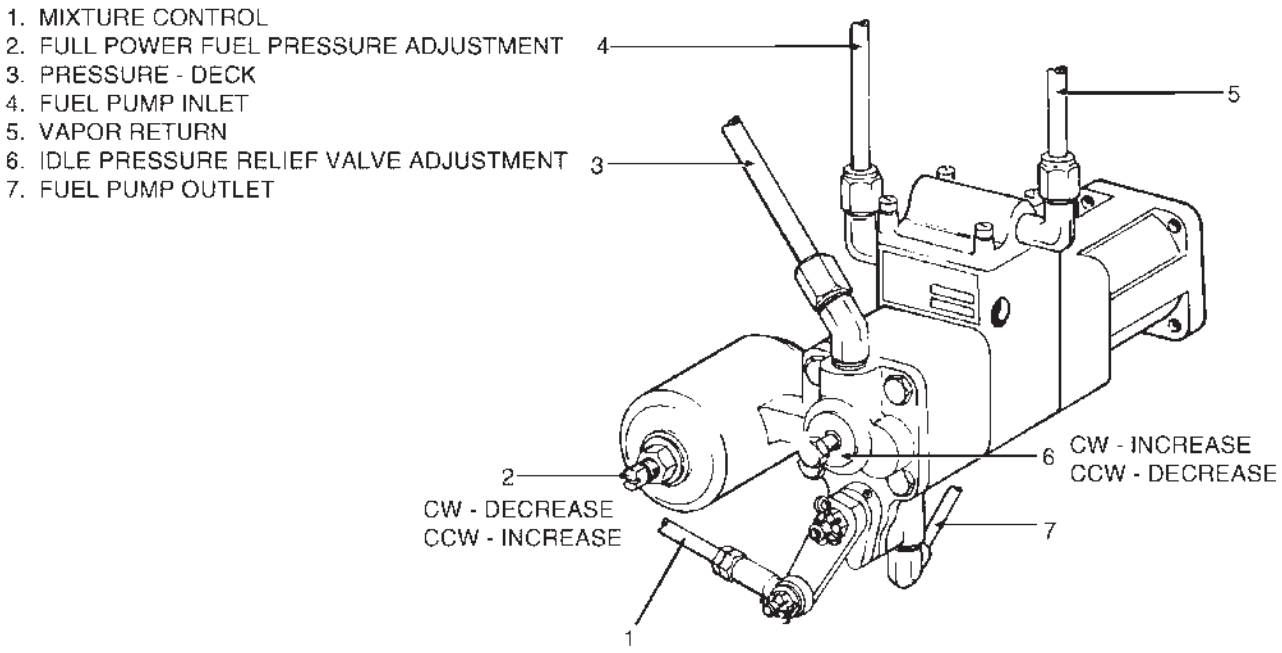
(a) Run both engines at 39.8 to 40.0 in. Hg manifold pressure (overboost lights activated), and beat synchronize the engines at 2600 rpm using the propeller governor controls. Readjust the throttle controls as required to maintain 39.8 to 40.0 in. Hg manifold pressure on both engines.

(b) Fuel flow should be 22.8 to 23.5 gallons per hour (gph), for each engine with the mixture controls in the full rich position.

(c) Observe the 0-60 calibrated gauge to cross check performance. High unmetered pressure should be 36 to 40 psi.

(d) If adjustment is required, shut the engine down, loosen the jam nut on the adjusting screw located on the aneroid housing of the fuel pump. (Refer to “Figure 4” on page 76107, item 2.) Clockwise adjustment decreases fuel flow reading; counterclockwise adjustment increases fuel flow reading. Keep in mind that one full turn will cause a 1.0 to 1.5 gph change. Use CAUTION when loosening and tightening the jam nut so as not to change settings.

NOTE: A complete investigation of interface systems is required if other than minor adjustments are required to the fuel flow.



Sectional View of Altitude Compensating Fuel Pump Assembly (Seneca IV)
Figure 4

(e) Restart the engines and recheck the high end fuel flow.

(f) Recheck the idle settings as directed in the four previous subject paragraphs.

(g) Recheck Full Power Fuel Flow settings.

(h) With engines operating at 2600 rpm (39.8 to 40 in. Hg manifold pressure), lean the mixture to obtain 21 gph fuel flow readings. The unmetered fuel pressure on the calibrated pressure gauge should be 32 to 36 psi.

NOTE: The key to keeping engines and fuel systems matched is beat synchronizing of both engines and operating them together to keep temperatures equal. Adjusting engines singly seldom produces a good match.

(6) Fuel System Match Check

(a) Set propeller governors to maintain 1900 to 2000 rpm and open throttles slowly, increasing engine speed until reaching 40 in. Hg manifold pressure. Keep engine speeds beat synchronized.

(b) Slowly reduce manifold pressures, keeping needles matched and observe fuel flows. A properly adjusted system will track fuel flows within a needle's width of each other.

CHART 1
FUEL FLOW VS. ENGINE SPEED (SENECA IV)

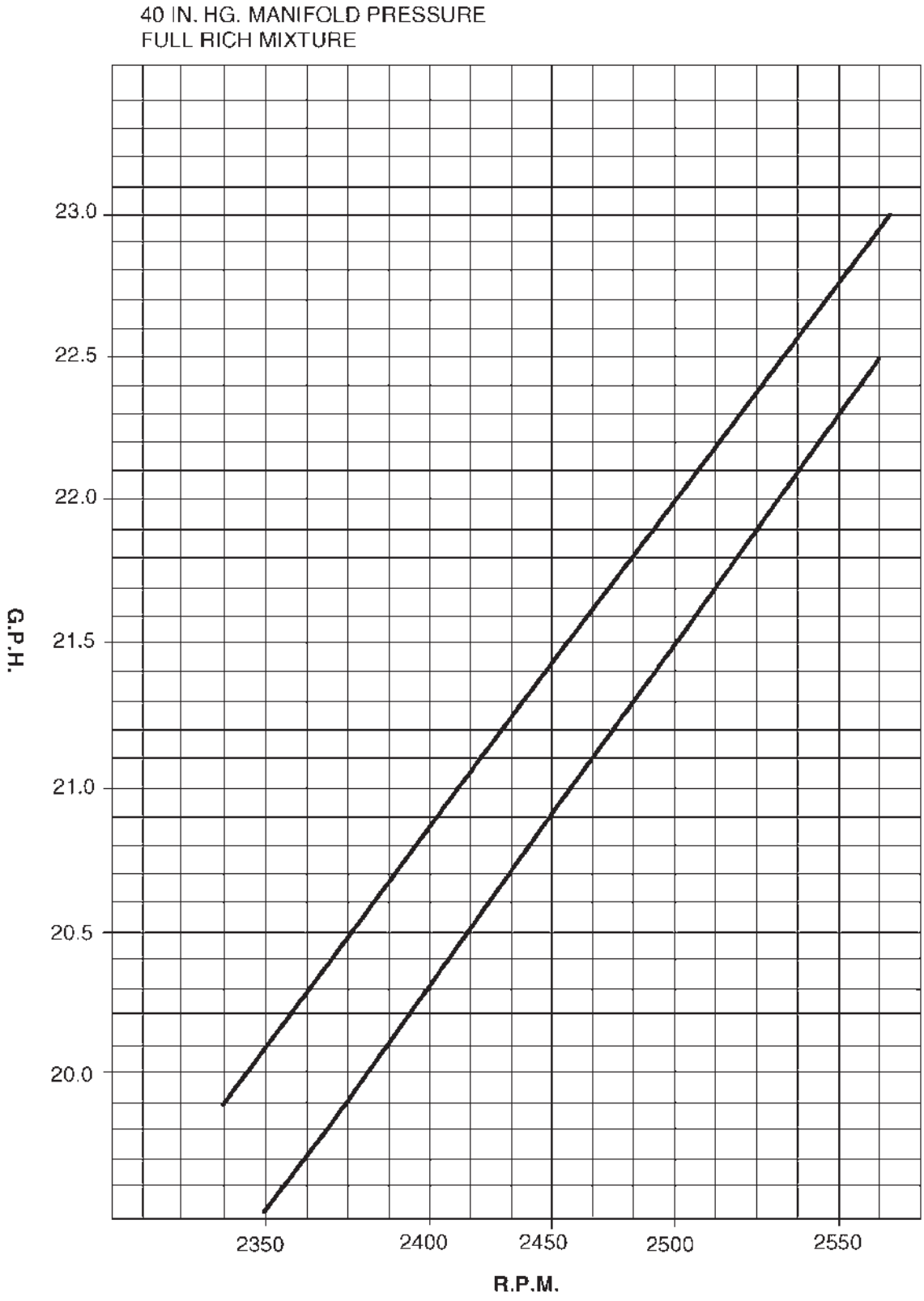


CHART 2 (Reference Only)
FUEL FLOW VS. FUEL PRESSURE (SENECA IV)

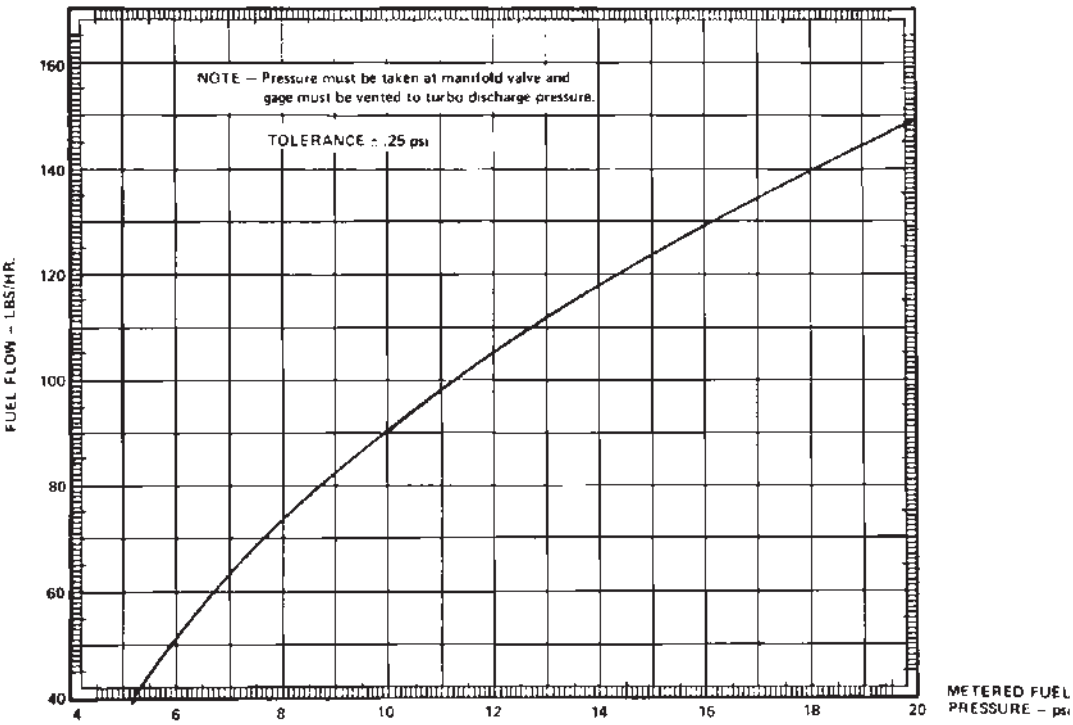
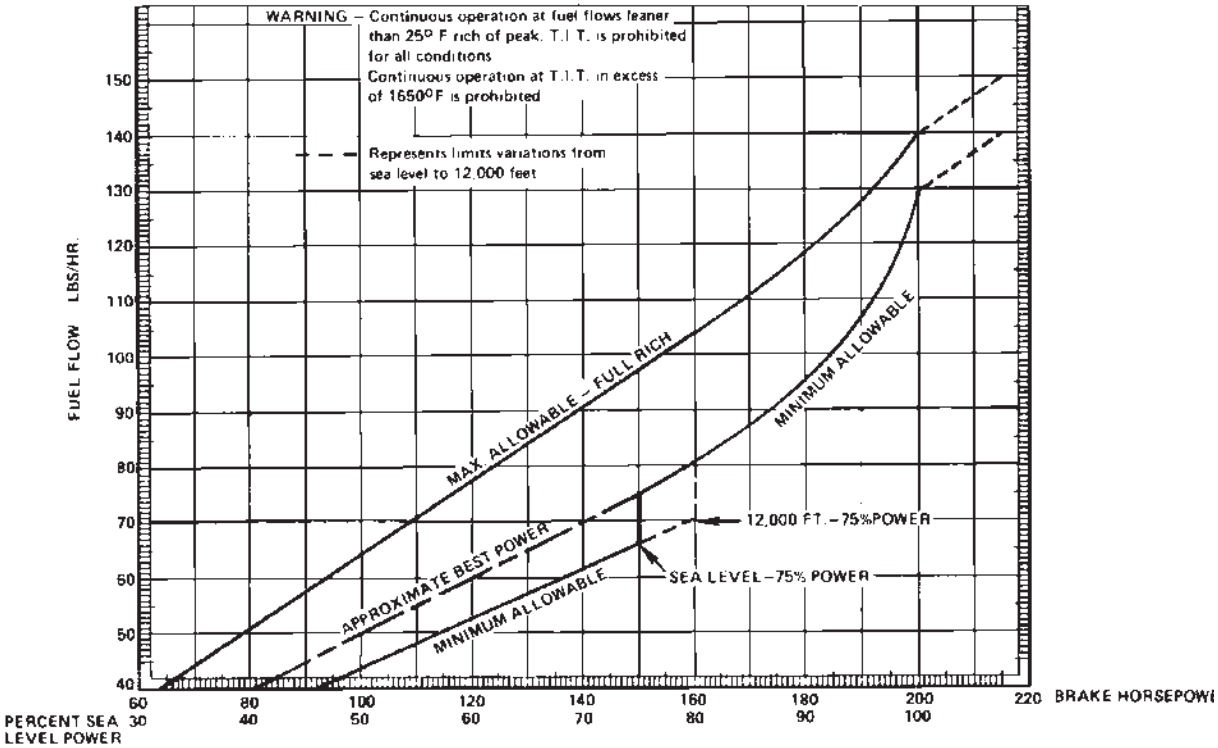


CHART 3 (Reference Only)
LIMITS - FUEL FLOW VS. BRAKE HP (SENECA IV)



(7) Post Setup Procedures

(a) Remove test equipment; safety wire the exhaust bypass screw and check nut to the bypass screw housing; reinstall the cap on the tee of the throttle body housing.

(b) The accuracy of the cockpit fuel flow gauge at maximum power can be checked against a calibrated gauge by connecting the calibrated gauge at the manifold valve and maintaining the gauge on the same level as the valve while checking pressure and using "Chart 1" on page 76108.

NOTE: The calibrated gauge fuel line must be purged of air and the reference side of the calibrated gauge vented to turbo discharge pressure.

C. Flight Test Procedures

(1) At 8,000 feet density altitude, set the engines to operate at 2450 ± 25 rpm and 31.0 to 32.0 in. Hg manifold pressure.

(2) Lean each engine to 25°F rich of peak exhaust gas temperature (EGT). (Peak EGT may not be the same for both engines; however, the difference should not exceed 50°F.)

(3) Fuel flow at these conditions should be 11.0 to 12.0 gph.

(4) Place the aircraft in a climb attitude at 92 KIAS with the mixture at full rich, cowl flaps half open, full throttle (2600 ± 25 rpm) and a manifold pressure 39.0-40.0 in. Hg (overboost annunciator lights illuminated).

(5) Continue to climb until overboost annunciator lights go out (indicating critical altitude). As the lights go out note fuel flow, indicated altitude and OAT.

(6) Fuel flow at critical altitude should be 23.0-25.0 gph and density altitude 11,500 minimum to 12,500 maximum.

(7) If a discrepancy in critical altitude was noted, adjust the exhaust bypass valve. (Turning the exhaust bypass valve screw one full turn will alter the critical altitude in excess of 1,000 feet.)

(a) On turbochargers equipped with fine thread adjustment screws:

1) One flat turn in will increase critical altitude approximately 200 feet.

2) One flat turn out will decrease critical altitude approximately 200 feet.

(b) On turbochargers equipped with course thread adjustment screws:

1) One flat turn in will increase critical altitude approximately 300 feet.

2) One flat turn out will decrease critical altitude approximately 300 feet.

Adjustments of critical altitude in excess of 500 feet may require retrimming of the fuel flows at 100% power. Critical altitude should not differ more than 500 feet between engines.

(8) With full rich mixture, cowl flaps open, 2600 ± 25 rpm, 92 KIAS airspeed, and 1,000 to 3,000 feet density altitude, check the operation of the manifold pressure relief valve. Slowly advance one throttle to the wide open position. The manifold pressure shall stabilize between 42.0 and 44.0 in. Hg; there shall be no loss of power, and fuel flow indication shall be well over the red line. Do not exceed 40.0 in. Hg manifold pressure for more than ten seconds. Repeat this check on the other engine.

NOTE: Idle speed and idle mixture indication is a function of engine temperatures. Therefore, at normal ground idle temperatures (cylinder and oil temperature indications may or may not be "in the green") idle speed win be approximately 700 ± 25 rpm, and the idle mixture check will result in a 25 to 50 rpm increase in engine speed.

2. Seneca V

A. Rigging

Refer to “Figure 5” on page 761012.

(1) Throttle Controls

(a) Move the affected throttle lever to idle position.

(b) At the appropriate engine, remove the upper cowl and disconnect the cable from the throttle valve arm.

(c) With the Engine Throttle Control at full throttle, rig the cable to provide a .03 minimum clearance between the Quadrant Throttle Lever and Engine Control Cover. The Quadrant Throttle Lever must have aft clearance of .03 minimum with the Engine Control Cover at the Idle Stop.

(d) Refer to 32-60-00 and check the rigging of the “Gear Warning Throttle Switch,” if so equipped.

(e) Install cowling.

(2) Propeller Controls

(a) If not already done so, move the particular propeller control to full increase (full forward).

(b) At the affected engine, remove the upper left cowl panel and disconnect the control cable from the governor control arm.

(c) Refer to “Figure 5” on page 761012 and check the travel of the control arm. The Hart 95° travel requirement.

(d) Move the governor control arm to max speed or full increase rpm and rig cable to provide 0.03 to 0.06 inch (0.762 to 1.524 mm) clearance or cushion between the appropriate prop lever and quadrant stop.

(e) Install cowl panel and ensure alignment of the corresponding control for the other engine.

(3) Mixture Controls

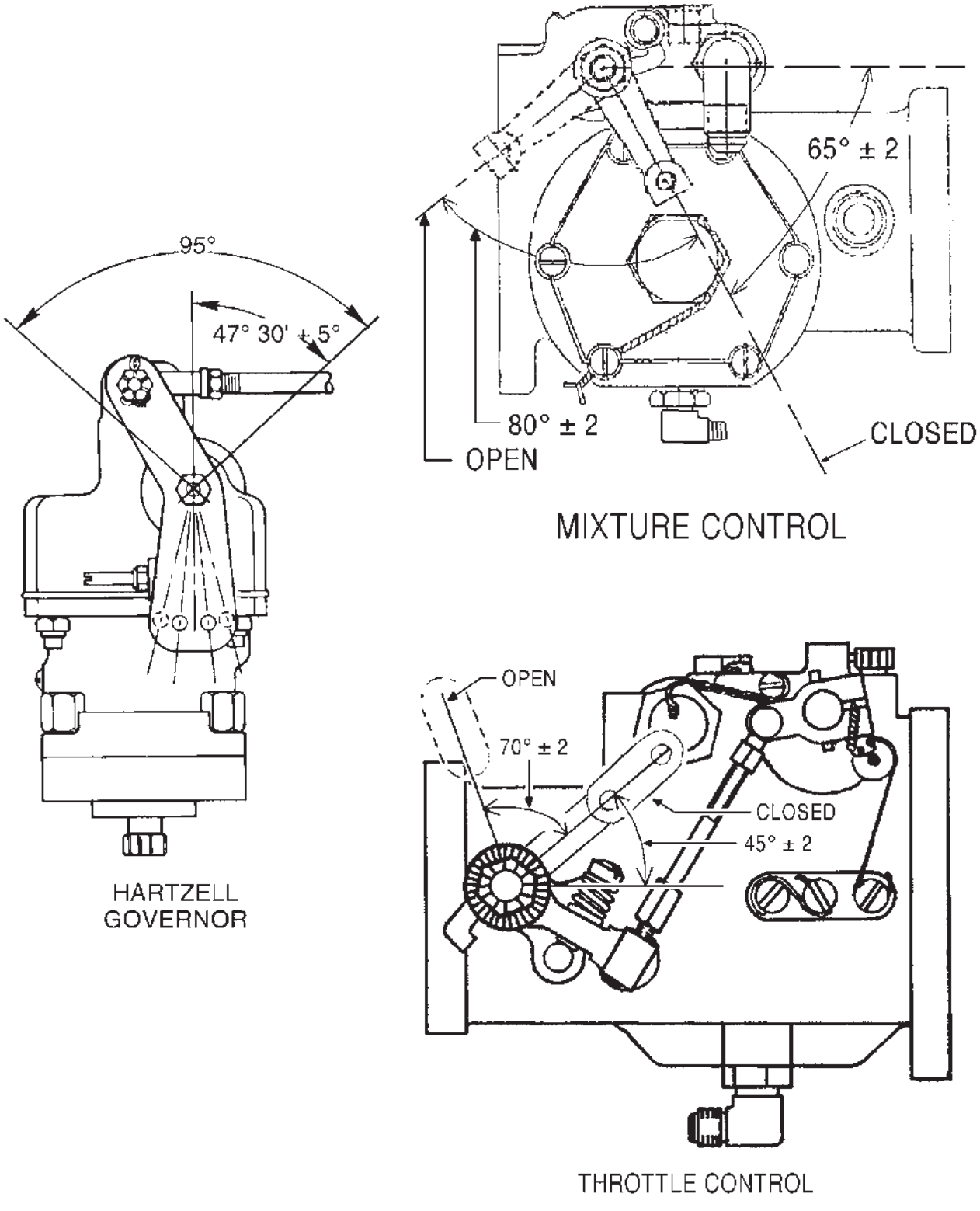
(a) Move the affected mixture control to its full rich (full forward) position.

(b) Remove the upper cowl panel on the appropriate engine and disconnect the control cable from the mixture control arm on the fuel injection servo.

(c) Move or ensure the arm is to its full rich (full travel) position. Refer to “Figure 5” on page 761012 for arm travel specifications.

(d) With the Engine Mixture Control at full rich, rig cable to provide a .03 minimum clearance between the Quadrant Mixture Lever and Engine Control Cover. The Quadrant Mixture Lever must have aft clearance of .03 minimum with the Engine Control Cover at the Idle Cutoff Stop.

(e) Check the alignment of the corresponding mixture lever and install removed panels and cowl panel.



Engine Controls (Seneca V)
Figure 5

B. Engine Setup Procedures

The following procedures should be used to check and adjust the power plants to maintain the required operating limits and ensure obtaining good setup results. It is important that the following checks be made to both engines before proceeding with any actual system adjustments:

(1) Preparations

Check all fuel lines, instrument lines, and electrical connections for proper routing and security of fittings.

(a) Manifold Pressure Gauge Check

Perform the following check:

1) Disconnect the manifold pressure hose at the bulkhead fitting on the firewall for both the LH and RH engines.

2) Connect two lines from a “tee” fitting on a pressurized air source to the LH & RH bulkhead fittings on the firewall. DO NOT EXCEED 40 INCHES OF MERCURY (ABSOLUTE).

3) The maximum difference in the static readings on the LH and RH manifold pressure gauges shall not exceed 0.50 In. Hg.

(b) Fuel line purge

Disconnect the calibrated pressure gauge which was connected to the fuel line upstream of the servo on the previous test. Pump out approximately one quart of aviation fuel. Reconnect the fuel line to the fuel servo.

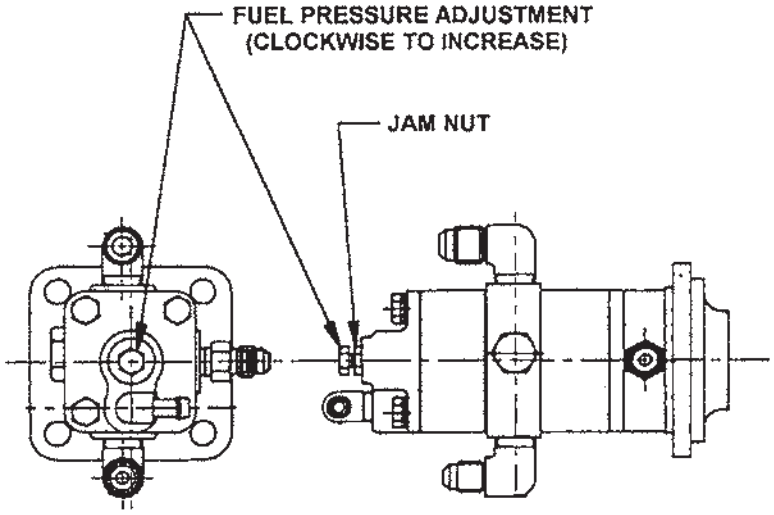
(2) Fuel pressure

(a) Emergency fuel pump operation check. Check the electric fuel pump as follows:

1) Install a calibrated fuel pressure gauge (range 0 to 75 psig) in the fuel line at the fuel flow transducer located just upstream of the inlet to fuel servo.

2) Using auxiliary power supplied to the aircraft such that the aircraft voltmeter shows 28.0 +.5, - .0 volts, engage the electric emergency fuel pump and verify that the “zero-flow” fuel pressure is between 44 and 60 psig. If the fuel pressure does not fall between these values, the pump must be replaced. There is no method of adjusting the fuel pump discharge pressure.

3) Disconnect external power from the aircraft.

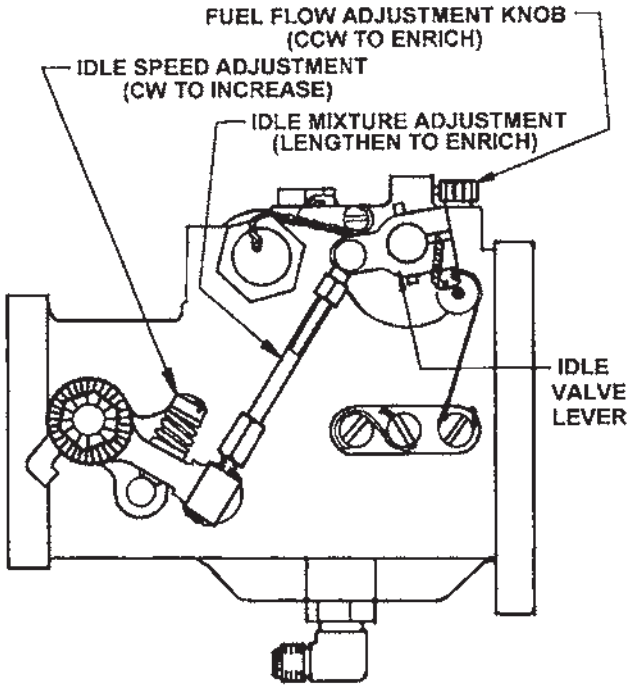


Fuel Pump Adjustment (Seneca V)
Figure 6

(b) Engine driven fuel pump check. Check the engine driven fuel pump as follows:

With the engines operating at 2575 - 2600 rpm and 37.5 - 38.0 inches Hg. manifold pressure mixtures set to 24.0 to 25.5 gallons per hour, and the emergency fuel pumps in the off position, verify that the engine-driven fuel pump discharge pressure is set to 40 to 50 psig (measured between the fuel flow transducer and the fuel servo inlet). If the engine driven fuel pump pressure does not fall within this range, the fuel pressure may be adjusted by the following method:

1) Loosen “jam nut” (see “Figure 6”) on fuel pressure adjustment screw located on end of fuel pump.
2) Adjust screw clockwise to increase pressure or counter clockwise to decrease fuel pressure.
3) Repeat static run-up and readjust fuel pressure as required to obtain fuel pump discharge pressure of 40 to 50 psig.



Fuel Servo Adjustment (Seneca V)
Figure 7

(3) Idle Performance

See “Figure 7”.

(a) Idle Mixture

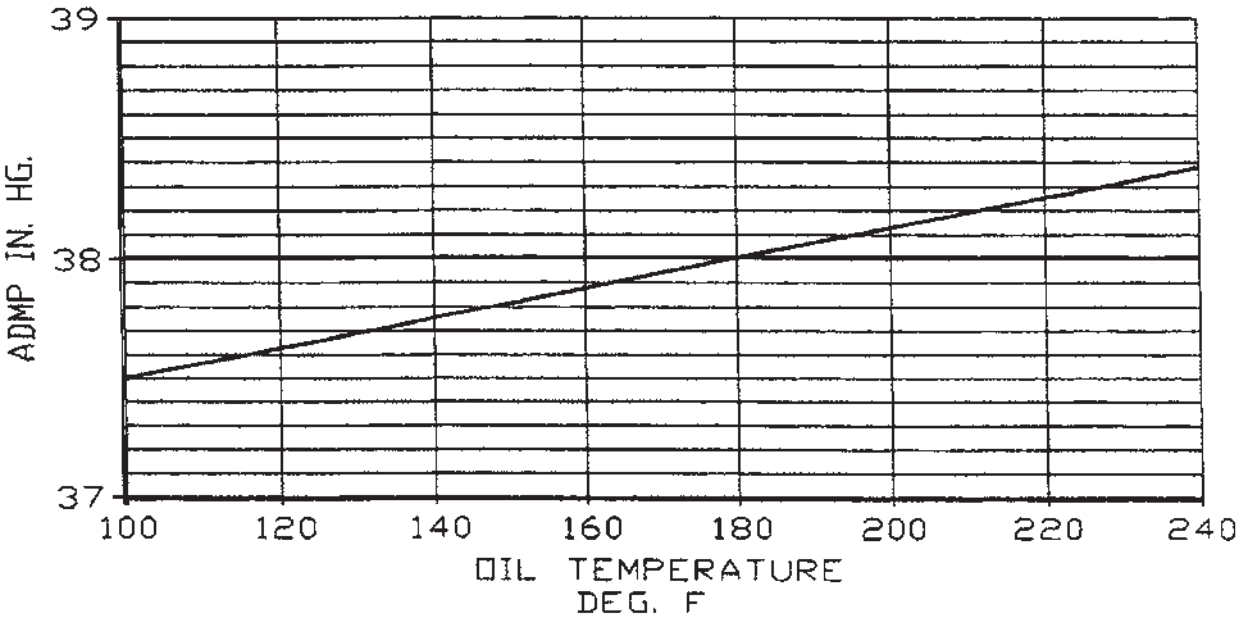
1) Check and adjust idle mixture as follows:
2) Operate the engine at 1500 to 1800 rpm until cylinder head temperatures are 250 °F to 350 °F and the oil temperature is 160 °F to 180 °F.
3) Reduce the engine speed and stabilize it at 775 ± 25 rpm.
4) Slowly, but positively, move the mixture control from full rich to idle cut-off. The engine speed shall increase 25 to 50 rpm before beginning to drop toward zero.
5) If the engine speed increase is less than 25 rpm, adjust the idle mixture adjustment to enrich the mixture. If the engine speed increase is more than 50 rpm, adjust the idle mixture to lean the mixture. The idle mixture adjustment is made by lengthening (richening) or shortening (leaning) of the linkage between the throttle lever and the idle valve lever. Recheck as required to ensure the idle mixture is adjusted within the limits specified above. Each time an adjustment is made, clear the engine by running it up to approximately 2000 rpm before making another mixture check.

(b) Idle Speed

Check and adjust idle speed as follows:

1) Operate the engine at 1500 to 1800 rpm until cylinder head temperatures are 250 °F to 350 °F and the oil temperature is 160 °F to 180 °F.
2) Reduce the engine speed and stabilize it at 775 ± 25 rpm.
3) Adjust the idle speed screw located on the aft side of the throttle lever until contact is made with the throttle arm stop.

CHART 4
MANIFOLD PRESSURE SETUP (SENECA V)



(4) Full Power Performance (PIR-PPS50026-6, Rev. C.)

Check and adjust full power performance as follows:

(a) With the aircraft pointed in the direction of the prevailing wind, run both engines at 1500 to 1800 rpm until the oil temperature is 160 °F to 180 °F.

(b) With the propeller controls set to 2575 - 2600 rpm (using a digital handheld tachometer), set the manifold pressure with the throttles in the full forward position to the value shown in Chart 4 (- 0.5, + 0.0 In. Hg.) by turning the wastegate actuator linkage clockwise (shortening) to decrease critical altitude and counter-clockwise (lengthening) to increase critical altitude. Re-install & secure cotter pin.

NOTE: 1 ½ turns = approximately 500 FT.

(c) Set up the propeller governor as follows: With the manifold pressure set to 38 inches of manifold pressure and the propeller controls in the full forward position, adjust the stop screw on the governor and associated rod end if necessary to obtain 2600 rpm (+0, -25 RPM).

(d) With the rpm set at 2575 - 2600 rpm and the manifold pressure set at 37.5 - 38.0 In. Hg. and the mixture controls in the full rich position adjust the fuel flows to 24.0 to 25.0 gallons per hour. The fuel flow can be adjusted by turning the adjustment knob located on the top left-hand side of the RSA-5 fuel servo (See “Figure 7” on page 761015). Turning the knob in the clockwise direction will result in a decrease in fuel flow while turning the knob in the counter-clockwise direction will result in an increase in fuel flow. Seven “clicks” of the adjustment knob will result in a fuel flow change of approximately one gallon per hour. Do not exceed twenty “clicks” of adjustment counterclockwise measured from the fully engaged position.

(5) Recheck

Recheck as follows:

(a) Recheck idle settings as specified in Idle Performance, above, and adjust as required.

(b) Recheck full power settings as specified in Full Power Performance, above, and adjust as required.

(6) Clean-up

After all ground checks and adjustments have been completed, check to ensure all lines which were disconnected during the setup have been reconnected and torqued to their appropriate values. Safety wire the mixture adjustment knob. Tighten “jam nut” on controller adjustment screw to prevent the screw setting from changing. Tighten “jam nut” on the engine driven fuel pump pressure adjustment screw.

(7) Flight Checks

Perform the following flight checks as follows:

(a) Full Power Fuel Flow Check

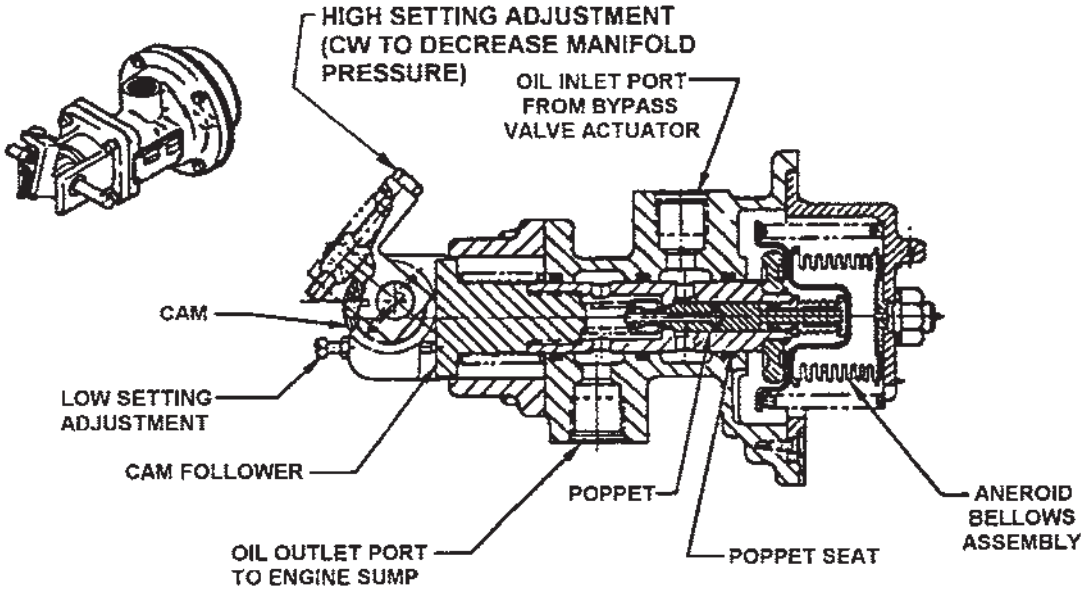
Conduct a full power climb to critical altitude. The fuel flows shall fall within the limits specified in “Chart 5” on page 761018.

NOTE: The maximum “split” between the LH and RH engines is not to exceed 1.0 GPH.

(b) Critical Altitude Check

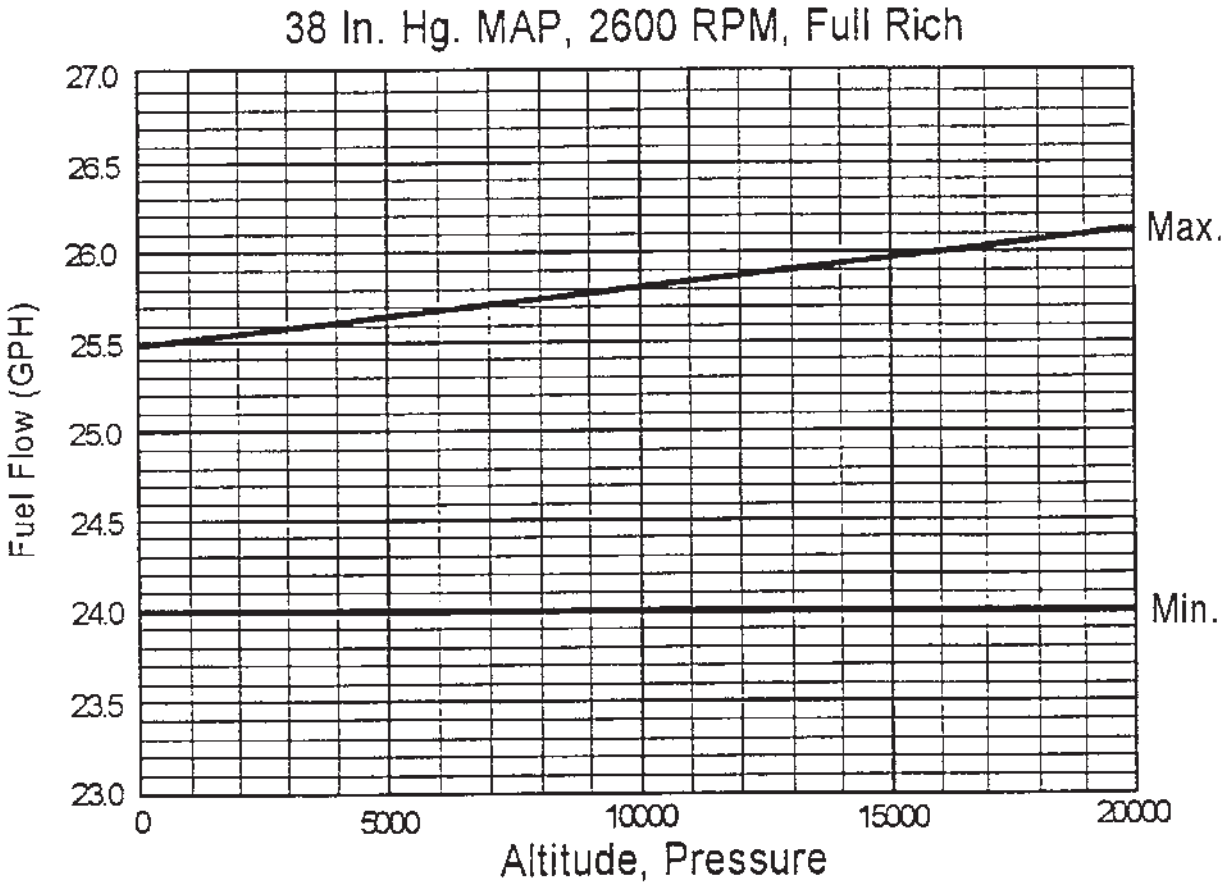
See “Figure 8” and “Chart 5” on page 761018.

The critical altitude for the left and right engines must be at least 19,500 ft (density) and may be adjusted down to meet the critical altitude of the lower engine. The “split” between the LH & RH engines shall be no more than 1500 ft. The engine with the higher critical altitude shall be adjusted down to meet the lower engine. The critical altitude may be adjusted by turning the high setting adjustment screw counterclockwise to increase manifold pressure or clockwise to decrease manifold pressure. One turn of the adjustment screw is approximately 1.0 In. Hg.



Controller Adjustment (Seneca V)
Figure 8

CHART 5
FUEL FLOW VS. PRESSURE ALTITUDE (SENECA V)



CHAPTER

77

ENGINE INDICATING

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CHAPTER 77

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CHAPTER 77 - ENGINE INDICATING

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Avidyne MFD or Garmin G1000/G1000 NXi EIS		1

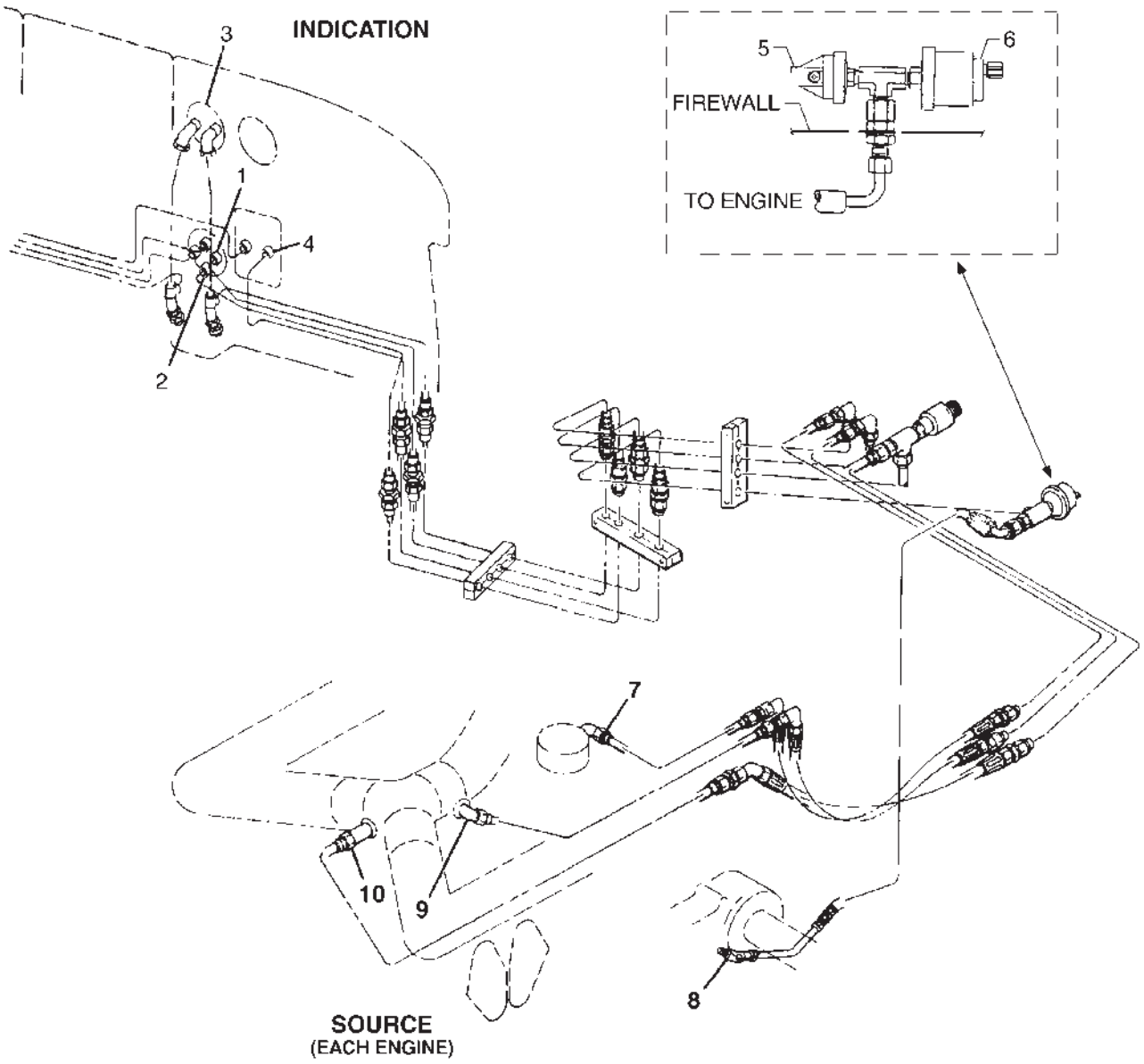
CHAPTER 77 - ENGINE INDICATING

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GENERAL

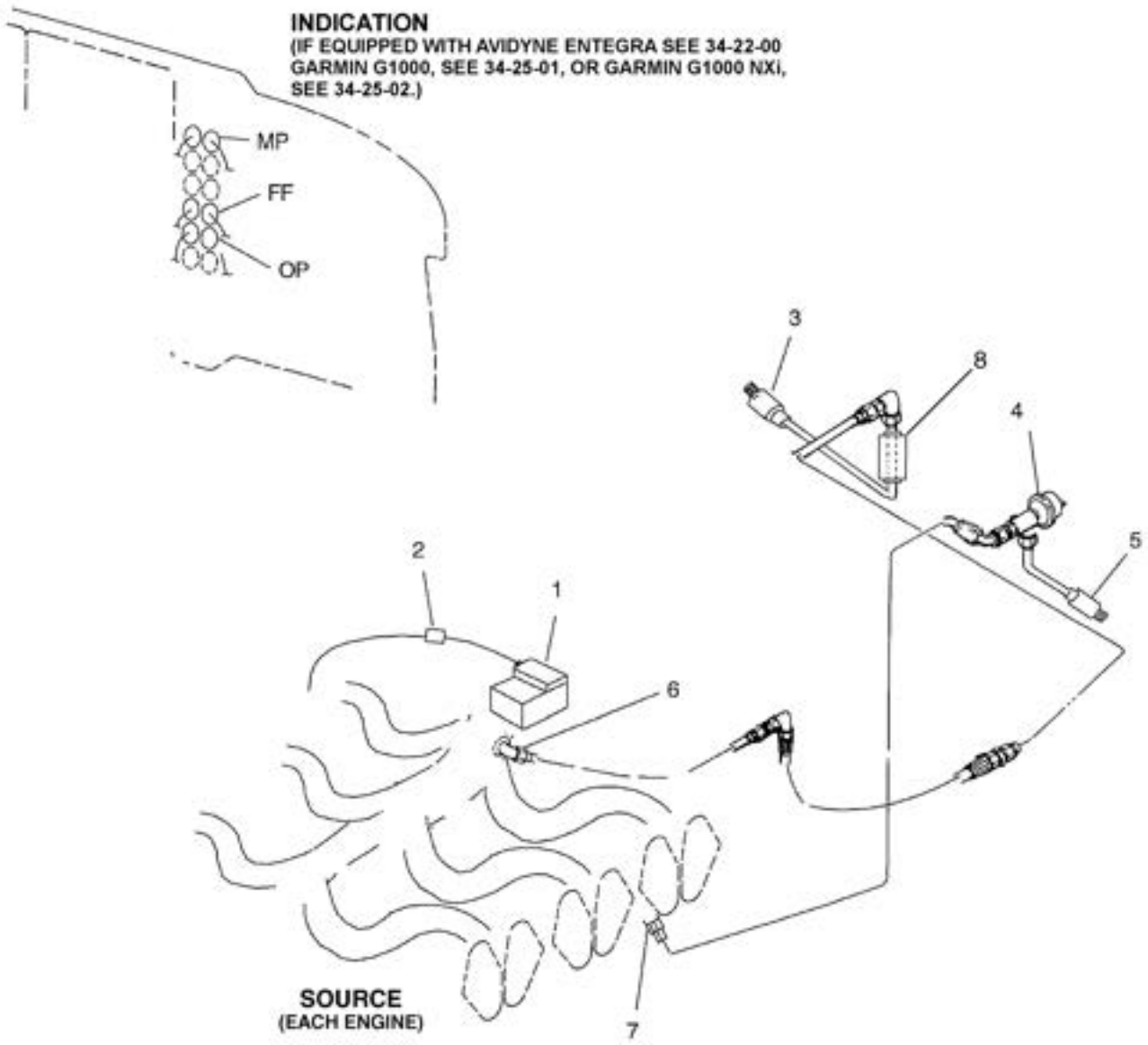
Engine Instrument Lines Installation



1. FUEL FLOW GAUGE PRESSURE PORT
2. FUEL FLOW GAUGE VENT PORT
3. MANIFOLD PRESSURE GAUGE
4. OIL PRESSURE GAUGE
5. OIL PRESSURE SWITCH
6. OIL PRESSURE SENDER
7. FUEL MANIFOLD (FUEL FLOW PRESSURE)
8. OIL GALLERY FITTING (OIL PRESSURE)
9. FUEL FLOW VENT RETURN
10. MANIFOLD PRESSURE FITTING

Engine Instrument Lines Installation
Figure 13 (Sheet 1 of 2)

Effectivity
Seneca IV



1. RSA FUEL SERVO
2. FUEL FLOW TRANSDUCER
3. MANIFOLD PRESSURE TRANSDUCER
4. OIL PRESSURE SWITCH
5. OIL PRESSURE TRANSDUCER
6. MANIFOLD PRESSURE FITTING
7. OIL GALLERY FITTING
8. MANIFOLD PRESSURE SWITCH (S/N'S 3449161, 3449301 AND UP)

Effectivity
Seneca V

Engine Instrument Lines Installation
Figure 1 (Sheet 2 of 2)

POWER

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

1. Seneca IV

A. Manifold Pressure Gauge

(1) Description

The dual manifold pressure gauge is a vapor proof, absolute pressure type instrument, calibrated from 10 to 50 inches of mercury. Incorporated in the gauge are switches that complete the annunciator panel circuit whenever engine manifold pressure exceeds 39.5 inches of mercury. The manifold pressure lines have drain valves located behind and below the fuel manifold pressure gauge. This allows any moisture which may have collected from condensation to be pulled into the engines. This is accomplished by depressing the two valves for 5 seconds while operating the engines at 1000 rpm.

NOTE: Do not depress the valves when manifold pressure exceeds 20 inches Hg.

(2) Troubleshooting

See “Chart 1”.

CHART 1
TROUBLESHOOTING MANIFOLD PRESSURE INDICATOR (SENECA IV)

Trouble	Cause	Remedy
Excessive error at existing barometric pressure.	Pointer shifted.	Replace instrument.
Excessive error when engine is running.	Line leaking.	Tighten line connections.
Sluggish or jerky pointer movement.	Defective instrument.	Replace instrument.
Dull or discolored marking.	Age.	Replace instrument.
Incorrect reading.	Moisture or oil in line.	Depress line valves and/or disconnect lines at instrument and blow out.

B. Tachometer

The electric tachometer system makes use of two tachometer generators (one on each engine) and a dual tachometer indicator. Each generator is mounted to the tachometer drive pad of the respective engine's accessory section. The generators are electrically interconnected with the tachometer secured to the instrument panel in the column of engine instruments.

The system functions through the reaction of the engine operating its tachometer generator. As the generator is activated by the engine, a pulse pattern (based on "4" pulses for each revolution of the generator) is sent to the tachometer which reacts to the incoming signals to indicate the engine rpm.

(1) Troubleshooting

See "Chart 2".

(2) Indicator

The tachometer indicator is mounted to the instrument panel just below the dual manifold pressure gauge in the column of engine instruments. Refer to "Figure 1" on page 77103.

(a) Removal

1) Disconnect harness from back of instrument.
2) Support the instrument and remove the four retaining screws.
3) Withdraw the instrument.

(b) Installation

1) Position the instrument in place.
2) Secure with the four retaining screws.
3) Connect harness to back of instrument.

(3) Generator

A tachometer generator is mounted to the tachometer drive pad of the each engine's accessory section.

(a) Removal

1) Remove the left and right cowl panels and upper support panel.
2) Locate the unit and follow the leads back to the connection and separate the connector.
3) Remove safety wire and unscrew the unit from the tachometer drive.

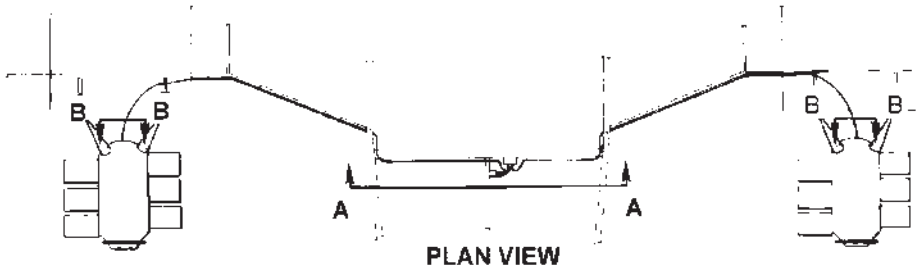
CHART 2
TROUBLESHOOTING TACHOMETER (SENECA IV)

Trouble	Cause	Remedy
Both pointers inoperative.	Circuit breaker tripped.	Reset.
One pointer inoperative.	Defective generator.	Replace generator.
One pointer inaccurate.	Defective indicator.	Replace tachometer indicator.

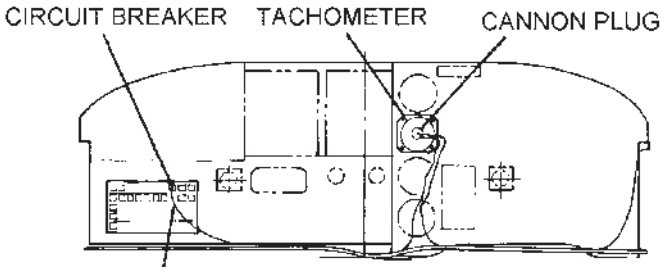
CAUTION: THE ADJUSTMENTS WITHIN THE TACHOMETER INDICATOR MUST BE MADE ONLY BY A QUALIFIED INSTRUMENT REPAIR FACILITY.

NOTE: With a tachometer generator removed from the engine, and electrical power applied, spinning the shaft with fingers should cause the pointer to deflect. This generally proves out the wiring. The tachometer generators receive an excitation voltage from the indicator.

The generators can be exchanged from one engine to the other for troubleshooting.

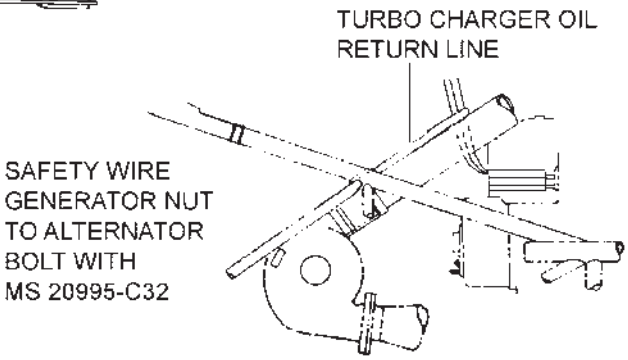


PLAN VIEW



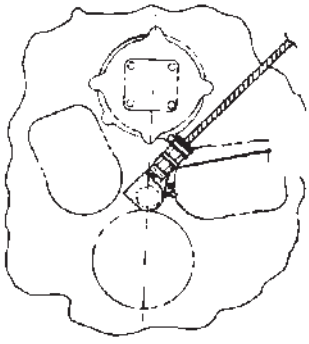
VIEW A - A

-- NOTE --
See Chapter 91 for
Electrical Schematic



TO CHECK CALIBRATION, SET GENERATOR AND READ TACHOMETER AS SHOWN.

TACH. INDICATOR (RPM)	TACH. GENERATOR (RPM)
500 ±50	619
1000 ±50	1238
1500 ±50	1856
2000 ±25	2475
2500 ±25	3094
2600 ±25	3218
2800 ±25	3465
3000 ±25	3713
3500 ±50	4331



VIEW B - B

Tachometer Installation
Figure 1

Effectivity
Seneca IV

(b) Installation

1) Screw the unit into the tachometer drive. Secure with safety wire.
2) Connect the electrical connector.
3) Reinstall the left and right cowl panels and upper support panel.

2. Seneca V

Manifold pressure and engine RPM information are displayed as follows:

Airplanes equipped with Standard avionics (1997 and up), Meggitt Single-side EFIS (2003–2005), or Garmin G600 (2010 and up) use the Engine Instrument and Digital Display Monitoring Panel (DDMP) System described in “Integrated Engine Instrument Systems” on page 77401.

In airplanes equipped with Avidyne FlightMax Entegra (2005–2010), engine data is collected by a Data Acquisition Unit (DAU) and display on a Multi-Function Display (MFD) or Primary Flight Display (PFD). See 34-22-00 and 91-77-40 for more information.

In airplanes equipped with Garmin G1000/G1000 NXi (2013 and later), engine data is collected by an Engine/Airframe Unit (GEA) and is normally displayed on the Multi-Function Display (MFD). In the event of a MFD failure, all engine instruments can be displayed on the Primary Flight Displays (PFD’s). See 34-25-01 or 34-25-02 and 91-77-40 for more information.

Accordingly, information in this section describes only the sensors and transducers used to send the engine information to the appropriate system described above.

A. Manifold Pressure Transducer

A manifold pressure transducer located on the aft side of the firewall on each engine provides manifold pressure data to the DDMP or MFD.

(1) Removal

(a) Remove the nacelle hatch cover.
(b) Locate the transducer on the aft side of the firewall, right of center.
(c) Disconnect the electrical leads from the transducer.
(d) Loosen the P-clamp securing the transducer to the firewall.
(e) Loosen the fitting connecting the transducer to the manifold pressure line and remove the transducer.

(2) Installation

(a) Place the transducer in the P-clamp and connect to the manifold pressure line.
(b) Torque the manifold pressure line fitting per label on the transducer.
(c) Tighten the P-clamp to secure the transducer to the firewall.
(d) Connect electrical leads.
(e) Reinstall the nacelle hatch cover.

(3) Test

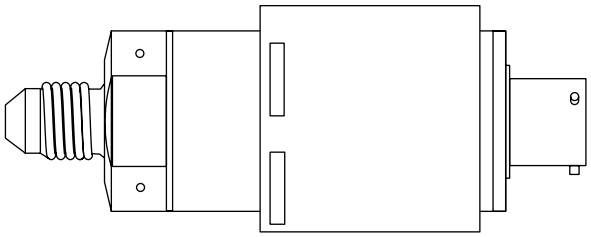
The manifold pressure transducer (P/N 599-592) can be tested as follows:

(a) Basic

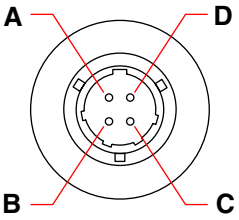
1) Turn aircraft electrical power on to energize cockpit instruments.
2) Prior to engine start, confirm the accuracy of the manifold pressure displayed by the cockpit instrumentation by comparing the indicated manifold pressure against the actual, current barometric pressure, as reported by the local airport authority, or another known good instrument.

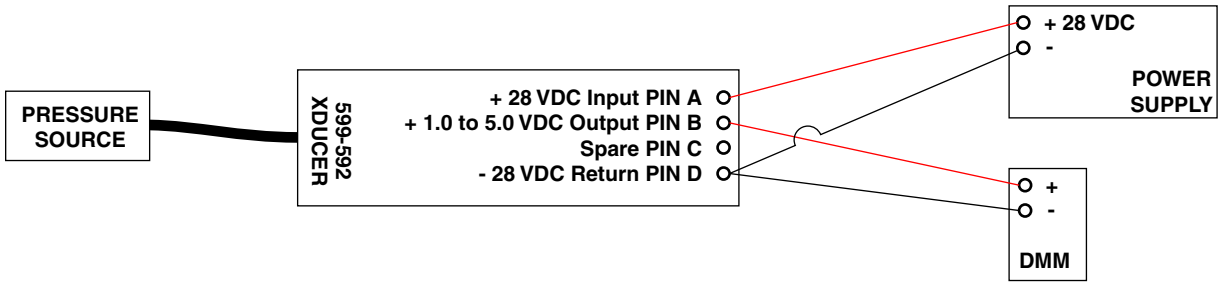
CHART 3
MANIFOLD PRESSURE TRANSDUCER SERVICE TEST SPECIFICATIONS

Pressure Applied (PSI)		P/N 599-592 Nominal ± 2% (VDC)
0		1.00
5		1.80
10		2.60
15		3.40
20		4.20
25	Maximum	5.00
20		4.20
15		3.40
10		2.60
5		1.80
0		1.00



P/N 599-592





Manifold Pressure Transducer Test Set-up
Figure 2

Effectivity
Seneca V

3) With a properly functioning Manifold Pressure Transducer, the indicated manifold pressure should agree with the actual barometric pressure within 0.5 inHg (one-half inches of mercury). If an erroneous reading is suspected, continue with detailed test, below.

(b) Detailed

1) Remove the transducer per Removal, above.

2) Using a calibrated regulated pressure source and a 28 VDC power source, setup the transducer for testing as shown in “Figure 2” on page 77105

3) Slowly applying regulated pressure, record the voltage shown on the digital multimeter (DMM) at each stage as shown in “Chart 3” on page 77105, until a maximum pressure of 25 PSI is reached.

4) Then, slowly decrease the regulated pressure and, again, record the voltage shown on the digital multimeter (DMM) at each stage as shown in “Chart 3” on page 77105, until zero (0) PSI is achieved.

5) The transducer is good if it meets the specifications shown in “Chart 3” on page 77105, Any other result is cause for transducer replacement.

6) Upon test completion, reinstall a good transducer per “Installation” on page 77104.

B. Tachometer Magnetic Sensor

Engine RPM information is provided by a magnetic sensor on the left magneto of each engine. As the magneto rotates, a signal is generated and sent to the DDMP or MFD. Refer to “Figure 3” on page 77107.

(1) Removal

(a) Remove the upper cowling and upper support panel.

(b) Locate the left magneto and the magnetic sensor. Follow the sensor leads back to the connection and separate the connector.

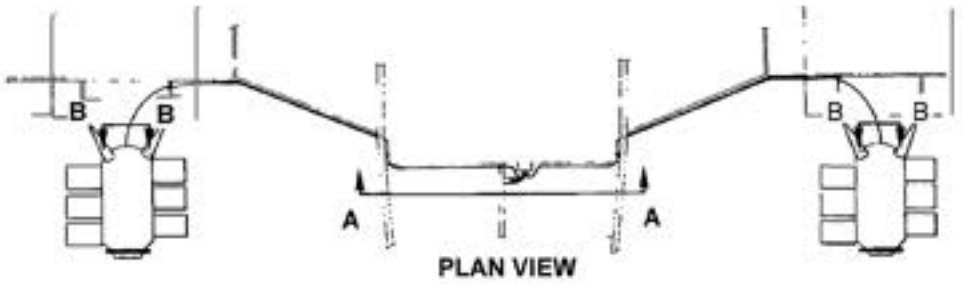
(c) Unscrew the sensor from the magneto.

(2) Installation

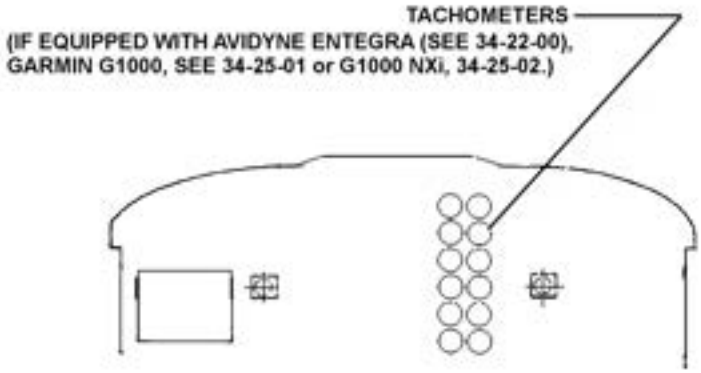
(a) Screw the sensor into the left magneto.

(b) Connect the sensor leads to the harness connector.

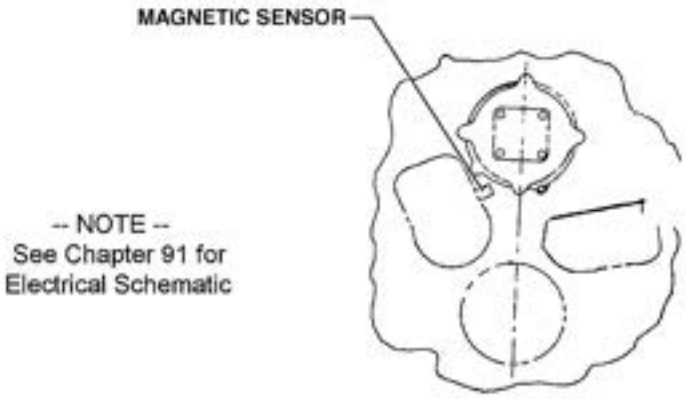
(c) Reinstall the upper support panel and upper cowling.



PLAN VIEW



VIEW A - A



VIEW B - B

Tachometer Installation
Figure 3

Effectivity
Seneca V

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TEMPERATURE

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

NOTE: Seneca V models incorporate a Turbine Inlet Temperature (TIT) gauge. For information on the TIT gauge installed in Seneca V models, see 77-40-00.

1. Exhaust Gas Temperature (EGT) Gauge and Probes (Seneca IV)

   A. Description

      This instrument, which is commonly referred to as EGT, is used to aid the pilot in setting the economical fuel-air mixture for cruising flight at a power setting of 75% or less. It is a sensing deviceto monitor the temperature of exhaust gases leaving the engine cylinders. If it is found defective after checking with troubleshooting chart, it should be replaced. If the leads to the gauge are defective in any way, they must be replaced. When replacing leads, it is necessary to use the same type and length of wire, because the resistance of the leads is critical for the proper operation of this gauge.

   B. Troubleshooting

      See “Chart 1”.

   C. Removal

      (1) Disconnect wires from the EGT gauge at the instrument panel.
      (2) Remove four bolts which secure the gauge to the instrument panel and remove the gauge.
      (3) Remove wires from the wire harness going to the engine.
      (4) Loosen the nut which secures the EGT probe to the exhaust transition area of the exhaust system, and remove the probe.

   D. Installation

      (1) Install the probe into the hole in the transition area of the exhaust system, and secure with locknut.
      (2) Route thermocouple wires along with the existing wire harness to the instrument panel.

CHART 1
TROUBLESHOOTING EXHAUST GAS TEMPERATURE GAUGE (SENECA IV)

Trouble	Cause	Remedy
Gauge inoperative.	Defective gauge, probe or wiring.	Check probe and lead wires for chafing, breaks or shorting between wires and/or metal structure.
	Adjusting potentiometer turned off scale.	Reset potentiometer.
Fluctuating reading.	Loose, frayed or broken electrical leads or faulty connections.	Clean and tighten connections. Repair or replace defective leads.

(3) Install the EGT gauge into the instrument panel, and secure with four bolts.

(4) Connect the thermocouple wires to the rear of the EGT gauge.

E. Cleaning and Inspection

Unless mechanical damage is evident such as broken glass, bent or broken pointer, or broken case, the following checks should be performed before removing the instrument:

CAUTION: DO NOT CONNECT OHMMETER ACROSS METER; IT WILL BURN OUT THE MOVEMENT OF THE METER.

(1) Remove probe from exhaust transition area and check for broken weld (at tip end) or burnt off end. Measured resistance of probe should be .8 ohms. Clean the connections with steel wool before reassembly.

(2) Disconnect lead wires at instrument. Resistance with lead wires connected to probe should be 3.3 ohms. Clean connections with steel wool before reassembly.

(3) With leads connected to instrument, heat probe with propane torch to dull red. The meter should read up to the fourth graduation or approximately 1500°F. If meter still does not read, replace it. Adjustments should be made by qualified shop.

2. Cylinder Head Temperature (CHT) Gauge (Seneca IV)

NOTE: For information on the Cylinder Head Temperature (CHT) gauge installed in Seneca V models, see 77-40-00.

A. Description

The cylinder head temperature gauge is part of the combination engine gauge which also includes the oil temperature gauge and the oil pressure gauge. Each cylinder head temperature gauge measures the cylinder head temperature using a sender located in a cylinder head. The cylinder head used is determined by the engine manufacturer. This gauge is an electrical instrument and is wired through the instrument's circuit breaker.

B. Troubleshooting

See "Chart 2".

CHART 2
TROUBLESHOOTING CYLINDER HEAD TEMPERATURE GAUGE (SENECA IV)

Trouble	Cause	Remedy
Instrument shows no indication.	Engine is cold.	Warm up engine.
	Power supply wire open.	Repair wire.
	Defective sender.	Replace sender.
	Defective instrument.	Replace instrument.
	Open circuit breaker.	Troubleshoot for fault.
Instrument goes all the way to upper stop.	Wire grounded between sender and gauge.	Repair wire.
	Defective sender.	Replace sender.

INTEGRATED ENGINE INSTRUMENT SYSTEMS

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

CAUTION: STATIC SENSITIVE DEVICES, HANDLE ONLY AT STATIC SAFE WORK STATIONS.

1. Description

A. Engine Instrument and Digital Display Monitoring Panel (DDMP)

This system is installed in Seneca Vs equipped with Standard, Meggitt Single Side EFIS, or Garmin G600 avionics, and is addressed in the remainder of this section. See “Figure 1” on page 77402.

The Engine Instrument and Digital Display Monitoring Panel (DDMP) form a complete state of the art engine monitoring system. Each analog instrument accepts signals from a variety of engine sensors. The DDMP communicates with each instrument via a digital interface, constantly receiving data regarding the status of each instrument parameter. In addition, the DDMP receives data directly from sensors that monitor Outside Air Temperature, Cabin Air Temperature, Electrical System Parameters, and Cabin Static Pressure. The DDMP provides a digital readout of each parameter, continually monitors for out of range (exceedance) conditions, and utilizes the data provided for auxiliary functions including Fuel Management and Engine Percent Power Calculations.

B. Avidyne Multi-Function Display (MFD) (Optional)

In airplanes equipped with the Avidyne FlightMax Entegra IFDS, the Multi-Function Display (MFD) provides all engine instrumentation. See the Pilot’s Operating Handbook (POH) Supplement and Avidyne Entegra Integrated Flight Display System (IFDS) in 34-22-00, for more information.

C. Garmin G1000/G1000 NXi Engine Indication System (EIS)

In airplanes equipped with the Garmin G1000 Integrated Avionics System, the Engine Indication System (EIS) displays electrical, fuel, engine, and flap position information. Engine data is collected by the Engine/Airframe Unit and is displayed on the Multi-Function Display (MFD) and Primary Flight Display (PFD). This is addressed in 34-25-01 or 34-25-02.

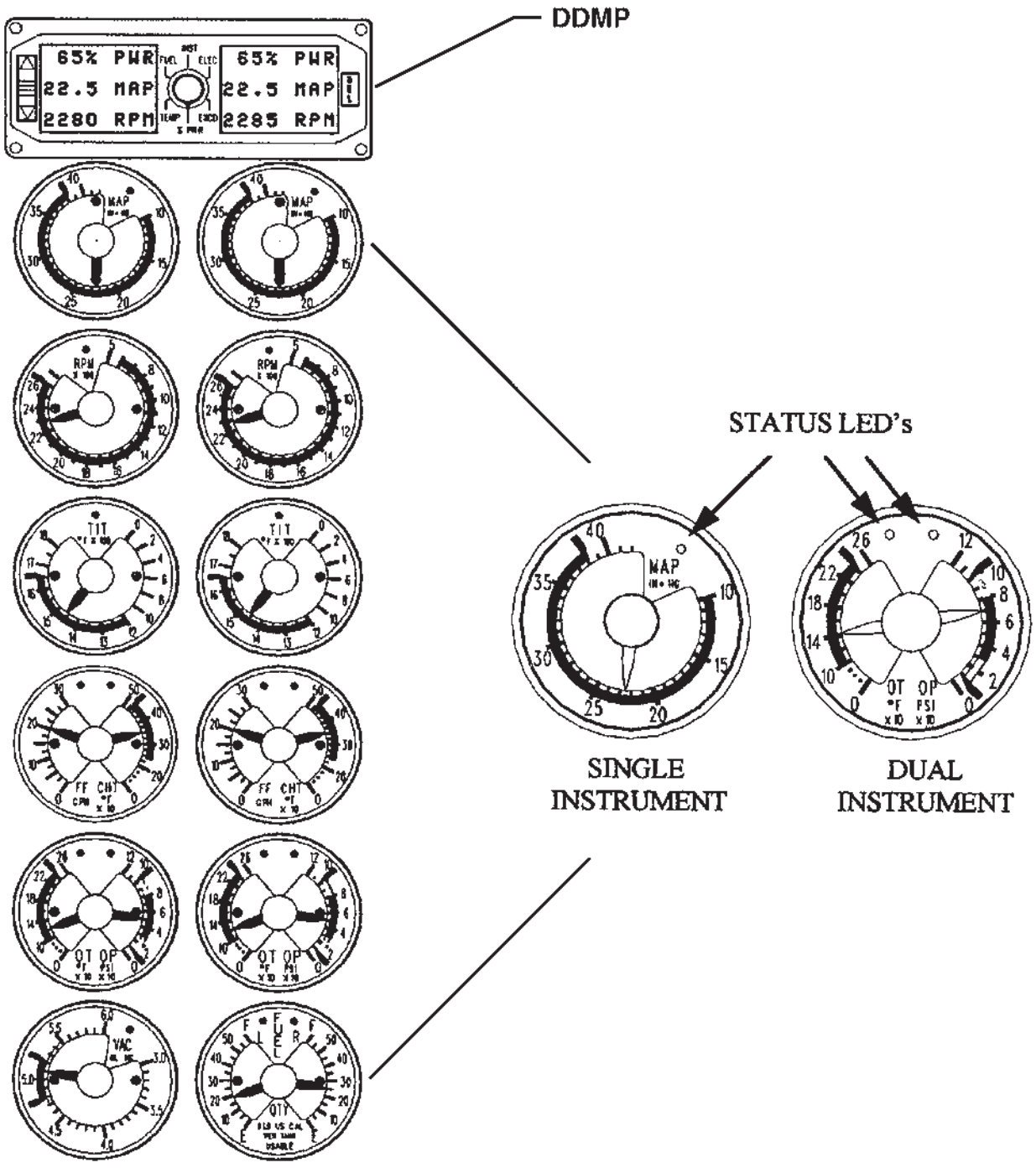
2. Troubleshooting

A. Engine Instrument and Digital Display Monitoring Panel (DDMP)

See “Chart 1” on page 77404 and “Chart 2” on page 77405.

B. Avidyne MFD or Garmin G1000/G1000 NXi EIS

See 34-22-00, 34-25-01 or 34-25-02, as appropriate.



Engine Instrument & Digital Display Monitoring Panel
Figure 1

3. Engine Instrument and Digital Display Monitoring Panel (DDMP)

A. Analog Instruments

The system uses conventional two (2) inch round engine instruments with either a single or dual pointer configuration. Instrument readings are displayed using a familiar rotating pointer against a fixed scale plate. An in-line pointer configuration is provided to minimize parallax and maximize viewing angle.

(1) Removal

(a) From front of instrument panel, loosen the larger of the two screws next to the instrument. (No need to remove screw completely.)

(b) Pull instrument out to gain access to connector on back of instrument.

(c) Twist connector on back of instrument to disconnect connector from instrument.

(2) Installation

CAUTION: REPLACEMENT OF THE FUEL QUANTITY INDICATOR REQUIRES THAT THE NEW INDICATOR BE CALIBRATED PER THE PROCEDURE IN 28-40-00.

NOTE: Calibration is not required if the existing instrument is simply removed and reinstalled.

(a) Holding instrument in front of proper position at instrument panel, connect connector to back of instrument.

(b) Insert instrument completely into instrument panel.

(c) Tighten screw.

(3) Instrument Self Test

Each instrument is microprocessor based and performs a power on self test, a continuous self test, and a continuous sensor validity test. A two color (green/red) status LED is provided for each instrument Upon initial power up, each instrument performs a power on self test. During this test, and prior to assuming normal operation, the status LED glows red then green and the pointer is driven to the full scale position, followed by the off scale zero position. The alarm audible alert is energized for one second at the end of the power on test.

(4) Instrument Status LED

The status LED provides an instant indication of the instrument status as follows:

(a) No LED Indication: Instrument is functioning normally. (If pointer is parked off scale low, no power to instrument)

(b) Green LED glows constantly: The instrument is being displayed in digital form on the DDMP.

(c) Red LED glows constantly: The parameter being measured is in an exceedance condition. (An Alarm Message is also displayed on the DDMP. See DDMP: Alarm Mode)

(d) Red LED flashes quickly (4 times per second): The input sensor has failed, or is providing erroneous information.

(e) Red LED flashes slowly (2 times per second): Self Test has identified a problem within the instrument. Instrument Failure.

CHART 1
TROUBLESHOOTING ENGINE INSTRUMENT AND DIGITAL DISPLAY MONITORING PANEL (DDMP)

Symptom	Solution
Red LED Flashing on Instrument (4 times per second)	• Input sensor providing erroneous information. Check input sensor.
Red LED Flashing on Instrument (2 times per second)	• Self test has determined problem within the instrument. Cycle power and check if instrument resets.
Instrument pointer parked off scale	• No status LED. Instrument not receiving power. Check power connection and circuit breakers. Cycle power to determine if instrument self test is initiated. • Red LED flashing 4 times per second. Input-sensor providing erroneous information. Check input sensor. • Red LED flashing 2 times per second. Self test has determined problem within the instrument. Cycle power and check if instrument resets.
“----” appears on DDMP display in Temperature or Electrical Mode	• Check sensor inputs
“----” appears on DDMP display in Instrument Mode	• Indicates Sensor, Instrument, or Communications Failure. Use status LED on Instrument to determine cause and verify as described above. • If status LED is off, and instrument appears to be operating properly, Run Self Test to check communications.
“----” appears on DDMP in % Power Mode	• If weight on wheels, DDMP functioning properly • Check MAP, RPM and Fuel Flow Instruments for proper operation. • Check communications using Self Test • Check Pressure Altitude in the DDMP Maintenance Mode. • Check OAT display in Temperature Mode
“----“ appears on DDMP in Fuel Mode	• Verify a valid file load was entered. • Check Fuel Flow Instrument for proper operation. • Check communications using Self Test. • Check GPS Input
Instrument Fail message on DDMP following Power On Self Test. i.e “LT RPM FAIL”	• Verify power to instrument. • Check connection between Instrument and DDMP • Run self test from Maintenance Mode

CHART 2
DDMP ERROR DECODER

"Error Code" Reason
"Error Showing Exceed Press SEL to Continue" Might show up in exceedance mode if there is a problem reading the EEPROM. Try deleting exceedances through the aux. Comm port to reset the state of the EEPROM.
"Commun Error Err 256" There was bad communication between the DDMP and the Instrument during calibration. Check the connection between the two.
"Data Not Valid Err 512" The instrument data was invalidated by the DDMP. Instrument and sensor failures can invalidate data. Old data eventually is marked as invalid. Check that the input to the instrument and connection to the DDMP are working.
"Non-Valid Index Err 8" The instrument was asked to calibrate to a point for which it cannot. For example, if a Seneca DDMP is mistakenly being used to calibrate a Saratoga fuel gauge - the Saratoga instrument cannot calibrate 61 gallons.
"Write Fail Bit Err 16" There was an error in the instrument's attempt to write to it's own EEPROM.
"Command Disabled Err 128" The command the instrument was asked to perform is disabled.
"Sensor Overflow Err 2" The raw sensor input is causing an overflow. The sensor signal is carefully offset and scaled in the instrument; check for correct input to the instrument. There are a few messages indicating that the available range of trim for Fuel Quantity, MAP, TIT, either null or span has been exceeded. Check that the input to the instrument is correct for the adjustment being made.
"DDMP FAIL OP SYS SHUTDOWN CODE 1" This appears only if the OS attempts to interrupt itself. This could happen if a periodic timer value was corrupted (and as a result ran too fast) or the task that is running is taking too long to complete its job. This is a protection against non-terminating loops in the task code.
"DDMP FAIL OP SYS SHUTDOWN CODE 2" This message appears when a task has been "blocked" too long. The OS has a prioritization scheme built in which selects the most important task waiting to be run and the runs it. If a lot of high priority task requests happen they could conceivably preempt a lower priority task from running. The system is designed for this not to happen but this message is the result of the "catch-all" code to deal with this occurrence, should it arise. To summarize, the two Op Sys Shutdown messages (above) are indications of some kind of software timing problem; either a piece of software is taking too long to execute (CODE 1) or waiting too long to be executed (CODE 2).

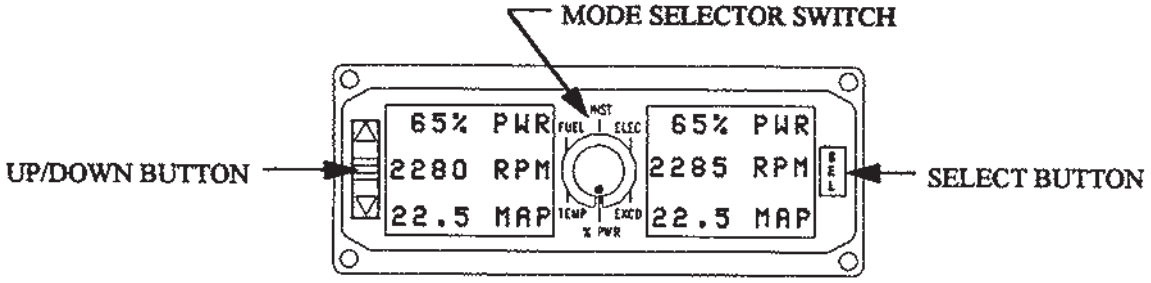
B. Digital Display Monitoring Panel (DDMP)

See “Figure 2”.

The DDMP continuously monitors each analog instrument via a digital communications link. Direct sensor inputs are provided for Outside Air Temperature, Cabin Air Temperature, Electrical System Inputs, and Cabin Static Pressure. A Global Positioning System input allows the DDMP to communicate with the aircraft GPS system to provide enhanced fuel management capabilities. The DDMP operates in one of six operating modes as explained below, and continuously monitors for exceedance conditions in the background. In the event that an exceedance condition is detected, an exceedance message will temporarily override the display of the selected mode of operation.

(1) Removal
  (a) From front of instrument panel, loosen four screws securing DDMP to the instrument panel.
  (b) Pull DDMP out to gain access to connectors on back of DDMP.
  (c) Twist connectors on back of DDMP to disconnect connectors from DDMP.

(2) Installation
  (a) Holding DDMP in front of proper position at instrument panel, connect connectors to back of DDMP.
  (b) Insert DDMP completely into instrument panel.
  (c) Tighten screws.



Digital Display Monitoring Panel (DDMP)
Figure 2

(3) Self Test

Upon power on, the DDMP will perform an internal self test, and will test communications with each of the instruments. If failure codes are displayed, or the system does not appear to be functioning properly, see the error codes, and instructions on initiating a self test from a power on state.

(4) DDMP Controls

The operating mode of the DDMP is selected using the rotary Mode Selector Switch located at the center of the instrument. Within a given mode, a cursor “>” is moved throughout different display locations to provide an easy to understand user interface. The Up, Down and Select Buttons are used in a multi-function manner as described in the following sections.

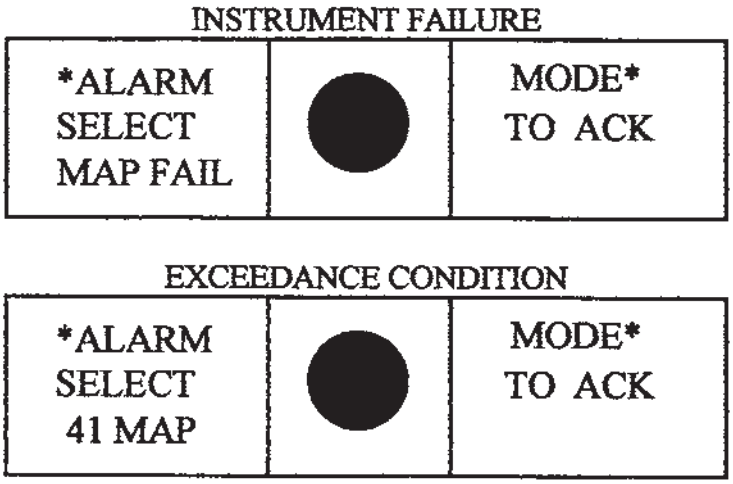
(5) Alarm Mode

The DDMP goes into the Alarm Mode if an exceedance or instrument failure is detected. In the Alarm Mode, the alarm overrides the current DDMP display and a display similar to one of the ones illustrated in “Figure 3” on page 77408 is shown.

The instrument will be identified on the side of the DDMP corresponding with right or left instrument failure.

To return to normal operation, press the SELECT Button to acknowledge the alarm. The DDMP will return to normal operation, and will not return to the ALARM MODE unless a new exceedance is detected. If an exceedance is still occurring when it is acknowledged, the Red LED will continue to glow on the instrument in exceedance, the DDMP will continue to record the exceedance, but no Alarm Message will appear on the DDMP until a new alarm condition occurs. To view the current exceedance on the DDMP, simply turn the Mode Selector Switch to “EXCD” (See DDMP: EXCEEDANCE MODE).

See “Chart 3” on page 77408 for possible alarm conditions.



Instrument Failure/Exceedance Condition
Figure 3

CHART 3
DDMP ALARM MODES

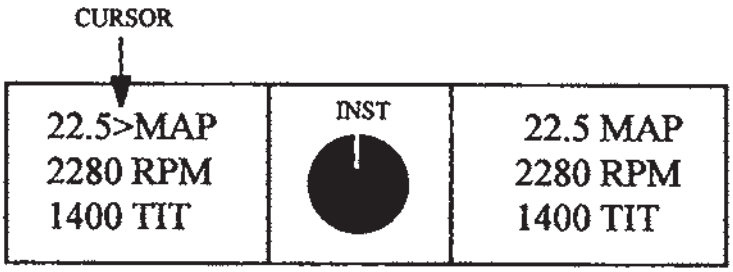
CONDITION	VISUAL ALARM	AUDIBLE ALARM
Instrument Exceedance	DDMP Display Red LED Glows on Instument	Horn until acknowledged
DDMP Exceedance	DDMP Display	Horn for 3 seconds
Instrument Failure	DDMP Display Red LED flashes twice per second on instrument	Horn for 3 seconds
Sensor failure	DDMP Display Red LED flashes four times second on instrument	Horn for 3 seconds

(6) Instrument Mode

In the Instrument Mode, each of the DDMP's two displays can be setup to display any three of the analog instrument values in a digital form. The left hand DDMP display can display any three of the instruments in the left column of instruments, plus the Left Fuel Quantity. The right hand DDMP display can display any three of the instruments in the right column of instruments, except for the Left Fuel Quantity.

To use the Instrument Mode, turn the Mode Selector Switch to "INST". On initial power up, the DDMP will default to the screen shown in "Figure 4".

To change the configuration, use the SELECT Button to move the cursor to the location of the display you wish to modify. Press the UP or DOWN Buttons to scroll through the options for that display. Continue as above to select the configuration for the remaining five displays as desired. After completing your configuration, the DDMP will default to that configuration until power is turned off.

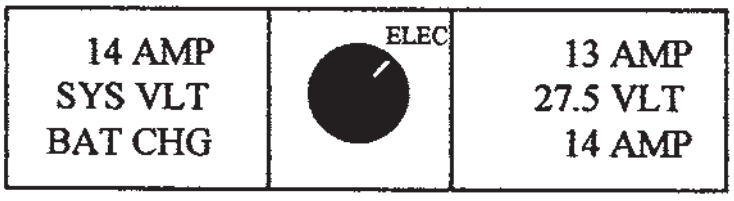


Instrument Mode
Figure 4

(7) Electrical Mode

In the Electrical Mode, the DDMP displays System Voltage, Left Alternator Current, Right Alternator Current, and Battery Charge Current. To access the Electrical Mode, turn the Mode Selector Switch to "ELEC". The display shown in "Figure 5" will appear.

NOTE: A special maintenance mode is accessed from the Electrical Mode using a button sequence password. The UP, DOWN, and SELECT Buttons serve no other purpose in the electrical mode. If a button is inadvertently pressed, an error message "INVALID KEY SEQUENCE SEL TO CONTINUE" will appear. Press SELECT to continue.



Electrical Mode
Figure 5

(8) Exceedance Mode

Each instrument continually monitors its input for out of limit, or exceedance conditions, and alerts the DDMP to such conditions. Over 200 exceedance events are stored in the DDMP memory. The exceedance mode is used to view and/or delete the exceedances that are stored in memory.

To view exceedances, turn the Mode Selector Switch to “EXCD”. The DDMP automatically defaults to the most recent or current exceedance. A display similar to that shown in “Figure 6” will appear.

To view additional exceedances, use the UP or DOWN Buttons to scroll through the exceedances stored in memory.

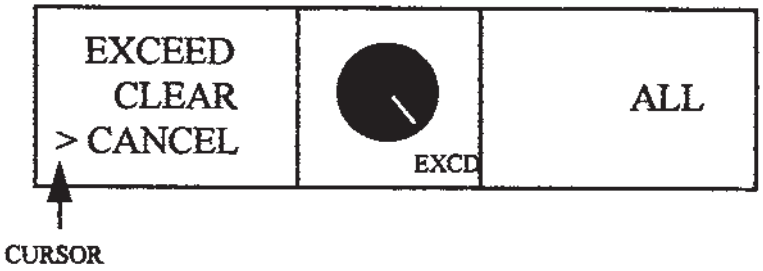
To delete exceedances, press the SELECT Button and the screen shown in “Figure 7” appears.

Press the UP Button to move the cursor to CLEAR ALL and press SELECT to delete all exceedances. Move the cursor back to the CANCEL Position if you decide not to delete all exceedances.

After SELECT is pressed, the DDMP will default back to the original exceedance screen. If the exceedances were deleted, the message NO EXCEEDS will appear, otherwise the original exceedance screen will appear.



Exceedance Mode
Figure 6



Exceedance Mode
Figure 7

The following abbreviations are used in the Exceedance Mode:

LO VLT	Low System Voltage
HI VLT	High System Voltage
LT MAP	High Left Manifold Pressure
RT MAP	High Right Manifold Pressure
LT RPM	High Left RPM
RT RPM	High Right RPM
LT TIT	High Left Turbine Inlet Temperature
RT TIT	High Right Turbine Inlet Temperature
LT CHT	High Left Cylinder Head Temperature
RT CHT	High Right Cylinder Head Temperature
LT OT	High Left Oil Temperature
RT OT	High Right Oil Temperature
LT LOP	Low Left Oil Pressure
RT LOP	Low Right Oil Pressure
LT HOP	High Left Oil Pressure
RT HOP	High Right Oil Pressure
LO VAC	Low Vacuum
HI VAC	High Vacuum
LFQ	Low Left Fuel Quantity
RFQ	Low Right Fuel Quantity

(9) Percent Power Mode

The DDMP displays the current Percent Power in 5% increments, and makes recommendations of desired MAP and Fuel Flow settings based on the current RPM to achieve the pilot's desired power setting.

To display the current Percent Power, turn the Mode Selector Switch to "% PWR". A display similar to that shown in "Figure 8" will appear.

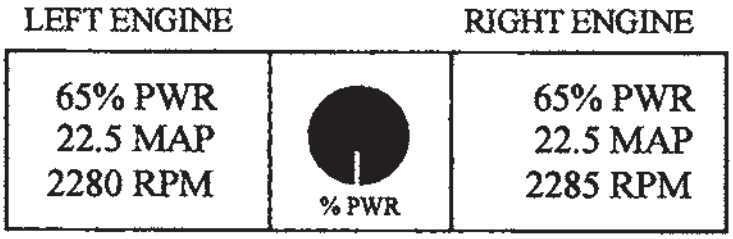
To estimate new engine settings for a given % Power, press the SELECT Button. The DDMP rounds the current settings to the nearest 5% Power and displays as shown in "Figure 9".

Press the UP or DOWN Button to change the % Power setting and the DDMP estimates new MAP and FF settings for the desired % Power and current RPM.

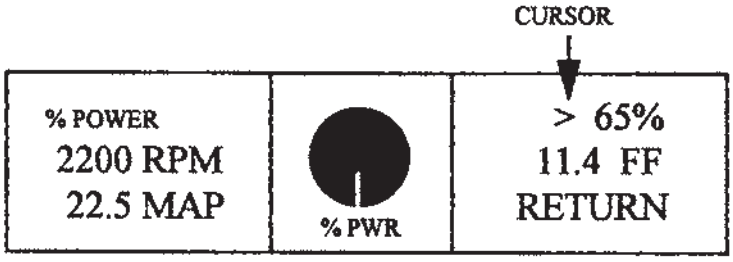
If the selected % Power is not obtainable at the current RPM setting, the DDMP will change the RPM to the nearest value within the range of the desired % Power. To select a new RPM value, press the SELECT Button to move the cursor to the RPM position. Using the UP or DOWN Button, select the desired value.

Adjust the engine MAP and FF settings to the recommended values and press SELECT two more times to return to the actual % Power screen.

The Percent Power function is only enabled when the aircraft is airborne. If the Percent Power Mode is entered while on the ground, or if an attempt is made to calculate estimated engine settings based on a combination of Fuel Flow, RPM, MAP, and Static Pressure values that are out of range, "----" will appear on the display.



Percent Power Mode
Figure 8



Percent Power Mode
Figure 9

(10) Temperature Mode

In the Temperature Mode, the DDMP displays Outside Air Temperature and Cabin Air Temperature. Turn the Mode Selector Switch to “TEMP” and the display shown in “Figure 10” appears.

Press the SELECT Button to toggle between Degrees Fahrenheit and Degrees Celsius.

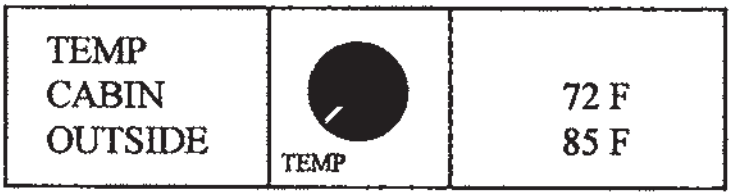
(11) Fuel Mode

In the Fuel Mode the DDMP interfaces with the aircraft’s Global Positioning System to calculate and display Flight Endurance, Fuel Remaining, Fuel Used, Nautical Miles Per Gallon, Estimated Fuel Required to Destination, and Estimated Fuel Remaining at Destination.

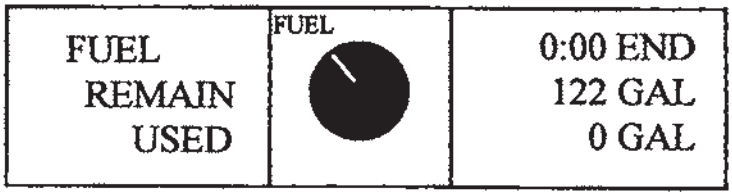
CAUTION: ALL OF THE FUEL CALCULATIONS ARE BASED ON THE PILOT MANUALLY ENTERING THE PROPER FUEL LOAD AFTER REFUELING. A MANUAL FUEL LOAD MUST BE ENTERED INTO THE DDMP FOR THE FUEL MODE TO FUNCTION PROPERLY. THERE IS NO CONSISTENCY CHECKING BETWEEN THE ENTERED FUEL LOAD AND THE ACTUAL FUEL QUANTITY.

After entering the fuel load, the DDMP performs all fuel calculation based on information from the Fuel Flow Instrument.

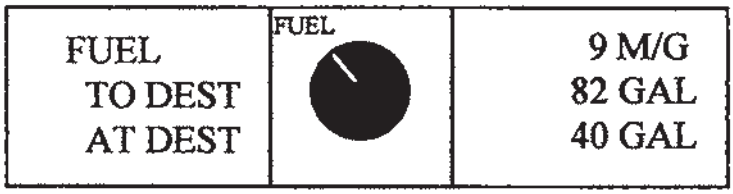
To enter the Fuel Mode, turn the Mode Selector Switch to “FUEL”. The screen shown in “Figure 11” will be displayed. Pressing SELECT will cycle the view to the next screen as shown in “Figure 12” on page 774014 and “Figure 13” on page 774014.



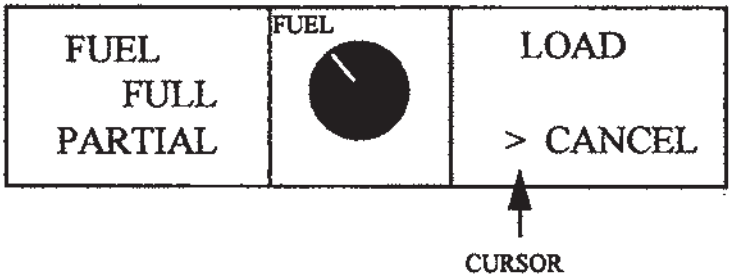
Temperature Mode
Figure 10



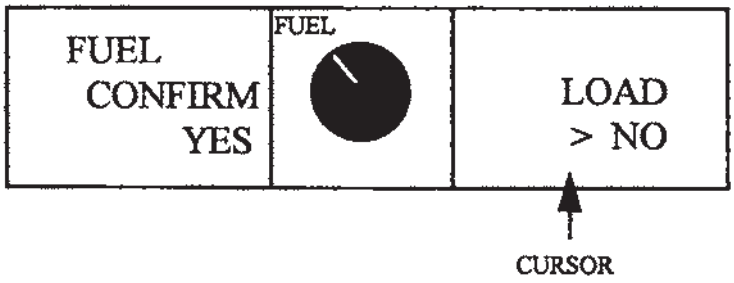
Fuel Mode
Figure 11



Fuel Mode
Figure 12



Fuel Mode
Figure 13



Fuel Mode
Figure 14

To enter a Fuel Load, use the UP/DOWN Button to position the cursor next to FULL or PARTIAL and press SELECT. FULL defaults to 122 gallons and allows the pilot to adjust the value starting with a full load. PARTIAL defaults to 0 gallons and allows the pilot to adjust the value starting with no load.

NOTE: Fuel load is the total fuel for both tanks, not a per tank value.

Press SELECT when the proper value has been entered and the screen shown in “Figure 14” will appear.

Using the UP Key move the cursor to YES and press SELECT to accept the new value or press SELECT to reject the new value.

After SELECT the display will revert back to the first Fuel screen.

(12) RS-232 Interface

An RS-232 interface is available to connect to the DDMP. It can be used to connect the ship's system to a computer to download exceedance data, or to connect a laptop system to log data during flight. The RS-232 connector is located under the pilot's side of the instrument panel.

(a) Hyperterminal Setup

The DDMP will communicate with any standard PC using the computer serial port, and any standard terminal emulation software such as Windows Hyperterminal, or Procomm. The following procedure uses Windows Hyperterminal.

1) Open Windows Hyperterminal (normally accessed by locating the Start Menu, then proceeding through Programs, and then to the Accessories folder).
2) Name the Connection "DDMP".
3) In the "Connect to" Window, change the "Connect Using" box to the proper serial port (normally COM1).
4) In the "COM1 Settings" Window, establish the following settings:
   a) 9600 Baud
   b) 8 Data bits
   c) No Parity
   d) 1 Stop bit
   e) No flow control
5) Click on "File" and "Properties".
6) In the "DDMP Properties" Window, select the "Settings" Tab, and "ASCII Setup".
7) Under ASCII Receiving, check "Append Line Feeds to Incoming Line Ends", and "Wrap Lines that Exceed Terminal Width".
8) Save Connection.

(b) Establishing Connection

1) Connect Aircraft RS-232 connections to Computer Serial Port.
2) Open "DDMP" Connection in Hyperterminal.
3) Apply Power to DDMP.
4) Press the Enter Key on the computer and the DDMP Menu should appear.

C. Instrument Calibration

See "Chart 4" on page 774016 and 28-40-00.

CHART 4
INTEGRATED ENGINE INSTRUMENTATION CALIBRATION

INDICATOR TYPE	SENSOR FULL SCALE OUTPUT (UNLESS NOTED)	CALIBRATION POINTS EACH INDICATOR SHALL MEET THE CRITERIA SPECIFIED BELOW WITH ITS APPLICABLE SENDER UNIT				
OAT	84.23 TO 121.32 OHMS (- 40 TO 55 °C)	TEST POINT °C MAXIMUM OHMS MINIMUM OHMS	- 40 84.62 83.84	0 100.39 99.61	20 108.20 107.42	55 121.72 120.93
CAT [1]	- 1.527 TO 2.229 mV (- 40 TO 55.°C)	TEST POINT °C MAXIMUM mV MINIMUM mV	- 40 - 37 - 43	0 3 - 3	20 23 17	55 58 52
VOLTS	10 TO 32 VDC	TEST POINT VDC MAXIMUM VDC MINIMUM VDC	10 10.2 9.8	16 16.2 15.8	22 22.2 21.8	28 28.2 27.8	32 32.2 31.8
BATTERY AMPS [2]	- 50 TO 50 mV	TEST POINT AMPS MAXIMUM mV MINIMUM mV	- 99 - 48.5 - 50.5	- 50 - 24 - 26	 1 - 1	50 26 24	99 50.5 48.5
MAP [3]	1.0 TO 5.0 VDC (O TO 25 PSIG)	TEST POINT IN HG(ABS) MAXIMUM VDC MINIMUM VDC	10 1.811 1.761	20 2.597 2.547	30 3.383 3.333	38 4.012 3.962	42 4.327 4.277
RPM [4]	12.50 HZ TO 67.50 HZ (500 TO 2700 RPM)	TEST POINT RPM MAXIMUM HZ MINIMUM HZ	500 13.05 11.95	1400 35.55 34.45	2000 50.55 49.45	2500 63.05 61.95	2700 68.05 66.95
TIT	O TO 1800 °F - 0.692 mV TO 40.581 mV	TEST POINT °F MAXIMUM °F MINIMUM °F	200 218 182	1200 1218 1182	1600 1618 1582	1650 1668 1632	1800 1818 1782
FUEL FLOW	0 TO 201.84 HZ (O TO 30 GPH)	TEST POINT GPH MAXIMUM HZ MINIMUM HZ	7.5 55.374 51.186	15 106.694 102.658	25 172.204 168.368	30 203.707 199.970	
CHT	THERMISTOR CURVE Q 200 °F = 748 OHMS 500 °F = 32 OHMS	TEST POINT ° F MAXIMUM OHMS MINIMUM OHMS	200 762.96 733.04	460 41 44	500 32.64 31.36		
OIL TEMPERATURE	THERMISTOR CURVE E 100 °F = 499 OHMS 250 °F = 34 OHMS	TEST POINT °F MAXIMUM OHMS MINIMUM OHMS	100 508.98 489.02	160 145.86 140.14	200 73.44 70.56	240 37.74 36.26	
OIL PRESSURE	1.0 TO 5.0 VDC (0 TO 150 PSIG)	TEST POINT PSI MAXIMUM VDC MINIMUM VDC	 1.032 0.968	30 1.832 1.768	70 2.899 2.835	110 3.966 3.902	120 4.232 4.168
VAC [5]	3.0 TO 5.0 VDC (3.0 TO 6.0 IN HG)	TEST POINT IN HG MAXIMUM VDC MINIMUM VDC	3.0 3.020 2.980	4.8 4.220 4.180	5.0 4.353 4.313	5.2 4.487 4.447	6.0 5.020 4.980
FUEL QUANTITY	SEE SECTION 28-40-00						REF. 100624 Z

1. Use temperature compensated calibrator to check calibration test points.
2. On 28 VDC common mode signal.
3. When the MAP indicator receives a signal from 37.1 IN HG to 38.9 IN HG, the digital indicator will snap to 38 IN HG. The analog MAP indicator will continue to display the actual IN HG.
4. When the RPM indicator receives a signal from 2560 RPM to 2640 RPM, the digital indicator will snap to 2600 RPM. The analog RPM indicator will continue to display the actual RPM.
5. When the vacuum indicator receives a signal below 3.0 IN HG, the digital indicator will snap to 0 IN HG.

NOTE: Fuel Quantity calibration specifications and procedures are in 28-40-00.

CHAPTER

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OIL

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CHAPTER 79

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CHAPTER 79 - OIL

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GENERAL

The oil system is a wet sump, force feed system with a capacity of 8 quarts. A conventional dipstick is provided for determining the oil quantity. On Seneca V models, in order to remove the dipstick, it must be rotated 90° due to a D-shaped retaining ring at the top of the dipstick.

When the engine is running, oil is drawn through a screen and pick up tube which extends from the sump to a port in the crankcase. Oil then flows to the inlet of the gear type, engine driven oil pump and is forced under pressure through the pump outlet. A pressure relief valve prevents excessive oil pressure by allowing excess oil to be returned to the sump. After leaving the pump, the oil under pressure enters a full flow filter and is passed onto the oil cooler. If the filter element becomes blocked, a bypass relief valve will open to permit unfiltered oil to flow to the engine. An oil temperature control unit allows oil to bypass the oil cooler when the oil is cold. Some oil flows through the cooler to prevent congealing in cold weather. When the oil temperature reaches approximately 170°F, the oil temperature control unit actuates to close off the cooler bypass, forcing the oil to flow through the cooler.

From the oil cooler, oil enters the crankcase where it is directed to the bearing surfaces and other engine components requiring lubrication and cooling. The propeller governor boosts engine oil pressure for operation of the propeller. A tap in the side of the crankcase supplies oil pressure for lubrication of the turbocharger bearings. Oil is carried to the turbocharger through an external line. After lubricating the turbocharger bearings, it is drawn into a scavenge pump and forced back to the oil sump. Oil within the engine drains, by gravity, back into the sump.

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DISTRIBUTION

Oil Filter

Replacement

Refer to “Figure 1”.

The oil filter element should be replaced after each 50 hours of engine operation. The filter element is mounted on the lower portion of the engine accessory case. Replace the filter element as follows:

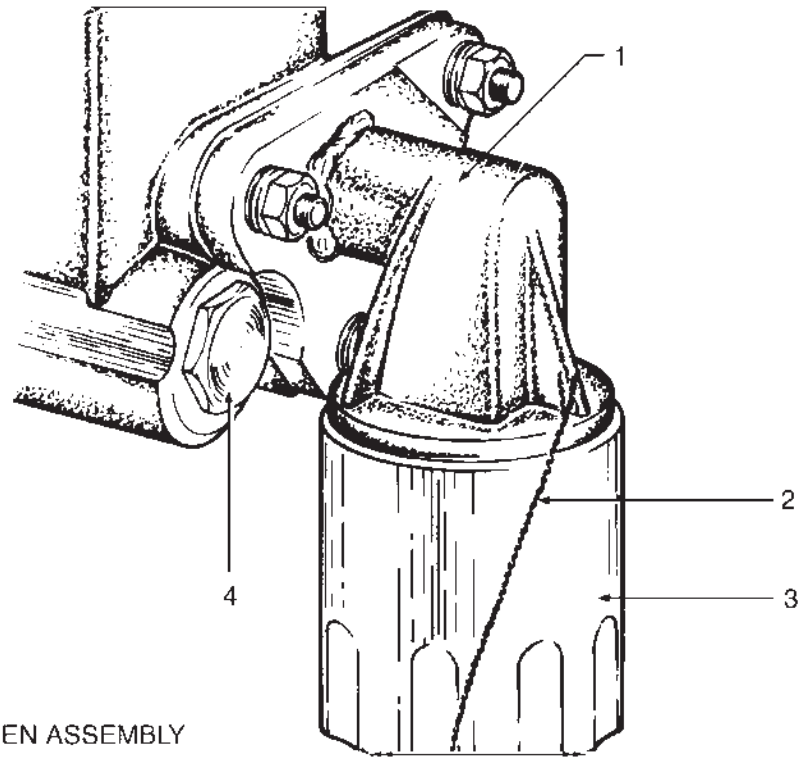
(1) Remove the lockwire between the nut on the filter and the oil filter adapter, and unscrew the filter element.

(2) Before installing a new filter, lubricate the gasket on the filter with clean engine oil.

CAUTION: DO NOT OVER TORQUE.

(3) Torque the filter 18 to 20 foot-pounds or 3/4 to 1 full turn after the gasket makes contact.

(4) Run the engine and check for oil leaks; then install lockwire between nut on filter and adapter.



1. OIL FILTER ADAPTER
2. SAFETY WIRE
3. OIL FILTER ELEMENT
4. OIL SCAVENGE SCREEN ASSEMBLY

Oil Filter Installation
Figure 1

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INDICATING

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

1. Engine Oil Pressure (Seneca IV)

NOTE: For information on the engine oil pressure gauge installed in Seneca V models, see 77-40-00.

A. Description

On Seneca IV models, the oil pressure gauge is part of the combination engine gauge which also includes the oil temperature gauge and the cylinder head temperature gauge. The oil pressure is transmitted to the gauge by means of an electrical transducer installed in the engine compartment.

Each oil pressure gauge is directly connected to the pressure side of its engine turbocharger oil supply line. See Figure 1 in 77-00-00.

B. Troubleshooting

See “Chart 1”.

CHART 1
TROUBLESHOOTING ENGINE OIL PRESSURE GAUGES (SENECA IV)

Trouble	Cause	Remedy
Excessive error at zero.	Pointer loose on shaft. Overpressure or seasoning of bourdon tube (if applicable).	Replace instrument.
Excessive scale error.	Improper calibration adjustment.	Replace instrument.
Excessive pointer oscillation.	Air in line (if applicable) or rough engine relief.	Disconnect line and fill with light oil (if applicable). Check for leaks (if applicable). If trouble persists, clean and adjust relief valve.
	Gauge malfunction.	Check gauge is properly grounded.
		Replace gauge.
Sluggish operation of pointer, or pressure fails to build up.	Engine relief valve open.	Clean and check.
NOTE: Gauge will take longer to indicate in cold weather.	Line restriction to instrument (if applicable).	Clean and check.
	Transducer (if applicable) gauge.	Replace transducer or or gauge malfunction.
	Loss of oil in engine, or other engine failure.	Shut down engine. See “Chart 2” on page 71003.

2. Engine Oil Temperature (Seneca IV)

NOTE: For information on the engine oil temperature gauge installed in Seneca V models see 77-40-00.

A. Description

On Seneca IV models, the oil temperature gauge is part of the combination engine gauge which also includes the oil pressure gauge and the cylinder head temperature gauge. Each oil temperature gauge provides a temperature indication through a temperature bulb-mounted in the left side of its respective engine.

B. Troubleshooting

See “Chart 2”.

CHART 2
TROUBLESHOOTING ENGINE OIL TEMPERATURE GAUGES (SENECA IV)

Trouble	Cause	Remedy
Instrument fails to show any reading.	Broken or damaged bulb, or open wiring.	Check engine unit and wiring to gauge.
Excessive scale error.	Improper calibration adjustment.	Check calibration. Repair or replace.
Pointer fails to move as engine is warmed up.	Broken or damaged bulb, or open wiring.	Check engine unit and wiring.
Dull or discolored marking.	Age.	Replace instrument.

CHAPTER

80

STARTING

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CHAPTER 80 - STARTING

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CRANKING

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

1. Description and Operation

Each starting motor consists of five major components; the commutator and head assembly, brush set and plate assembly, armature, and the drive end head assembly. When the starting circuit is energized, battery current is applied to the starting motor terminal. As current flows through the field coils, a strong magnetic field is created. At the same time, current flows through the brushes to the commutator through the armature windings to ground. The magnetic force created in the armature combined with that in the field windings turns the armature.

The starters are located on the back of each engine. Each starter is connected to the engine accessory gear box through a 90° gear box. Access to the starter must be made by removing the upper cowl panels.

2. Troubleshooting

See “Chart 1” on page 80102

3. Starter

See “Figure 1” on page 80104.

A. 100 Hour Inspection

The starting circuit should be inspected at regular intervals, the frequency of which should be determined by the amount of service and the condition under which the aircraft is operated. It is recommended that such inspection be made at each 100 hours and include the following:

(1) Battery

The battery should be checked with a hydrometer to be sure it is fully charged and filled to the proper level with approved water. A load test should be made to determine battery condition. If dirt and corrosion have accumulated on the battery, it should be cleaned with a solution of baking soda and water. Be sure none of the solution enters the battery cells.

(2) Voltage Loss Test

The starting circuit wiring should be inspected to be sure that all connections are clean and tight, and that the insulation is sound. A voltage loss test should be made to locate any high resistance connections that would affect starting motor efficiency. This test is made with a low reading voltmeter while cranking the engine or at approximately 100 amperes, and the following limits should be used:

(a) Voltage loss from insulated battery post to starting motor terminal -0.3-volt maximum.

(b) Voltage loss from battery ground post to starter frame -0.1-volt maximum.

NOTE: If voltage loss is greater than the above limits, additional tests should be made over each part of the circuit to locate the high resistance connections.

NOTE: If a solderless terminal on an aluminum cable is loose, corroded or otherwise unsatisfactory, it is recommended that the complete cable assembly be replaced instead of replacing or repairing the solderless terminal. Should replacement of the complete assembly not be practical, it is permissible to replace the aluminum cable assembly with a copper cable assembly which is two sizes smaller (EX: An AL-1 aluminum cable assembly is replaced with an AN-3 copper cable assembly). The new cable should be installed in accordance with AC-43-13-2A.

CHART 1 (Sheet 1 of 2)
TROUBLESHOOTING STARTER

Trouble	Cause	Remedy
Starter fails to operate.	Low battery charge.	Check and recharge if necessary.
	Defective or improper wiring or loose connections.	Refer to electrical wiring diagram and check all wiring.
	Defective starter solenoid or control switch.	Replace faulty unit.
	Binding, worn, or improperly seated brush, or brushes with excessive side play.	Brushes should be a free fit in the brush boxes without excessive side play. Binding brushes and brush boxes should be wiped clean with a gasoline (undoped) moistened cloth. A new brush should be run in until at least 50% seated; however, if facilities are not available for running in brushes, then the brush should be properly seated by inserting a strip of number 000 sandpaper between the brush and commutator, with the sanded part next to the brush. Pull sandpaper in the direction of rotation, being careful to keep it in the same contour as the commutator.
CAUTION: DO NOT USE COARSE SANDPAPER OR EMERY CLOTH. AFTER SEATING, CLEAN THOROUGHLY TO REMOVE ALL SAND AND METAL PARTICLES TO PREVENT EXCESSIVE WEAR. KEEP MOTOR BEARING FREE FROM SAND OR METAL PARTICLES.		
	Dirty commutator.	If commutator is rough or dirty, smooth and polish with number 0000 sandpaper. If too rough and pitted, remove and turn down. Blow out all particles.
	Shorted, grounded, or an open armature.	Remove and replace with armature known to be in good condition.
	Grounded or open field circuit.	Test, repair if possible or replace with a new part.

CHART 1 (Sheet 2 of 2)
TROUBLESHOOTING STARTER

Trouble	Cause	Remedy
Low motor and cranking speed.	Worn, rough, or improperly lubricated gearing.	Disassemble, clean, inspect, and relubricate, replacing motor or starter ball bearings if worn.
	Same electrical causes as listed under “Motor Fails to Operate.”	Same remedies listed for these troubles.
Excessive arcing of motor brushes.	Binding. worn or improperly seated brush. or brushes with excessive side play.	See previous page for information dealing with this trouble.
	Dirty commutator rough pitted or scored.	Clean as outlined above.
Excessive wear and arcing of motor brushes.	Rough or scored commutator.	Remove and turn commutator down on a lathe.
	Armature assembly not concentric.	Reface commutator.

(3) Lubrication

No lubrication is required on the starting motor except at the time of overhaul. Soak new absorbent bronze bearings in SAE 20 oil before installation. Saturate the felt oiling pad in the commutator end head with SAE 20 oil. Allow excess oil to drain out before installing end head on motor. Put a light film on Lubriplate 777 on the drive end of the armature shaft before and after installing the drive end head.

(4) Starter Drive Check

The starting motor should be operated for a few seconds with the ignition switch off. This is to determine that the starter engages properly and that it turns freely without binding or excessive noise. Start the engine two or three times to check the starter drive assembly.

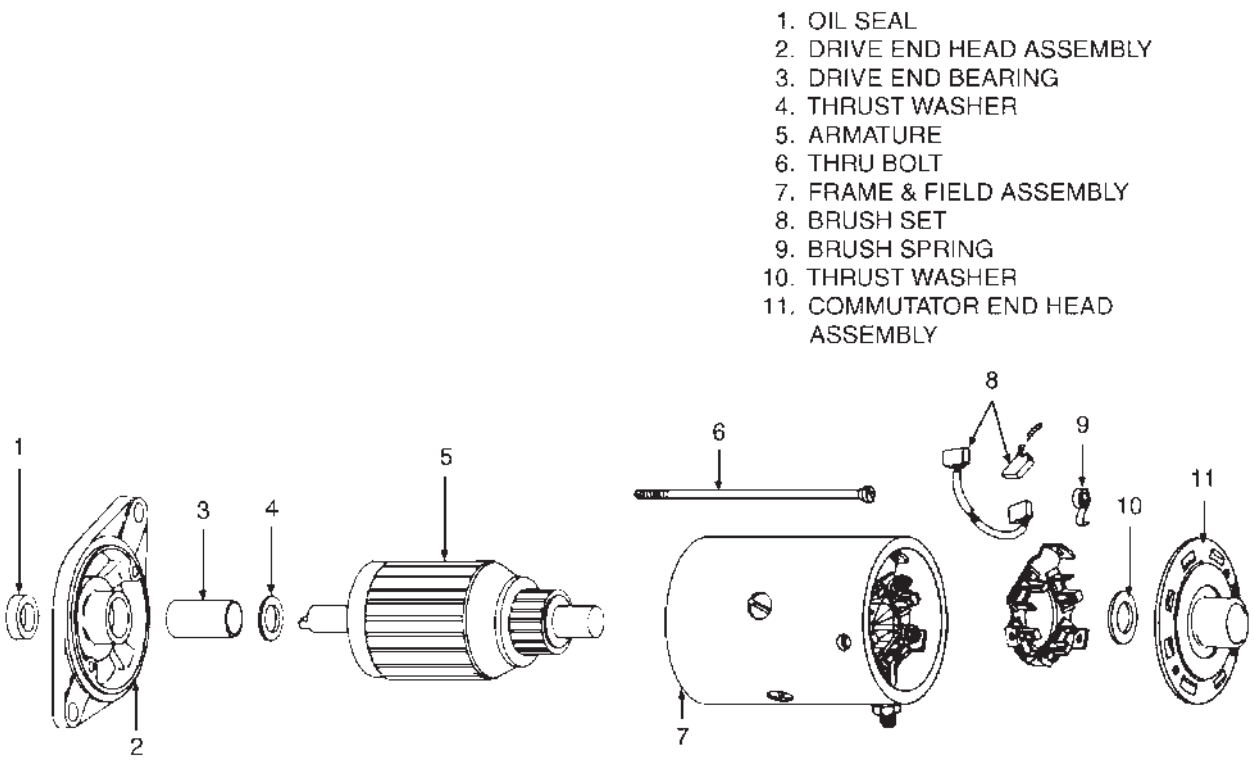
NOTE: Refer to the engine manufacturer’s service information concerning the starter drive mechanism.

B. Removal

To remove the starting motor from the engine, first disconnect the ground cable from the battery post to prevent short circuiting. Disconnect the lead from the starting motor terminal, then remove retaining nuts from studs. The motor can then be pulled off and taken to the bench for overhaul.

C. Overhaul

If during the above inspection any starting motor difficulty is noted, the starting motor should be removed from the engine for cleaning and repair. Refer to Continental Service Manual, Starting Motor section.



Exploded View of Starting Motor (Typical)
Figure 1

D. Starting Motor Service Test Specifications

See “Chart 2”.

CHART 2
STARTING MOTOR TEST SPECIFICATIONS (TCM 6427)

No-Load Test (75°F)	
Volt	20
Amps	30-50
RPM	4300

CHAPTER

81

TURBINES

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CHAPTER 81 - TURBINES

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TURBO-SUPERCHARGER

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

1. Description and Operation

On Seneca IV models, the turbocharger system (“Figure 2” on page 81206 Sheet 1 of 2) consists of a turbine and compressor assembly, ground adjustable exhaust bypass screw, and the necessary hoses and engine air intake ducts. The ground adjustable exhaust bypass screw allows exhaust gas to bypass the turbine and flow directly overboard. In the closed position, the bypass screw diverts the exhaust gases into the turbine. The turbocharger requires little attention between overhauls. However, the items outlined in the Inspection Report, 5-20-00, must be checked periodically.

On Seneca V models, the turbocharger system (“Figure 2” on page 81207 Sheet 2 of 2) consists of a turbine and compressor assembly, hydraulic wastegate and controller, and the necessary hoses and engine air intake ducts. The hydraulic wastegate allows exhaust gas to bypass the turbine and flow directly overboard. The turbocharger requires little attention between overhauls. However, the items outlined in the Inspection Report, 5-20-00, must be checked periodically.

2. Troubleshooting

See “Chart 1” on page 81202.

3. Nomenclature

Many unfamiliar terms may appear on the following pages of this manual. An understanding of these will be helpful, if not necessary, in performing maintenance and troubleshooting. “Chart 2” on page 81203 is a list of commonly used terms and names as applied to turbocharging and a brief description.

4. Turbocharger

CAUTION: BEFORE REMOVING, CLEAN THE ENTIRE TURBOCHARGER, AIR PIPING, AND OIL LINE CONNECTIONS WITH A STIFF BRUSH OR WHISK BROOM, FOLLOWED BY WIPING WITH A CLOTH DAMPENED WITH CLEANING SOLVENT. THIS PRECAUTION IS NECESSARY TO PREVENT ENTRANCE OF FOREIGN MATERIALS INTO THE ENGINE AND TURBOCHARGER SYSTEM AFTER REMOVAL.

A. Removal

(1) Remove the Upper Cowling.
(2) Disconnect the oil supply and return lines from the center section of the turbo.
(3) Disconnect the air ducts from the compressor inlet and outlet and the exhaust system from the turbine inlet and outlet.
(4) Remove the bolts that attach the turbocharger to the mounting bracket and remove the turbocharger assembly.

CHART 1
TROUBLESHOOTING TURBOCHARGER

Trouble	Cause	Remedy
Smoking engine exhaust, loss of engine power, low boost pressure.	Dirty air cleaner, undersize air cleaner.	Clean or replace air cleaner as required.
	Restricting intake manifold or piping.	Remove restriction.
	Foreign matter or dirt accumulation on impeller.	Clean impeller. See “Major Inspection and Cleaning.”
	Damaged impeller or turbine wheel.	Rebuild unit.
	Excessive oil leakage from seals.	Rebuild unit.
	Leaking intake or exhaust manifold connections.	Tighten all connections and replace gaskets where required.
	Excess back-pressure on turbine outlet.	Reduce restriction in exhaust ducting.
Noisy rotating assembly.	Damaged bearing or other components, causing rotating assembly to rub against housing.	Rebuild unit.
Excess oil in intake manifold or exhaust stack.	Excessive oil leakage from from seals.	Rebuild unit.
Boost pressure low, power low, clean exhaust.	Insufficient fuel supply to engine.	Check fuel system or reset fuel pump.
Engine knock (gasoline).	Improper fuel.	Use recommended fuel.
	Oil leakage from compressor seal.	Rebuild unit.
	Ignition timing incorrect.	Reset to specified timing.

CHART 2 (Sheet 1 of 2)
TURBO-SUPERCHARGER NOMENCLATURE

Term	Definition
Supercharge	To increase the air pressure (density) above or higher than ambient conditions.
Supercharger	A device that accomplishes the increase in pressure.
Turbo-supercharger	More commonly referred to as a “Turbocharger” this device is driven by a turbine. The turbine is spun by energy extracted from the engine exhaust gas.
Compressor	The portion of a turbocharger driven by the turbine that takes in ambient air and compresses it before discharging it to the engine.
Turbine	The exhaust driven end of the turbocharger unit.
Ground Boosted or Ground Turbocharged	These phrases indicate that the engine depends on a certain amount of turbocharging at sea level to produce the advertised horsepower. An engine that is so designed will usually include a lower compression ratio to avoid detonation.
Deck Pressure	The pressure measured in the area downstream of the turbo compressor discharge and upstream of the engine throttle valve. This should not be confused with manifold pressure.
Manifold Pressure	The pressure measured downstream of the engine throttle valve and is almost directly proportional to the engine power output.
Normalizing	If a turbocharger system is used only to regain power losses caused by decreased air pressure of high altitude, it is considered that the engine has been “normalized.”
Overboost	An overboost condition means that manifold pressure is exceeding the limits at which the engine was tested and FAA certified, and can be detrimental to the life and performance of the engine. Overboost can be caused by malfunctioning controllers or improperly operating wastegate in the automatic system, or by pilot error in a manual controlled system.
Overshoot	Overshoot is a condition of the automatic controls not having the ability to respond quickly enough to check the inertia of the turbocharger speed increase with rapid engine throttle advance. Overshoot differs from overboost in that the high manifold pressure lasts only for a few seconds. This condition can usually be overcome by smooth throttle advance. A good method for advancing the throttle is as follows: After allowing the engine oil to warm up to approximately 140°F, advance the throttle to 28” to 30” manifold pressure, hesitate 1 to 3 seconds and continue advancing to full throttle slow and easy. This will eliminate any overshoot due to turbocharger inertia.

CHART 2 (Sheet 2 of 2)
TURBO-SUPERCHARGER NOMENCLATURE

Term	Definition
Bootstrapping	This is a term used in conjunction with turbo machinery. If you were to take all the air coming from a turbocharger compressor and duct it directly back into the turbine of the turbocharger, it would be called a bootstrap system and if no losses were encountered, it would theoretically run continuously. It would also be very unstable because if for some reason the turbo speed would change, the compressor would pump more air to drive the turbine faster, etc. A turbocharged engine above critical altitude is similar to the example mentioned above, except now there is an engine placed between the compressor discharge and turbine inlet. Slight system changes cause the exhaust gas to change slightly, which causes the turbine speed to change slightly, which causes the compressor air to the engine to change slightly, which in turn again affects the exhaust gas, etc.
Critical Altitude	Altitude at which the engine can no longer maintain rated horsepower.

B. Installation

(1) Seneca IV

NOTE: The turbocharger is properly installed with the large diameter center clamp loose to allow rotation of the turbine housing during installation for proper alignment with the exhaust system inlet connection. Tighten clamp after alignment.

(a) Position turbocharger assembly on the mounting bracket and secure with attaching hardware.

(b) Align exhaust system manifold to the turbine inlet and secure with mounting bolts.

WARNING: WHEN TIGHTENING ANY OF THE THREE V-BAND CLAMPS, TAP THE CLAMP ALL AROUND ITS CIRCUMFERENCE TO ENSURE PROPER SEATING. DO NOT RELY ON TIGHTENING ALONE FOR PROPER CLAMP SEATING.

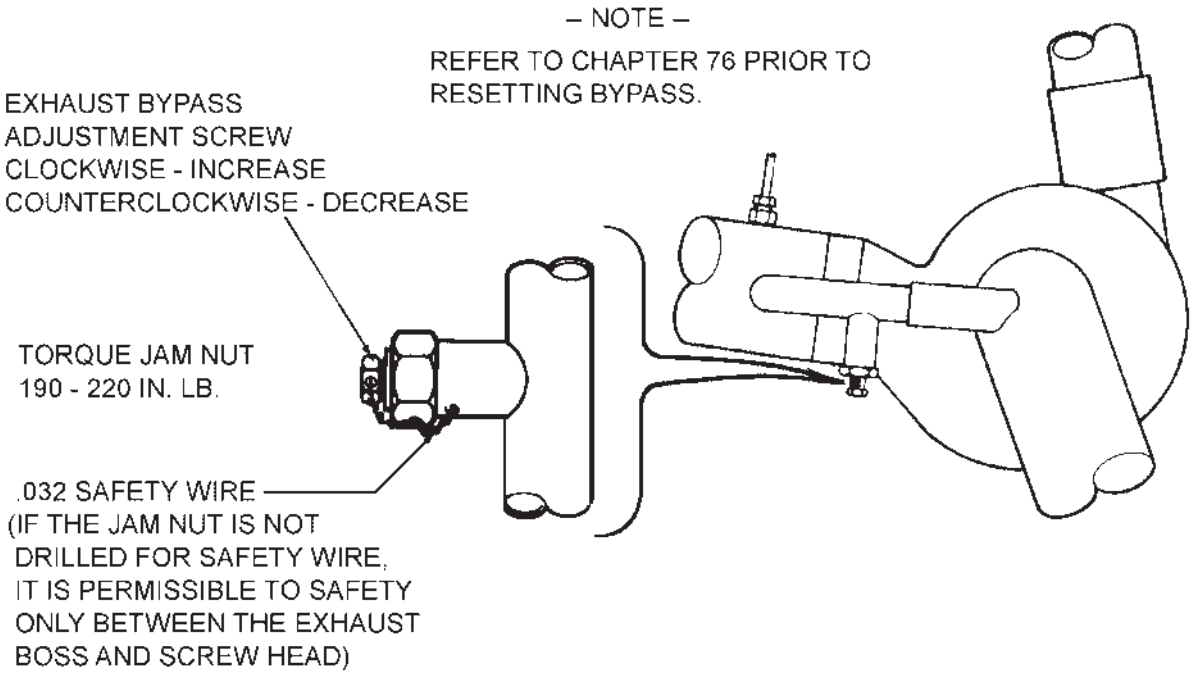
(c) Tighten the large diameter center clamp securing the turbine housing to the turbocharger.

(d) Place turbine housing insulation blanket in proper position and safety blanket to turbocharger attaching hardware.

(e) Position the exhaust tail pipe and exhaust bypass screw to the turbine outlet, aligning the tail pipe with the hole cut out in the lower cowl provided for it.

NOTE: Check position of the exhaust bypass adjustment screw (“Figure 1” on page 81205). If 8 minimum, 9 maximum threads are showing below jam nut, no adjustment is required.

(f) Tighten both turbine housing inlet clamps. (Refer to previous warning on tightening V-band clamps.)



Exhaust Bypass Screw (Seneca IV)
Figure 1

(g) Position the engine induction tube to turbocharger compressor outlet connector, and the induction air inlet tube to the turbocharger compressor inlet connector, and tighten the clamps.

(h) If previously removed, install the overboost valve assembly as follows:

1) Install a new O-ring on the overboost mounting flange of the induction tube.

2) Position the overboost valve assembly on the mounting flange, with the holes in the valve aligning with the holes in the flange.

3) Install the four bolts and secure with plain washers and self-locking nuts.

(i) Connect the oil supply and return lines to the turbocharger center housing. Connect the oil pressure cockpit gauge line if it was previously disconnected.

(j) Perform engine ground run-up and check for normal engine functioning, excessive exhaust manifold leakage and oil leaks. Repair as necessary.

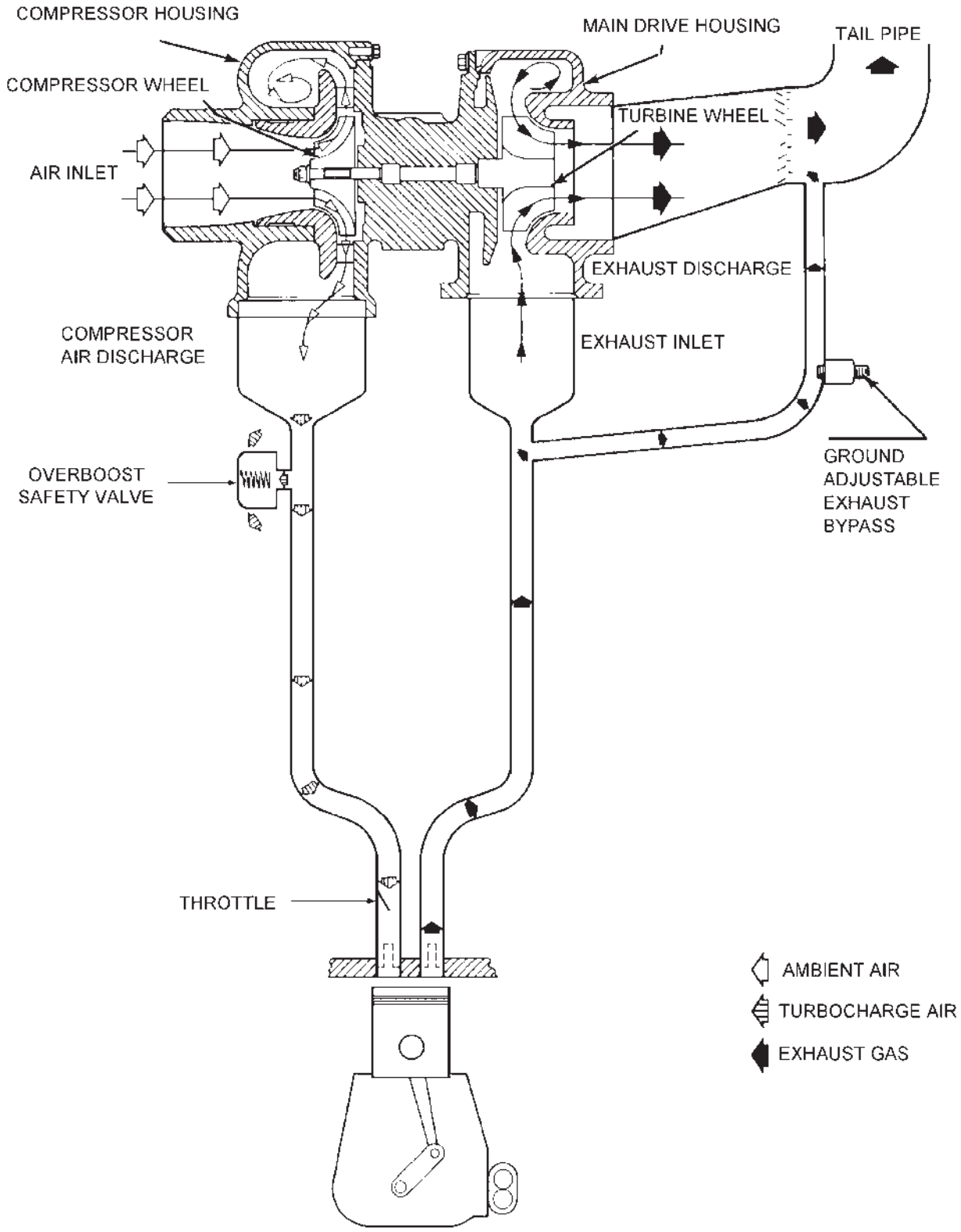
(k) Install the upper cowling.

(2) Seneca V

NOTE: The turbocharger is properly installed with the large diameter center clamp loose to allow rotation of the turbine housing during installation for proper alignment with the exhaust system inlet connection. Tighten clamp after alignment.

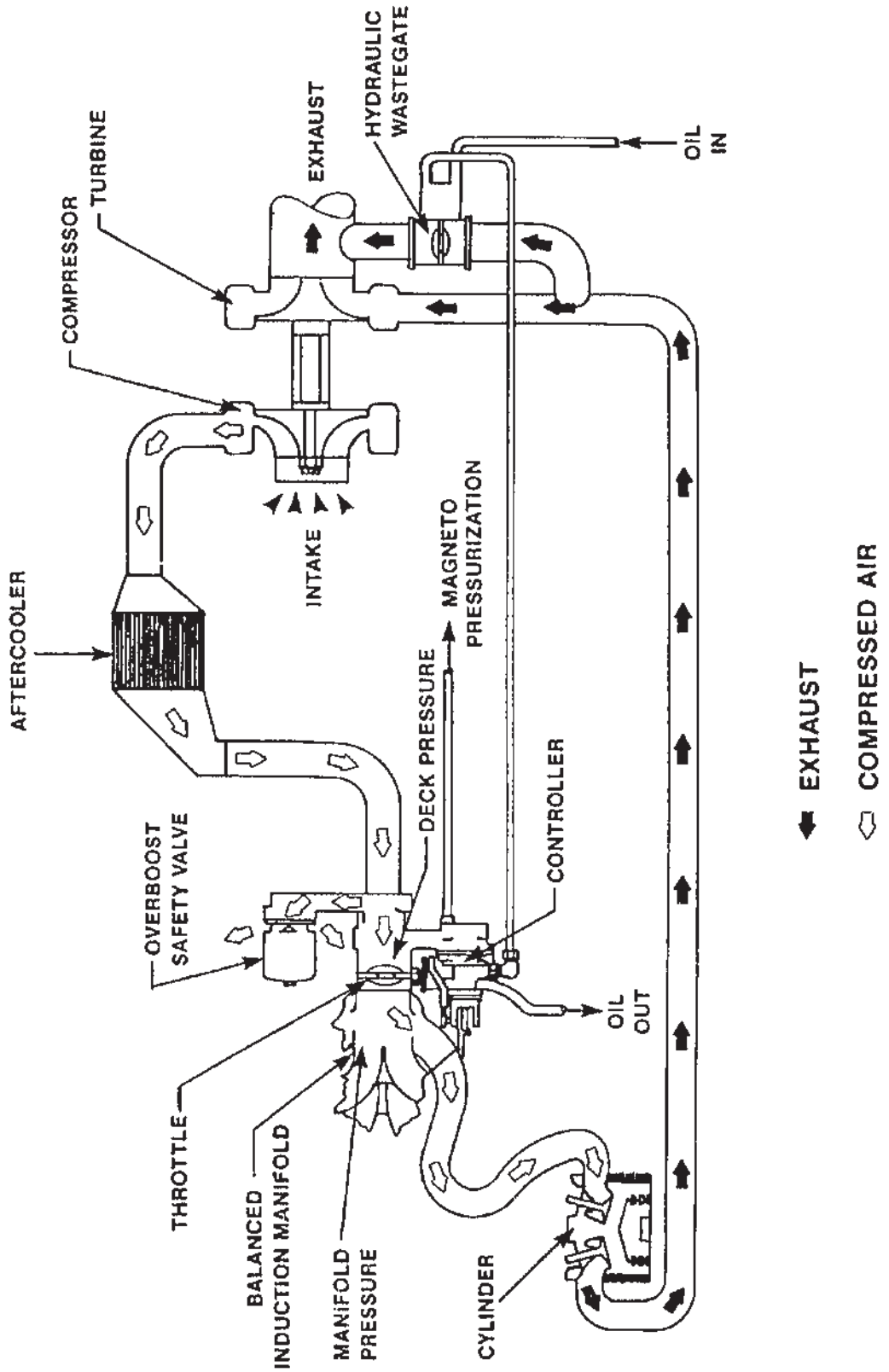
(a) Verify that the turbocharger compressor and turbine housings are properly oriented to the center housing and locked.

(b) Position turbocharger assembly on the mounting bracket and secure with attaching hardware.



Effectivity
Seneca IV

Turbocharger System Functional Schematic
Figure 2 (Sheet 1 of 2)



Turbocharger System Functional Schematic
Figure 2 (Sheet 2 of 2)

Effectivity
Seneca V

(c) Align exhaust system manifold to the turbine inlet and secure with mounting bolts.

WARNING: WHEN TIGHTENING ANY OF THE THREE V-BAND CLAMPS, TAP THE CLAMP ALL AROUND ITS CIRCUMFERENCE TO ENSURE PROPER SEATING. DO NOT RELY ON TIGHTENING ALONE FOR PROPER CLAMP SEATING.

(d) Position the exhaust tail pipe to the turbine outlet, aligning the tail pipe with the hole cut out in the lower cowl provided for it.

(e) Position the engine induction air duct to turbocharger compressor outlet connector, and the induction air inlet duct to the turbocharger compressor inlet connector, and tighten the clamps.

(f) Connect the oil supply and return lines to the turbocharger center housing.

(g) Perform engine ground run-up and check for normal engine functioning, excessive exhaust manifold leakage and oil leaks. Repair as necessary.

(h) Install the upper cowling.

C. Inspection

(1) Routine Inspection

Whenever routine service of the engine is performed, inspect the turbocharger as follows.

(a) Inspect the hoses and tubing of the air intake system between the air cleaner and turbocharger, and from the turbocharger to the intake manifold. Check for leakage due to cracks, damaged gaskets, loose clamps or connections, and restrictions due to kinks, collapsed hoses, or dented tubing.

(b) Inspect impeller blades on inlet and discharge side.

(c) Inspect for exhaust leakage from a cracked exhaust manifold, damaged gaskets, or loose turbocharger mounting.

(d) Inspect the oil lines and fittings for kinks, damage, and leakage. (On Seneca V models, include wastegate and controller.)

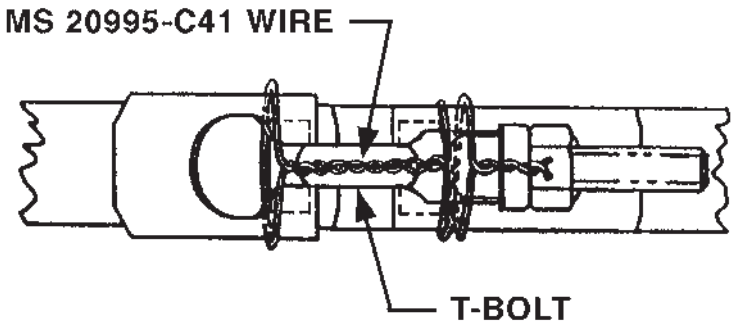
(e) Note any unusual noises or vibration which would warrant further inspection of the turbocharger.

(f) Observe engine exhaust. Excessive smoke may indicate a restricted air cleaner or intake piping, over fueling, or faulty turbocharger operation.

(2) V-Band Coupling 100 Hour Inspection

See “Figure 3” on page 81209.

Each 100 hours, inspect lockwiring on V-band couplings for condition and security. If lockwiring is found broken, inspect T-bolt for stretching, cracking, or any other damage. Replace coupling as required.



Lockwiring V-Band Couplings
Figure 3

(3) 1000 Hour Inspection

WARNING: ON SENECA IV MODELS, MAJOR INSPECTION AND CLEANING MUST BE COMPLETED BY REFERENCING THE APPLICABLE RAJAY OVERHAUL MANUAL.

WARNING: ON SENECA V MODELS, MAJOR INSPECTION AND CLEANING MUST BE COMPLETED BY REFERENCING THE APPLICABLE GARRETT OVERHAUL MANUAL.

After every 1,000 hours of operation, or particularly if trouble is suspected in the turbocharger, a major inspection of the turbocharger should be performed. This requires removal of the turbocharger from the engine.

(a) Remove the air cleaner piping from the turbocharger compressor housing inlet. Observe the condition of the impeller and housing. Carefully check the leading edges of the impeller blades for damage and for evidence of interference with the compressor housing.

(b) Disconnect the oil lines and the intake manifold piping from the turbocharger. Support the turbocharger and remove the turbine housing clamp that secures the turbine housing to the bearing house assembly. Remove the turbocharger from the turbine housing, leaving the turbine housing mounted on the engine. Cover all openings to prevent the entrance of foreign materials.

(c) Inspect the turbine wheel for cracks, erosion, nicked blade tips, and broken or missing blades. Inspect the turbine shield for warpage, rubbing, scoring, and erosion. Check for accumulation of carbon behind the turbine wheel and check for other defects that could interfere with proper turbocharger operation.

NOTE: The shield must be depressed against the tension of the spring ring to check for free rotation.

(d) If the turbine and impeller do not rotate freely when the turbine shield is depressed away from the turbine wheel, the parts may be damaged or there may be interference due to foreign material. These conditions will necessitate the disassembly of the turbocharger for inspection.

D. Adjustment

See 76-10-00, Seneca IV, “Engine Setup Procedures” on page 76102 and Seneca V, “Engine Setup Procedures” on page 761013

NOTE: A complete inspection of the power plant system should be performed before any turbo adjustments are made.

5. Overboost Valve

A. Removal

(1) Remove the four self-locking nuts, plain washers and bolts.

(2) Lift the overboost valve assembly from the induction tube (Seneca IV) or from the induction wing assembly (Seneca V).

(3) Remove the O-ring from the seating surface of the overboost mounting flange on the induction tube.

B. Installation

(1) Install a new O-ring on the overboost mounting flange of the induction tube.

(2) Position the overboost valve assembly on the mounting flange with the holes in the valve aligning with the holes in the flange.

(3) Install the four bolts and secure with plain washers and self-locking nuts.

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91

CHARTS AND WIRING DIAGRAMS

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CHARTS

1. Torque Requirements

   See 20-00-00.

2. Conversion Tables

   The following charts contain various conversion data that may be useful when figuring capacities, lengths, temperatures, and various weights and measures from the English system to the metric system or back again:

   Torque Conversion – “Chart 1” on page 91102

   Decimal Conversions – “Chart 2” on page 91103

   Temperature Conversion – “Chart 3” on page 91104

   Weights and Measures Conversion – “Chart 4” on page 91105

   Metric Conversion – “Chart 5” on page 91106

   Drill Sizes – “Chart 6” on page 91107

3. Hose Specifications

   See “Chart 7” on page 91108.

4. Consumable Materials

   General – “Chart 8” on page 91109

   Sealants – “Chart 9” on page 911017

5. Vendor Contact Information

   See “Chart 10” on page 911021.

6. Electrical Wire Coding

   See “Chart 11” on page 911025.

7. Electrical Symbols

   See “Chart 12” on page 911028.

8. Electrical / Electronic Component Reference Designation Codes

   See “Chart 13” on page 911030.

CHART 1
TORQUE CONVERSION

INCH-POUNDS (IN-LB) TO CENTIMETER KILOGRAMS (CMKG)
CENTIMETER KILOGRAMS (CMKG) TO INCH-POUNDS (IN-LB)
FOOT-POUNDS (FT-LB) TO METER KILOGRAMS (MKG)
METER KILOGRAMS (MKG) TO FOOT-POUNDS (FT-LB)

IN-LB	CMKG
5	5.76
10	11.52
15	17.28
20	23.04
25	28.80
30	34.56
35	40.32
40	46.08
45	51.84
50	57.60
55	63.36
60	69.12
65	74.88
70	80.64
75	86.40
80	92.16
85	97.92
90	103.68
95	109.44
100	115.20
105	120.96
110	126.72
115	132.48
120	138.24

CMKG	IN-LB
50	43.4
100	86.8
150	130.2
200	173.6
250	217.0
300	260.4
350	303.8
400	347.2
450	390.6
500	434.0
550	477.4
600	520.8
650	564.2
700	607.6

FT-LB	MKG
2.5	.346
5	.691
7.5	1.037
10	1.383
12.5	1.728
15	2.074
17.5	2.419
20	2.765
22.5	3.111
25	3.456
27.5	3.802
30	4.148
32.5	4.493
35	4.839
37.5	5.185
40	5.530
42.5	5.876
45	6.222
47.5	6.567
50	6.913
52.5	7.258
55	7.604
57.5	7.950
60	8.295
62.5	8.641
65	8.987
67.5	9.332
70	9.678
72.5	10.024
75	10.369
77.5	10.715
80	11.060
82.5	11.406
85	11.752
87.5	12.097
90	12.443
92.5	12.789
95	13.134
97.5	13.480
100	13.826
105	14.517
110	15.208

FT-LB	MKG
115	15.900
120	16.591
125	17.282
130	17.974
135	18.665
140	19.356
145	10.047
150	20.739
155	21.430
160	22.121
165	22.813
170	23.504
175	24.195
180	24.887
185	25.578
190	26.269
195	26.960
200	27.652
105	28.343
210	29.034
215	29.726
220	30.417
225	31.108
230	31.800
235	32.491
240	33.182
245	33.873
250	34.565
255	35.256
260	35.947
265	36.639
270	37.330
275	38.021
280	38.713
285	39.404
290	40.095
295	40.786
300	41.478

MKG	FT-LB
1	7.23
2	14.46
3	21.69
4	28.98
5	36.16
6	43.39
7	50.63
8	57.86
9	65.09
10	72.32
11	79.56
12	86.79
13	94.02
14	101.26
15	108.49
16	115.72
17	122.95
18	130.19
19	137.42
20	144.65
21	151.89
22	159.12

CHART 2
DECIMAL CONVERSIONS

4ths	8ths	16ths	32nds	64ths	TO 3 PLACES	TO 2 PLACES	M.M. EQUIV
				1/64	.016	.02	.397
			1/32		.031	.03	.794
				3/64	.047	.05	1.191
		1/16			.062	.06	1.587
				5/64	.078	.08	1.984
			3/32		.094	.09	2.381
				7/64	.109	.11	2.778
	1/8				.125	.12	3.175
				9/64	.141	.14	3.572
			5/32		.156	.16	3.969
				11/64	.172	.17	4.366
		3/16			.188	.19	4.762
				13/64	.203	.20	5.159
			7/32		.219	.22	5.556
				15/64	.234	.23	5.593
1/4					.250	.25	6.350
				17/64	.266	.27	6.747
			9/32		.281	.28	7.144
				19/64	.297	.30	7.540
		5/16			.312	.31	7.937
				21/64	.328	.33	8.334
			11/32		.344	.34	8.731
				23/64	.359	.36	9.128
	3/8				.375	.38	9.525
				25/64	.391	.39	9.922
			13/32		.406	.41	10.319
				27/64	.422	.42	10.716
		7/16			.438	.44	11.112
				29/64	.453	.45	11.509
			15/32		.469	.47	11.906
				31/64	.484	.48	12.303
					.500	.50	12.700

4ths	8ths	16ths	32nds	64ths	TO 3 PLACES	TO 2 PLACES	M.M. EQUIV
				33/64	.516	.52	13.097
			17/32		.531	.53	13.494
				35/64	.547	.55	13.891
		9/16			.562	.56	14.288
				37/64	.578	.58	14.684
			19/32		.594	.59	15.081
				39/64	.609	.61	15.478
	5/8				.625	.62	15.875
				41/64	.641	.64	16.272
			21/32		.656	.66	16.669
				43/64	.672	.67	17.065
		11/16			.688	.69	17.462
				45/64	.703	.70	17.859
			23/32		.719	.72	18.256
				47/64	.734	.73	18.653
3/4					.750	.75	19.050
				49/64	.766	.77	19.447
			25/32		.781	.78	19.844
				51/64	.797	.80	20.241
		13/16			.812	.81	20.637
				53/64	.828	.83	21.034
			27/32		.844	.84	21.431
				55/64	.859	.86	21.828
	7/8				.875	.88	22.225
				57/64	.891	.89	22.622
			29/32		.906	.91	23.019
				59/64	.922	.92	23.416
		15/16			.938	.94	23.812
				61/64	.953	.95	24.209
			31/32		.969	.97	24.606
				63/64	.984	.98	25.003
					1.000	1.00	25.400

CHART 3
TEMPERATURE CONVERSION

CENTIGRADE - FAHRENHEIT

Example: To convert 20°C to Fahrenheit, find 20 in the center column with the heading °F - °C, then read 68.0°F in the column to the right (°F). To convert 20°F to Centigrade, find 20 in the center column and read -6.67°C in the column to the left (°C).

°C	°F - °C	°F	°C	°F - °C	°F
-56.7	-70	-94.0	104.44	220	428.0
-51.1	-60	-76.0	110.00	230	446.0
- 45.6	-50	-58.0	115.56	240	464.0
- 40.0	-40	-40.0	121.11	250	482.0
- 34.0	-30	-22.0	126.67	260	500.0
- 28.9	-20	-4.0	132.22	270	518.0
- 23.3	-10	14.0	137.78	280	536.0
- 17.8	0	32.0	143.33	290	554.0
-12.22	10	50.0	148.89	300	572.0
-6.67	20	68.0	154.44	310	590.0
-1.11	30	86.0	160.00	320	608.0
4.44	40	104.0	165.56	330	626.0
10.00	50	122.0	171.11	340	644.0
15.56	60	140.0	176.67	350	662.0
21.11	70	158.0	182.22	360	680.0
26.67	80	176.0	187.78	370	698.0
32.22	90	194.0	193.33	380	716.0
37.78	100	212.0	198.89	390	734.0
43.33	110	230.0	204.44	400	752.0
48.89	120	248.0	210.00	410	770.0
54.44	130	266.0	215.56	420	788.0
60.00	140	284.0	221.11	430	806.0
65.56	150	302.0	226.67	440	824.0
71.00	160	320.0	232.22	450	842.0
76.67	170	338.0	237.78	460	860.0
82.22	180	356.0	243.33	470	878.0
87.78	190	374.0	248.89	480	896.0
93.33	200	392.0	254.44	490	914.0
98.89	210	410.0	260.00	500	932.0

CHART 4
WEIGHTS AND MEASURES CONVERSION

MULTIPLY	BY	TO OBTAIN
CENTIMETERS	0.3937	IN.
	0.03281	FT.
CU. CENTIMETERS	0.001	LITERS
	0.06102	CU. IN.
	0.0002642	U.S. GAL.
CU. FT.	28.320	CU. CM.
	1.728	CU. IN.
	7.481	U.S. GAL.
	28.32	LITERS
CU. IN.	16.39	CU. CM.
	0.01639	LITERS
	0.004329	U.S. GAL.
	0.01732	QUARTS
CU. METERS	1000000	CU. CM.
	35.314	CU. FT.
	61.023	CU. IN.
	264.17	GAL.
	999.97	LITERS
FEET	0.3048	METERS
	12.000	MILS
	304.8	MM.
	0.3333	YARDS
FT-LB	0.1383	M-KG
	0.001285	BTU
	0.000000376	KW-HR
FLUID OZ.	8	DRAM
	29.6	CU. CM.
GAL., IMPERIAL	277.4	CU. IN.
	1.201	U.S. GAL.
	4.546	LITERS
GAL., U.S. DRY	268.8	CU. IN.
	0.1556	CU. FT.
	1.164	U.S. GAL., LIQ.
	4.405	LITERS
GAL., U.S. LIQ.	231.0	CU. IN.
	0.1337	CU. FT.
	3.785	LITERS
	0.8327	IMPERIAL GAL.
	128	FLUID OZ.
IN.	2.540	CM.
	.08333	FT.
JOULES	0.000948	BTU
	0.7376	FT-LB

MULTIPLY	BY	TO OBTAIN
KILOGRAMS	2.205	LB
	35.27	OZ
	1000	GRAMS
LITERS	1000	CU. CM.
	61.03	CU. IN.
	0.03532	CU. FT.
	0.2642	U.S. GAL.
	0.22	IMPERIAL GAL.
	1.057	QUARTS
METERS	39.37	IN.
	3.281	FT
	1000	MM
METER-KILOGRAM	7.233	FT-LB
	9.807	JOULES
OUNCES, AVDP	0.0625	LB, AVDP
	28.35	GRAMS
	437.5	GRAINS
OUNCES, FLUID	29.57	CU. CM.
	1.805	CU. IN.
LB, AVDP	453.6	GRAMS
	7000	GRAINS
	16.0	OUNCES
SQUARE INCH	6.4516	SQ. CM.
POUND PER SQUARE INCH (PSI)	0.0703	KG-CM SQUARED
STATUTE MILE	1.609	KILOMETER
	0.8684	NAUTICAL MILE
NAUTICAL MILE	1.151	STATUTE MILE
QUART	.9463	LITER
MILLIMETER	1000	MICRON
MICRON	0.001	MILLIMETER
	0.000039	INCH
INCH POUNDS	11.521	METER GRAMS
INCH OUNCES	0.72	METER GRAMS
POUNDS	0.453	KILOGRAMS

CHART 5
METRIC CONVERSION

Example: Convert 1.5 inches to millimeters.

(1) Read down inches column to 1. inches.
(2) Read across top inch column to 0.5.
(3) Read down and across to find millimeters (1.5 inches is 38.10 millimeters).

INCHES TO MILLIMETER

INCHES	0.0000	0.0001	0.0002	0.0003	0.0004	0.0005	0.0006	0.0007	0.0008	0.0009
	MILLIMETER									
0.000		0.0025	0.0050	0.0076	0.0101	0.0127	0.0152	0.0177	0.0203	0.0228
0.001	0.0254	0.0279	0.0304	0.0330	0.0355	0.0381	0.0406	0.0431	0.0457	0.0482
0.002	0.0508	0.0533	0.0558	0.0584	0.0609	0.0635	0.0660	0.0685	0.0711	0.0736
0.003	0.0762	0.0812	0.0838	0.0863	0.0889	0.0914	0.0939	0.0965	0.0965	0.0990
0.004	0.1016	0.1041	0.1066	0.1092	0.1117	0.1143	0.1168	0.1193	0.1219	0.1244
0.005	0.1270	0.1295	0.1320	0.1346	0.1371	0.1397	0.1422	0.1447	0.1447	0.1498
0.006	0.1524	0.1549	0.1574	0.1600	0.1625	0.1651	0.1676	0.1701	0.1727	0.1752
0.007	0.1778	0.1803	0.1828	0.1854	0.1879	0.1905	0.1930	0.1955	0.1981	0.2006
0.008	0.2032	0.2057	0.2082	0.2108	0.2133	0.2159	0.2184	0.2209	0.2235	0.2260
0.009	0.2286	0.2311	0.2336	0.2362	0.2387	0.2413	0.2438	0.2463	0.2489	0.2514

INCHES	0.000	0.001	0.002	0.003	0.004	0.005	0.006	0.007	0.008	0.009
	MILLIMETER									
0.00		0.025	0.050	0.076	0.101	0.127	0.152	0.177	0.203	0.228
0.01	0.254	0.279	0.304	0.330	0.355	0.381	0.406	0.431	0.457	0.482
0.02	0.508	0.533	0.558	0.584	0.609	0.635	0.660	0.685	0.711	0.736
0.03	0.762	0.787	0.812	0.838	0.863	0.889	0.914	0.939	0.965	0.990
0.04	1.016	1.041	1.066	1.092	1.117	1.143	1.168	1.193	1.219	1.244
0.05	1.270	1.295	1.320	1.346	1.371	1.397	1.422	1.447	1.473	1.498
0.06	1.524	1.549	1.574	1.600	1.625	1.651	1.676	1.701	1.727	1.752
0.07	1.778	1.803	1.828	1.854	1.879	1.905	1.930	1.955	1.981	2.006
0.08	2.032	2.057	2.082	2.108	2.133	2.159	2.184	2.209	2.235	2.260
0.09	2.286	2.311	2.336	2.362	2.387	2.413	2.438	2.463	2.489	2.514

INCHES	0.00	0.01	0.02	0.03	0.04	0.05	0.06	0.07	0.08	0.09
	MILLIMETER									
0.0		0.254	0.508	0.762	0.016	1.270	1.524	1.778	2.032	2.286
0.1	2.540	2.794	3.048	3.302	3.556	3.810	4.064	4.318	4.572	4.826
0.2	5.080	5.334	5.558	5.842	6.096	6.350	6.604	6.858	7.112	7.366
0.3	7.620	7.874	8.128	8.382	8.636	8.890	9.144	9.398	9.652	9.906
0.4	10.160	10.414	10.668	10.922	11.176	11.430	11.684	11.938	12.192	12.446
0.5	12.700	12.954	13.208	13.462	13.716	13.970	14.224	14.478	14.732	14.986
0.6	15.240	15.494	15.748	16.002	16.256	16.510	16.764	17.018	17.272	17.526
0.7	17.780	18.034	18.288	18.542	18.796	19.050	19.304	19.558	19.812	20.066
0.8	20.320	20.574	20.828	21.082	21.336	21.590	21.844	22.098	22.352	22.606
0.9	22.860	23.114	23.368	23.622	23.876	24.130	24.384	24.638	24.892	25.146

INCHES	0.00	0.1	0.2	0.3	0.4	0.5	0.6	0.7	0.8	0.9
	MILLIMETER									
0.0		2.54	5.08	7.62	10.16	12.70	15.24	17.78	20.32	22.86
1.0	25.40	27.94	30.48	33.02	35.56	38.10	40.64	43.18	45.72	48.26
2.0	50.80	53.34	55.88	58.42	60.96	63.50	66.04	68.58	71.12	73.66
3.0	76.20	78.74	81.28	83.82	86.36	88.90	91.44	93.98	96.52	99.06
4.0	101.60	104.14	106.68	109.22	111.76	114.30	116.84	119.38	121.92	124.46
5.0	127.00	129.54	132.08	134.62	137.16	139.70	142.24	144.78	147.32	149.86
6.0	152.40	154.94	157.48	160.02	162.56	165.10	167.64	170.18	172.72	175.26
7.0	177.80	180.34	182.88	185.42	187.96	190.50	193.04	195.58	198.12	200.66
8.0	203.20	205.74	208.28	210.82	213.36	215.90	218.44	220.98	223.52	226.06
9.0	228.60	231.14	233.68	236.22	238.76	241.30	243.84	246.38	248.92	251.46

CHART 6
DRILL SIZES

Decimal/Millimeter Equivalents of Drill Sizes From 1/2" to No. 80

Size	Decimal Equiv.	Millimeter Equiv.	Size	Decimal Equiv.	Millimeter Equiv.	Size	Decimal Equiv.	Millimeter Equiv.	Size	Decimal Equiv.	Millimeter Equiv.
1/2	0.500	12.7000	G	0.261	6.6294	5/32	0.1562	3.9687	51	0.067	1.7018
31/64	0.4843	12.3031	F	0.257	6.5278	23	0.154	3.9116	52	0.0635	1.6129
15/32	0.4687	11.9062	E-1/4	0.250	6.3500	24	0.152	3.8608	1/16	0.0625	1.5875
29/64	0.4531	11.5094	D	0.246	6.2484	25	0.1495	3.7973	53	0.0595	1.5113
7/16	0.4375	11.1125	C	0.242	6.1468	26	0.147	3.7338	54	0.055	1.397
27/64	0.4218	10.7156	B	0.238	6.0452	27	0.144	3.6576	55	0.052	1.3208
Z	0.413	10.4902	15/64	0.2343	5.9531	9/64	0.1406	3.5719	3/64	0.0468	1.1906
13/32	0.4062	10.3187	A	0.234	5.9436	28	0.1405	3.5687	56	0.0465	1.1811
Y	0.404	10.2616	1	0.228	5.7912	29	0.136	3.4544	57	0.043	1.0922
X	0.397	10.0838	2	0.221	5.6134	30	0.1285	3.2639	58	0.042	1.0668
25/64	0.3906	9.9212	7/32	0.2187	5.5562	1/8	0.125	3.1750	59	0.041	1.0414
W	0.386	9.8044	3	0.213	5.4102	31	0.120	3.048	60	0.040	1.016
V	0.377	9.5758	4	0.209	5.3086	32	0.116	2.9464	61	0.039	0.9906
3/8	0.375	9.5250	5	0.2055	5.2197	33	0.113	2.8702	62	0.038	0.9652
U	0.368	9.3472	6	0.204	5.1816	34	0.111	2.8194	63	0.037	0.9398
23/64	0.3593	9.1262	13/64	0.2031	5.1594	35	0.110	2.794	64	0.036	0.9144
T	0.358	9.1281	7	0.201	5.1054	7/64	0.1093	2.7781	65	0.035	0.899
S	0.346	8.7884	8	0.199	5.0546	36	0.1065	2.7051	66	0.033	0.8382
11/32	0.3437	8.7300	9	0.196	4.9784	37	0.104	2.6416	1/32	0.0312	0.7937
R	0.339	8.6106	10	0.1935	4.9149	38	0.1015	2.5781	67	0.032	0.8128
Q	0.332	8.4328	11	0.191	4.8514	39	0.0995	2.5273	68	0.031	0.7874
21/64	0.3281	8.3337	12	0.189	4.8006	40	0.098	2.4892	69	0.029	0.7366
P	0.323	8.2042	3/16	0.1875	4.7625	41	0.096	2.4384	70	0.028	0.7112
O	0.316	8.0264	13	0.185	4.699	3/32	0.0937	2.3812	71	0.026	0.6604
5/16	0.3125	7.9375	14	0.182	4.6228	42	0.0935	2.3749	72	0.025	0.635
N	0.302	7.6708	15	0.180	4.572	43	0.089	2.2606	73	0.024	0.0696
19/64	0.2968	7.5387	16	0.177	4.4958	44	0.086	2.1844	74	0.0229	0.58166
M	0.295	7.4930	17	0.173	4.3942	45	0.082	2.0828	75	0.021	0.5334
L	0.290	7.3660	11/64	0.1718	4.3656	46	0.081	2.0574	76	0.020	0.508
9/32	0.2812	7.1425	18	0.1695	4.3053	47	0.0785	1.9939	77	0.018	0.4572
K	0.281	7.1374	19	0.166	4.2164	5/64	0.0781	1.9844	1/64	0.0156	0.3969
J	0.277	7.0358	20	0.161	4.0894	48	0.076	1.9304	78	0.016	0.4064
I	0.272	6.9088	21	0.159	4.0386	49	0.073	1.8542	79	0.0145	0.3683
H	0.266	6.7564	22	0.157	3.9878	50	0.070	1.778	80	0.0135	0.3429
17/64	0.2656	6.7462									

DRILL SIZES AVAILABLE

Drill may be obtained in regular sizes to a 4 inch diameter, and increase in 64th of an inch. The regular metric drills vary from 2 to 76mm and increase in 0.5mm variations.

CHART 7
HOSE SPECIFICATIONS

SINGLE WIRE BRAID FABRIC COVERED

MIL PART NO.	TUBE SIZE O.D.	HOSE SIZE I.D.	HOSE SIZE O.D.	RECOMMENDED OPER. PRESS	MIN BURST PRESS	MAX PROOF PRESS	MIN BEND RADIUS
MIL-H-8794- 3-L	3/16	1/8	.45	3,000	12,000	6,000	3.00
MIL-H-8794- 4-L	1/16	3/16	.52	3,000	12,000	6,000	3.00
MIL-H-8794- 5-L	5/16	1/4	.58	3,000	10,000	5,000	3.38
MIL-H-8794- 6-L	3/8	5/16	.67	2,000	9,000	4,500	4.00
MIL-H-8794- 8-L	1/2	13/32	.77	2,000	8,000	4,000	4.63
MIL-H-8794-10-L	5/8	1/2	.92	1,750	7,000	3,500	5.50
MIL-H-8794-12-L	3/4	5/8	1.08	1,500	6,000	3,000	6.50
MIL-H-8794-16-L	1	7/8	1.23	800	3,200	1,600	7.38
MIL-H-8794-20-L	1 1/4	1 1/8	1.50	600	2,500	1,250	9.00
MIL-H-8794-24-L	1 1/2	1 3/8	1.75	500	2,000	1,000	11.00
MIL-H-8794-32-L	2	1 13/16	2.22	300	1,400	700	13.25
MIL-H-8794-40-L	2 1/2	2 3/8	2.88	200	1,000	300	24.00
MIL-H-8794-48-L	3	3	3.56	200	800	300	33.00

Construction: Seamless synthetic rubber inner tube reinforced with one fiber braid, one braid of high tensile steel wire and covered with an oil resistant rubber impregnated fiber braid.

Identification: Hose is identified by specification number, size number, quarter year and year, hose manufacturer's identification.

Uses: Hose is approved for use in aircraft hydraulic, pneumatic, coolant, fuel and oil systems.

Operating Temperatures:

Sizes -3 thru -12: Minus 65°F. to plus 250°F.

Sizes -16 thru -48: Minus 40°F to plus 275°F.

NOTE: Maximum temperatures and pressures should not be used simultaneously.

MULTIPLE WIRE BRAID RUBBER COVERED

MIL PART NO.	TUBE SIZE O.D.	HOSE SIZE I.D.	HOSE SIZE O.D.	RECOMMENDED OPER. PRESS	MIN BURST PRESS	MAX PROOF PRESS	MIN BEND RADIUS
MIL-H-8788- 4-L	1/4	7/32	.63	3,000	16,000	8,000	3.00
MIL-H-8788- 5-L	5/16	9/32	.70	3,000	14,000	7,000	3.38
MIL-H-8788- 6-L	3/8	11/32	.77	3,000	14,000	7,000	5.00
MIL-H-8788- 8-L	1/2	7/16	.86	3,000	14,000	7,000	5.75
MIL-H-8788-10-L	5/8	9/16	1.03	3,000	12,000	6,000	6.50
MIL-H-8788-12-L	3/4	11/16	1.22	3,000	12,000	6,000	7.75
MIL-H-8788-16-L	1.00	7/8	1.50	3,000	10,000	5,000	9.63

Hose Construction: Seamless synthetic rubber inner tube reinforced with one fabric braid, two or more steel wire braids, and covered with a synthetic rubber cover (for gas applications, request perforated cover).

Identification: Hose is identified by specification number, size number, quarter year and year, hose manufacturer's identification.

Uses: High pressure hydraulic, pneumatic, coolant, fuel and oil.

Operating Temperature:

Minus 65°F to plus 200°F.

CHART 8 (Sheet 1 of 7)
CONSUMABLE MATERIALS - GENERAL

Material	Specification	Product	Vendor
ABS-Solvent/ Cements		Solarite, #11 Series	Solar Compounds Corp.
Adhesive		EC 801 EC 807 EC 1357 Scotch Grip 210 (Rubber Adhesive)	3M (Minnesota Mining and Manufacturing) Adhesive Coating and Sealers Division
		Hysol EA 9446 (Piper P/N 279-129)	Hensel Corp.
Adhesive, Deice Boot		EC-1300L (Piper P/N 179-929)	3M
Adhesive, Fuel Barrier		A-851-B (Piper P/N 279-137)	Goodrich Corp.
Anti-Galling Solution	MIL-PRF-907	Ease-Off	Taxacone Company
Anti-Seize Compound (Graphite Petrolatum)	SAE-AMS-2518	Armite Product	Armite Laboratories
		Anti-Seize Compound	Exxon Oil Company
		Royco 44	Royal Lubricants Co.
Anti-Seize Compound	TT-A-580 (TT-S-1732)	Armite Product	Armite Laboratories
Anti-Seize Thread Compound “HIGH TEMPERATURE”		Fel-Pro C5-A	Fel-Pro Incorporated
Cleaner and Polish, Plexiglas	P-P-560	Part Number 403D	Permatex Co., Inc. Kansas City, Kansas 66115
Cleaners, General		Fantastic Spray Perchlorethylene VM&P Naphtha (Lighter Fluid)	Local Supplier
Cloth, Clean	SAE-AMS-3819	Cloth, Class 1 or 2 Grade A (Piper P/N 280-686)	Local Supplier
Compound, Buffing and Rubbing		Automotive Type Axalta #7	Axalta Coating Systems
		Ram Chemical #69	Ram Chemicals
Compound, Polishing		Mirror Glaze	Mirror Bright Polish Co., Incorporated

CHART 8 (Sheet 2 of 7)
CONSUMABLE MATERIALS - GENERAL

Material	Specification	Product	Vendor
Corrosion Retardant Compounds	MIL-PRF-16173E	LPS-3 Heavy Duty Rust Inhibitor (Piper P/N 197-508)	LPS Laboratories
		Metal Parts Protector Protector Flex (Piper P/N 197-509)	Chemi-Cap Chemical Packaging Corp.
Deicer Boot Surface Coatings		Agemaster	Goodrich Corp.
Dry Lubricant		MS-122AD	Miller-Stephenson
Filler (Edge)		AC-665 B½ (Piper P/N 279-065) WS8070 B½ Type II (Piper P/N 279-472)	Royal Adhesives and Sealants
Gasket Cement		Permatex No. 2	Permatex Company, Inc.
Grease, Actuator		2196-74-1	Dukes Astronautics Co.
Grease, Aircraft Instrumentation, Gear and Actuator Screw (-73 to +121°C) (-94 to +250°F)	MIL-PRF-23827C (See Note at end.)	Supermil Grease No. A72832	Amoco
		Royco 27A	Royal Lubricants Co.
		Shell 6249 Grease	Shell Oil Company
		RR-28	Socony Mobil Oil Co.
		Castrolease A1	Burmah-Castrol LTD.
		Low-Temp. Grease E.P.	Texaco Incorp.
		5114 E.P. Grease AV55	Standard Oil of Calif.
		Aeroshell Grease 7 Braycote 627S	Shell Oil Company
		Mobilgrease 33	ExxonMobil Corporation
		B.P. Aero Grease 31B	B.P. Trading Limited
Grease, Ball and Roller Bearing	DOD-G-24508	Regal ASB-2 Formula TG-10293	Texaco Incorporated
		Andok B Code 1-20481, Darina Grease 1 XSG-6213 Code 71-501, Darina Grease 2 XSG-6152 Code 71-502, Alvania Grease 2 XSG-6151 Code 71-012, Cyprina Grease 3 XSG-6280 Code 71-003	Exxon Company, U.S.A. Shell Oil Company

CHART 8 (Sheet 3 of 7)
CONSUMABLE MATERIALS - GENERAL

Material	Specification	Product	Vendor
Grease, General Purpose Wide Temperature	MIL-PRF-81322G	Marfax All Purpose	Texaco Incorporated
		Aeroshell No. 6	Shell Oil Company
(-54 to +177 °C)		Mobilgrease 28	ExxonMobil Corporation
(-65 to +350 °F)		Royco 22	Royal Lubricants Co.
		Aeroshell No. 22	Shell Oil Company
Grease, Lubricating, Molybdenum Disulfide, Low and High Temperature	MIL-G-21164	Aeroshell Grease No. 17	Shell Oil Company
		Royco 64C	Royal Lubricants Co.
		Castrolease MSA (c)	Burmah Castrol LTD.
Grease, Plug Valve, Gasoline and Oil Resistant	SAE-AMS-G-6032	Royco 32	Royal Lubricants Co.
		Castrolease PV	Burmah Castrol LTD.
		Parker Fuel Lube 44	Parker Seal Company
		B.P. Aero Grease 32	B.P. Trading Limited
		L-237	Lehigh Tenneco Chemicals Co., Inc.
		Rockwell 950	Rockwell International
Grease, Waterproof, High and Low Temperature		Mobilgrease 33 (Piper P/N 279-657)	ExxonMobil Corporation
“Hot Melt” Polyamide Adhesives and “Hot Melt” Guns	Stick form 1/2 in. diameter, 3 in. long		Local supplier; most hardware or arts and crafts stores.

CHART 8 (Sheet 4 of 7)
CONSUMABLE MATERIALS - GENERAL

Material	Specification	Product	Vendor
Hydraulic Fluid	MIL-PRF-5606	Brayco 756D	Bray Oil Company
		TL-5874	Texaco Incorporated
		PED 3565	Standard Oil Company of California
		Aircraft Hydraulic Oil AA	Texaco Incorporated
		RPM Aviation Oil No. 2 Code PED 2585 PED 3337	Standard Oil Company of California
		Aeroshell Fluid 4, SL-7694	Shell Oil Company
		Mobil Aero HF	ExxonMobil Corporation
		Royco 756, 756A and 756B	Royal Lubricants Co.
Isopropyl Alcohol	Fed. Spec. TT-I-735		Local Supplier
Isocryl Tape		(PMS-C1012-2)	Schnee Moorehead Chemicals, Incorporated
Kevlar		Kevlar	Kevlar Special Products
Leak Detector Solution for Oxygen Systems	MIL-PRF-25567	ALPHA 73 Oxygen Leak Detector Type 1	U.S. Gulf Corporation
		Leak Tec #16-OX	American Gas and Chemical Co. Ltd.
Loctite	ASTM-D-5363	Loctite 290 (Red)	Loctite Corporation
		Loctite 222 (Brown)	
		Loctite 27121 (10 ml) (Piper P/N 279-128)	
Masking Tape 1”		Scotch No. 2308 or equivalent	3M
Methyl Ethyl Ketone	Fed. Spec. TT-M-261		Local Supplier
Molybdenum Disulfide	SAE-AMS-M-7866	Molykote-Type G (Paste)	Dow Corning Corp.
		Molykote - Type 2 (Powder)	

CHART 8 (Sheet 5 of 7)
CONSUMABLE MATERIALS - GENERAL

Material	Specification	Product	Vendor
Oil, Air Conditioner, R12		Frigidaire #525	Virginia Chemical
		Suniso #5	Sun Oil Company of Pennsylvania
		Texaco Capilla “E”	Texaco Incorporated
Oil, Air Conditioner, HFC-134a, POE		Ester-25065 (Piper P/N 197-511)	
Oil Lubricating, General Purpose, Low Temperature	MIL-PRF-7870C	Caltex Low Temp. Oil	Caltex Oil Products Company
		Sinclair Aircraft Orbit Lube	Sinclair Refining Company
		1692 Low Temp Oil	Texaco Incorporated
		Aviation Instrument Oil	Standard Oil Company of California
		Royco 363	Royal Lubricants Co.
Pad, Abrasive, Type A		3M Scotch-Brite 7447 (Piper P/N 279-235)	3M
Patching Compound Epoxy		Solarite #400	Solar Compounds Corp.
Primer, Fluid Resistant Epoxy		EWDE072A/B (Piper P/N 279-179)	PPG Aerospace PRC-DeSoto
		10P8-10NF / EC-283 (Piper P/N 279-506)	Akzo Nobel Aerospace Coatings
		10P30-5 / EC-275 (Piper P/N 279-108)	
		44GN036	PPG Aerospace
		Axalta Epoxy 13550S (Piper P/N 279-240)	Axalta Coating Systems
Primer, Non-Chromated Corrosion Inhibiting		Aeroglaze 9743A (Piper P/N 279-456)	Lord Corporation
		Aeroglaze 9700B (Piper P/N 279-455)	
Rain Repellent	FSCM 50150	Repcon	Unelco Corporation
Refrigerant, A/C HFC-134a	1,1,1,2-Tetrafluoroethane and/or CAS# 811-97-2		Local Supplier
Safety Walk Pressure Sensitive		Flextred 300	Wooster Products, Incorporated

CHART 8 (Sheet 6 of 7)
CONSUMABLE MATERIALS - GENERAL

Material	Specification	Product	Vendor
Sealant	See “Chart 9”.		
Sealer		PR 1321 B-1/2	Products Research Company
Sealer, Deice Boot		A-56-B-R1 (Piper P/N 279-142)	Goodrich Corp.
Sealing Compound, Gasket and Joint		Tite-Seal	Radiator Specialty Co.
Silicone Compound -54 to +204 °C -65 to +400 °F	SAE-AS-8660	DC-4, DC-6 Compound (Piper P/N 279-149)	Dow Corning
		G-624	General Electric Co. Silicone Products Department
Solvents		Methyl Ethyl Ketone	Local Supplier
		Methylene Chloride	
	ASTM D329	Acetone (Piper P/N 193-035)	
		Y2900	Union Carbide; Plastic Division
	Fed. Spec. PD 680 Type I - Stoddard Solvent		Local Supplier
	Type II - High Temperature		Local Supplier
Solvent, Cleaning, Propeller Slip Ring		CRC-2-26	Corrosion Reaction Consultants, Inc.
Thinner (Toulene)		ADC 108-88-3 (Piper P/N 179-451)	Ashland
Teflon Tape	.003” x .5” wide/-1		3M
			Shamban W.S. and Co.
	.003” x .25” wide/-2		Johnson & Johnson, Inc. Permacel Division

CHART 8 (Sheet 7 of 7)
CONSUMABLE MATERIALS - GENERAL

Material	Specification	Product	Vendor
Thread Lubricant, Oleo Strut, Air Valve	MIL-PRF-907E	Kopr-Kote	Jet Lube, Inc.
Thread Sealant for High Pressure Oxygen System	A-A-58092	Permacel 412	Johnson & Johnson, Inc. Permacel Division
Toluol	TT-M-261		Local Supplier
Trichlorethylene	MIL-T-7003	Perm-A-Clor	Dextrex Chemical Industries, Inc.
		Turco 4217	Turco Products, Inc.
Vinyl Foam	1 in. x 1/8 in.	530 Series, Type I	Norton Tape Division
Vinyl, Foam Tape	1/8 in. x 1 in.	501 Series, Type II	Norton Tape Division
Vinyl, Black Plastic	2 in. x 9 mil. and/or 1 1/2 in. x 9 mil.		

NOTE: Take precautions when using MIL-PRF-23827 and engine oil. These lubricants contain chemicals harmful to painted surfaces.

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CHART 9 (Sheet 1 of 3)
CONSUMABLE MATERIALS - SEALANTS

Sealant Type	Manufacturer Product (Specific Gravity & Cured Hardness)	Class/ Worklife (hrs)	Typical Time (Hrs) at 77°F/50%RH		Piper Code		
			Tack Free	Cure to 30A	2.5 oz Semco	6 oz Semco	Gallon Can Kit
Type 1 Fuel Tank and Fuselage Pressurized Structure See Note 1 and see also Type 6.	3M AC-350 (SG=1.40, ≥48A)	A-2	8	12	279-518	279-185	279-524
		B-½	2-3	2-3	279-519	279-186	---
		B-2	7-8	7-8	279-520	279-187	---
		B-4	32-36	32-36	279-521	279-183	279-525
	Flamemaster CS3204 (SG=1.66, ≥55A)	A-2	24	72	279-526	279-213	279-532
		B-½	8	30	279-527	279-217	---
		B-2	24	72	279-528	279-212	---
		B-4	36	90	279-529	279-220	279-533
	Royal WS-8020 (SG=1.52, ≥50A)	A-2	<24	<48	279-822	279-821	279-810
		B-½	<10	<24	279-534	279-557	---
		B-2	<24	<48	279-535	279-548	---
		B-4	<36	<90	279-823	279-824	279-807
	Naftoseal MC-780 (SG=1.10, ≥45A)	B-4	<16	<24	279-696 (2.0 oz)	279-696 (4.6 oz)	---
	Royal WS-8020RC (SG=1.45, ≥50A)	A-2	<10	<30	279-536	179-202	179-209
		B-½	<5	<5	279-537	179-203	---
		B-2	<9	<14	279-538	179-204	---
		B-4	<24	<26	279-539	179-205	179-210
	PPG PR-1440 (SG=1.57, ≥46A)	A-2	<36	72	279-540	279-545	279-544
		B-½	<4	8	279-541	279-546	---
		B-2	<8	11	279-542	279-122	---
Type 1LD Fuel Tank and Fuselage Pressurized Structure See Note 2	Royal WS-8032 (SG=1.10, ≥48A)	B-½	<4	<6	279-428	279-432	---
		B-2	<6	<9	279-429	279-433	---
	PPG PR-2007 (SG=1.10, ≥45A)	B-½	4	6	279-426	279-430	---
		B-2	<6	8	279-427	279-431	---

(PIR-PPS45012-1 Rev. BB.)

(1) Use accelerator below if desired.
(2) Low density. Use accelerator below if desired.

CHART 9 (Sheet 2 of 3)
CONSUMABLE MATERIALS - SEALANTS

Sealant Type	Manufacturer Product & Class (Specific Gravity & Cured Hardness)	Worklife/ Assembly Time (hrs)	Typical Time (Hrs) at 77°F/50%RH		Piper Code
			Tack Free	Cure to 30A	
Type 2 Access Doors and Panels in Pressurized Structure See Note 3	Flamemaster CS3330ci B-2 (SG=1.50, 35A)	2	18	30	279-820
	PPG PR-1428 B-2 (SG=1.50, 50A)	2	<8	24	279-036
	Royal WS8010 B-1/2 (SG=1.50, 53A)	30 min	<3	<4	279-497
	Royal WS8010 B-2 (SG=1.50, 53A)	2	<6	<12	279-496
Type 3 Fill and Drain Coating in Small Tanks	3M EC-776SR (SG=0.86)	-	20 min	24	279-229
	Flamemaster CS3600 (SG=0.88)	-	20 min	24-48	279-547
	PPG PR-1005-L (SG=0.85)	-	20 min	48	179-750
Type 4, Group I General Purpose Silicone See Note 4	Momentive RTV102 -White (SG=1.05, 30A)	-	20 min	24	179-758
	Momentive RTV103 - Black (SG=1.05, 30A)	-	20 min	24	179-759
	Momentive RTV108 - Clear (SG=1.05, 30A)	-	20 min	24	179-760
Type 4, Group III Engine Compartment Silicone See Note 5	Momentive RTV106 -Red (SG=1.07, 30A)	-	20 min	24	179-763
Type 5 Wet Fastener Installation in Pressurized Structure See Note 6	3M AC-350 C-8(24) (SG=1.40, 50A)	8(24)	96	96	279-523
	Royal WS -8032 C-8 (SG=1.17, 50A)	8(24)	<168	<168	279-550
	PPG P/S 890 C-8(20) (SG=1.54, 45A)	8(20)	<168	<168	279-531
	PPG PR-1440 C-20 (SG=1.57, 46A)	8(20)	<168	<168	279-192
	PPG PR-1776 C-8 (SG=1.29, 48A)	8(24)	<120	120	279-551
Type 6 Fuel Tank and Fuselage Pressurized Structure See Note 7	3M AC-250 A-1/6 (SG=1.61, 55A)	10 min	70 min	2	279-244
	3M AC-250 B-1/6 (SG=1.62, 55A)	10 min	1	2	279-245
	PPG PS 860 B-1/6 (SG=1.65, 45A)	20 min	<3	8	279-191
Type 7	Cytec DAPCO 2100 (SG=1.37, 50A)	10 min	15 min	24	279-522
	Cytec DAPCO 2200 (SG=1.37, 50A)	30 min	1	4	279-530
	Esterline Fastblock 100 (SG=1.25, 63A)	4	6	<48	279-543
Firewall See Note 8	Flamemaster CS1900 (SG=1.33, 80A)	2	<4	<60	279-226
	Momentive RTV133 (SG=1.23, 45A)	1	3-5	48	279-029
	PPG PR-812 (SG=1.33, 75A)	2	<24	48	279-549

(3) Low adhesion.
(4) Minimum exposure to fuel; not paintable.
(5) Minimal exposure to fuel, intermittent temperature up to to 600°F; not paintable.
(6) Use accelerator below if desired. AMS specifications for class C sealants contain no standard cure requirements.
(7) Short work life repairs.
(8) 15 minutes at 2,000°F.

CHART 9 (Sheet 3 of 3)
CONSUMABLE MATERIALS - SEALANTS

Sealant Type	Manufacturer Product & Class (Specific Gravity & Cured Hardness)	Worklife/ Assembly Time (hrs)	Typical Time (Hrs) at 77°F/50%RH		Piper Code
			Tack Free	Cure to 30A	
Type 8 Pressurized Windows or Deice Boot Edge Sealer	Flamemaster CS3247 B-4 (SG=1.63, 45A)	4	<36	<90	279-227
	PPG PR-1425CF B-2 (SG=1.44, 47A)	2	<8	12	279-019
	Flamemaster CS3247 B-2 (SG=1.63, 45A)	2	<24	48	279-560
Type 9 Accelerator See Note 9	PPG PR-716 Turquoise Liquid	-	<30 min	-	279-816
	PPG PR-717 Green Liquid	-	<30 min	-	279-552
	PPG P/S 815 Clear Aerosol	-	<2	-	279-553

(9) See above for applicability.

NOTE: For the shelf lives of these materials, refer to the manufacturers' recommendations.

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CHART 10 (Sheet 1 of 3)
VENDOR CONTACT INFORMATION

#

3M
3M Center
St. Paul, MN 55144
800-831-0658
https://www.3m.com/

A

Akzo Nobel Aerospace Coatings
East Water Street
Waukegan, IL 60085
847-625-3340
https://aerospace.akzonobel.com

American Gas and
Chemical Co. Ltd.
220 Pegasus Avenue
Northvale, NJ 07647
201-767-7300

Amoco Oil Co.
200 E. Randolph Drive
Chicago, IL 60601
312-856-5111

Armite Laboratories
1845-49 Randolph Street
Los Angeles, CA 90001
213-587-7744

Axalta Coating Systems
Suite 3600
2001 Market Street
Philadelphia, PA 19103
855-547-1461
https://www.axalta.com

B

BP Trading Limited
Moore Lane
Brittanic House
London E.C. 2
England

Bray Oil Company
1925 N. Marianna Avenue
Los Angeles, CA 98103
213-268-6171

Burmah - Castrol Inc.
30 Executive Avenue
Edison, NJ 08817
201-287-3140

C

California Texas Oil Corp.,
380 Madison Avenue
New York, NY 10017

Caltex Oil Products Co.
New York, NY 10020

Castrol North America, Inc.
1500 Valley Road
Wayne, NJ 07470
800-462-0835

CEE BEE Chemical Co.
9520 E. CEE BEE Drive
Box 400
Downey, CA 92041

Chemi-cap
Chemical Packaging Corp.
1100 N.W. 70th Street
Ft. Lauderdale, FL 33309
305-665-9059

ChevronTexaco Corporation
6001 Bollinger Canyon Rd.
San Ramon, CA 94583
925-842-1000

Conoco-Phillips
800-234-6603
https://www.phillips66aviation.com

Corrosion Reaction
Consultants, Inc.
Limekin Pike
Dresher, PA 19025

D

Dextrex Chemical
P. O. Box 501
Detroit, MI 48232

DINOL International
25200 Malvina
Box 1065
Warren, MI 48090

Dow Corning Corporation
Alpha Molykote Plant
64 Harvard Avenue
Stanford, CT 06902

Dukes Astronautics Co.
7866 Deering Avenue
Canoga Park, CA 91304

DuPont Company
Finishes Div.
DuPont Building
Wilmington, DE 19898
302-774-1000

E

ExxonMobil Corporation
22777 Springwoods Village Pkwy.
Spring, TX 77389
1-888-299-2883

F

Fel-Pro Incorporated
7450 N. McCormick Blvd.
Box C1103
Skokie, IL 60076
312-761-4500

G

General Electric Co.
Silicone Products Dept.
Waterford, NY 12188
518-237-3330

Goodrich Corp. - UTC Aerospace
Systems
1555 Corporate Woods Parkway
Uniontown, OH 44685
330-374-4011

H

H. S. Bancroft Corp.
One Rockhill
Industrial Park
Cherry Hill, NJ 08003
609-854-8000

I

ITW Polymers Sealants -
Corp. Headquarters
155 Harlem Avenue
Glenview IL 60025
800-878-7876
(International: 972-438-9111)
Fax: 972-554-3939
https://www.itwprobrands.com

CHART 10 (Sheet 2 of 3)
VENDOR CONTACT INFORMATION

J

Jet Lube, Inc.
P.O. Box 21258
Houston, TX 77226-1258
800-538-5823
https://www.jetlube.com

Johnson & Johnson, Inc.
Permacel Division
501 George Street
New Brunswick, NJ 08901
201-524-0400

K

Kevlar Special Products
E.I. DuPont de
Nemours & Co., Inc.
Textile Fibers
Department
Centre Road Building
Wilmington, DE 19898
302-999-3156

L

Lehigh - Tenneco Chemicals Co.
Chestertown, MD 21620
301-778-1991

Loctite Corporation
777 N. Mountain Road
Newington, CT 06111
800-243-8160
In CT 800-842-0225

LPS Laboratories
4647 Hugh Howell Rd.
Tucker, GA 30084
800-241-8334
https://www.lpslabs.com

LUBRIPLATE
Lubricants Co.
129 Lockwood Street
Newark, NJ 07105
800-733-4755
https://www.lubriplate.com

M

Miller-Stephenson
George Washington Hwy.
Danbury, CT 06810
PH: 203-743-4447
https://miller-stephenson.com

Minnesota Mining and Mfg. -
See 3M

Mirror Bright Polish Co., Inc.
Irvine Industrial Complex
P.O. Box 17177 Irvine, CA 92713
714-557-9200

Morton Inc.
7341 Anacona Ave
Garden Grove, CA 92641
724-373-2837
Fax 724-373-1913

N

Norton Tape Division
Department 6610
Troy, NY 12181
518-273-0100

P

Parker Hannifin Corp.
O-Ring Division
2360 Palumbo Drive
Lexington, KY 40509
PH: 859-269-2351
https://www.parker.com

Permatex Co., Inc.
P.O. Box 11915
Newington, CT 06111
203-527-5211

PPG Aerospace (PRC-DeSoto,
Deft, and other brands)
11601 United Street
Mojave, CA 93501
661-824-4532; 818-549-7999
https://www.ppgaerospace.com

R

Radiator Specialty Co.
P.O. Box 34689
Charlotte, NC 28234
704-377-6555

Ram Chemicals
201 E. Alondra Blvd.
Gardena, CA 90248
213-321-0710

Rockwell International
600 Grant Street
Pittsburgh, PA 15219
412-565-2000

Royal Lubricants Company
River Road
East Hanover, NJ 07936
201-887-3100

S

Shamban W.S. and Co.
1857 Centinela Avenue
Santa Monica, CA 90404
213-397-2195

Shell Oil Company
One Shell Plaza
Houston, TX 77003
713-220-6697

Sinclair Refining Co.
600 Fifth Avenue
New York, NY 10020

Socony Mobile Oil Co.
Washington 5, DC 20005

Solar Compounds Corp.
1201 W. Blancke Street
Linden, NJ 07036
201-862-2813

Standard Oil of California
225 Bush Street
San Francisco, CA 94104
415-894-7700

Sun Oil Company of Penna
5 Penn Center Plaza
Philadelphia, PA 19103
215-972-2000

CHART 10 (Sheet 3 of 3)
VENDOR CONTACT INFORMATION

T

Taxacone Company
P.O. Box 10823 TR
Dallas, TX 75208

Turco Products Inc.
24600 S. Main Street
Box 6200
Carson, CA 90749
213-835-8211

U

U.S. Gulf Corp.
P.O. Box 233
Stoney Brook, NY 11790
212-683-9221

Unelko Corporation
727 E. 110th Street
Chicago, IL 60628

Union Carbide; Plastic Div.
270 Park Avenue
New York, NY 10017
212-551-3763

V

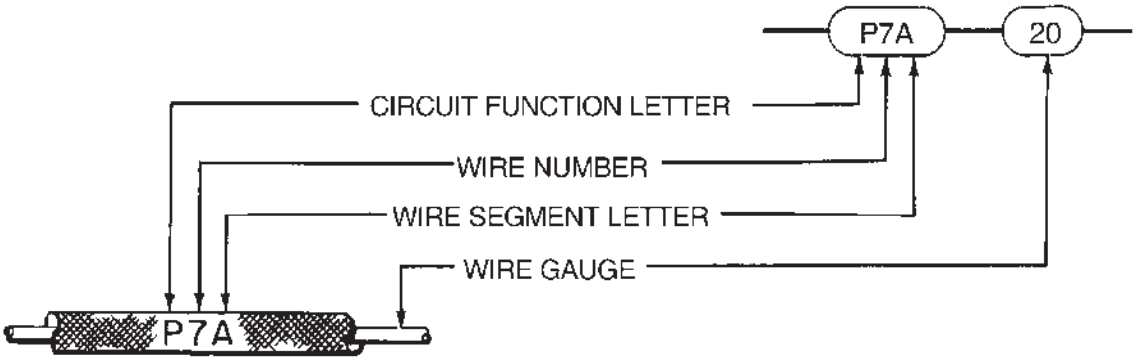
Virginia Chemical
3340 W. Norfolk Rd.
Portsmouth, VA 23703
703-484-5000

W

Wooster Products, Inc.
1000 Spruce Street
Wooster, OH 44691
800-321-4936
In OH: 216-264-2844

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CHART 11 (Sheet 1 of 3)
ELECTRICAL WIRE CODING



(PIR-PPS55006, Rev. AU)

FUNCTION LETTER (Circa 2010 and Prior)	FUNCTION LETTER (Circa 2010 and Later)	CIRCUIT
	C	Airbrakes
AC	H	Air Conditioning
	F	Airspeed Indicator
	W	Airspeed Warning
PF		Alternator Field Control and Power
	F	Altitude Indicator / Altimeter
	L	Anti-collision Lights
	F	Attitude Indicator
A	C	Autopilot
	H	Avionics Cooling
	L	Avionics Lights
	P	Battery
	H	Cabin Heating
	L	Cabin Lights
	D	Cabin Pressure Control and Indication
	H	Cabin Ventilation
	D	Clock
C		Control Surfaces
	E	DAU
	H	Deicing
	C	Electric Trim
	M	ELT Switch
K		Engine Control, Starter
	J	Engine Ignition
E		Engine Instruments
	Q	Engine Priming
	K	Engine Starting
	E	Engine Temperature
	E	Engine Torque
	M	Entertainment System
	W	Exceedance Horn
	C	Flap Position/Control
FD		Flight Director

CHART 11 (Sheet 2 of 3)
ELECTRICAL WIRE CODING

FUNCTION LETTER (Circa 2010 and Prior)	FUNCTION LETTER (Circa 2010 and Later)	CIRCUIT
	F	Flight Display / MFD
F		Flight Instruments
	E	Fuel Flow
	E	Fuel Pressure
	Q	Fuel Pumps
Q	Q	Fuel Quantity
	G	Gear Extension and Retraction
	G	Gear Position and Warning
GND		Ground
	T	Hazard Awareness (IHAS, Skywatch, etc.)
H		Heater, Ventilating
	H	Heat, Propeller
WH	H	Heat, Windshield
	D	Hours Meters
	G	Hydraulic Pump, Power, and Control “
J		Ignition
G		Landing Gear
	L	Landing Lights
L		Lighting
	E	Manifold Pressure
M		Misc. Equipment, Cigar Lighter, Hour Meter, etc.
	L	Navigation Lights
	E	Oil Pressure
Q		Oil Quantity
	E	Oil Temperature
OX	W	Oxygen
	L	Panel Lights
	H	Pitot Heat
	K	Propeller Control
	P	Power Distribution
	P	Power Generation
	P	Power Monitoring
P		Power, DC
X		Power, AC
	P	Power, External
PP		Propeller
	K	Propeller Control
	E	Propeller Overspeed Governor
	S	Radar Altimeter / Indicator
RC		Radio Cooling
RG		Radio Ground
RP		Radio Power
RZ		Radio Audio
	R	Radio, ADF
	R	Radio, Comm and Nav

CHART 11 (Sheet 3 of 3)
ELECTRICAL WIRE CODING

FUNCTION LETTER (Circa 2010 and Prior)	**FUNCTION LETTER** (Circa 2010 and Later)	**CIRCUIT**
	R	Radio, DME
	R	Radio, Headphone, Microphone and Speakers
	R	Radio, Transponder
	L	Recognition Lights
	C	Rudder Trim
	R	Satellite Radio
	W	Stall Detection and Warning
S		Stall Warning
	T	Stormscope
	L	Switch Lights
	E	Tachometer
	L	Taxi Lights
	R	Transponder
	D	Vacuum System
GB		Vent / Defogger
W		Warning
	S	Weather Radar
WH	H	Windshield Heat
WW		Windshield Wiper

Chart 12 (Sheet 1 of 2)
ELECTRICAL SYMBOLS

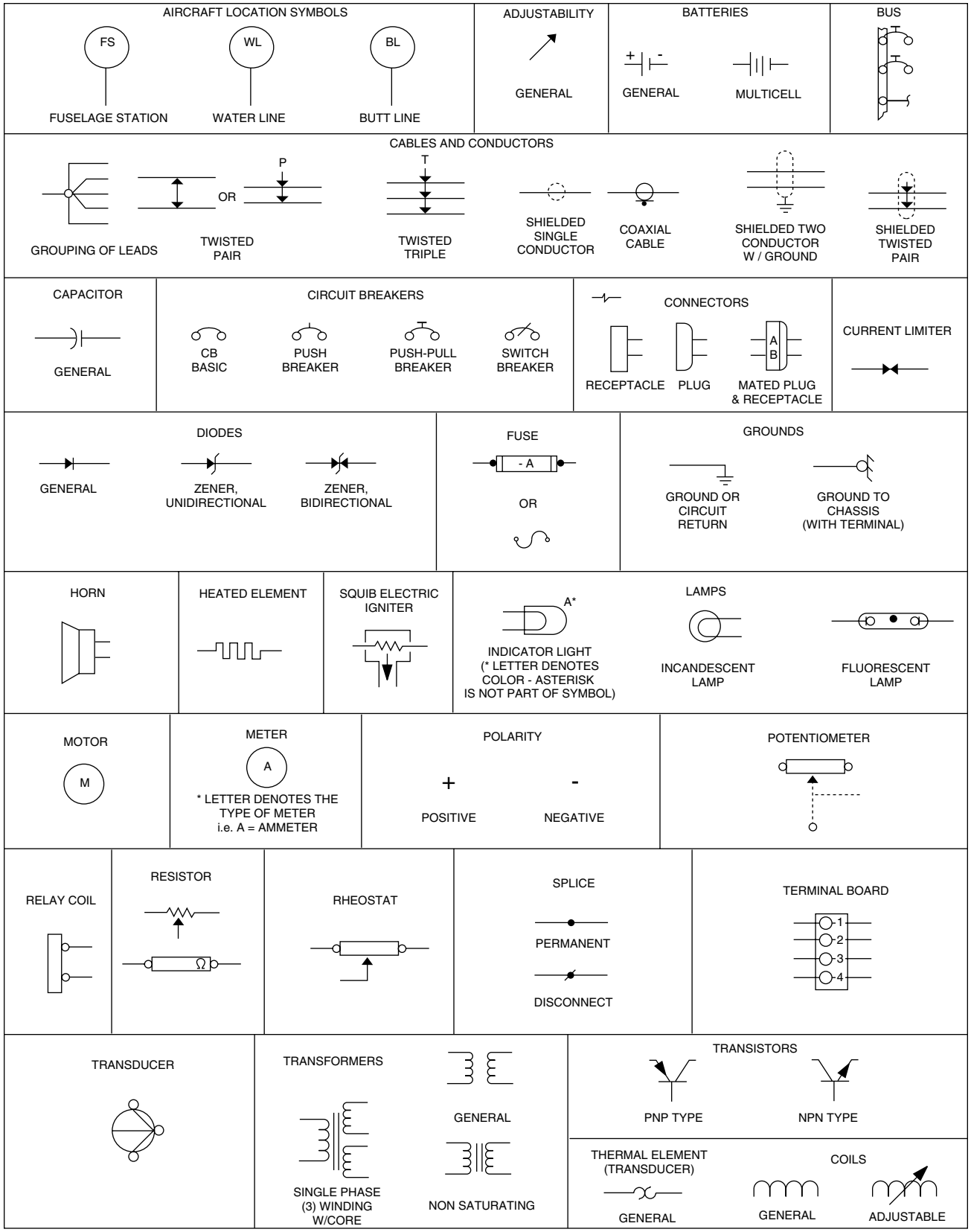
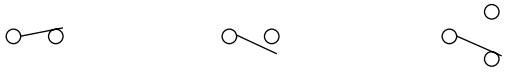


CHART 12 (Sheet 2 of 2)
ELECTRICAL SYMBOLS

CONTACT SWITCH ASSEMBLIES
BASIC



CLOSED CONTACT | OPEN CONTACT | TRANSFER

PUSH BUTTON



(MAKE) CIRCUIT CLOSING

MOMENTARY OR SPRING RETURN

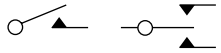


(BREAK) CIRCUIT OPENING



TWO CIRCUIT

NON-LOCKING

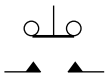


(MAKE) CIRCUIT CLOSING | (MAKE OR BREAK) CIRCUIT CLOSING OR OPENING



(BREAK) CIRCUIT OPENING

MOMENTARY OR SPRING RETURN



TWO CIRCUIT



TRANSFER

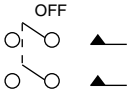
LOCKING AND NON-LOCKING



OFF

THREE POSITION ONE POLE

OFF



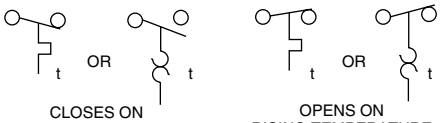
THREE POSITION TWO POLE

PRESSURE OR VACUUM ACTUATED SWITCH



CLOSES ON RISING PRESSURE | OPENS ON RISING PRESSURE

TEMPERATURE ACTUATED



CLOSES ON RISING TEMPERATURE | OPENS ON RISING TEMPERATURE

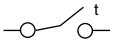
NOTE: "t" symbol shall be replaced by data giving the operating temperature of the device.

LIMIT SWITCH, DIRECTLY ACTUATED - SPRING RETURN

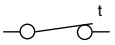


NORMALLY OPEN | NORMALLY OPEN HELD CLOSED | NORMALLY CLOSED | NORMALLY CLOSED HELD OPEN

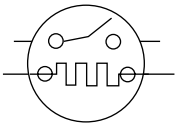
THERMAL SWITCHES



NORMALLY OPEN CLOSES ON RISING TEMPERATURE



NORMALLY CLOSED OPENS ON RISING TEMPERATURE



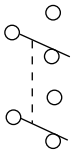
NORMALLY OPEN INTERNAL HEATER SHOWN

SELECTOR OR MULTI - POSITION SWITCH

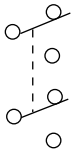


ANY NUMBER OF TRANSMISSION PATHS MAY BE SHOWN

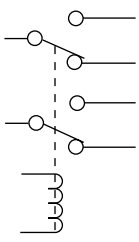
EXAMPLE
ON-ON-ON SWITCH ACUTATION



TOGGLE IN THE DOWN POSITION

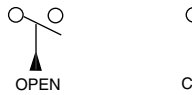


TOGGLE IN THE UP POSITION

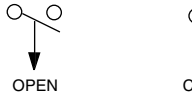


EXAMPLE OF RELAY

SWITCHES WITH TIME/DELAY FEATURE



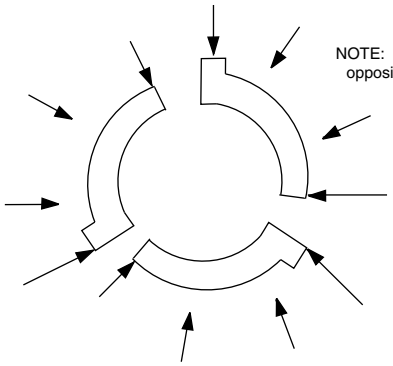
OPEN TIME-DELAY CLOSING | CLOSED TIME-DELAY OPENING



OPEN TIME-DELAY OPENING | CLOSED TIME-DELAY CLOSING

ARROW INDICATES DIRECTION OF SWITCH OPERATION IN WHICH CONTACT ACTION IS DELAYED

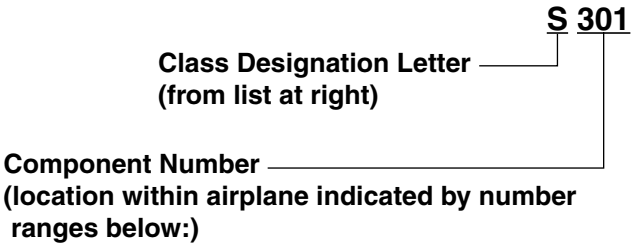
ROTARY SWITCH



NOTE: Viewed from end opposite control knob.

CHART 13
ELECTRICAL / ELECTRONIC COMPONENT REFERENCE DESIGNATION CODES

All electrical / electronic components are assigned a unique reference code which indicates component function and general location within the airplane.



100 to 199 -- Electrical - Left Wing and Nacelle
200 to 299 -- Electrical - Right Wing and Nacelle
300 to 399 -- Electrical - Fuselage area (inside cabin)
400 to 499 -- Electrical - Fuselage area (outside cabin)
500 to 599 -- Avionics - Left Wing and Nacelle
600 to 699 -- Avionics - Right Wing and Nacelle
700 to 799 -- Avionics - Fuselage area (inside cabin)
800 to 899 -- Avionics - Fuselage area (outside cabin)

So, the example above, S301, represents either a flasher, switch, or thermostat located in the left fuselage area inside the cabin.

Component	Class Letter
Alternator	G
Annunciator	DS
Antenna	E
Battery	BT
Blower / Fan / Motor	B
Capacitor	C
Circuit Breaker	CB
Clock	M
Compass	M
Contactor	K
Diode	D or CR
Flasher	S
Fuse	F
Generator	G
Heater	HR
Horn	LS
Hour Meter	M
Indicator Lamp	DS
Instrument	M
Jack	J
Light	L
Magneto	G
Miscellaneous Electrical Part	E
Plug	P
Potentiometer	R
Power Supply	PS
Pump	B
Receptacle	J
Rectifier	CR or D
Relay / Solenoid	K
Resistor	R
Sensor	A or MT
Shunt	R
Speaker	LS
Splice	SP
Starter	B
Switch	S
Terminal	E
Terminal Board	TB
Thermocouple	TC
Thermostat	S
Transducer	A or MT
Transformer	T
Transistor	Q
Voltage Regulator	VR
Zener Diode	D

WIRING DIAGRAM (SCHEMATICS) CROSS-REFERENCE

Each schematic in this manual contains a drawing index number in one of the corners of the schematic, showing the drawing number, revision level, and sheet number (i.e., 105553 A 23) of the Piper engineering drawing the schematic was created from.

Many schematics in this manual were created from harness assembly drawings which typically show only portions of a functional system's wiring on each sheet. In harness assembly drawings, internal cross-reference is to other numbered sheets within that same drawing - typically shown (SH 15) = sheet 15. The following provides sheet locating information by drawing number, sheet number and the Section Number, Figure Number, and Sheet Number where that harness assembly drawing sheet is used in this manual.

NOTE: To follow a drawing internal cross-reference found in a schematic, note the sheet number you are looking for, note the engineering drawing number from the drawing index number found in the corner of the schematic, compare both to the following information to determine where your cross-reference sheet is shown within this manual.

In addition, electrical / avionics component reference designator lists are provided to aid in identifying components and connectors.

Engineering		Maintenance Manual		
Drawing Number	**Sheet Number**	**Section Number**	**Figure Number**	**Sheet Number**
39654 (L) Air Conditioning Schematic	2	91-21-50	1	1-2
39656 (AH) Air Conditioning	1-2	91-21-50	1	1-2
83353 (G)	2	91-24-30	1	1
Electrical	3	91-24-30	2	1
Schematic	3.1	91-24-30	2	2
Seneca IV	4	91-31-50	1	1
	5	91-77-10	1	1
	6	91-79-30	1	
	7	91-28-40	1	1
	8	91-31-20	1	
	9	91-34-20	6	1
	10	91-74-10	1	1
	12	91-33-40	2	
	13	91-28-20	1	1
	14	91-28-20	1	2
	15	91-31-30	1	1
	16	91-21-20	1	
	17	91-31-50	3	1
	18	91-33-30	1	1
	19	91-24-40	1	1
	20	91-23-00	2	2
	21	91-33-10	3	1
	22	91-33-40	1	1

Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
83353 (G)	23	91-33-40	3	1
(cont.)	24	91-33-10	1	1
	25	91-33-10	2	1
	26	91-33-20	1	1
	27	91-33-10	4	1
	28	91-27-30	1	1
	29	91-23-00	1	1
	30	91-80-10	1	1
	31	91-32-60	1	1
	32	91-21-40	1	1
	33	91-27-50	1	1
84549 (AJ)	3	91-22-10	4	3
Switch Harness				
85186 (AD)	3	91-30-10	1	1
Ice Protection	2	91-30-40	1	1
(Complete)		91-30-80	1	1
87291 (Y)	1	91-30-80	1	1
Stall Warn, Heat,	2	91-30-80	1	1
Wing Ice Light				
87719 (AC)	24	91-61-20	1	1
Synchrophaser	26	91-61-20	1	2
	32	91-61-20	2	1
89426 (J)	2	91-30-60	1	1
Propeller	4	91-30-60	1	2
Ice Protection				
Seneca IV				
100651 (T)	2	91-24-30	1	2
Electrical	3	91-31-50	4	
Schematic	4.0-4.3	91-31-50	1	4-7
Seneca V	5	91-37-20	1	
	6.0-6.2	91-77-40	1	1-3
	7	91-28-40	1	2
	8	91-31-20	2	1
	9	91-34-20	6	1
	10	91-74-10	1	2
	13	91-28-20	1	3
	14	91-28-20	1	4
	15	91-31-30	1	2
	16	91-21-20	1	
	17	91-31-50	3	2
	18	91-33-30	1	2

Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
100651 (T)	19	91-24-40	1	2
(cont.)	20	91-23-00	2	2
	21	91-33-10	3	2
	22	91-33-40	1	2
	23	91-33-40	3	2
	24	91-33-10	1	1
	25	91-33-10	2	2
	26	91-33-20	1	2
	27	91-33-10	4	2
	28	91-27-30	1	2
	29	91-23-00	1	2
	30	91-80-10	1	2-3
	31	91-32-60	1	2
	32	91-21-40	1	2
	33	91-27-50	1	2
	34	91-77-20	1	1
	35	91-77-20	1	2
	36	91-77-10	1	2
	37	91-77-10	1	3
	38	91-77-10	2	1
	39	91-77-10	2	2
	40	91-77-20	2	1
	41	91-77-20	2	2
	42	91-79-30	2	1
	43	91-79-30	2	2
	44	91-25-20	1	
100735 (G) Propeller Ice Protection Seneca V	5	91-30-60	1	3-4
101261 (J) Harness Flux Detector Seneca V	1	91-34-20	5	
101288 (D)	2	91-24-30	1	2
Electrical	3	91-31-50	4	
Schematic	4.0-4.3	91-31-50	1	4-7
Seneca V	5	91-37-20	1	
	6.0-6.2	91-77-40	1	1-3
	7	91-28-40	1	2
	8	91-31-20	2	2
	9	91-34-20	6	1

Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
101288 (D)	10	91-74-10	1	2
(cont.)	13	91-28-20	1	3
	14	91-28-20	1	4
	15	91-31-30	1	2
	16	91-21-20	1	
	17	91-31-50	3	2
	18	91-33-30	1	2
	19	91-24-40	1	2
	20	91-23-00	2	2
	21	91-33-10	3	2
	22	91-33-40	1	3
	23	91-33-40	3	3
	24	91-33-10	1	1
	25	91-33-10	2	2
	26	91-33-20	1	2
	27	91-33-10	4	2
	28	91-27-30	1	2
	29	91-23-00	1	2
	30	91-80-10	1	2-3
	31	91-32-60	1	2
	32	91-21-40	1	2
	33	91-27-50	1	2
	34	91-77-20	1	1
	35	91-77-20	1	2
	36	91-77-10	1	2
	37	91-77-10	1	3
	38	91-77-10	2	1
	39	91-77-10	2	2
	40	91-77-20	2	1
	41	91-77-20	2	2
	42	91-79-30	2	1
	43	91-79-30	2	2
	44	91-25-20	1	
101479 (G)	2	91-34-20	1	
Harness Meggitt	3	91-34-20	2	1
EFIS	4	91-34-20	2	2
101753 (D)	1	91-34-20	5	
Harness MAG/OAT				
Entegra				

Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
101772 (L)	2	91-30-10	1	1
Ice Prot. & Vacuum				
Entegra				
101813 (G)	6	91-34-20	3	
Avionics Harness	7	91-34-20	4	
Entegra	8	91-34-50	1	
	9	91-34-50	2	
	10	91-34-50	3	
	11	91-23-50	1	
	12	91-23-50	2	
	13	91-34-50	4	
	14	91-34-40	1	
	15	91-34-50	5	1
	16	91-34-50	5	2
	17	91-34-40	2	
	18	91-34-40	3	1
	19	91-34-40	3	2
	20	91-22-10	1	1
	21	91-22-10	1	2
	22	91-22-10	1	3
	24	91-34-50	6	
	23	91-22-10	1	4
	25	91-23-15	1	
		91-34-40	4	
	26	91-24-30	3	1-4
	27	91-24-30	3	5-7
101815 (L)	7	91-34-20	4	
Electrical Harness	27	91-80-10	1	5
Entegra				
101840 (A)	2	91-24-30	1	4
Electrical Schematic	3	91-31-50	4	
Entegra	4.0-4.2	91-31-50	1	12-14
	4.4	91-31-50	5	
	6.0	91-34-20	6	2
	6.1	91-74-10	1	4
	6.2	91-32-60	2	
	7	91-20-20	1	5
	8	91-28-20	1	6
	9	91-31-30	1	2
	10	91-21-20	1	

Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
101840 (A)	11	91-31-50	3	3
(cont.)		91-33-30	1	2
	12	91-24-40	1	4
	13	91-23-00	2	4
	14	91-33-10	3	3
	15	91-33-40	1	4
	16	91-33-40	3	4
	17	91-33-10	1	1
	18	91-33-10	2	4
	19	91-33-20	1	3
	20	91-33-10	4	3
	21	91-27-30	1	3
	22	91-23-00	1	3
	23	91-80-10	1	4
			1	5
	24	91-32-60	1	4
	25	91-21-40	1	3
	26	91-27-50	1	3
	27	91-77-40	2	1
	28	91-77-40	2	2
	29	91-77-40	2	3
	30	91-77-40	2	4
	31	91-77-40	2	5
	32	91-77-40	2	6
	33	91-34-10	2	
104229 (C)	2	91-24-30	1	3
Electrical Schematic	3	91-31-50	4	
Seneca V	4.0-4.3	91-31-50	1	4-7
	5	91-37-20	1	
	6.0-6.2	91-77-40	1	1-3
	7	91-28-40	1	2
	8	91-31-20	2	2
	9	91-34-20	6	2
	10	91-74-10	1	3
	12	91-24-60	1	
	13	91-28-20	1	3
	14	91-28-20	1	4
	15	91-31-30	1	2
	16	91-21-20	1	
	17	91-31-50	3	3

Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
104229 (C)	18	91-33-30	1	2
(cont.)	19	91-24-40	1	3
	20	91-23-00	2	3
	21	91-33-10	3	2
	22	91-33-40	1	3
	23	91-33-40	3	3
	24	91-33-10	1	1
	25	91-33-10	2	3
	26	91-33-20	1	2
	27	91-33-10	4	2
	28	91-27-30	1	2
	29	91-23-00	1	2
	30	91-80-10	1	2-3
	31	91-32-60	1	2
	32	91-21-40	1	2
	33	91-27-50	1	2
	34	91-77-20	1	1
	35	91-77-20	1	2
	36	91-77-10	1	2
	37	91-77-10	1	3
	38	91-77-10	2	1
	39	91-77-10	2	2
	40	91-77-20	2	1
	41	91-77-20	2	2
	42	91-79-30	2	1
	43	91-79-30	2	2
	44	91-25-20	1	
104368 (K)	2	91-24-30	1	3
Electrical Schematic	3	91-31-50	4	
Seneca V	4.0-4.3	91-31-50	1	8-11
	4.4	91-31-50	2	
	5	91-37-20	1	
	6.0-6.2	91-77-40	1	1-3
	7	91-28-40	1	2
	8	91-31-20	2	2
	9	91-34-20	6	2
	10	91-74-10	1	3
	11	91-32-60	2	
	12	91-24-60	1	
	13	91-28-20	1	3

Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
104368 (K)	14	91-28-20	1	4
(cont.)	15	91-31-30	1	2
	16	91-21-20	1	
	17	91-31-50	3	3
	18	91-33-30	1	2
	19	91-24-40	1	3
	20	91-23-00	2	3
	21	91-33-10	3	2
	22	91-33-40	1	3
	23	91-33-40	3	3
	24	91-33-10	1	1
	25	91-33-10	2	3
	26	91-33-20	1	2
	27	91-33-10	4	2
	28	91-27-30	1	2
	29	91-23-00	1	2
	30	91-80-10	1	2-3
	31.0	91-32-60	1	2
	31.1	91-32-60	1	3
	32	91-21-40	1	2
	33	91-27-50	1	2
	34	91-77-20	1	1
	35	91-77-20	1	2
	36	91-77-10	1	2
	37	91-77-10	1	3
	38	91-77-10	2	1
	39	91-77-10	2	2
	40	91-77-20	2	1
	41	91-77-20	2	2
	42	91-79-30	2	1
	43	91-79-30	2	2
	44	91-25-20	1	
107230 (New)	4	91-24-30	4	1-2
Electrical Schematic	5	91-24-40	2	1-2
G1000	6	91-24-30	5	1-2
	7	91-24-30	6	1-2
	9	91-77-40	3	1
	10	91-80-10	2	1
	11	91-77-40	3	2
	12	91-80-10	2	2

Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
107230 (New)	13	91-32-60	3	1
(cont.)	14	91-32-60	4	1-2
	15	91-28-20	1	7
		91-28-40	2	1
	16	91-27-30	2	1
	17	91-27-50	2	1
	18	91-21-40	2	1-2
	19	91-21-50	2	1-2
	20	91-30-60	2	
	21	91-30-10	1	2
	22	91-30-40	1	2
		91-30-80	1	2
	23	91-61-20	2	2
	24	91-35-20	1	
	26	91-33-40	3	5-6
	27	91-33-40	1	5-6
	28	91-33-20	2	
		91-33-30	1	2
	29	91-33-10	4	4
	30	91-33-10	1	2
		91-33-10	2	5
	31	91-31-50	6	
	32	91-25-60	3	
		91-31-30	2	
	33	91-34-10	7	1
	34	91-34-20	9	1
	35	91-21-20	2	1
	36	91-34-10	5	1
	37	91-34-10	5	2
	38	91-34-10	6	1
	39	91-34-10	6	2
	40	91-34-50	10	1
	41	91-23-50	4	1
	42	91-23-50	4	3
	43	91-23-50	4	4
	44	91-22-10	3	1
	45	91-22-10	3	2
	46	91-22-10	3	3
	50	91-34-50	11	1
	51	91-23-15	3	1
	52	91-23-15	4	1

Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
107230 (New)	53	91-34-40	9	1
(cont.)	54	91-34-40	8	
	55	91-34-50	13	1
	56	91-34-50	12	1
107235 (E)	3	91-32-60	4	3-4
Harness				
Left Wing				
107237 (J)	4	91-37-40	9	1
Avionics Harness		91-31-50	6	
G1000	6	91-21-20	2	1
	7	91-34-50	11	1
	8	91-77-40	3	1
	9	91-77-40	3	2
	10	91-34-20	10	1
	11	91-34-20	10	2
		91-77-40	3	3
	12	91-34-20	10	3
	13	91-34-20	10	4-5
		91-77-40	3	4
	14	91-34-20	10	6
107238 (C)	2	91-34-10	5	1
AHRS & Magnetometer	3	91-34-10	5	2
G1000	4	91-34-10	5	3
	5	91-34-10	5	4
107239 (C)	2	91-34-10	6	1
Harness ADC/OAT	3	91-34-10	6	2
G1000	5	91-34-10	6	3
	6	91-34-10	6	4
107240 (New)	4	91-24-30	4	3-4
Electrical Schematic	5	91-24-40	2	1-2
GFC 700	6	91-24-30	5	1-2
	7	91-24-30	6	1-2
	9	91-77-40	3	1
	10	91-80-10	2	1
	11	91-77-40	3	2
	12	91-80-10	2	2
	13	91-32-60	3	1-2
	14	91-32-60	4	3-4
	15	91-28-20	1	7
		91-28-40	2	1
	16	91-27-30	2	2

Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
107240 (New)	17	91-27-50	2	2
(cont.)	18	91-21-40	2	1-2
	19	91-21-50	2	1-2
	20	91-30-60	2	
	21	91-30-10	1	2
	22	91-30-40	1	2
		91-30-80	1	2
	23	91-61-20	2	2
	24	91-35-20	1	
	26	91-33-40	3	5-6
	27	91-33-40	1	5-6
	28	91-33-20	2	
		91-33-30	1	2
	29	91-33-10	4	4
	30	91-33-10	1	2
	30	91-33-10	1	2
		91-33-10	2	5
	31	91-31-50	6	
	32	91-25-60	3	
		91-31-30	2	
	33	91-34-10	7	1
	34	91-34-20	9	1
	35	91-21-20	2	1
	36	91-34-10	5	1
	37	91-34-10	5	2
	38	91-34-10	6	3
	39	91-34-10	6	4
	40	91-34-50	10	2
	41	91-23-50	4	2
	42	91-23-50	4	3
	43	91-23-50	4	4
	44	91-22-10	4	1
	45	91-22-10	4	2
	46	91-22-10	4	3
	50	91-34-50	11	1
	51	91-23-15	3	2
	52	91-23-15	4	2
	53	91-34-40	9	1
	54	91-34-40	8	
	55	91-34-50	13	2
	56	91-34-50	12	1

Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
107241 (C)	2	91-34-10	7	1
Harness Stby. Instruments				
G1000				
107242 (D)	2	91-22-10	3	1
Harness Autopilot	3	91-22-10	3	3
S-TEC G1000	4	91-22-10	3	2
	5	91-22-10	3	3
107244 (A)	2	91-34-50	12	1
Harness DME	3	91-34-50	12	2
G1000/G1000 NXi				
107245 (D)	2	91-34-50	13	1-2
Harness ADF				
G1000				
107246 (New)	2	91-34-50	11	1
Harness TAS				
G1000				
107247 (New)	2	91-34-40	9	1
Harness Stormscope				
G1000				
107248 (H)	2	91-23-15	3	1-3
Harness XM				
G1000				
107249 (B)	2	91-23-15	4	1-2
Harness Iridium				
G1000				
107309 (E)	3	91-23-50	4	1-2
Harness Audio	4	91-23-50	4	3
G1000/G1000 NXi				
107313 (D)	2	91-34-20	7	1
Flight Instruments		91-34-20	8	
G600	3	91-34-20	7	2
107319 (E)	2	91-34-40	8	
Harness Stormscope				
G1000				
107322	2	91-34-10	3	1
AHRS G600	3	91-34-10	3	2
107332 (C)	2	91-34-10	4	1
Harness ADC/OAT	3	91-34-10	4	2
G600				

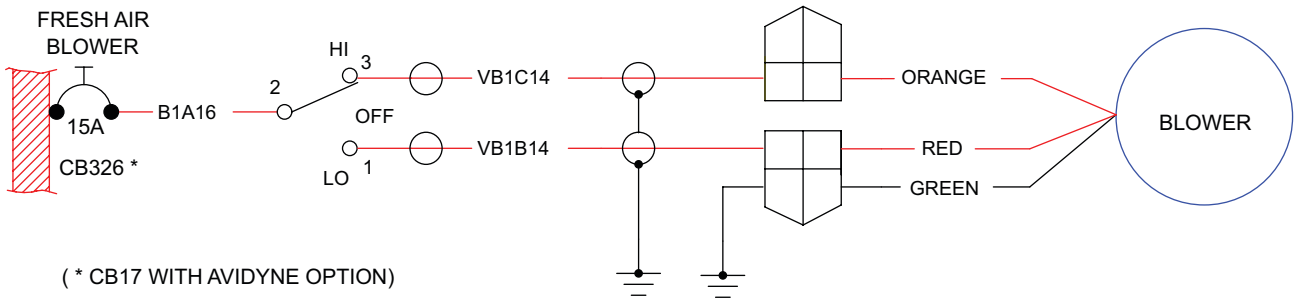
Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
107341	2	91-23-50	3	1
Audio Harness	3	91-23-50	3	2
G600	4	91-23-50	3	3
107342 (B)	2	91-34-50	7	1
COM/NAV/GPS	3	91-34-50	7	2
G600	4	91-34-50	7	3
		91-34-50	7	4
107343 (C)	2	91-34-50	8	
Harness Transponder				
G600				
107344	2	91-22-10	2	1
S-TEC Autopilot	3	91-22-10	2	2
G600		91-22-10	2	3
	4	91-22-10	2	4
	5	91-22-10	2	5
107345 (C)	2	91-34-50	9	
Harness ADF				
G600				
107346	2	91-34-40	5	1
TAS G600	3	91-34-40	5	2
107347 (D)	2	91-34-40	6	
Harness Stormscope				
G600				
107348 (C)	2	91-23-15	2	1-2
Harness XM				
G600				
107349 (B)	2	91-34-40	7	
Harness WX Radar				
G600				
107445	3	91-23-50	5	1
Audio - Aft PTT	4	91-23-50	5	2
Harness	5	91-23-50	5	3
Seneca V		91-23-50	5	5
	6	91-23-50	5	4
107865 (F)	3	91-22-10	4	1
Harness GFC 700	4	91-22-10	4	2
G1000/G1000 NXi		91-22-10	5	2
	5	91-22-10	4	3
		91-22-10	5	3

Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
34F24A001 (A)	7	91-33-40	1	7-8
Harness Fuselage	8	91-33-30	1	2
G1000 NXi				
34F24A002 (A)	3	91-32-60	4	5-6
Harness Instrument Panel		91-33-40	3	7-8
G1000 NXi				
34F24A004 (New)	2	91-24-30	6	3-4
Harness Circuit Breaker	3	91-24-30	6	3-4
G1000 NXi				
34F24A006 (New)	2	91-33-10	4	5
Harness Overhead Switch		91-33-40	3	7-8
Panel, G1000 NXi				
34F24A008 (A)	2	91-31-20	4	
CO Pulse Oximeter				
34F24A020 (New)	4	91-24-30	4	5-6
Electrical Schematic	5	91-24-40	2	3-4
G1000 NXi	6	91-24-30	5	3-4
	7	91-24-30	6	3-4
	8	91-77-40	3	5-6
	9	91-80-10	2	3
	10	91-77-40	3	7-8
	11	91-80-10	2	4
	12	91-32-60	4	5-6
	13	91-32-60	4	7-8
	14	91-28-20	1	7
		91-28-40	2	2
	15	91-27-30	2	3
	16	91-27-50	2	3
	17	91-21-40	2	3-4
	18	91-21-50	2	1-2
	19	91-30-60	2	
	20	91-30-10	1	2
	21	91-30-40	1	2
		91-30-80	1	2
	22	91-61-20	2	3
	23	91-35-20	1	
	24	91-24-60	2	
	25	91-33-40	3	7-8
	26	91-33-40	1	7-8

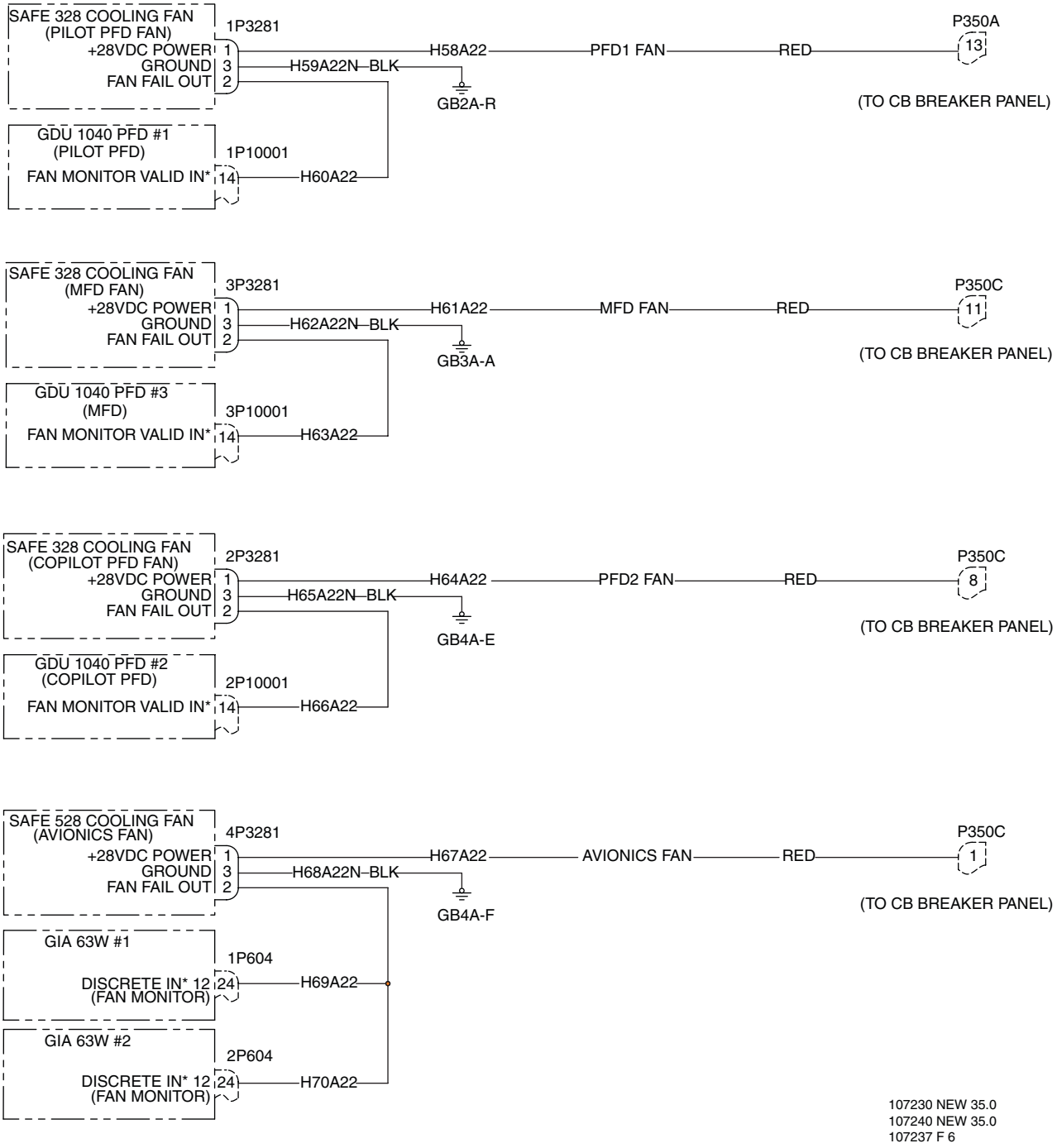
Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
34F24A020 (New)	27	91-33-20	2	
(cont.)		91-33-30	1	2
	28	91-33-10	4	5
	29	91-33-10	1	3
		91-33-10	2	5
	30	91-33-10	6	
	31	91-25-60	4	
		91-31-30	2	
	32	91-34-10	7	2
	33	91-34-20	9	2
	34	91-21-20	2	2
		91-34-20	11	
	35	91-34-10	5	5
	36	91-34-10	5	6
	37	91-34-10	6	5
	38	91-34-10	6	6
	39	91-34-20	10	7
	40	91-34-50	10	3
	41	91-23-50	4	2
	42	91-23-50	4	5
	43	91-23-50	4	6
	44	91-23-50	4	7
	45	91-22-10	5	1
	46	91-22-10	5	2
	47	91-22-10	5	3
	48	91-31-20	4	
	49	91-34-50	11	2
	50	91-34-50	11	3
	51	91-23-15	3	3
	52	91-23-15	4	2
	53	91-34-40	9	2
	54	91-34-40	8	
	55	91-34-50	13	3
	56	91-34-50	12	2
34F34A003 (New)	2	91-34-10	6	5
Harness ADC	3	91-34-10	6	6
G1000 NXi				
34F34A004 (New)	2	91-34-10	5	5
Harness AHRS	3	91-34-10	5	6
G1000 NXi				

Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
34W24A001 (New)	3	91-24-30	4	5-6
Harness Wing		91-28-20	1	7
G1000 NXi		91-32-60	4	7-8
		91-80-10	2	3
	4	91-27-30	2	3
		91-33-40	1	7-8
		91-33-40	3	7-8
	5	91-77-40	3	5-6
	6	91-28-40	2	2
		91-77-40	3	5-6
	7	91-24-30	4	5-6
		91-28-20	1	7
		91-32-60	4	7-8
		91-80-10	2	4
	8	91-77-40	3	7-8
	9	91-28-40	2	2
		91-77-40	3	7-8

83353 NEW/G 16.0
100651 NEW/T 16
101288 NEW/D 16
104229 C 16
104368 A/K 16.0
101840 NEW 10.0



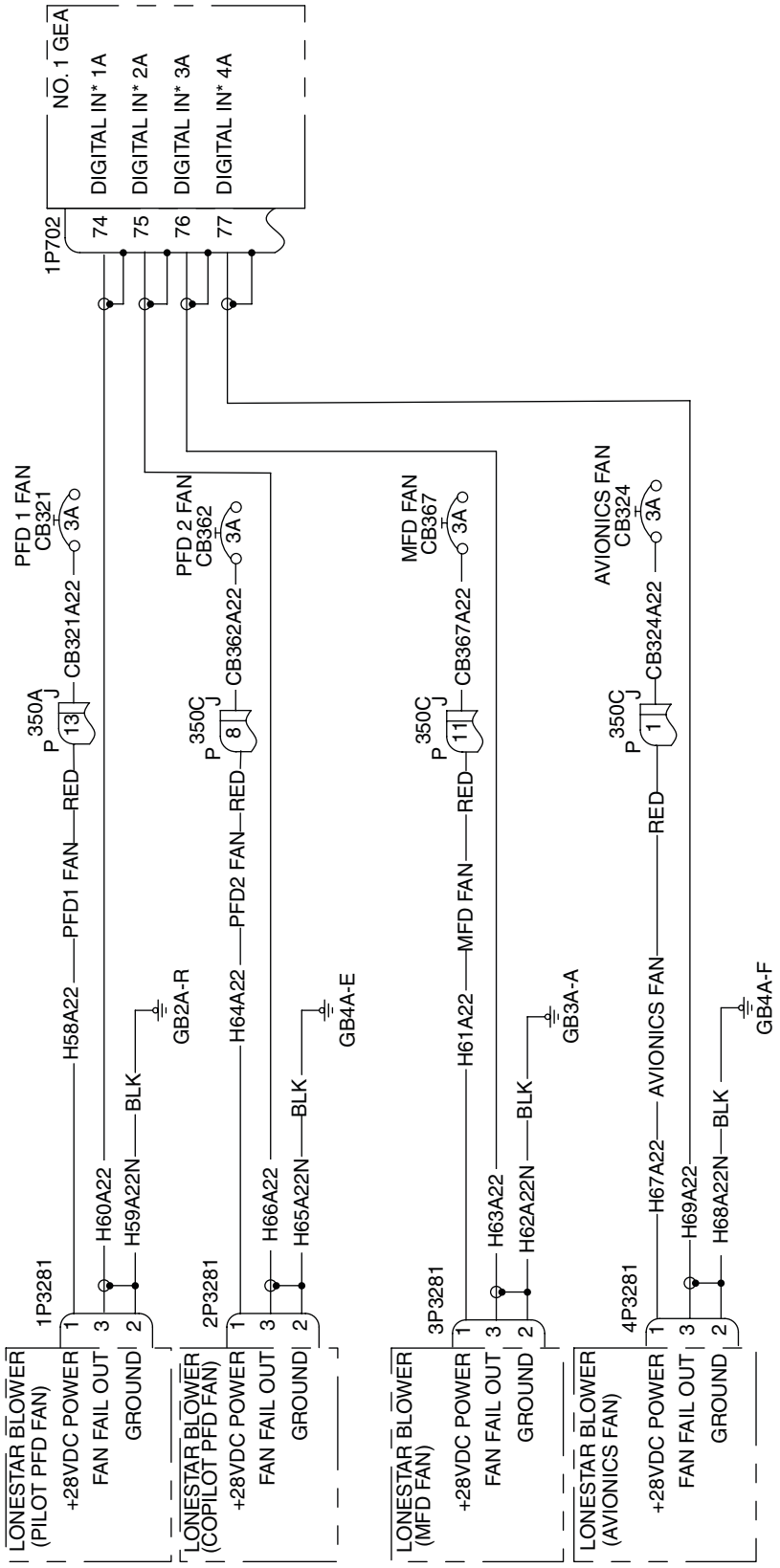
Fresh Air Blower
Figure 1



Effectivity
Seneca V
with Garmin G1000

Avionics Cooling
Figure 2 (Sheet 1 of 2)

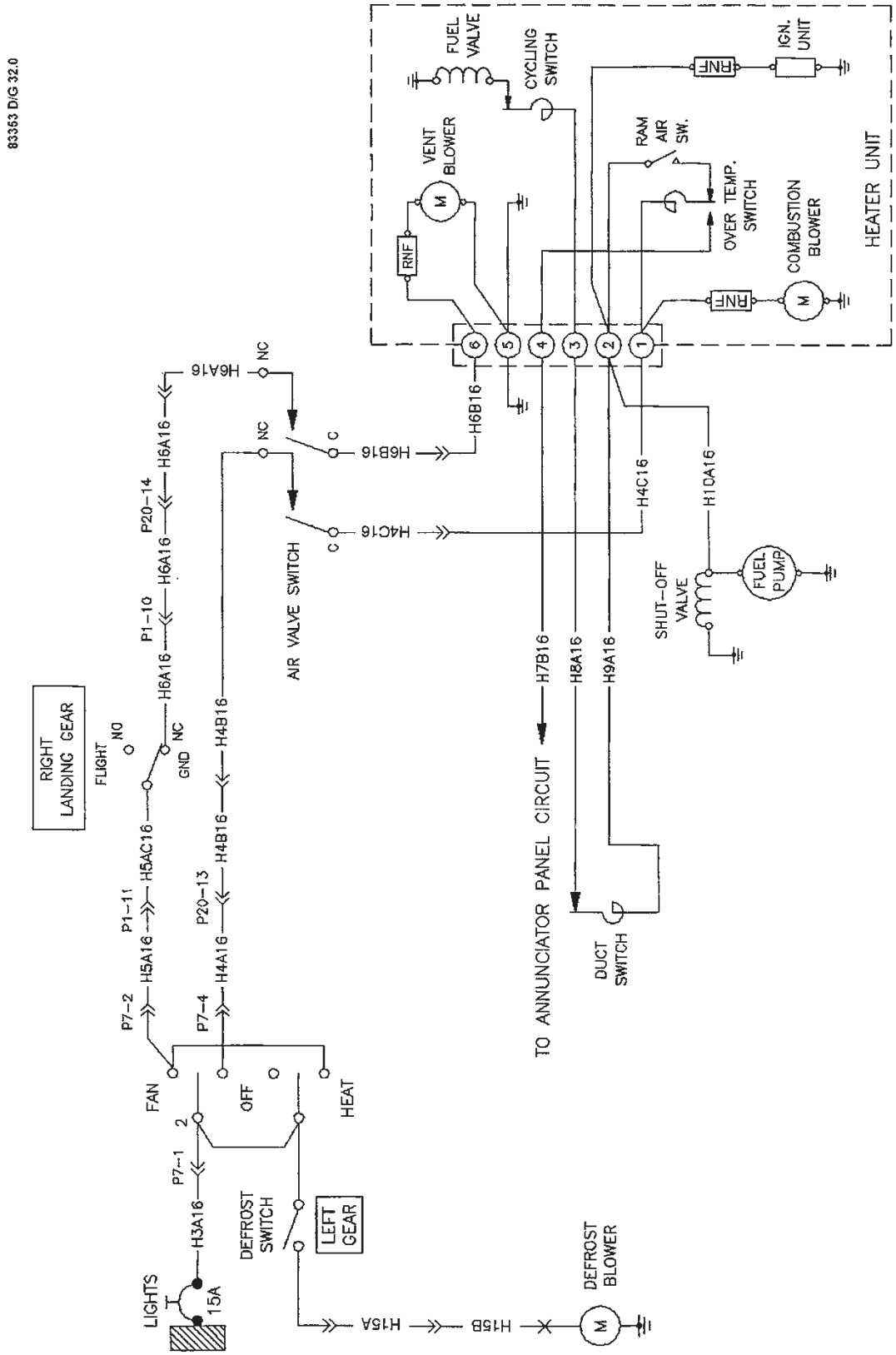
34F24A020 NEW 34.0



Avionics Cooling
Figure 2 (Sheet 2 of 2)

Effectivity
Seneca V
with Garmin G1000 NXi

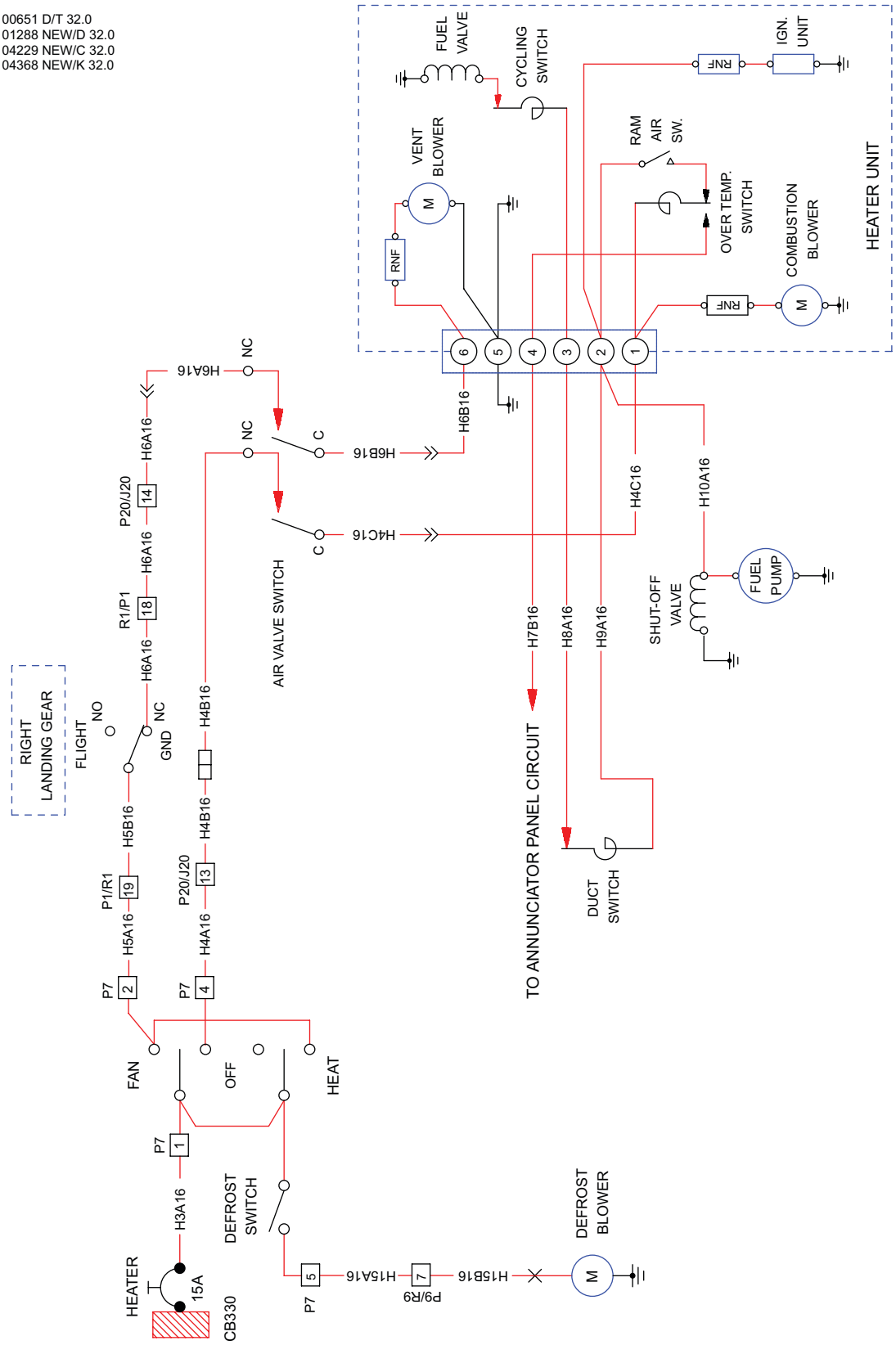
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Heating and Defrosting
Figure 1 (Sheet 1 of 3)

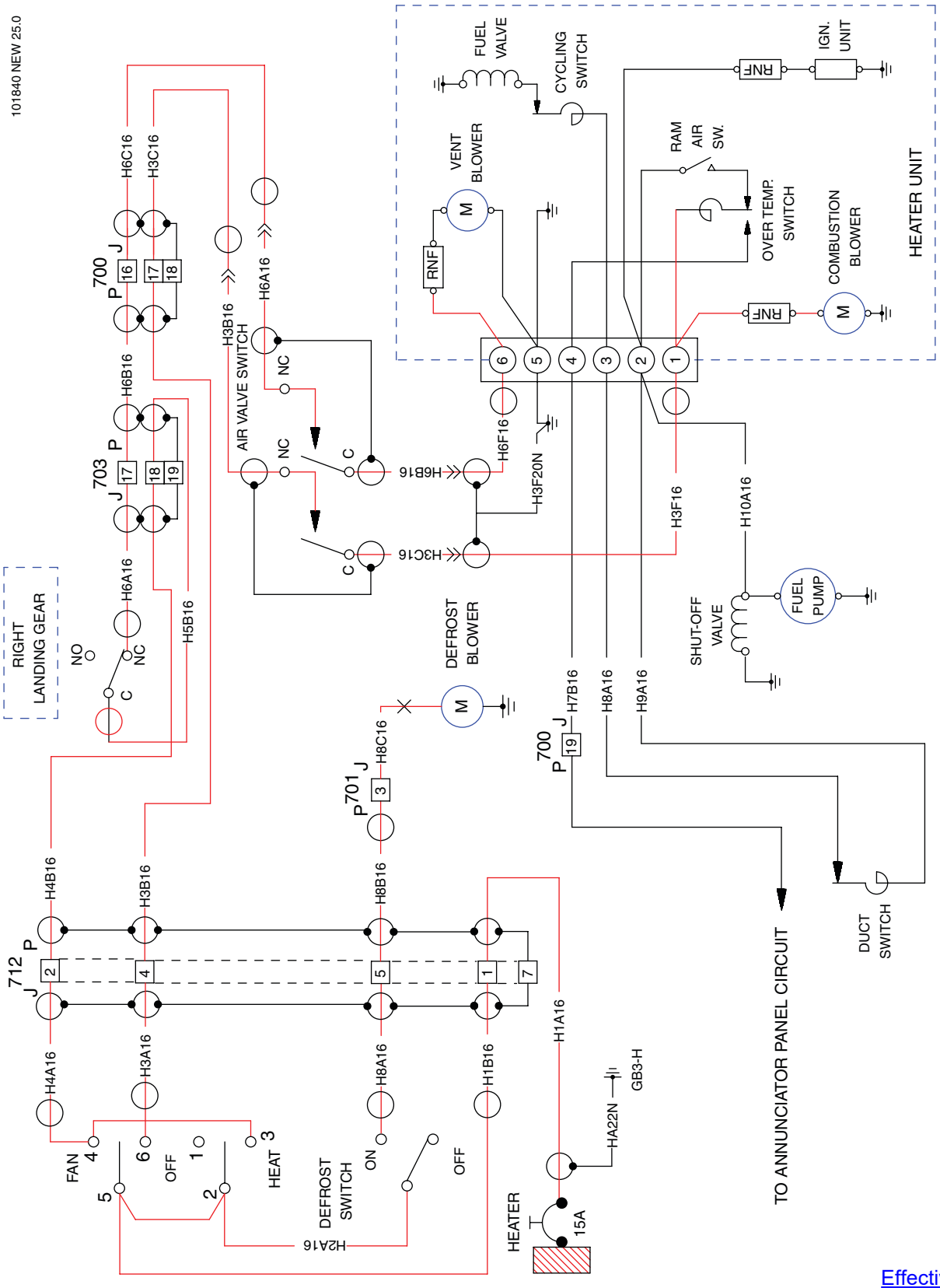
Effectivity
Seneca IV

100651 D/T 32.0
101288 NEW/D 32.0
104229 NEW/C 32.0
104368 NEW/K 32.0



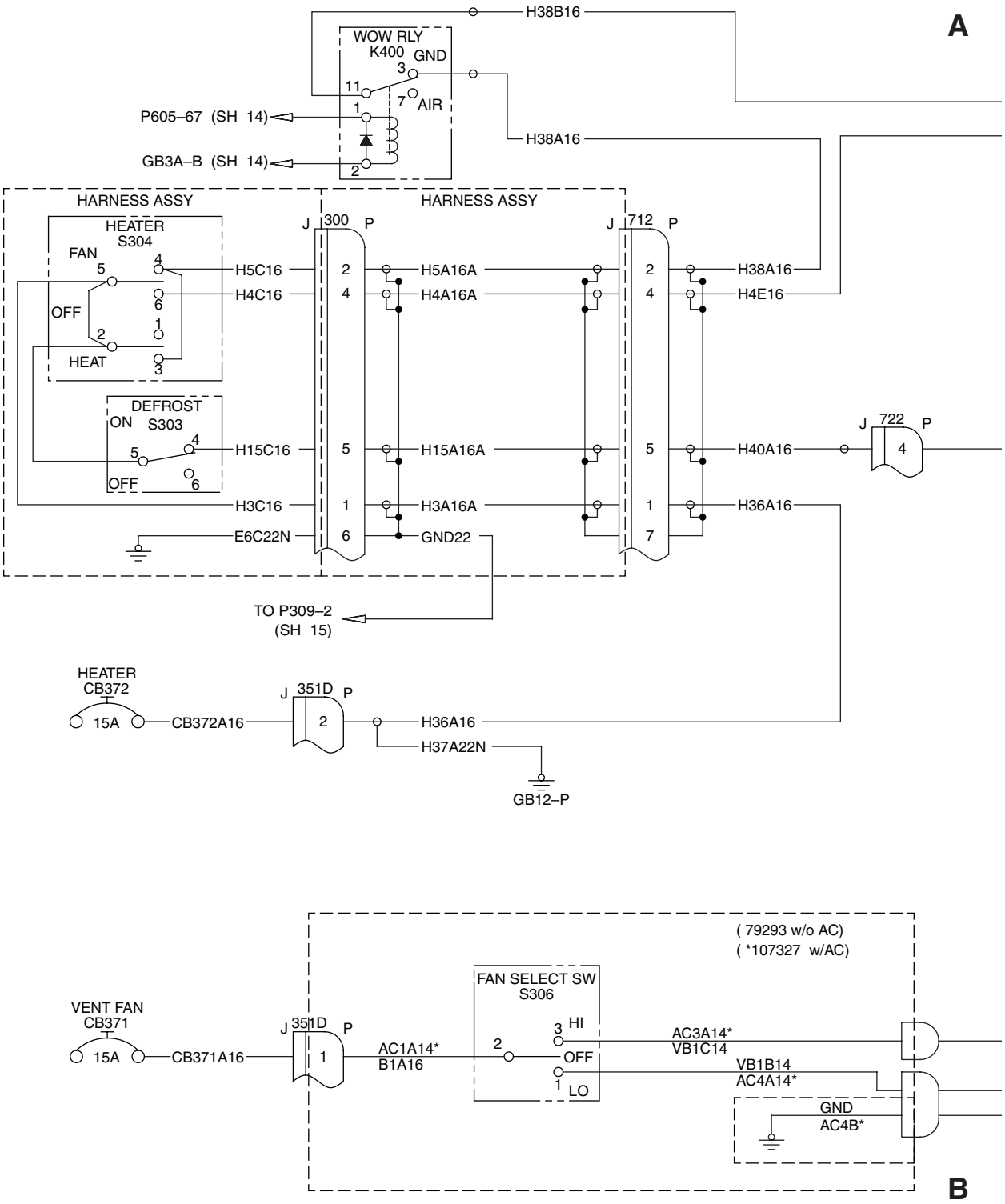
Effectivity
Seneca V

Heating and Defrosting
Figure 1 (Sheet 2 of 3)



Heating and Defrosting
Figure 1 (Sheet 3 of 3)

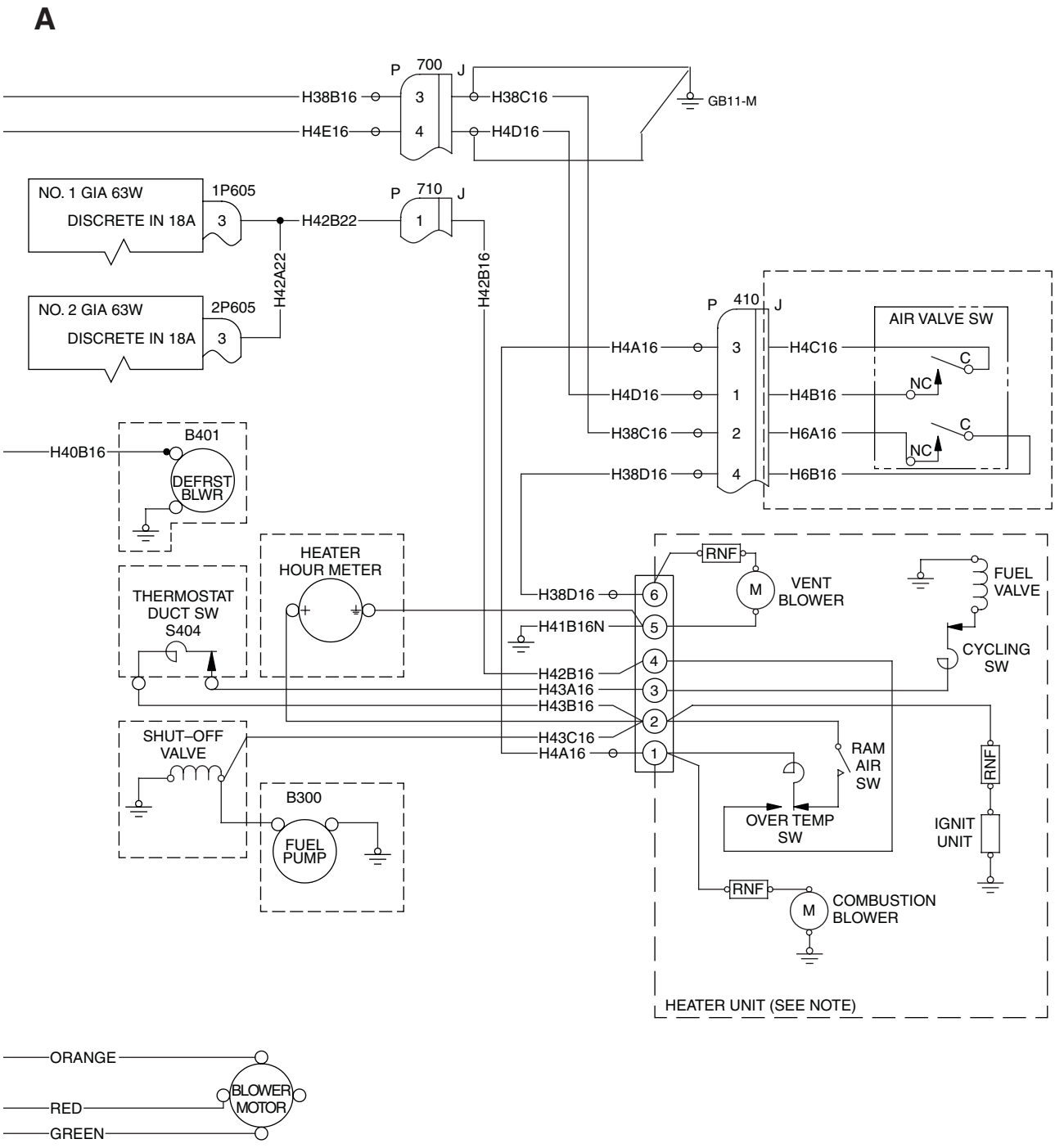
Effectivity
Seneca V
with Avidyne Entegra



107230 NEW 18.0
107240 NEW18.0

Effectivity
Seneca V
with Garmin G1000

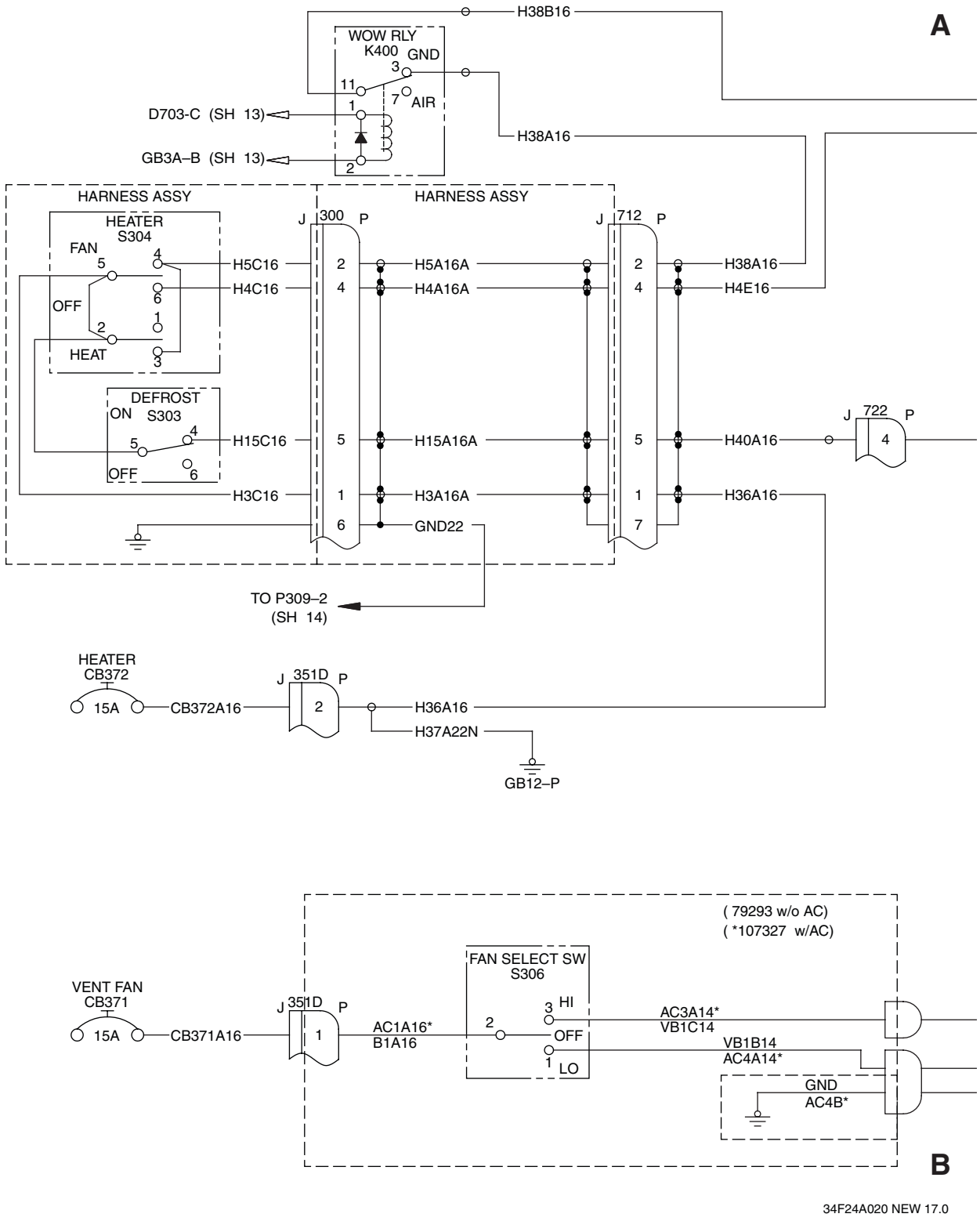
Heater and Defroster Vent Fan
Figure 2 (Sheet 1 of 4)



107230 NEW 18.0
107240 NEW 18.0

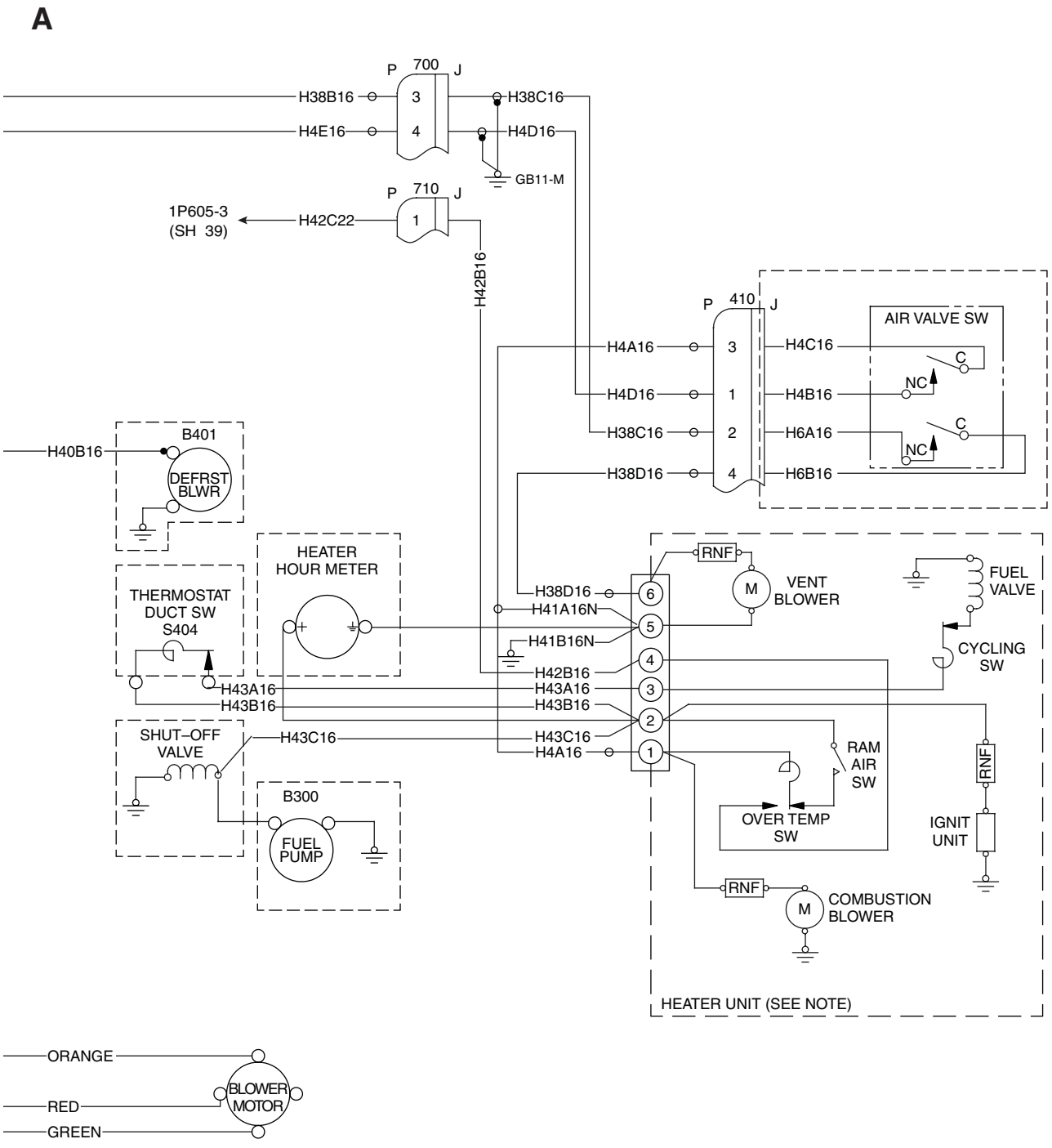
Heater and Defroster Vent Fan
Figure 2 (Sheet 2 of 4)

Effectivity
Seneca V
with Garmin G1000



Effectivity
Seneca V
with Garmin G1000 NXi

Heater and Defroster Vent Fan
Figure 2 (Sheet 3 of 4)

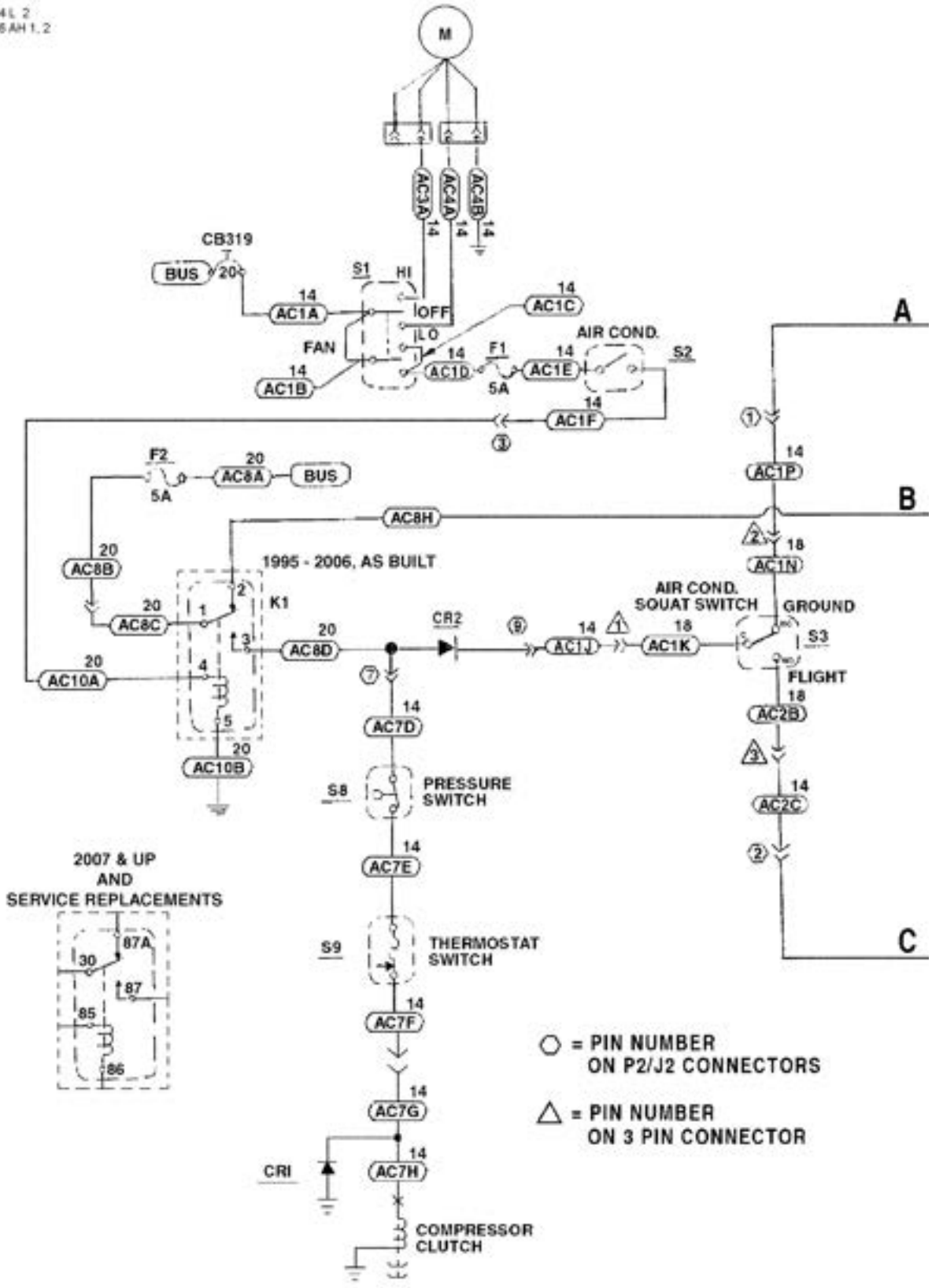


Heater and Defroster Vent Fan
Figure 2 (Sheet 4 of 4)

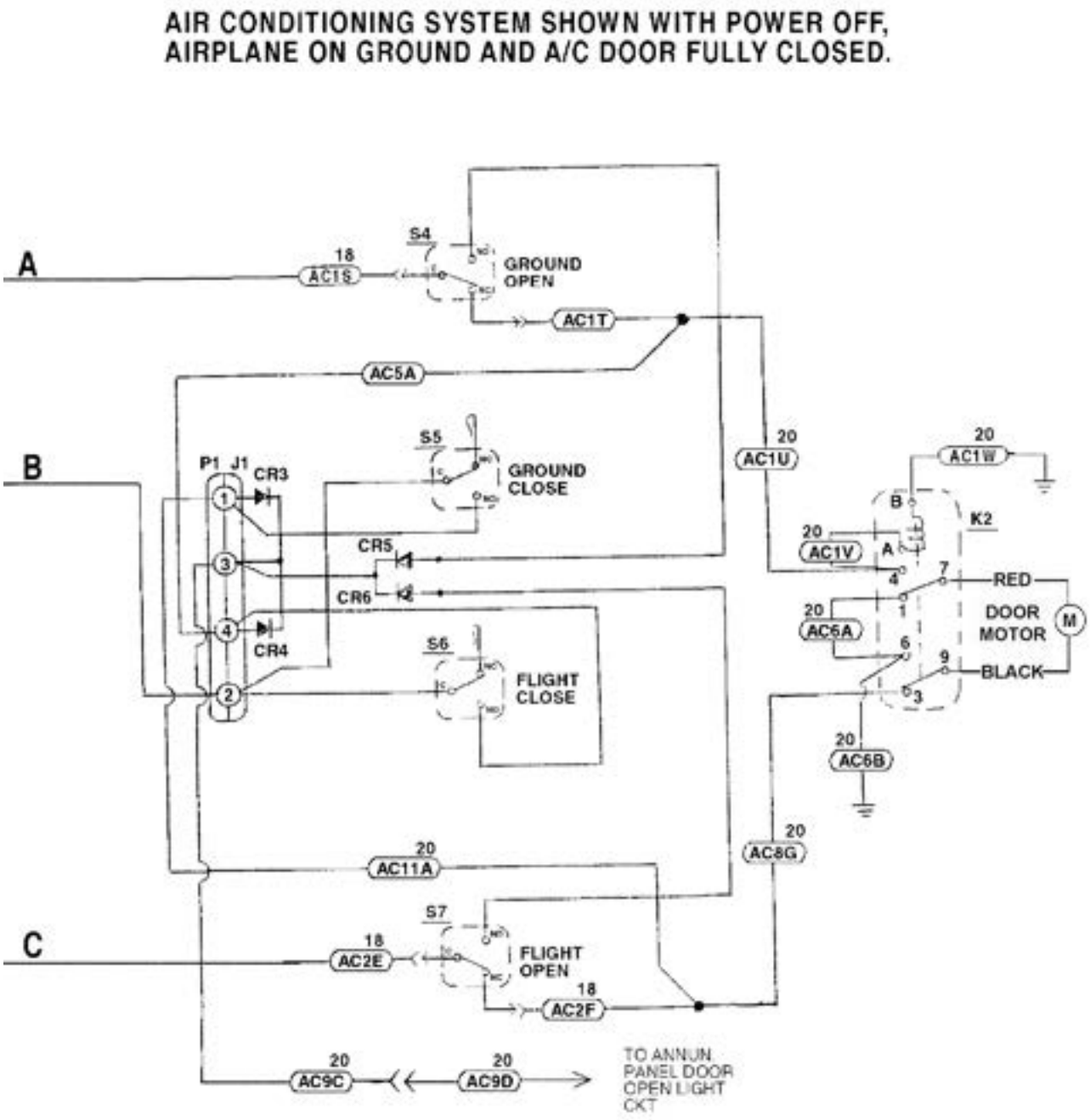
Effectivity
Seneca V
with Garmin G1000 NXi

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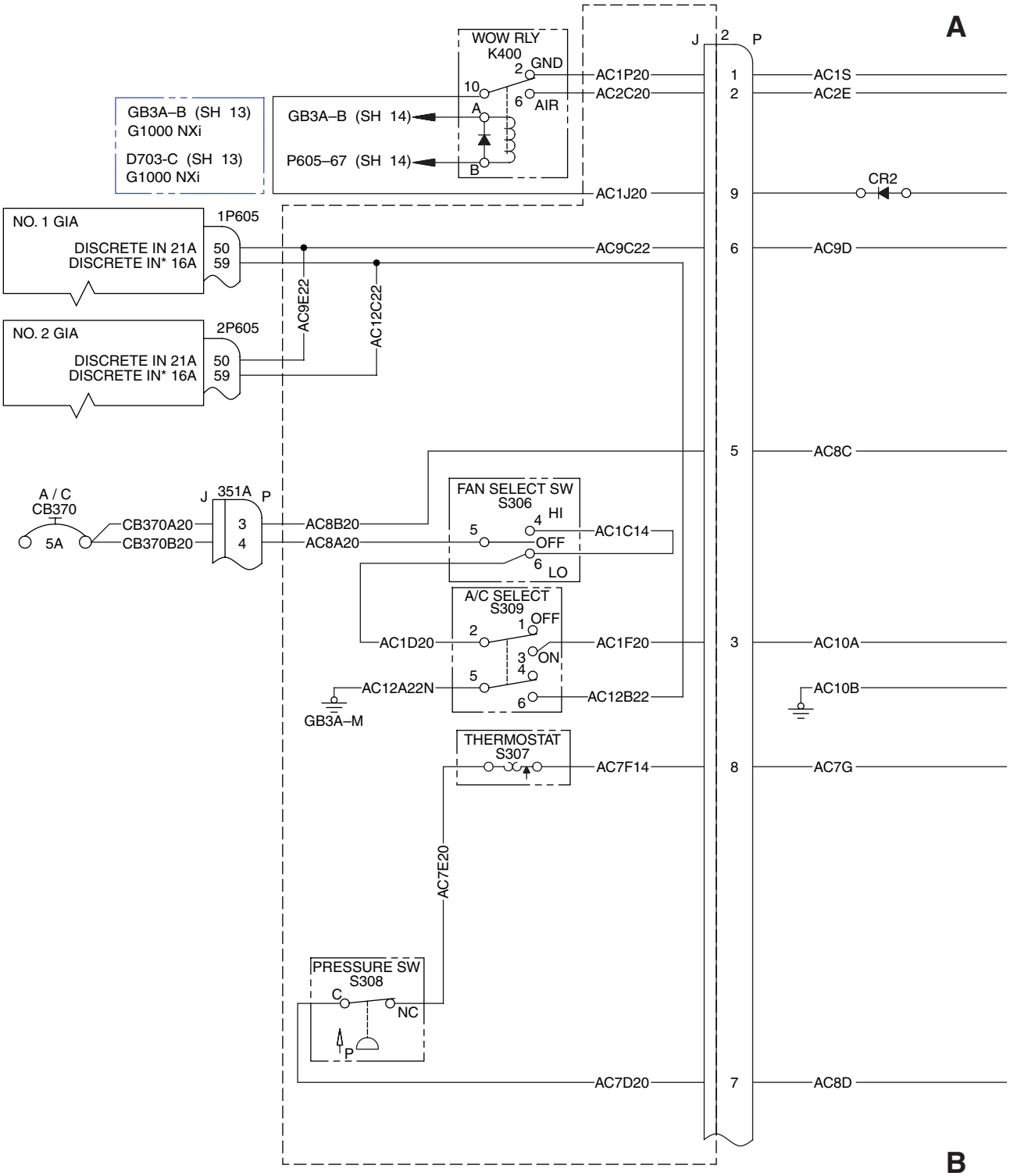
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Air Conditioning (Optional)
Figure 1 (Sheet 1 of 2)



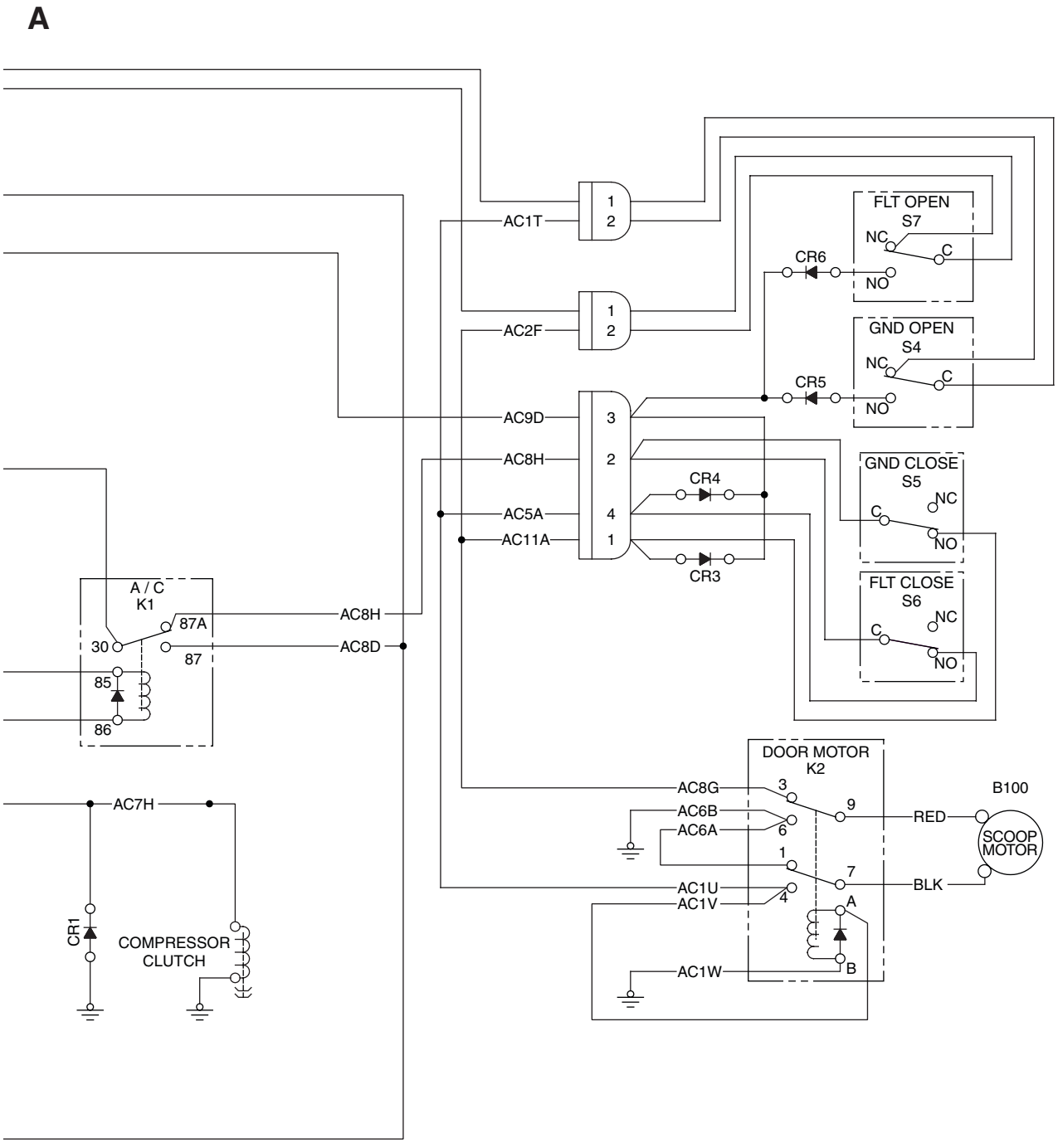
Air Conditioning (Optional)
Figure 1 (Sheet 2 of 2)



107230 NEW 19.0
107240 NEW 19.0
34F24A020 NEW 18.0

Effectivity
Seneca V
with Garmin G1000 / G1000 NXi

Air Conditioning
Figure 2 (Sheet 1 of 2)



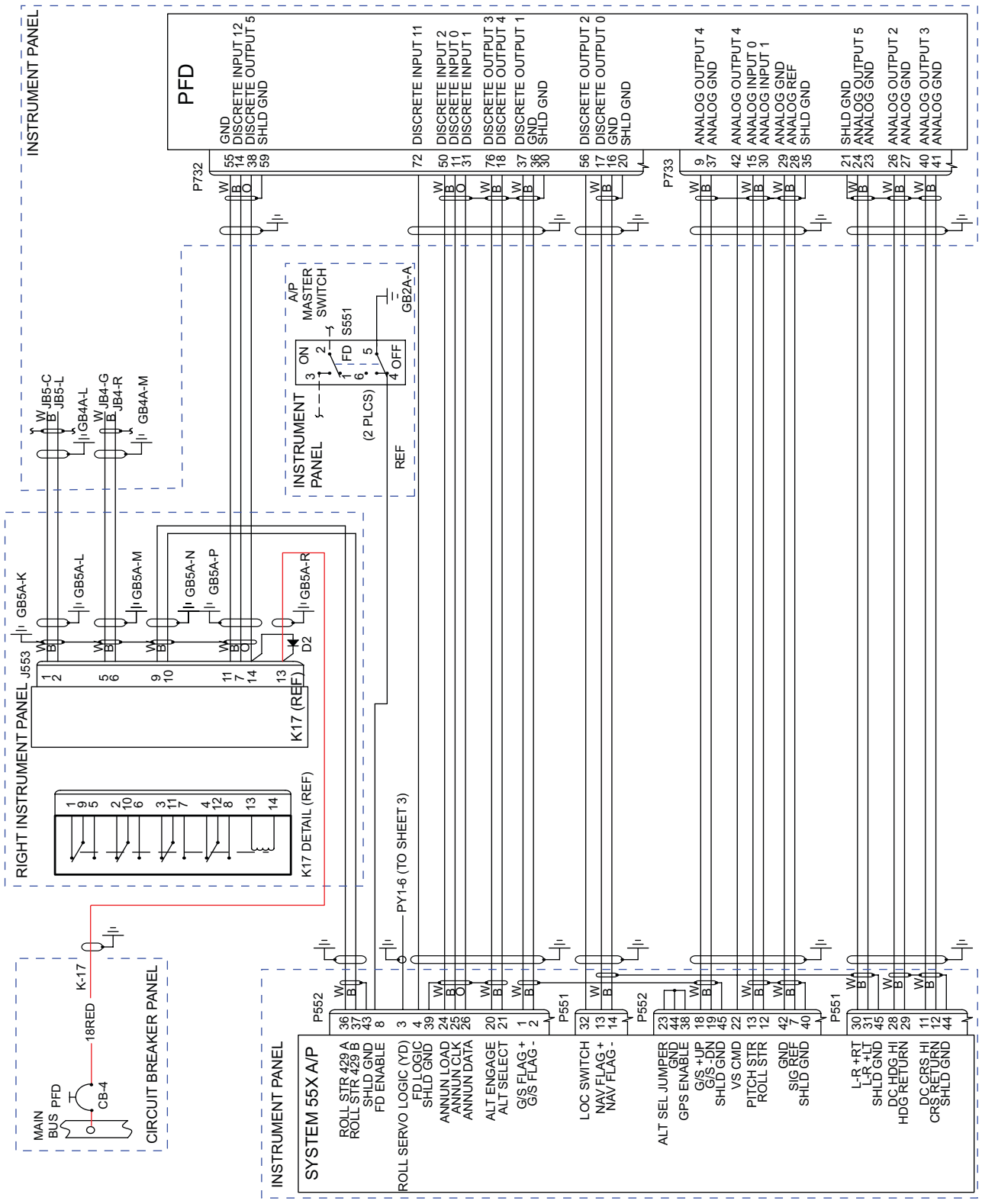
107230 NEW 19.0
107240 NEW 19.0
34F24A020 NEW 18.0

Air Conditioning
Figure 2 (Sheet 2 of 2)

Effectivity
Seneca V
with Garmin G1000 / G1000 NXi

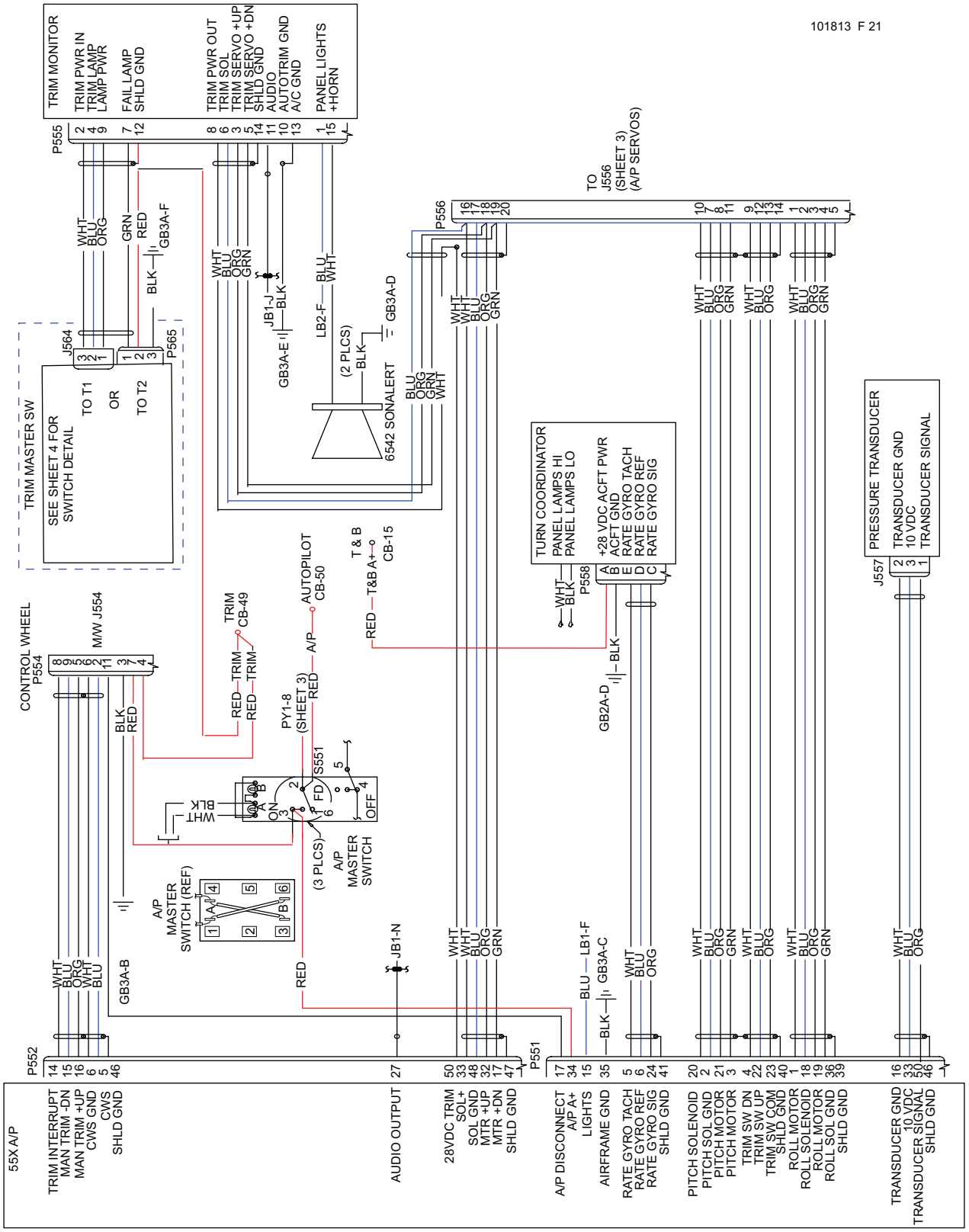
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101813 E 20



Autopilot
Figure 1 (Sheet 1 of 4)

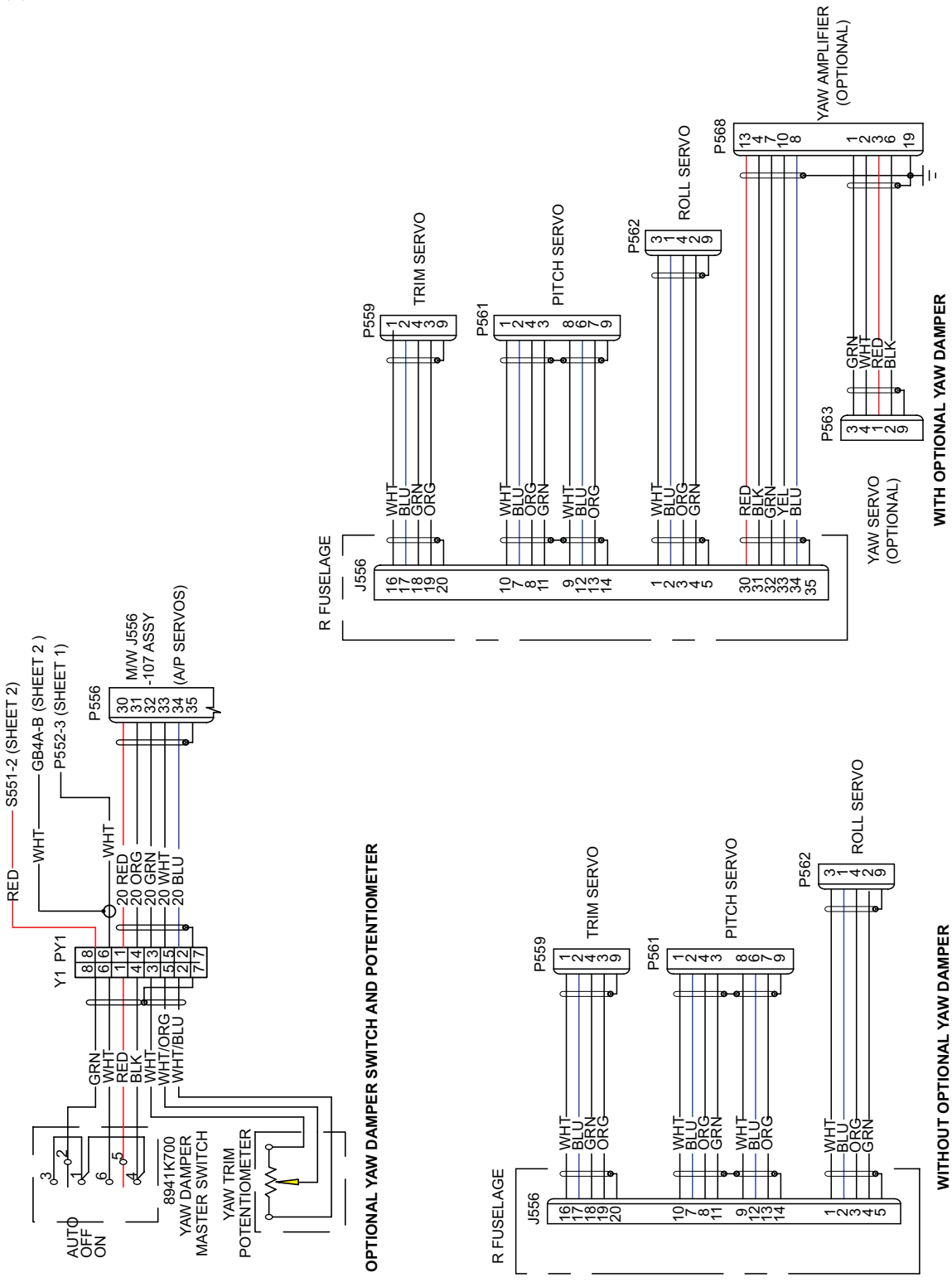
Effectivity
Seneca V
with Avidyne Entegra



Effectivity
Seneca V
with Avidyne Entegra

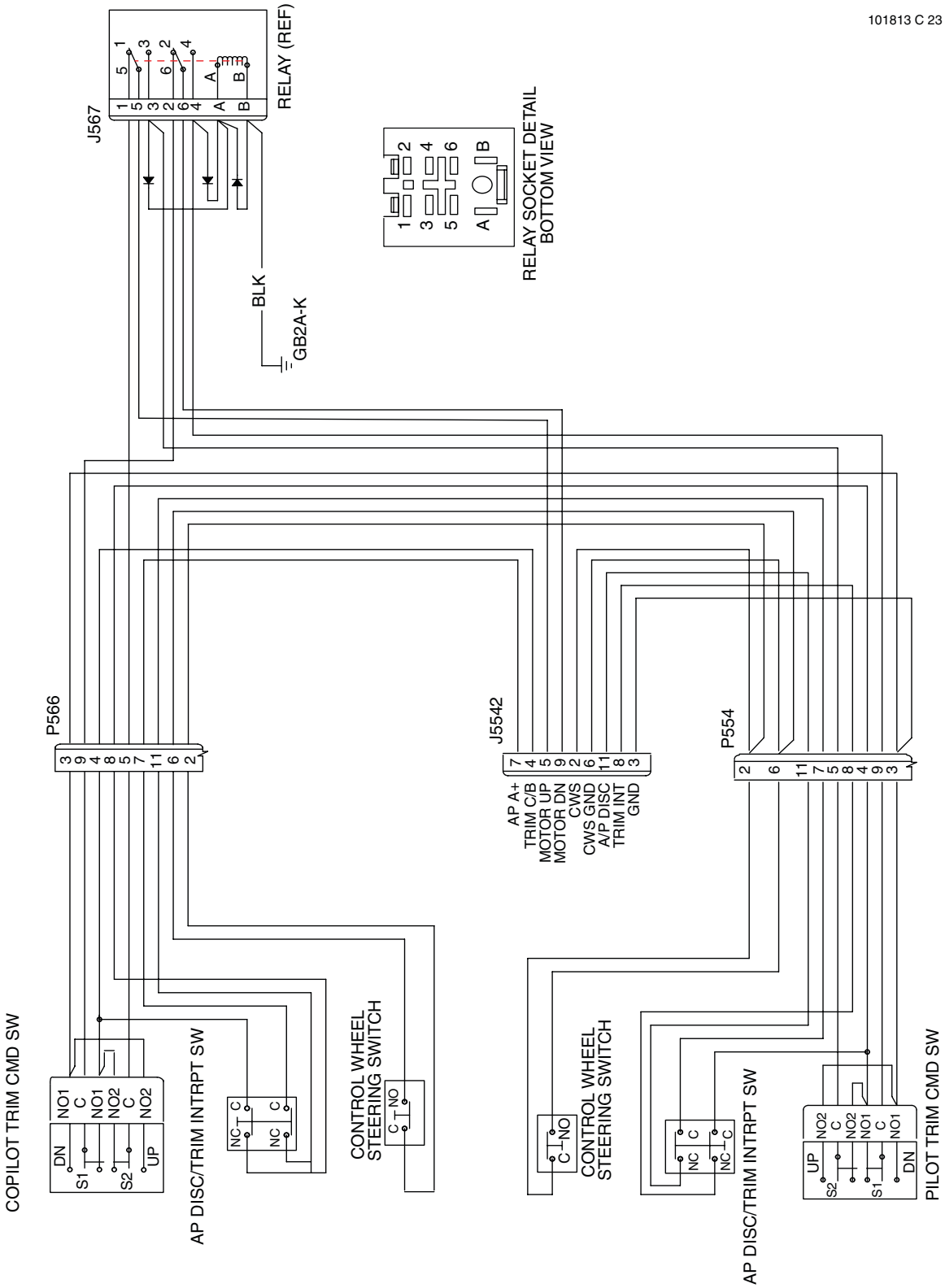
Autopilot
Figure 1 (Sheet 2 of 4)

101813 E 22



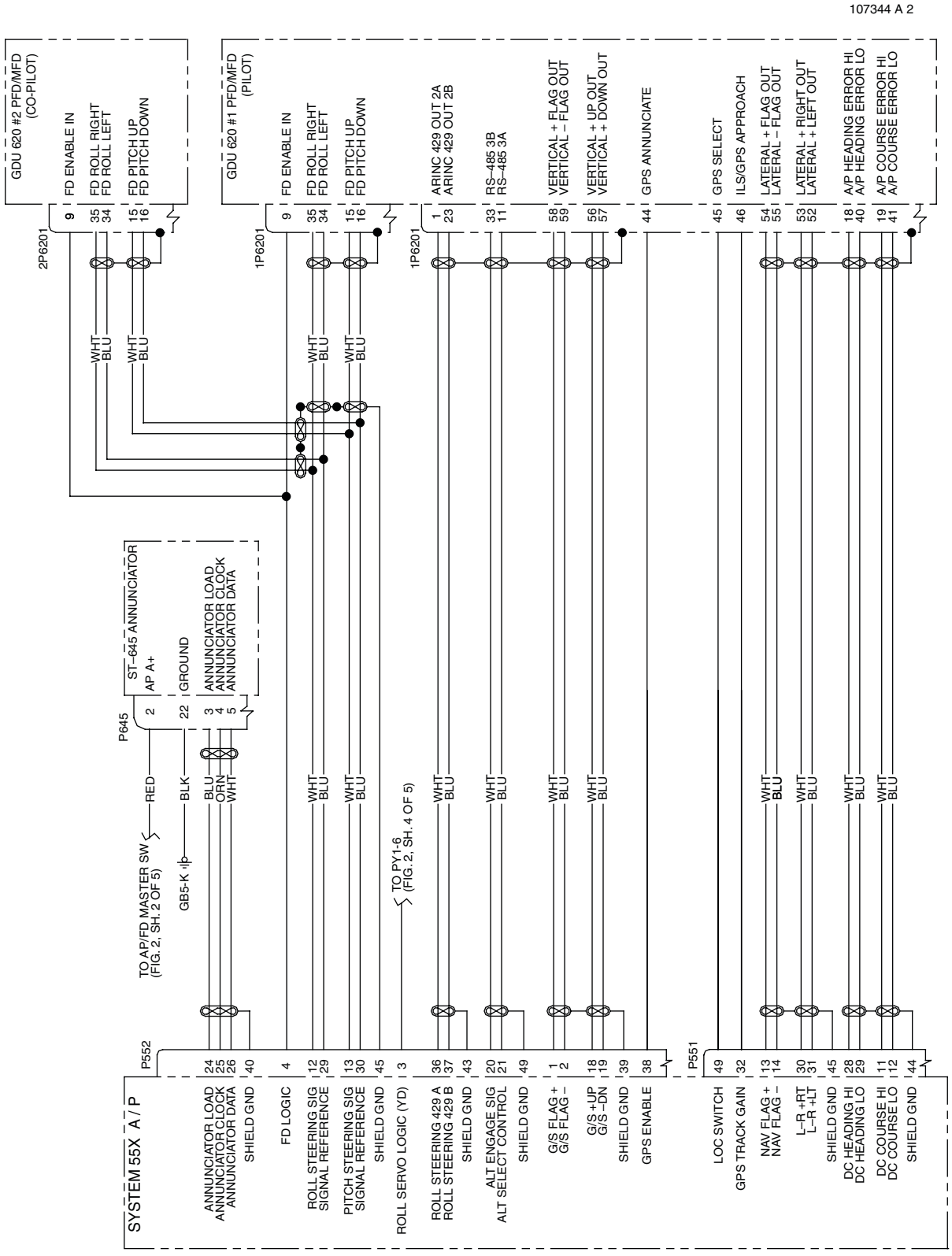
Autopilot
Figure 1 (Sheet 3 of 4)

Effectivity
Seneca V
with Avidyne Entegra



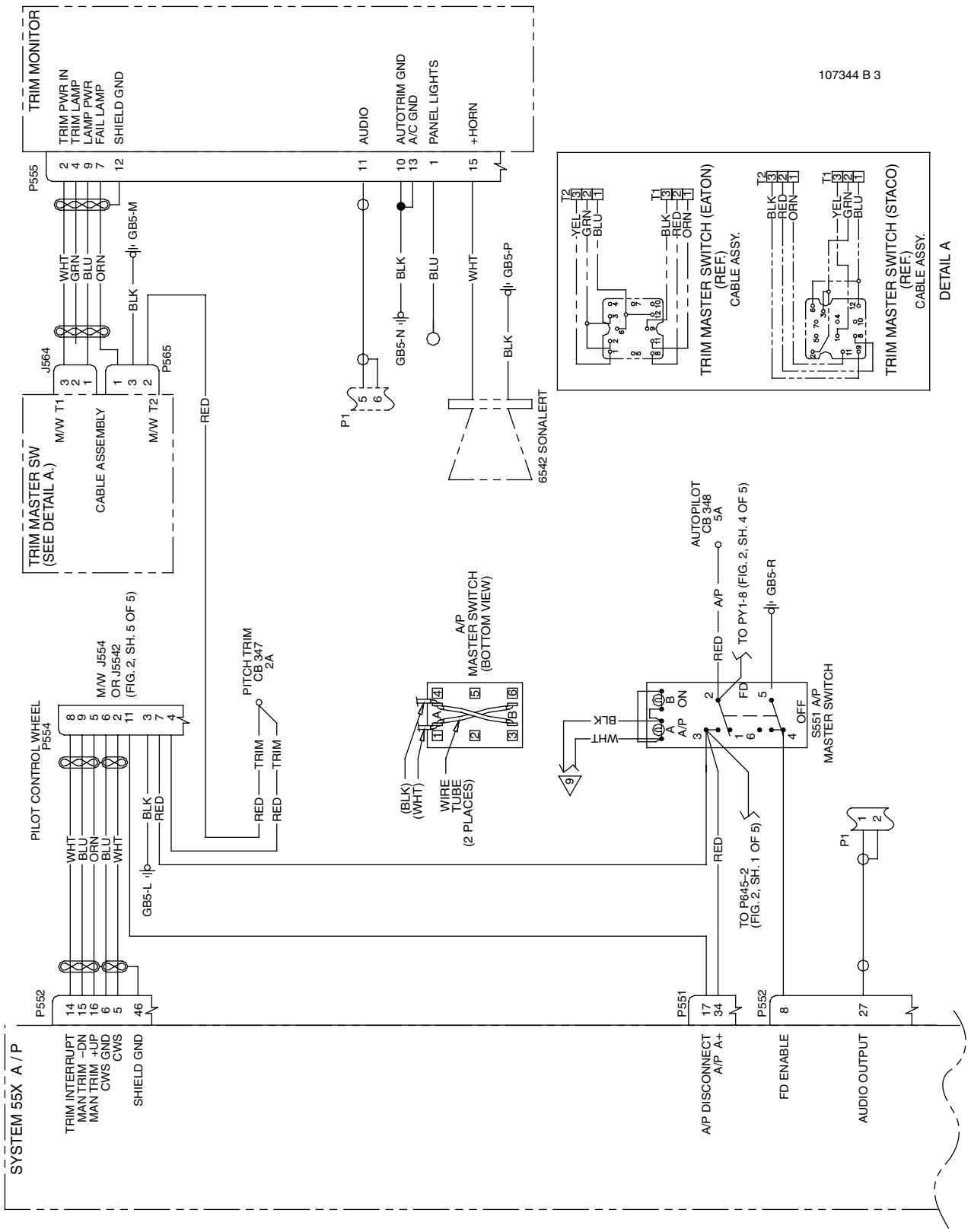
Effectivity
Seneca V
with Avidyne Entegra

Autopilot
Figure 1 (Sheet 4 of 4)



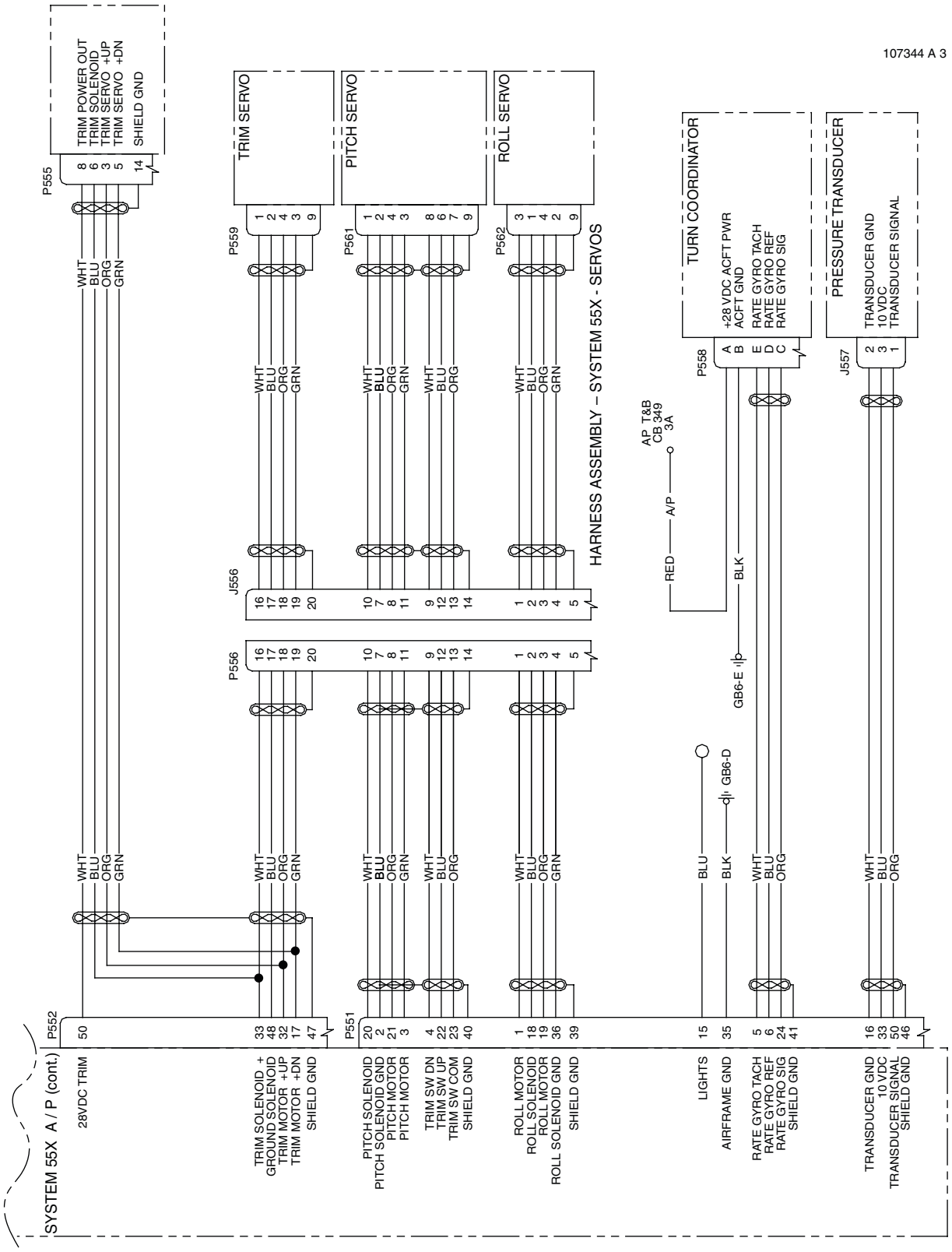
Autopilot
Figure 2 (Sheet 1 of 5)

Effectivity
Seneca V
with Garmin G600



Effectivity
Seneca V
with Garmin G600

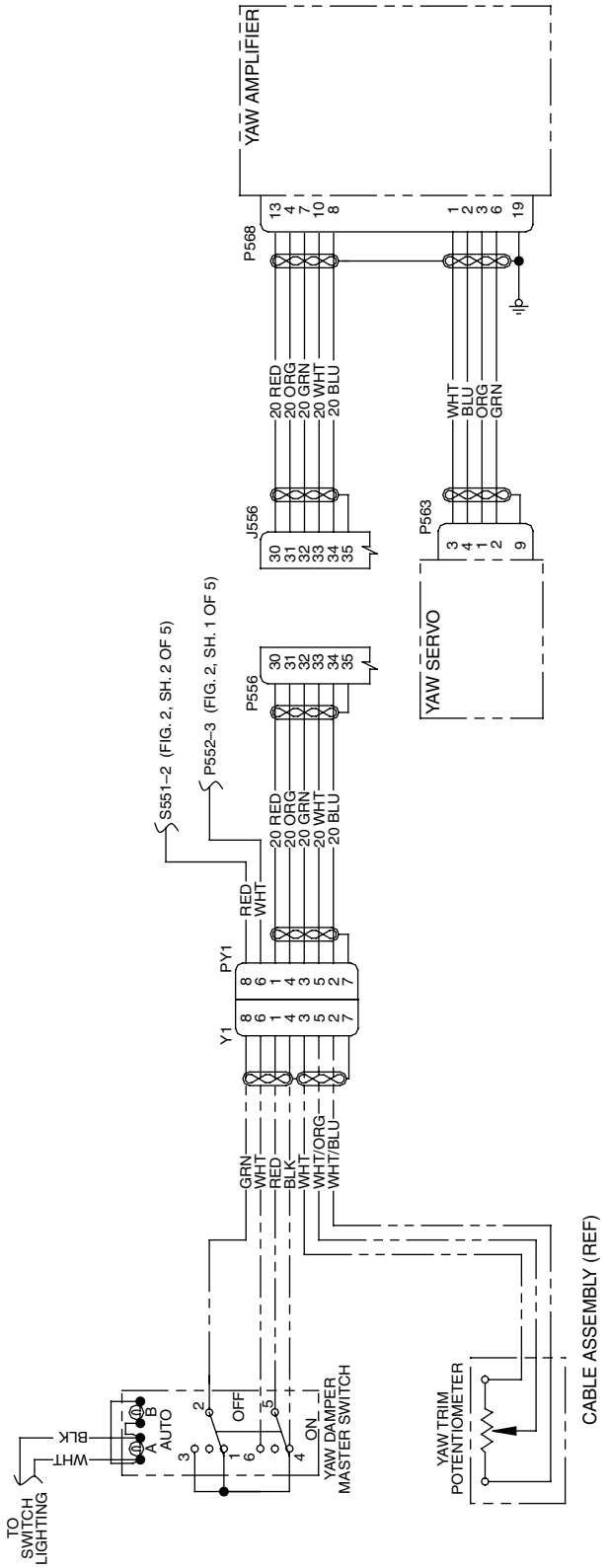
Autopilot
Figure 2 (Sheet 2 of 5)



Autopilot
Figure 2 (Sheet 3 of 5)

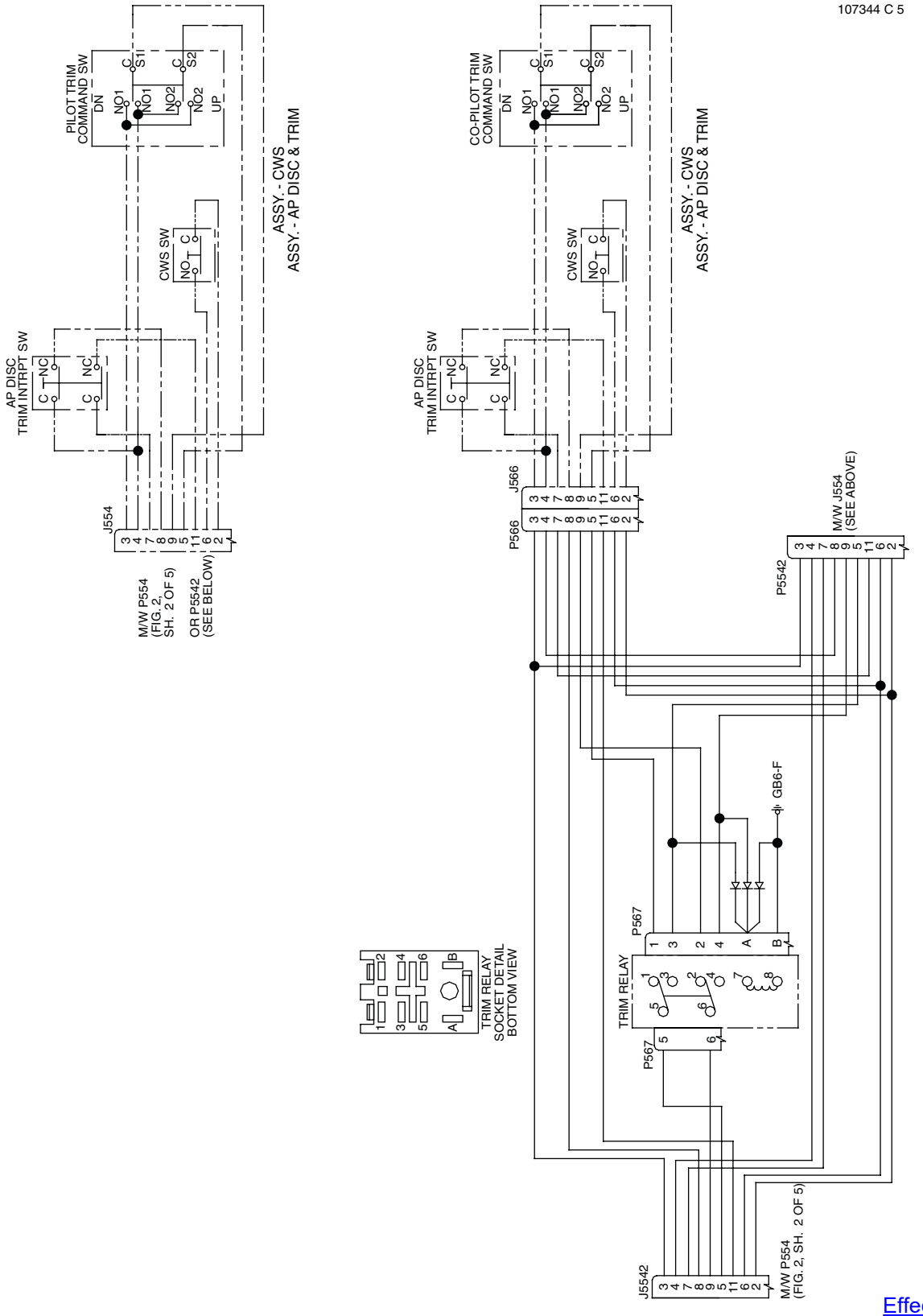
Effectivity
Seneca V
with Garmin G600

107344 A 4



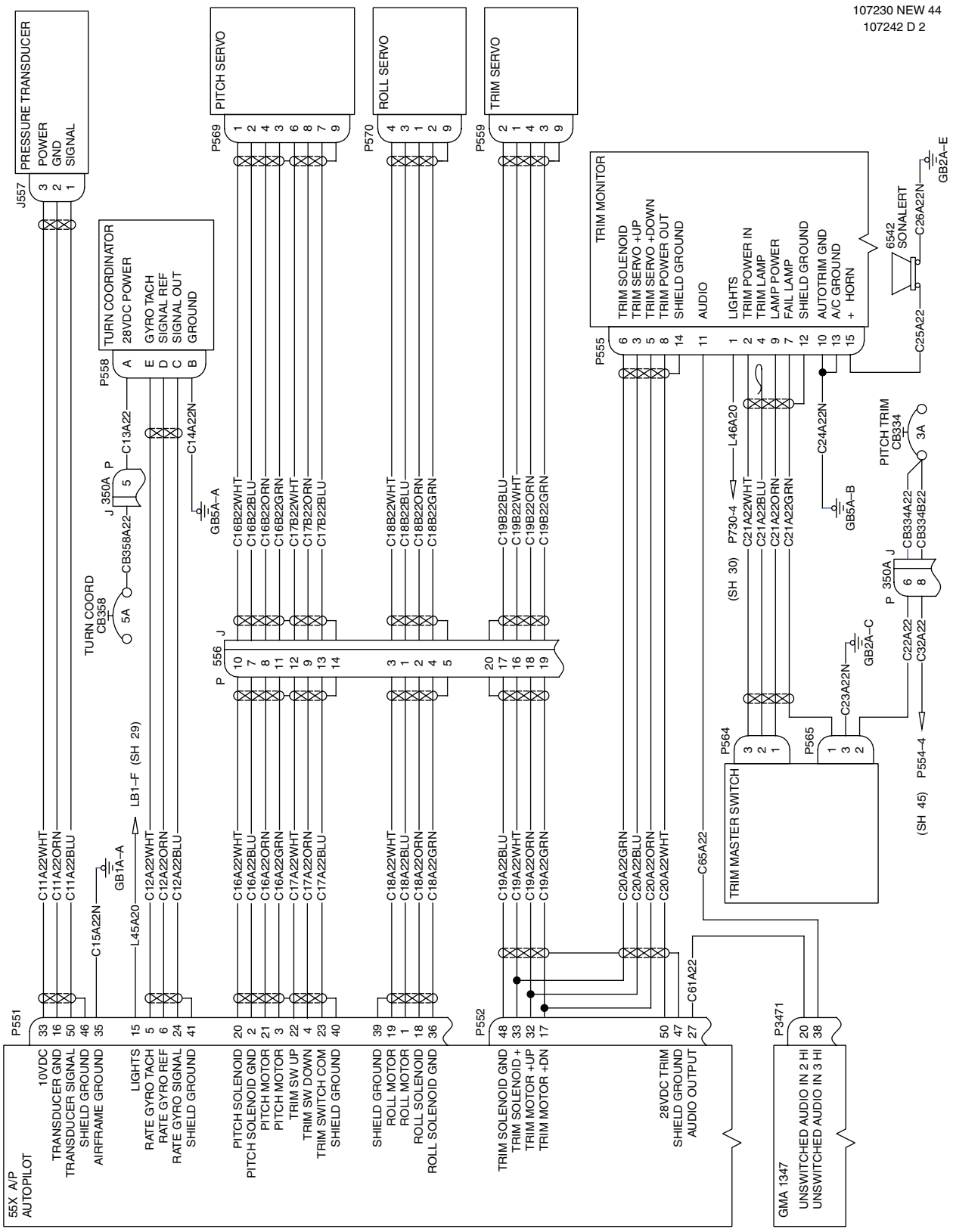
Effectivity
Seneca V
with Garmin G600

Autopilot
Figure 2 (Sheet 4 of 5)



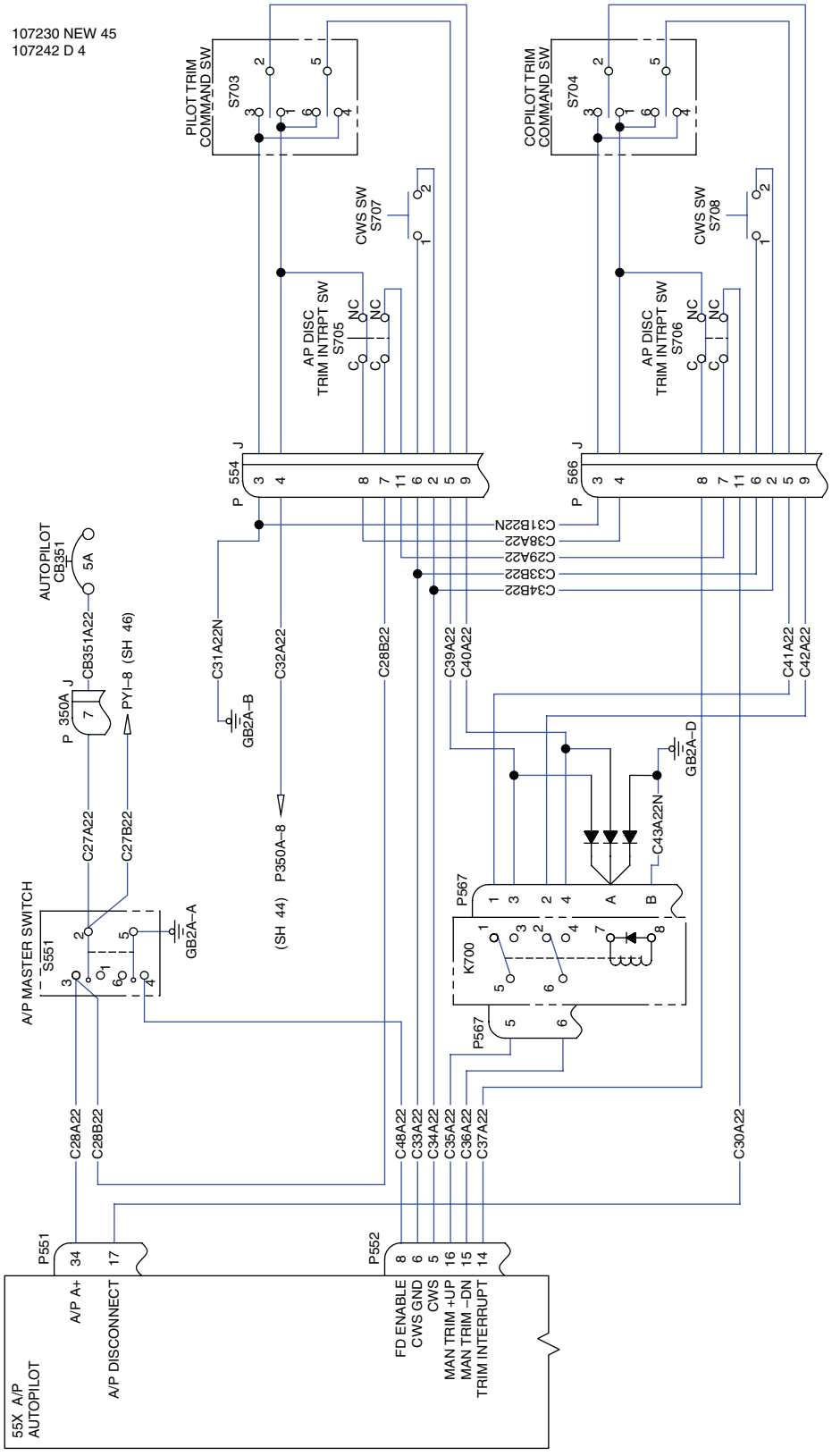
Autopilot
Figure 2 (Sheet 5 of 5)

Effectivity
Seneca V
with Garmin G600



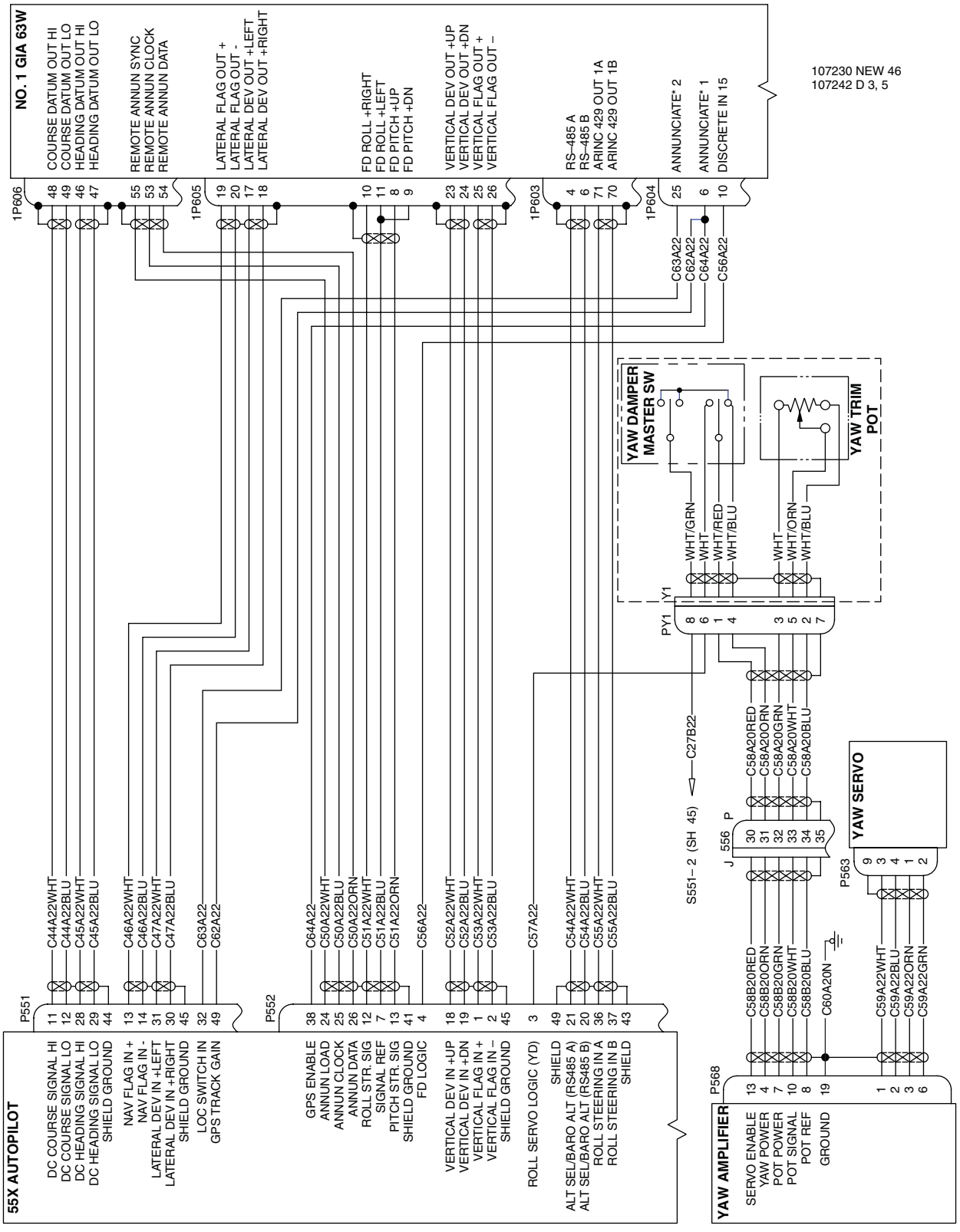
Effectivity
Seneca V
with Garmin G1000

Autopilot - S-TEC 55X
Figure 3 (Sheet 1 of 3)



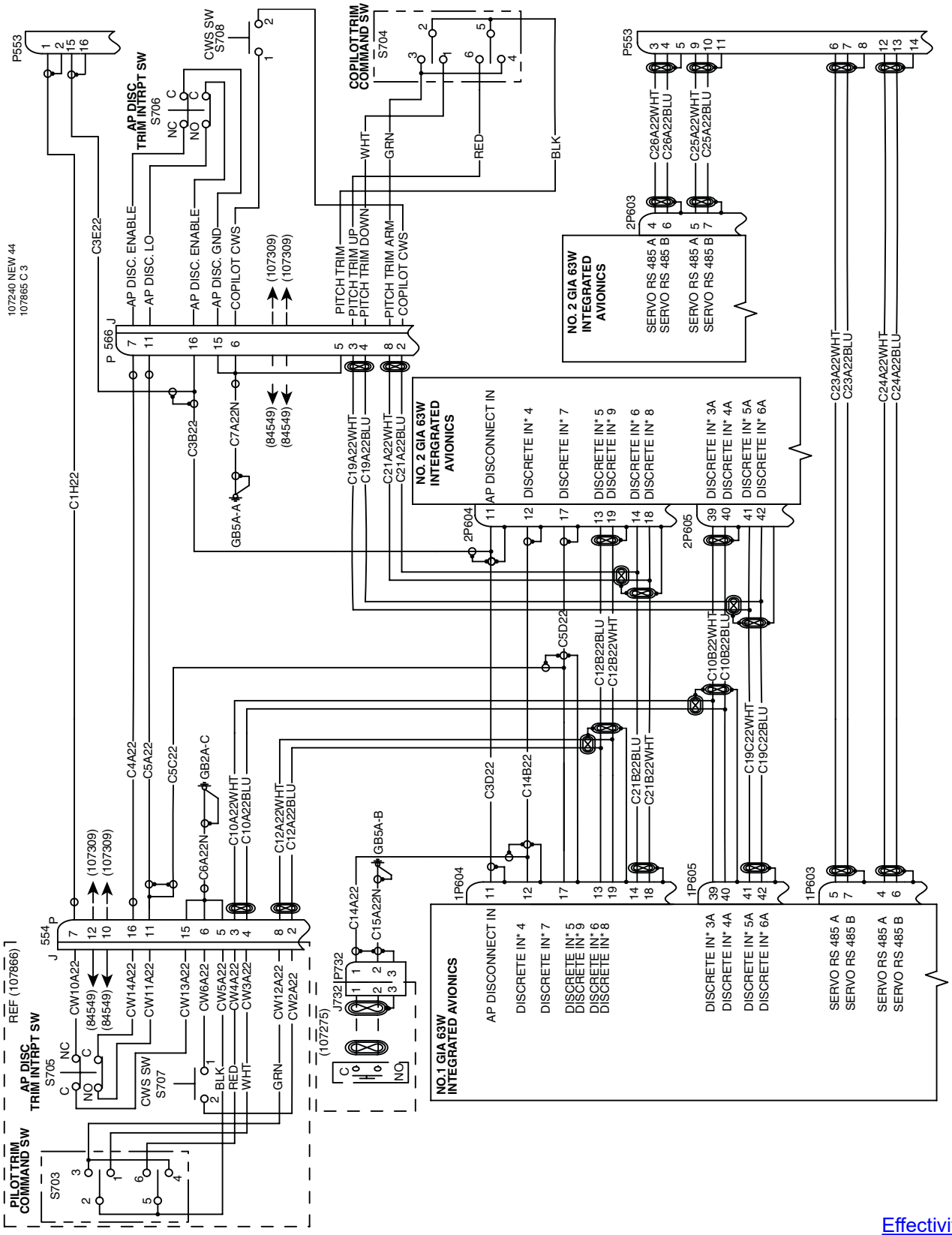
Autopilot - S-TEC 55X
Figure 3 (Sheet 2 of 3)

Effectivity
Seneca V
with Garmin G1000



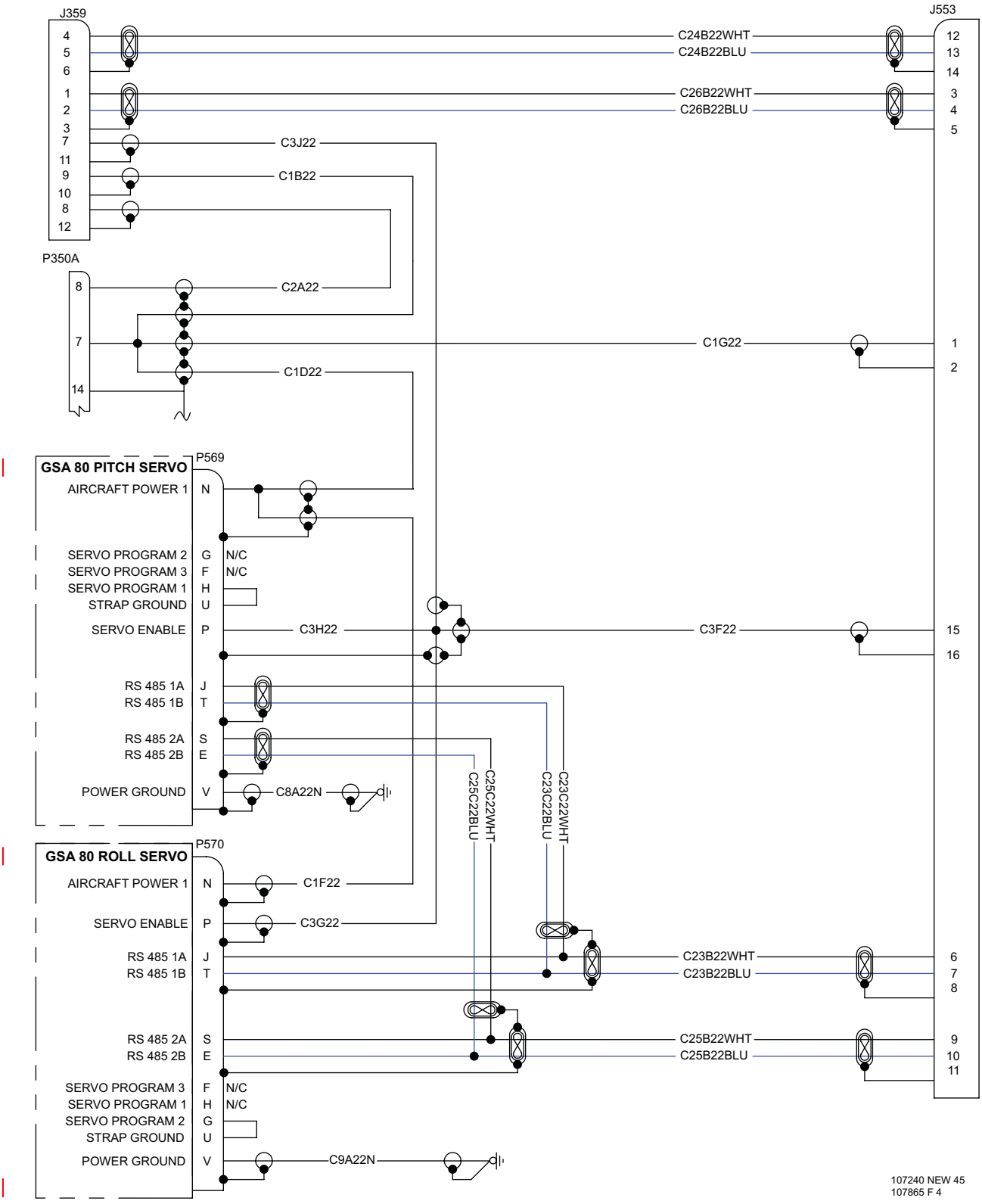
Effectivity
Seneca V
with Garmin G1000

Autopilot - S-TEC 55X
Figure 3 (Sheet 3 of 3)



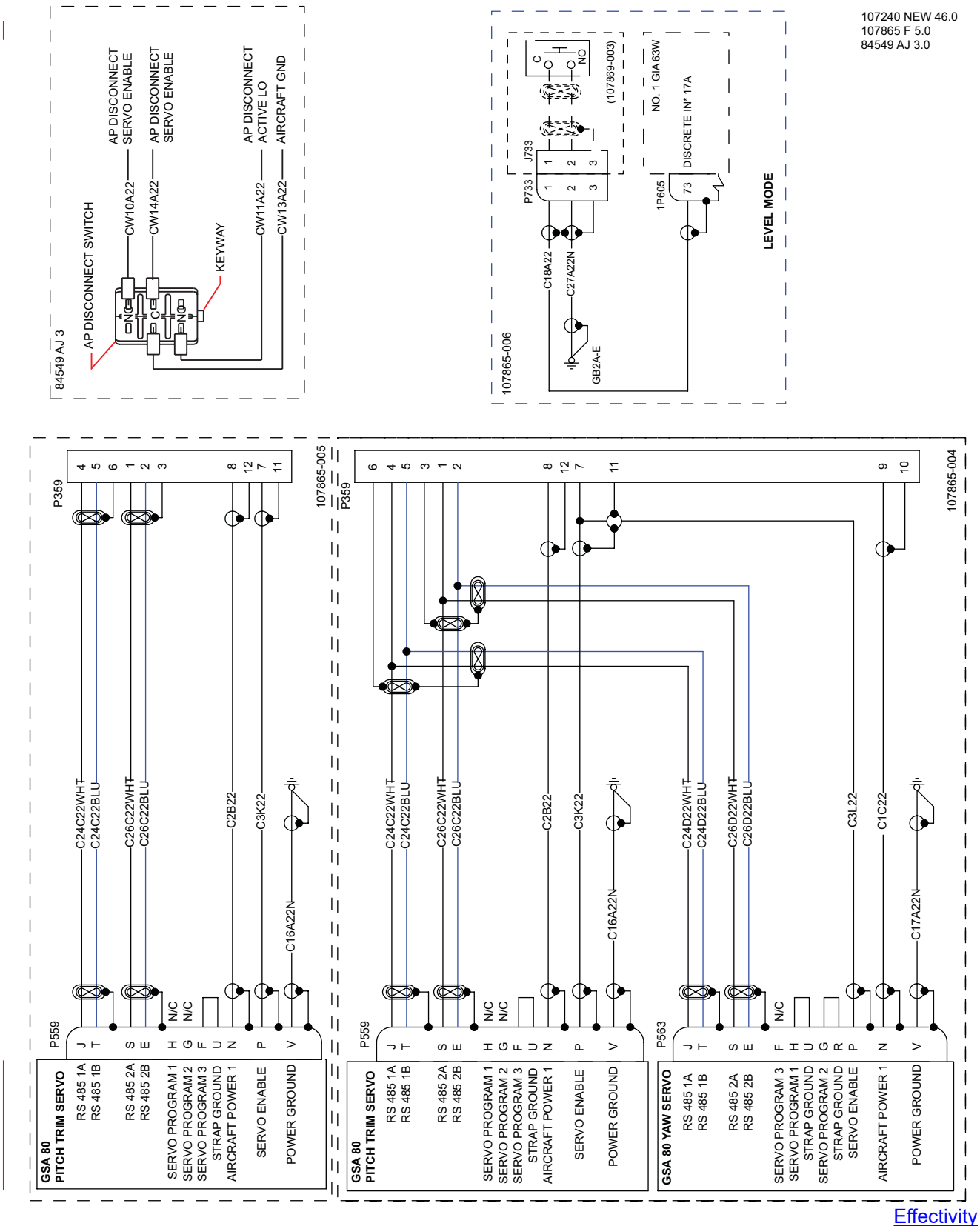
Autopilot - GFC 700
Figure 4 (Sheet 1 of 3)

Effectivity
Seneca V
with Garmin G1000



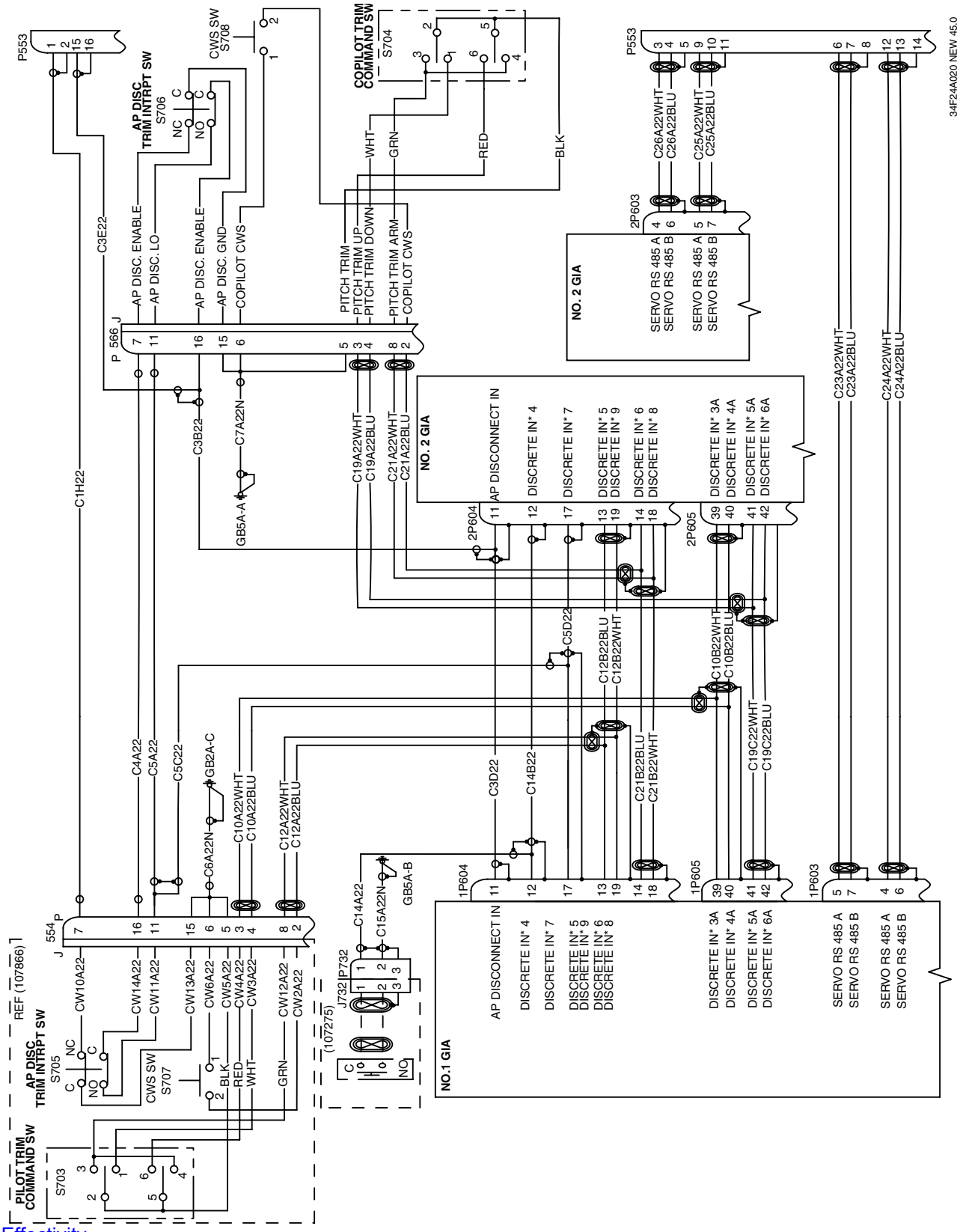
Effectivity
Seneca V
with Garmin G1000

Autopilot - GFC 700
Figure 4 (Sheet 2 of 3)



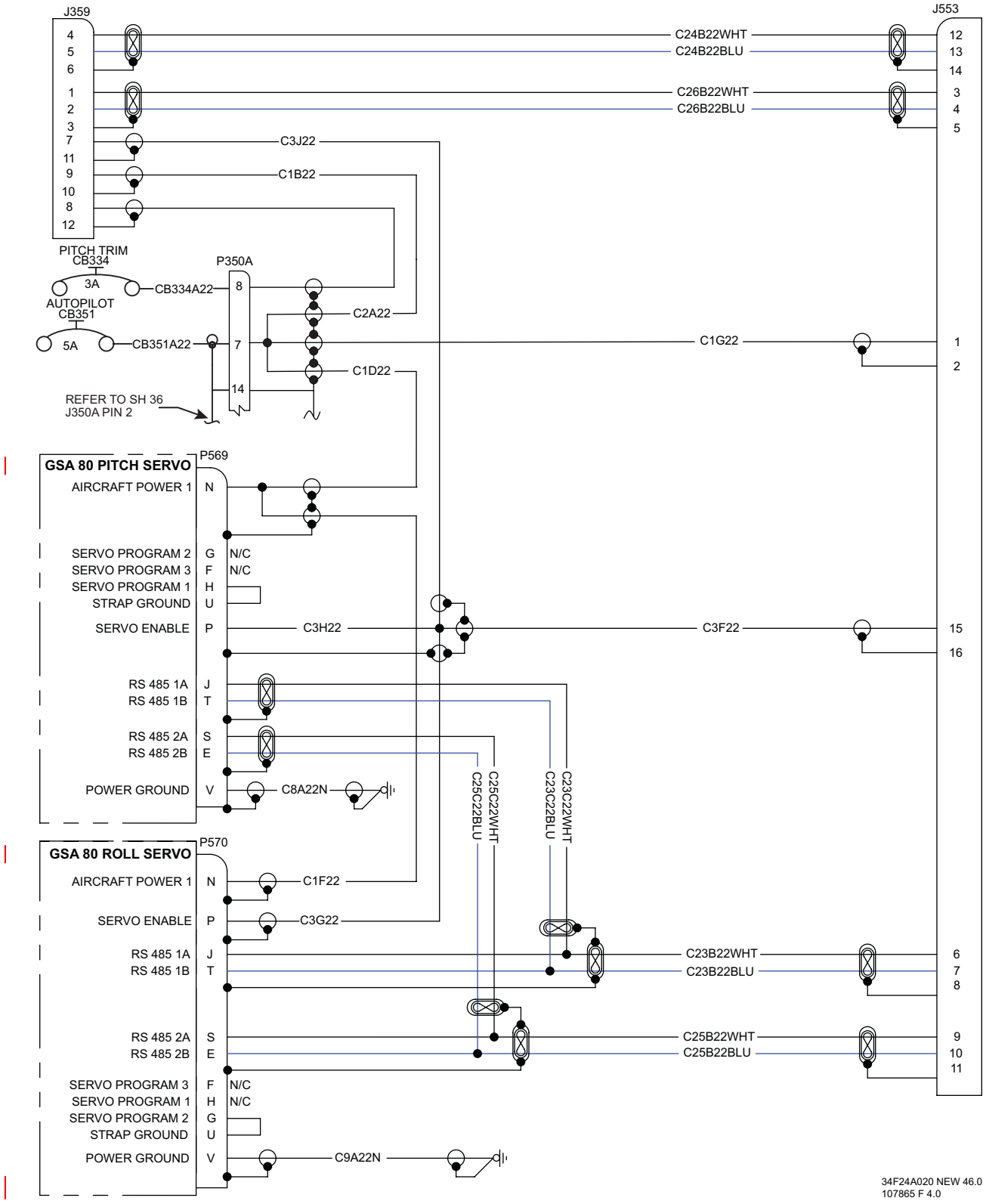
Autopilot - GFC 700
Figure 4 (Sheet 3 of 3)

Effectivity
Seneca V
with Garmin G1000



Effectivity
Seneca V
with Garmin G1000 NXi

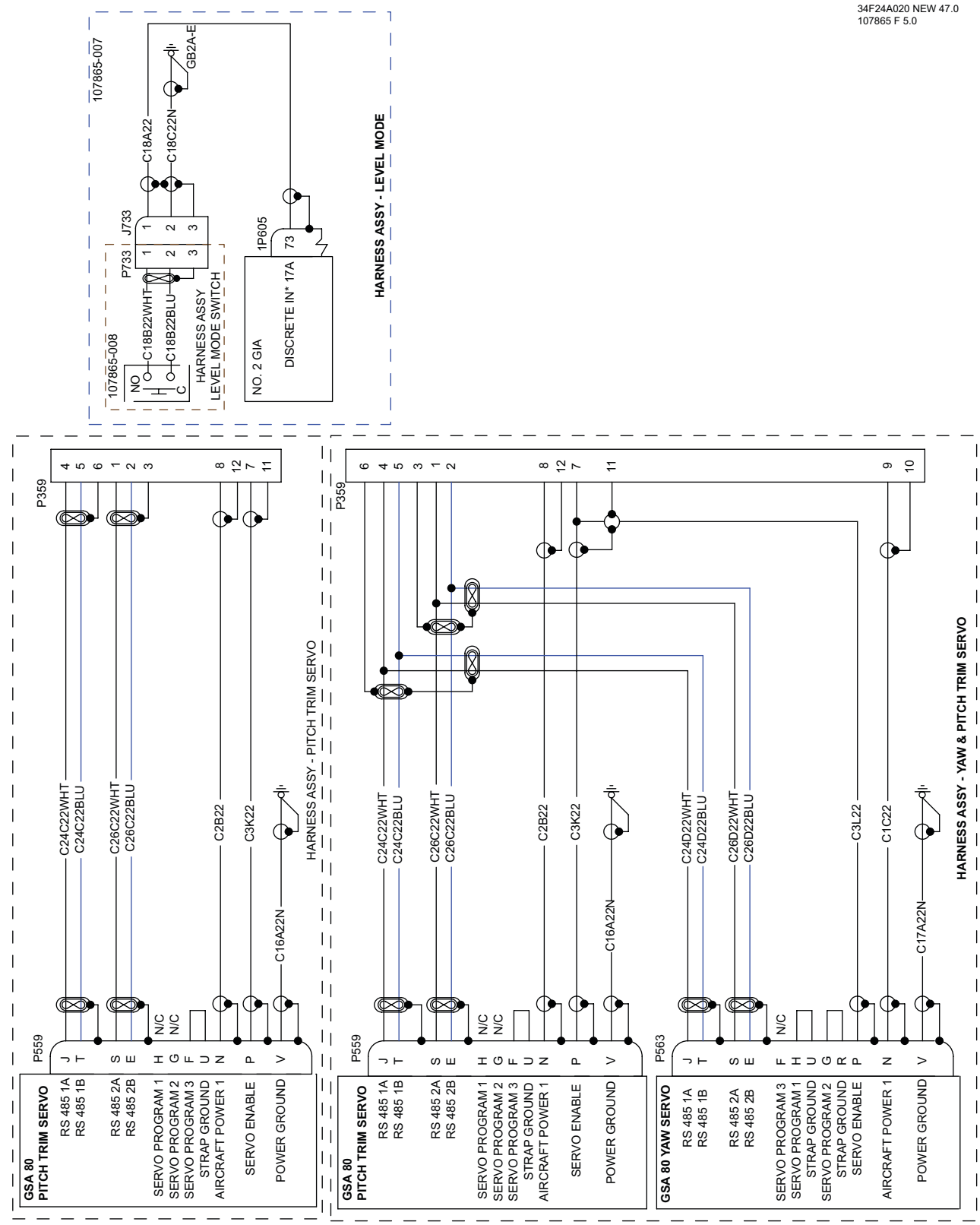
Autopilot - GFC 700
Figure 5 (Sheet 1 of 3)



Autopilot - GFC 700
Figure 5 (Sheet 2 of 3)

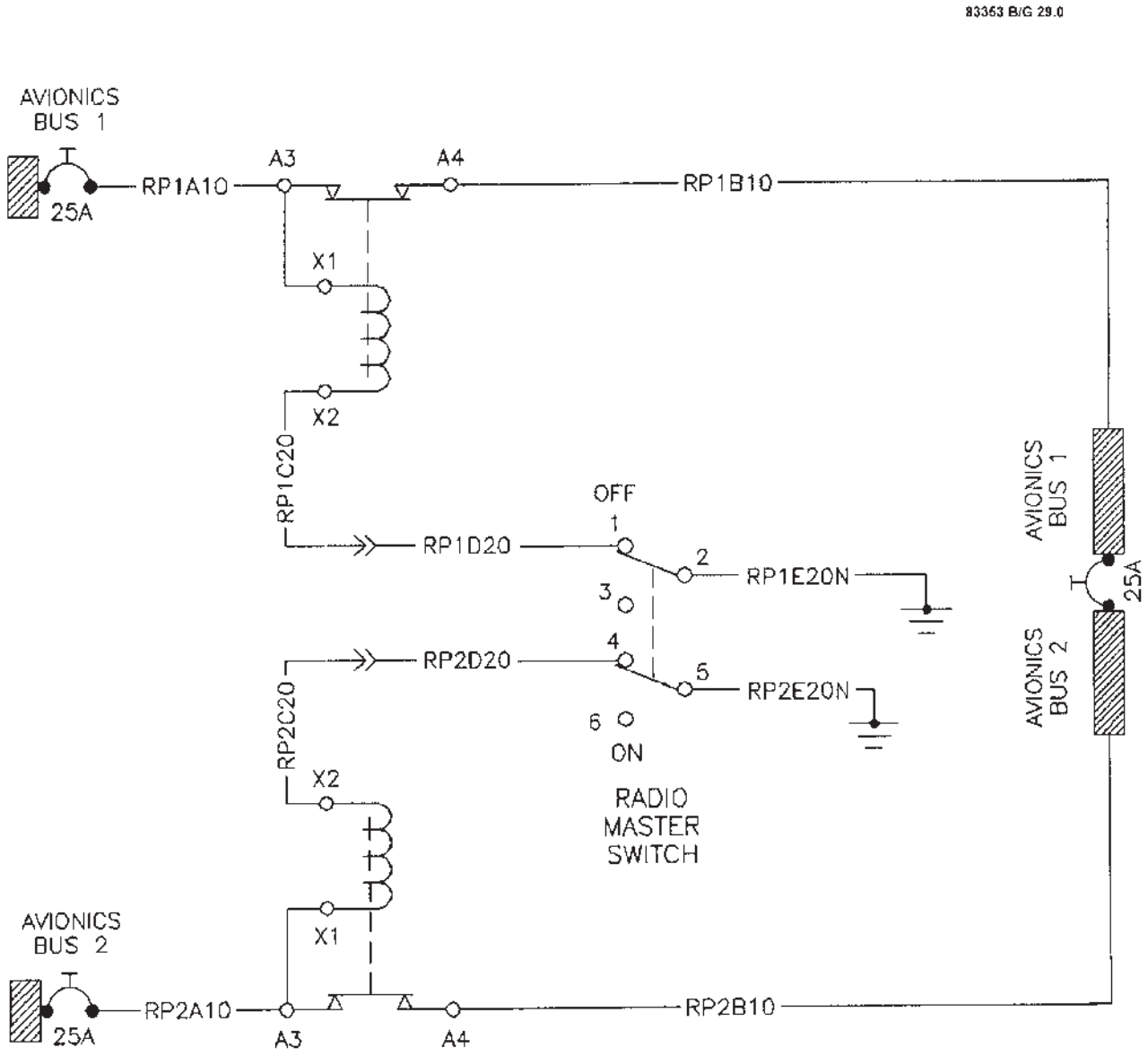
Effectivity
Seneca V
with Garmin G1000 NXi

34F24A020 NEW 47.0
107865 F 5.0



Effectivity
Seneca V
with Garmin G1000 NXi

Autopilot - GFC 700
Figure 5 (Sheet 3 of 3)

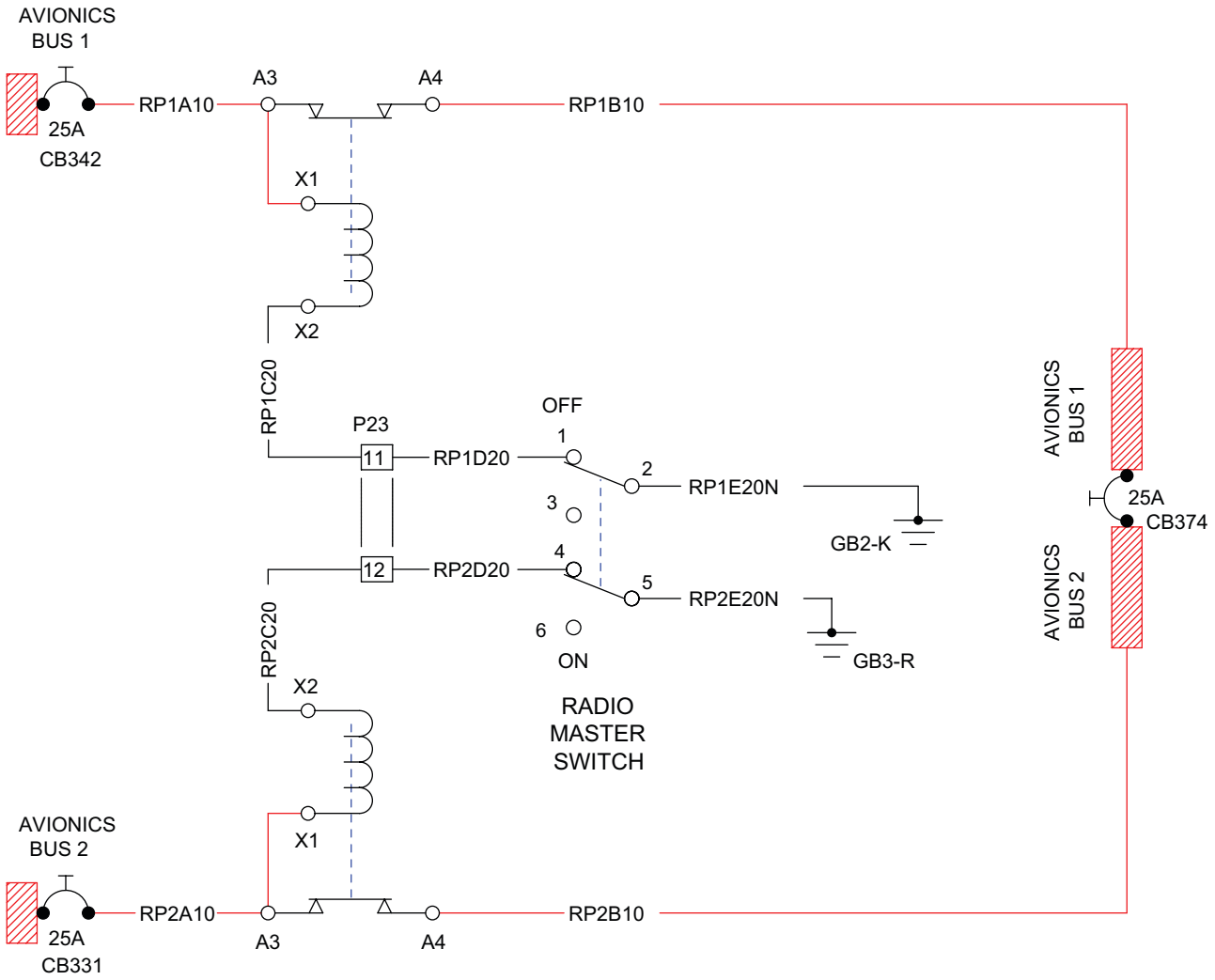


(SHOWN WITH POWER OFF)

Radio Master Switch
Figure 1 (Sheet 1 of 3)

Effectivity
Seneca IV

100651 NEW/T 29
101288 NEW/D 29
104229 C 29
104368 A/K 29

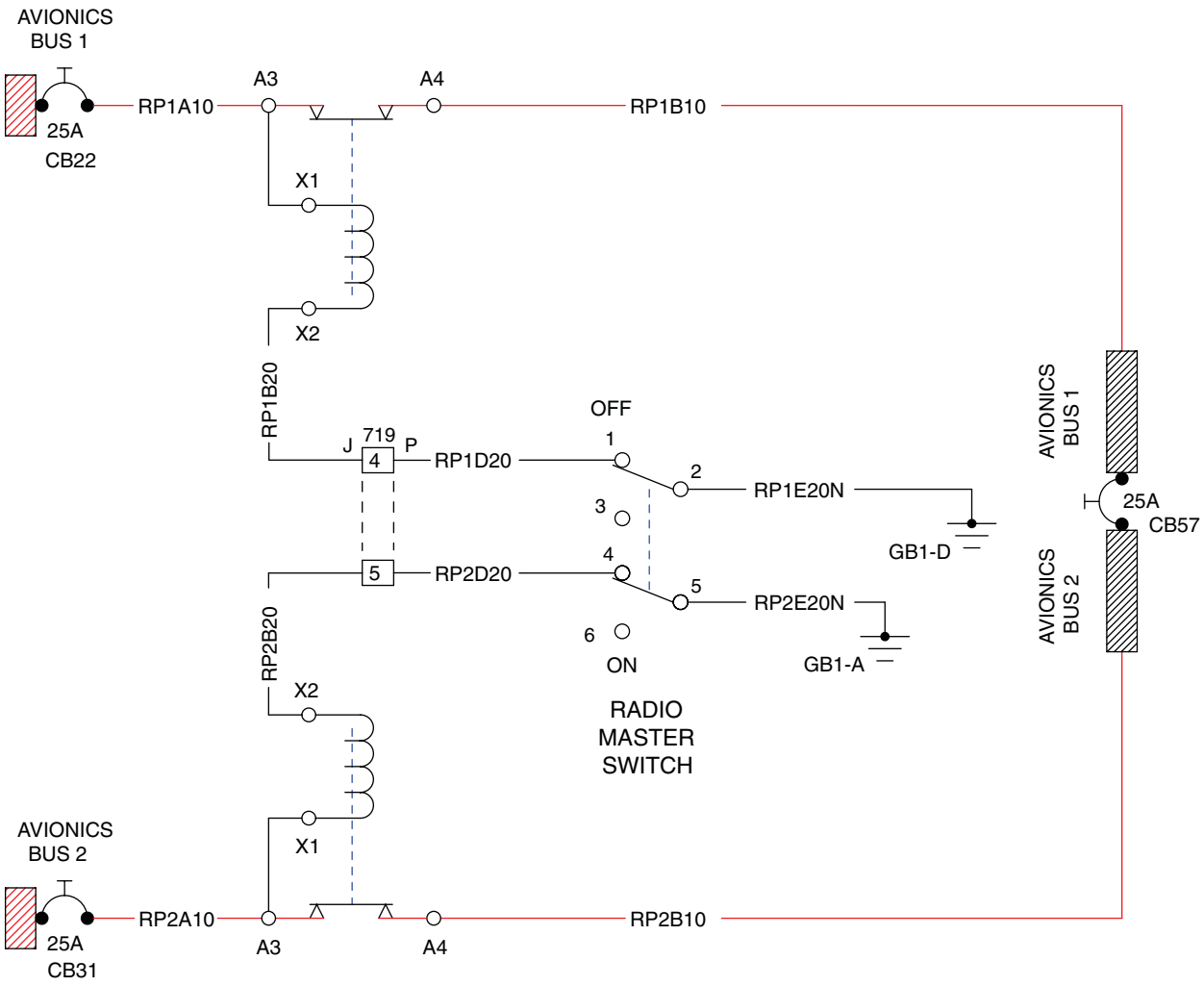


(SHOWN WITH POWER OFF)

Effectivity
Seneca V

Radio Master Switch
Figure 1 (Sheet 2 of 3)

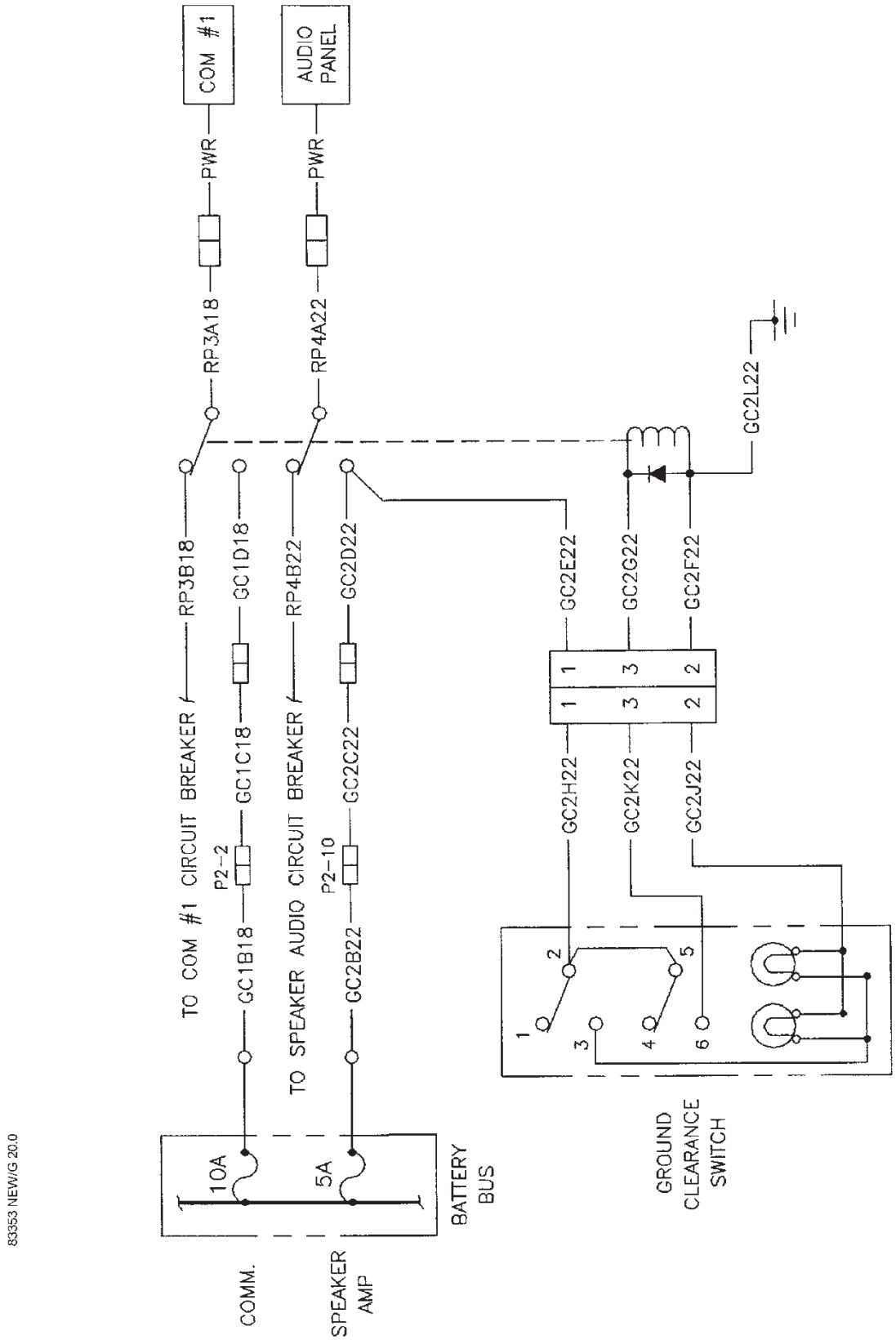
101840 NEW 22.0



Radio Master Switch
Figure 1 (Sheet 3 of 3)

Effectivity
Seneca V
with Avidyne Entegra

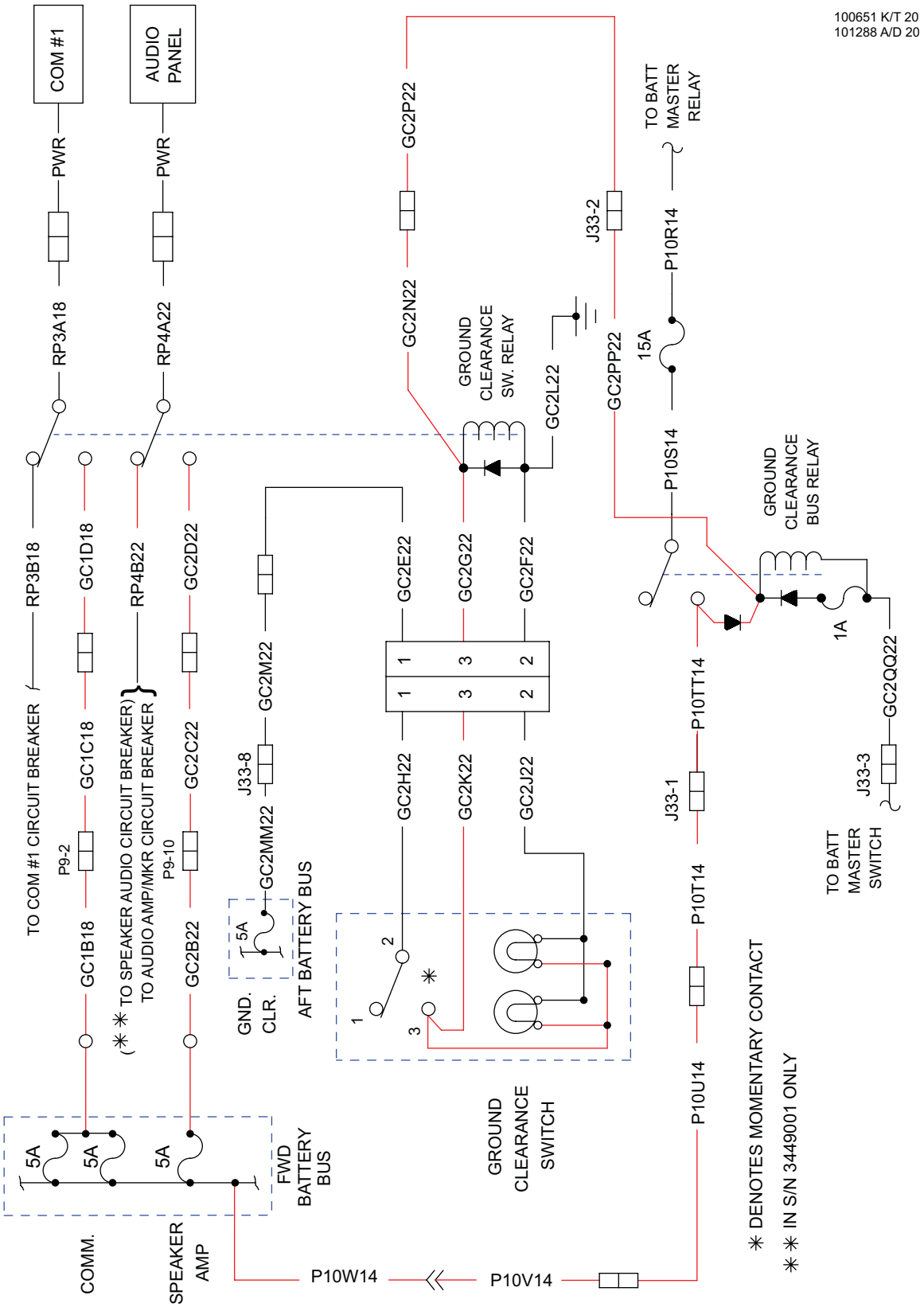
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Ground Clearance Switch
Figure 2 (Sheet 1 of 4)

Effectivity
Seneca IV

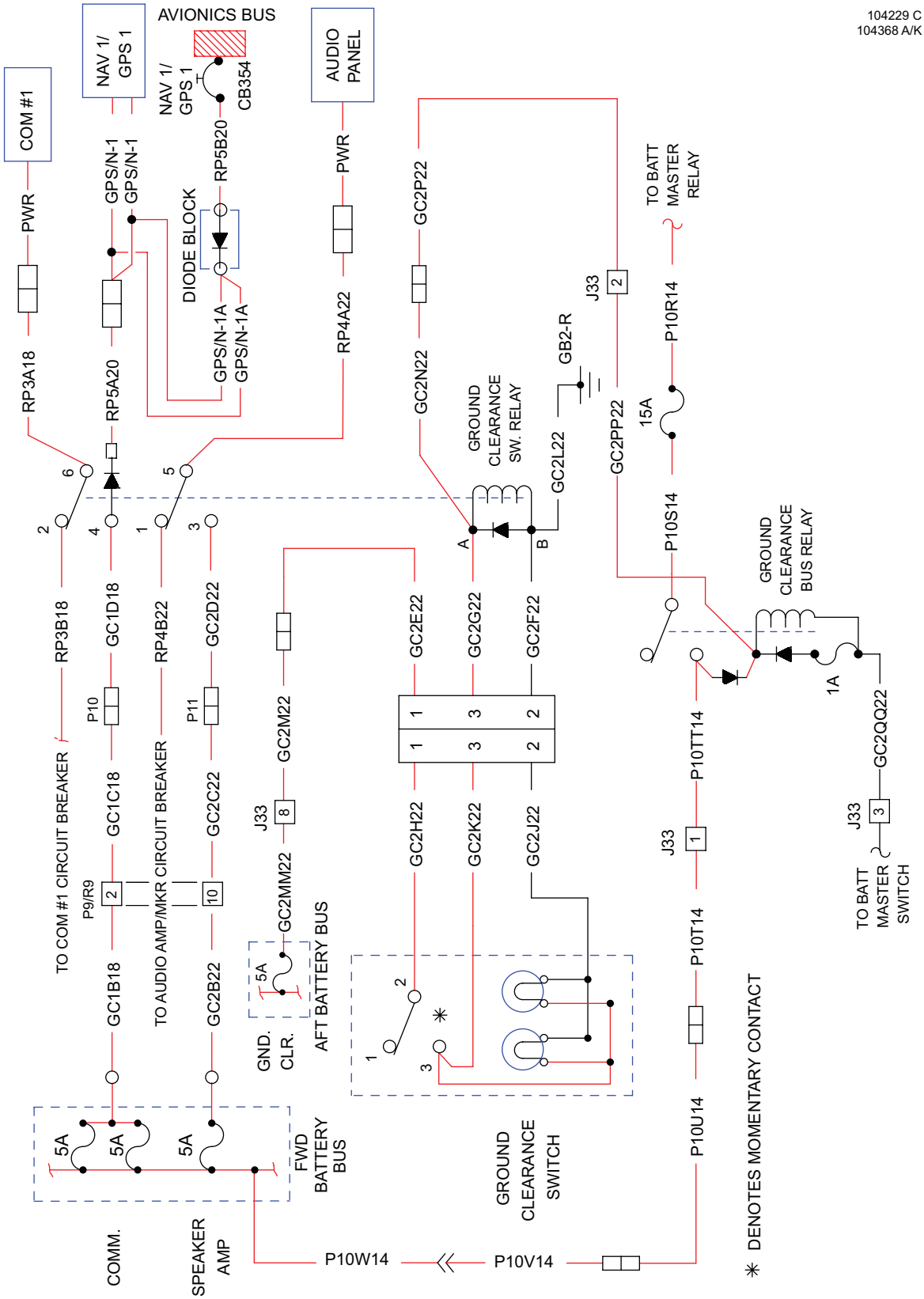
100651 K/T 20
101288 A/D 20



Effectivity
3449001 thru 3449151

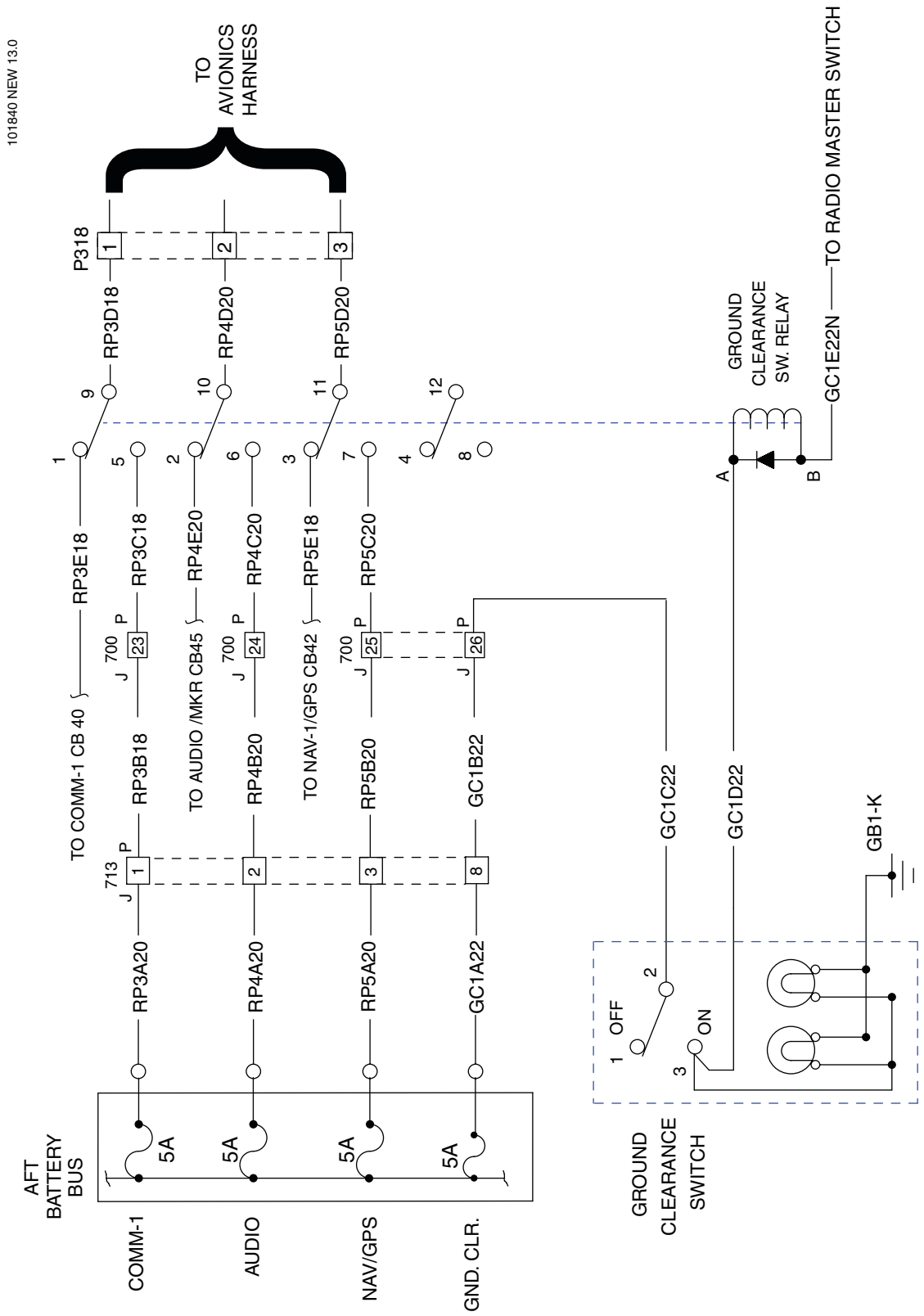
Ground Clearance Switch
Figure 2 (Sheet 2 of 4)

104229 C 20
104368 A/K 20



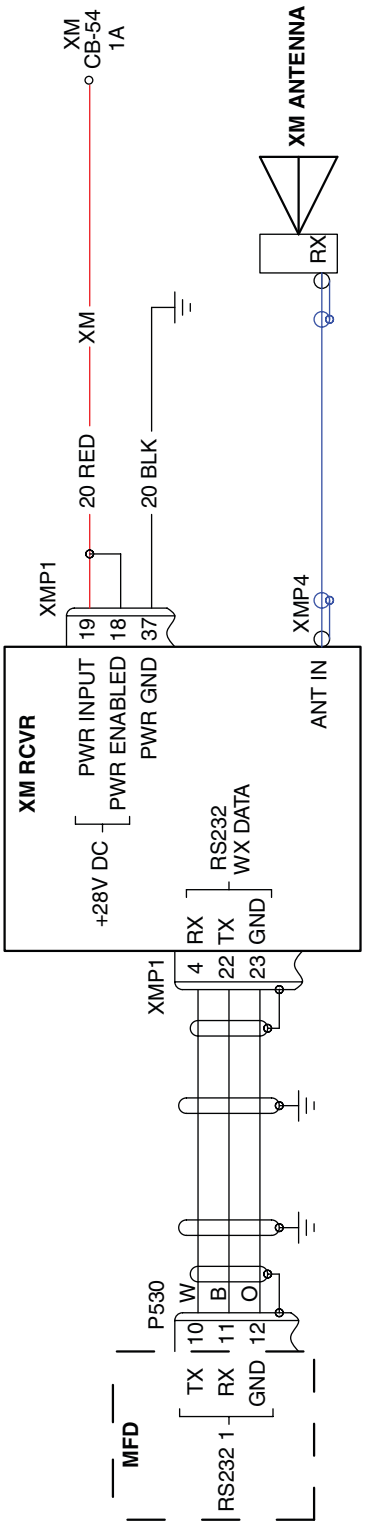
Ground Clearance Switch
Figure 2 (Sheet 3 of 4)

Effectivity
3449152 & up



Effectivity
Seneca V
with Avidyne Entegra

Ground Clearance Switch
Figure 2 (Sheet 4 of 4)

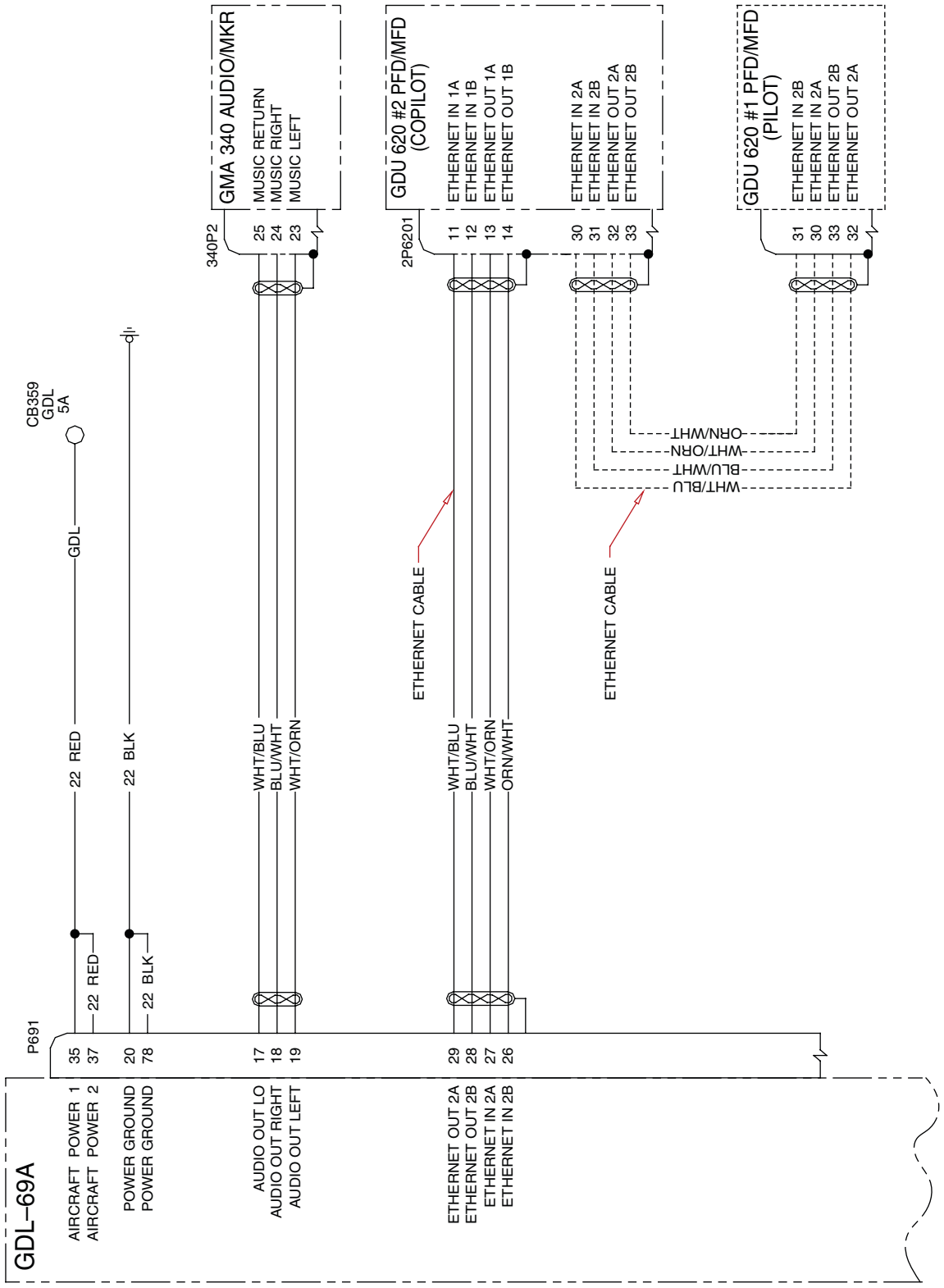


101813 B 25.0

XM Receiver (Optional)
Figure 1

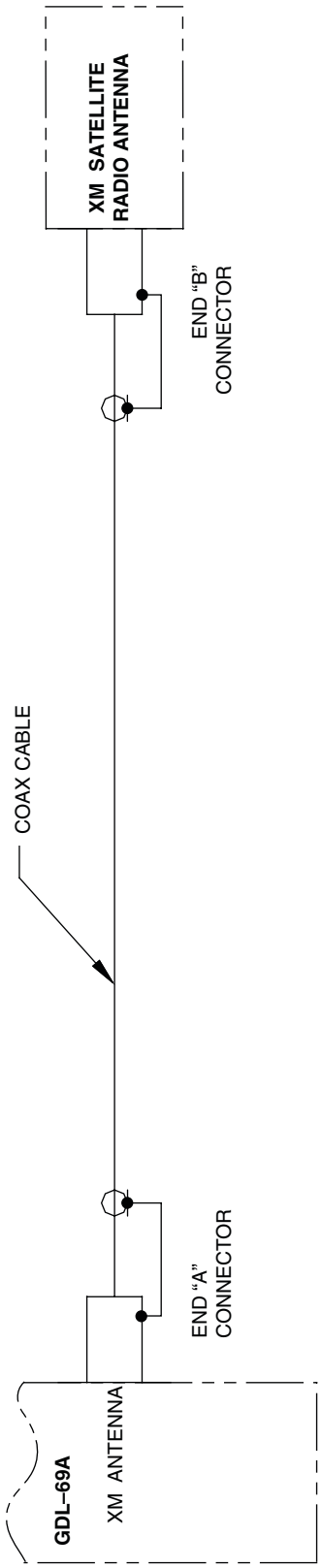
Effectivity
Seneca V
with Avidyne Entegra

107348 B 2



Effectivity
Seneca V
with Garmin G600

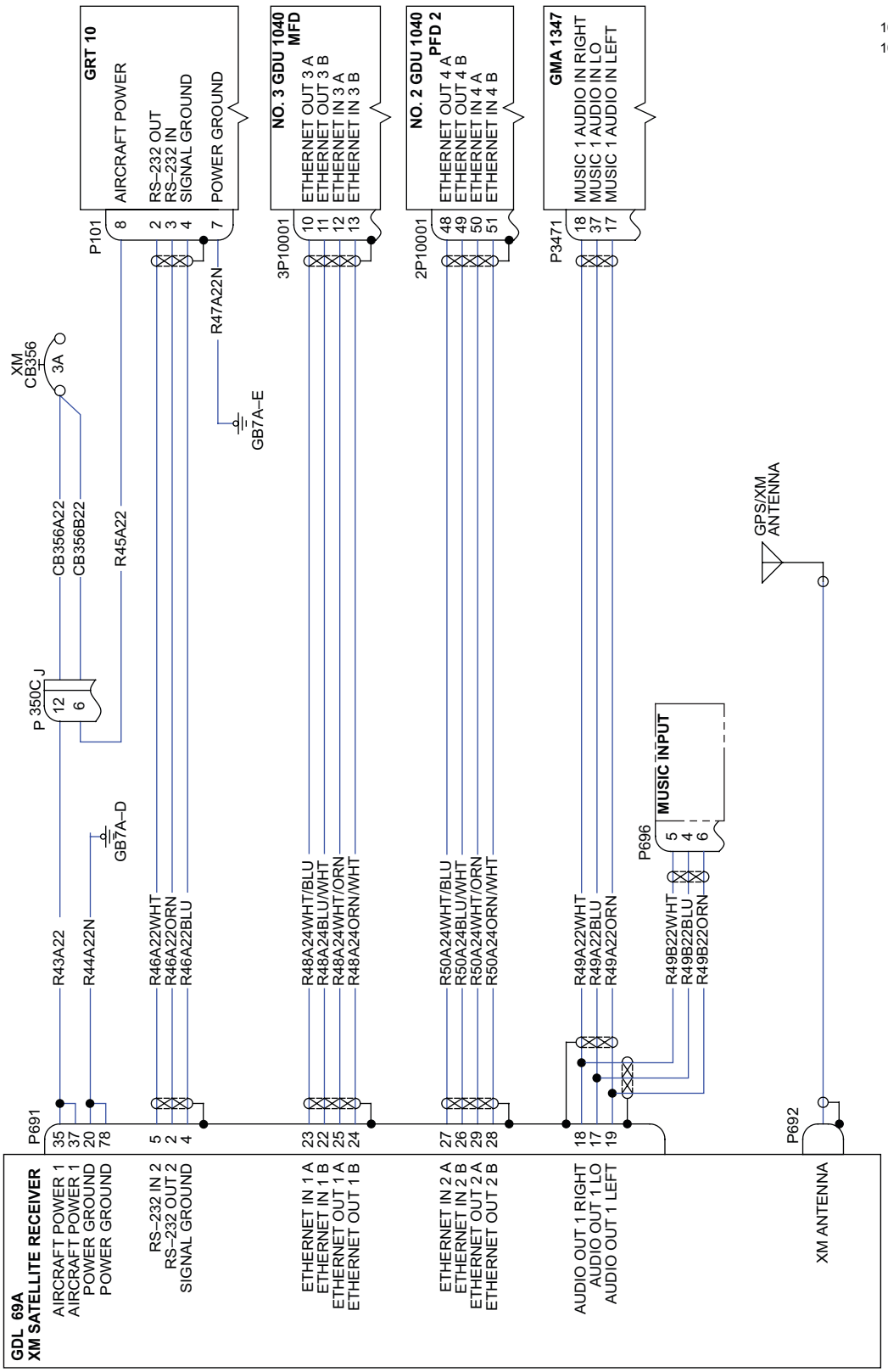
XM Receiver (Optional)
Figure 2 (Sheet 1 of 2)



XM Receiver (Optional)
Figure 2 (Sheet 2 of 2)

Effectivity
Seneca V
with Garmin G600

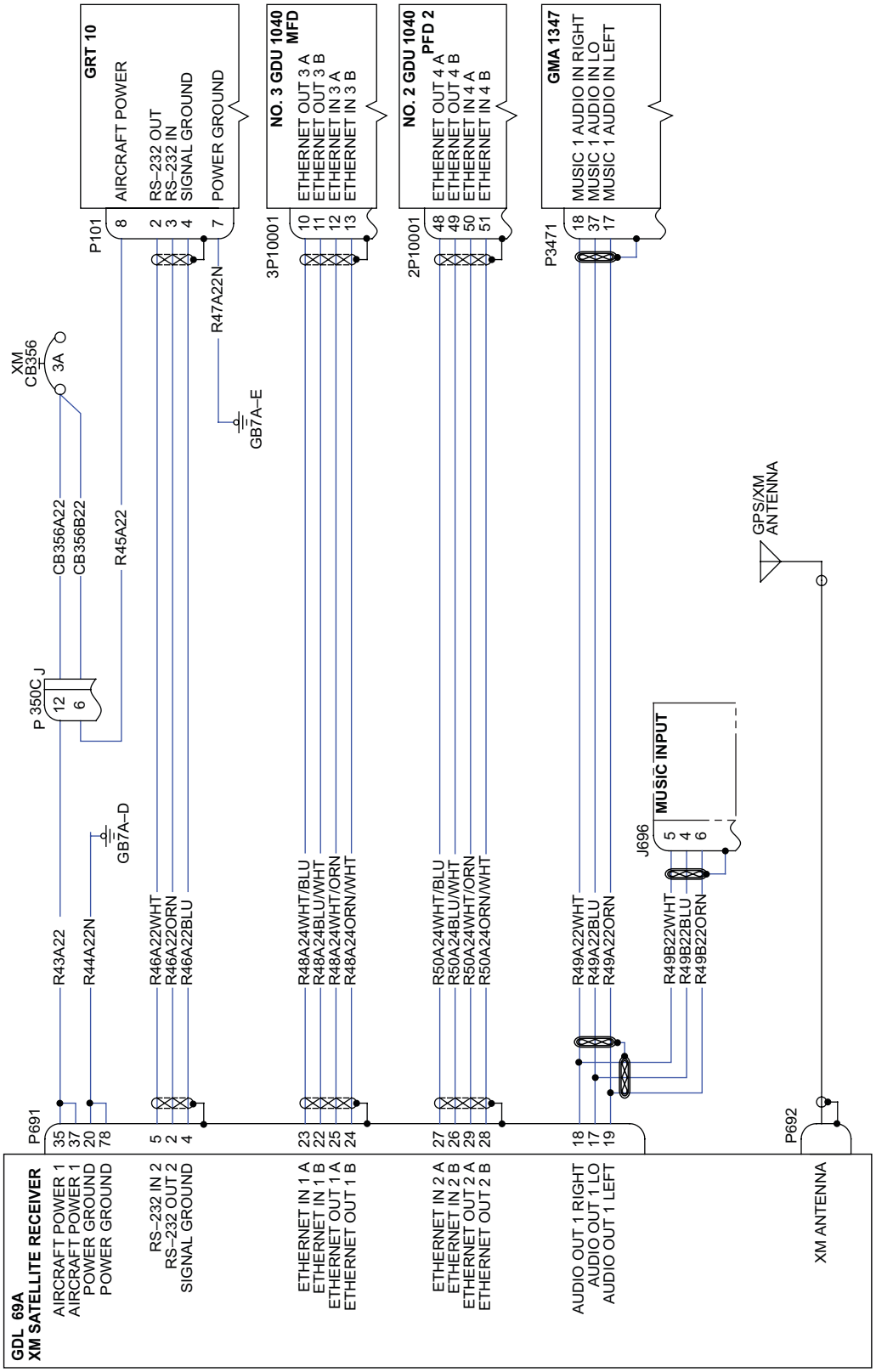
107230 NEW 51
107248 C 2



Effectivity
Seneca V
with Garmin G1000

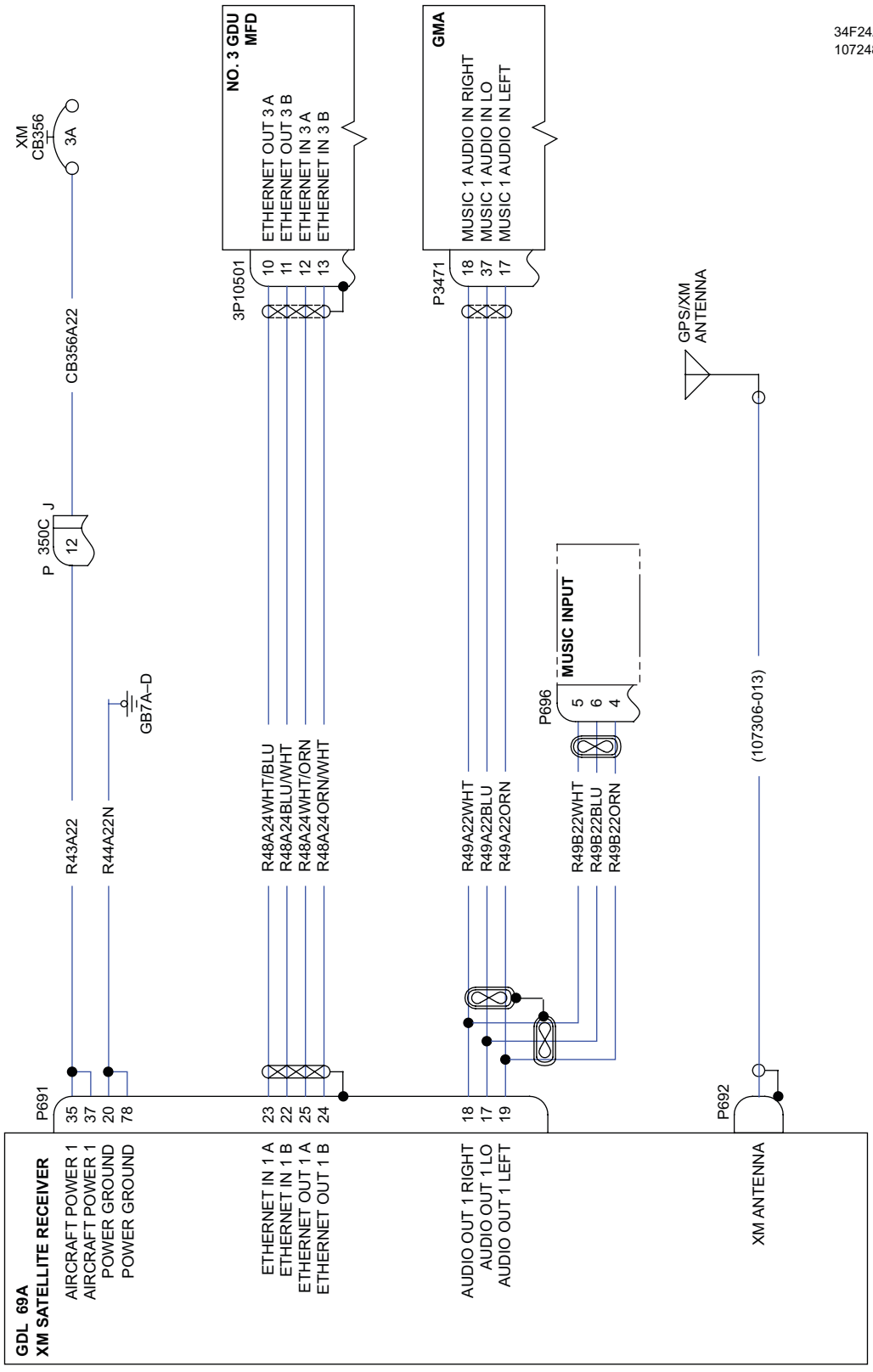
XM Receiver
Figure 3 (Sheet 1 of 3)

107240 NEW 51
107248 E 2



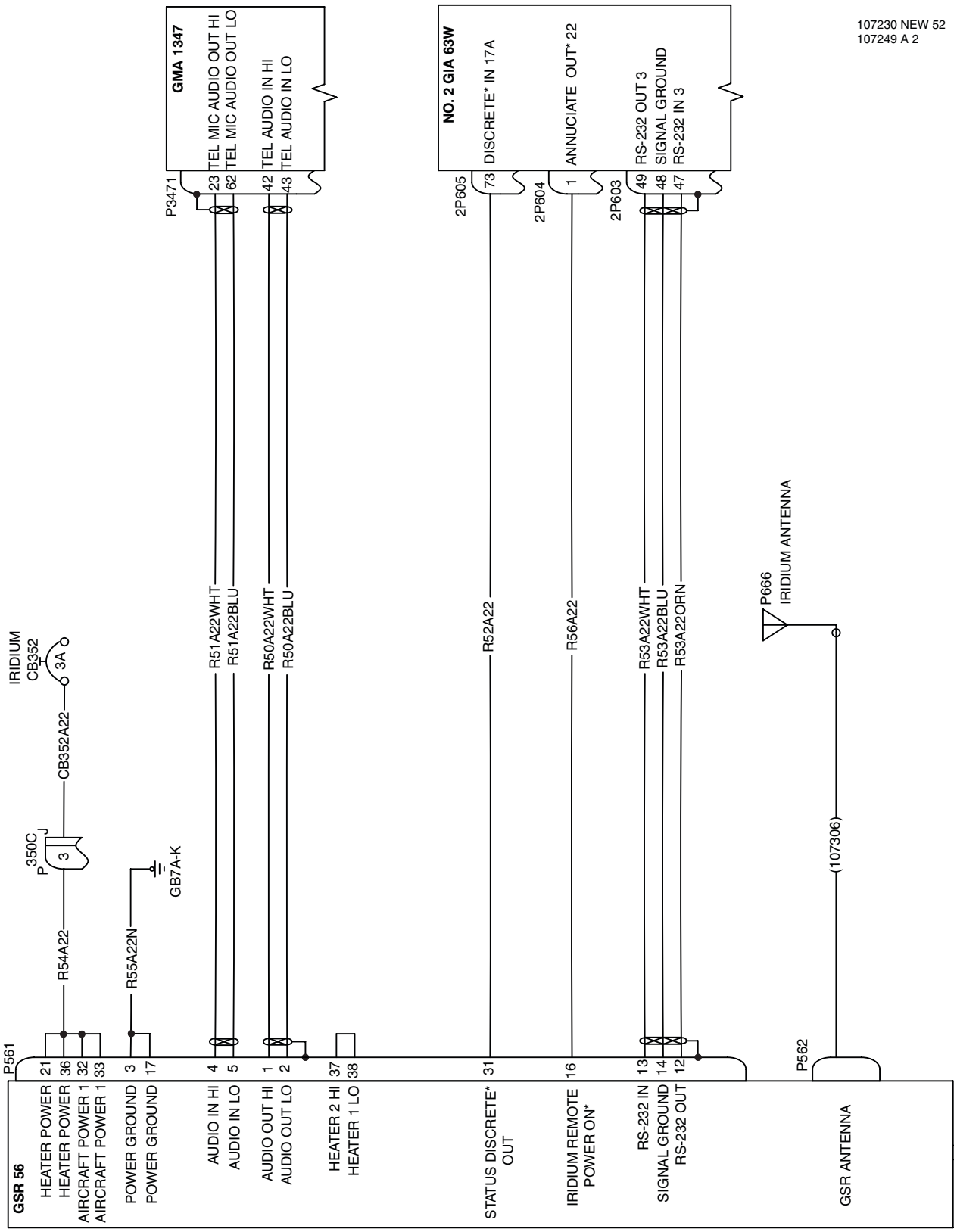
XM Receiver
Figure 3 (Sheet 2 of 3)

Effectivity
Seneca V
with Garmin G1000 / GFC 700



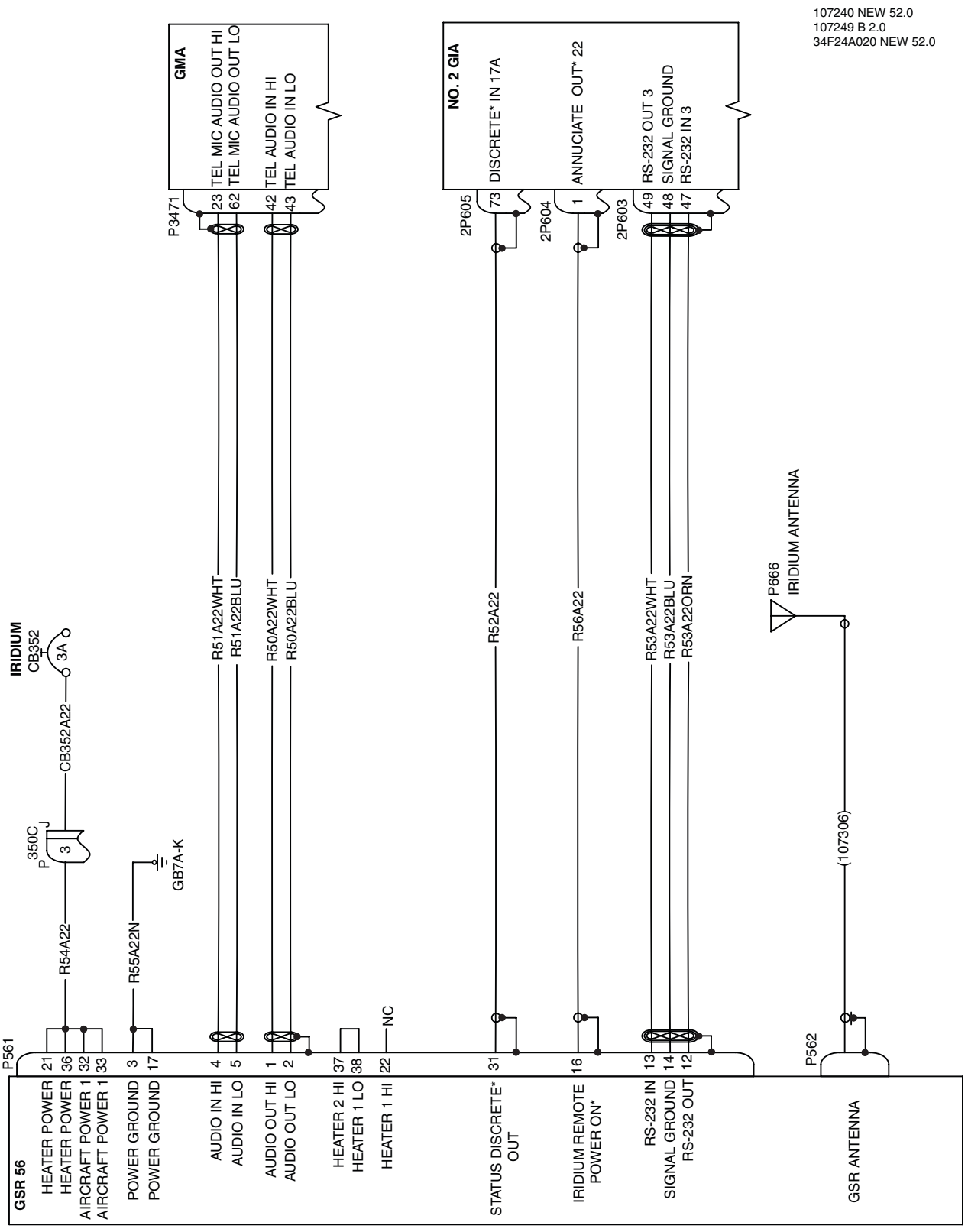
Effectivity
Seneca V
with Garmin G1000 NXi

XM Receiver
Figure 3 (Sheet 3 of 3)



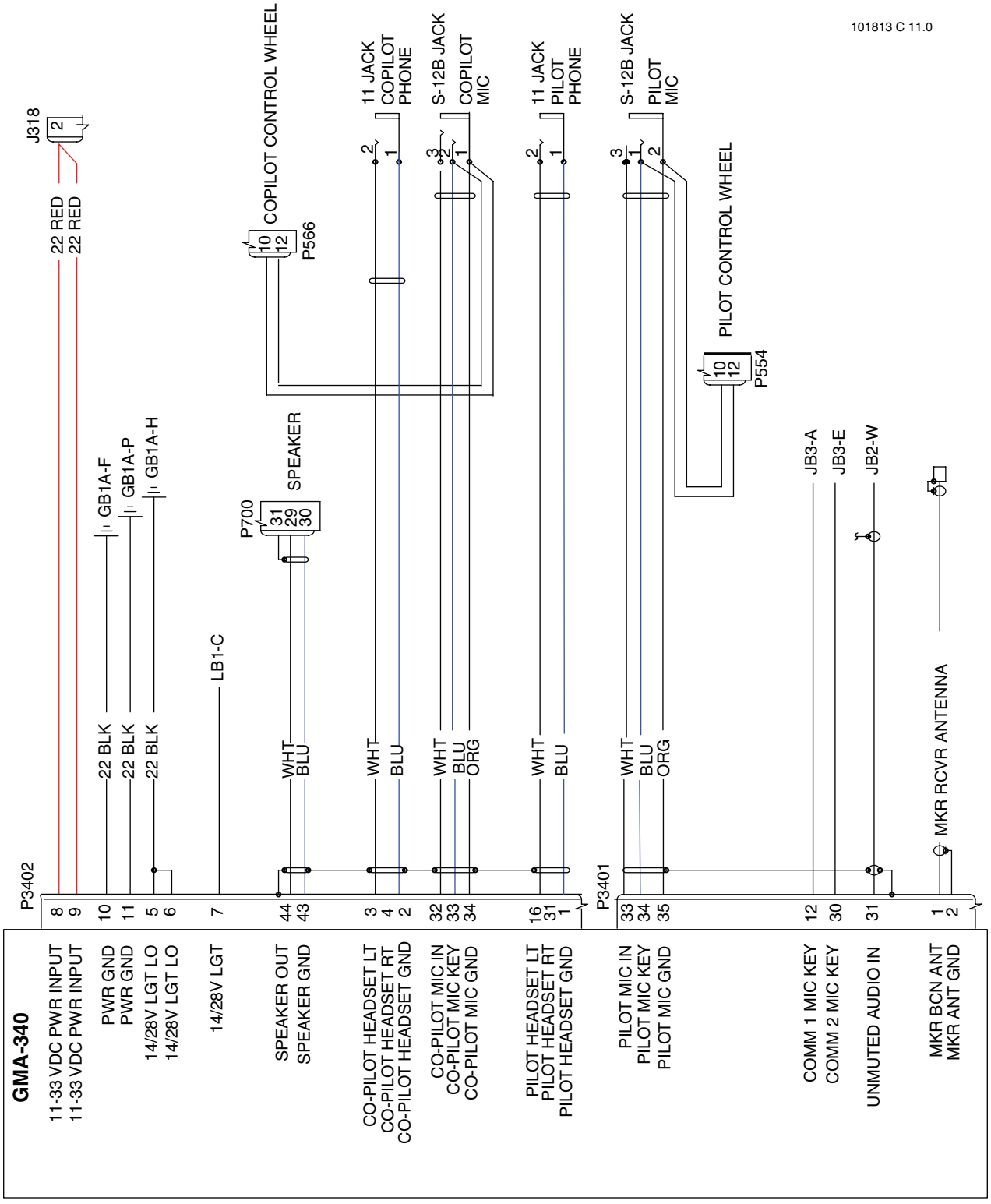
Iridium Receiver
Figure 4 (Sheet 1 of 2)

Effectivity
Seneca V
with Garmin G1000



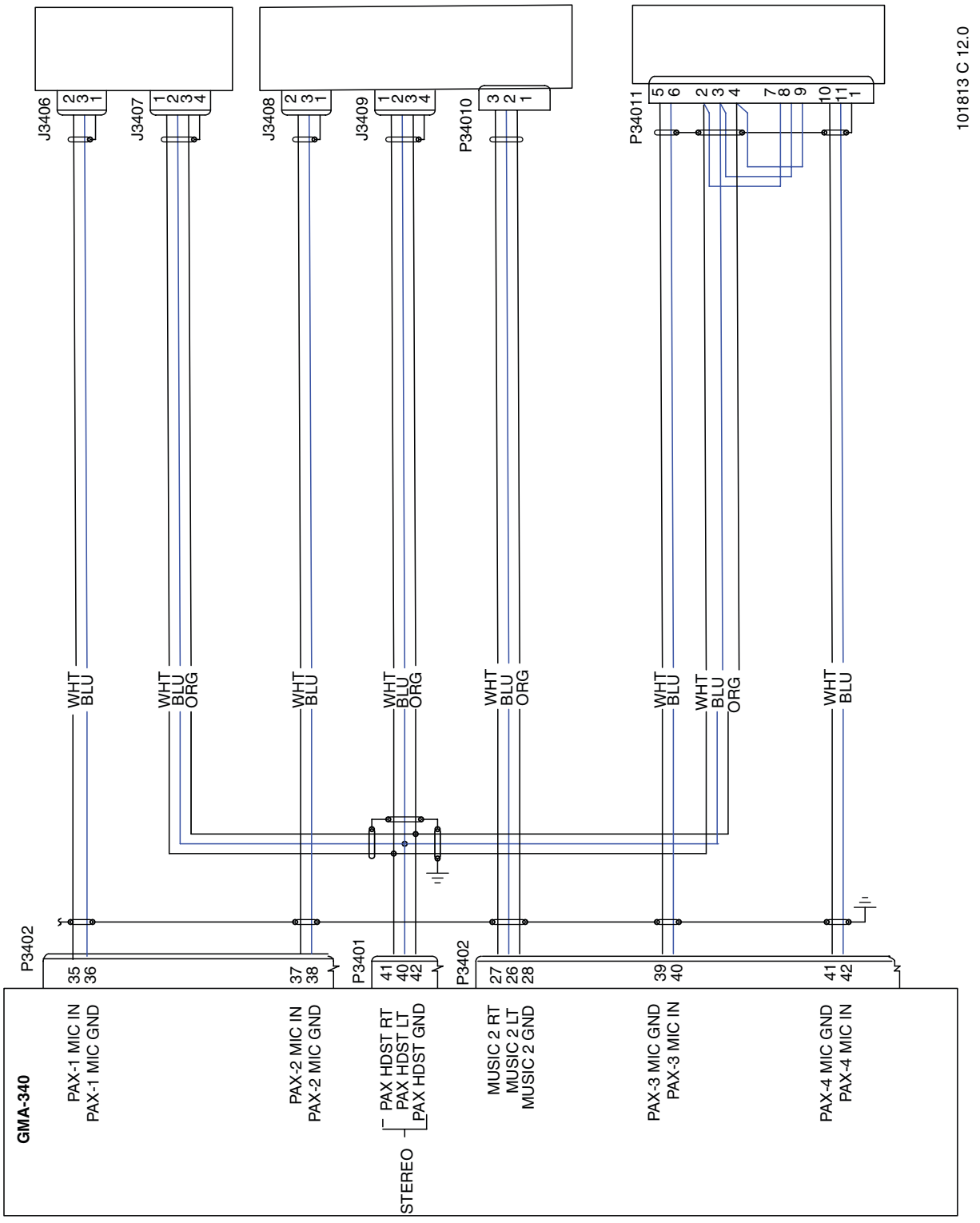
Effectivity
Seneca V
with Garmin GFC 700 / G1000 NXi

Iridium Receiver
Figure 4 (Sheet 2 of 2)



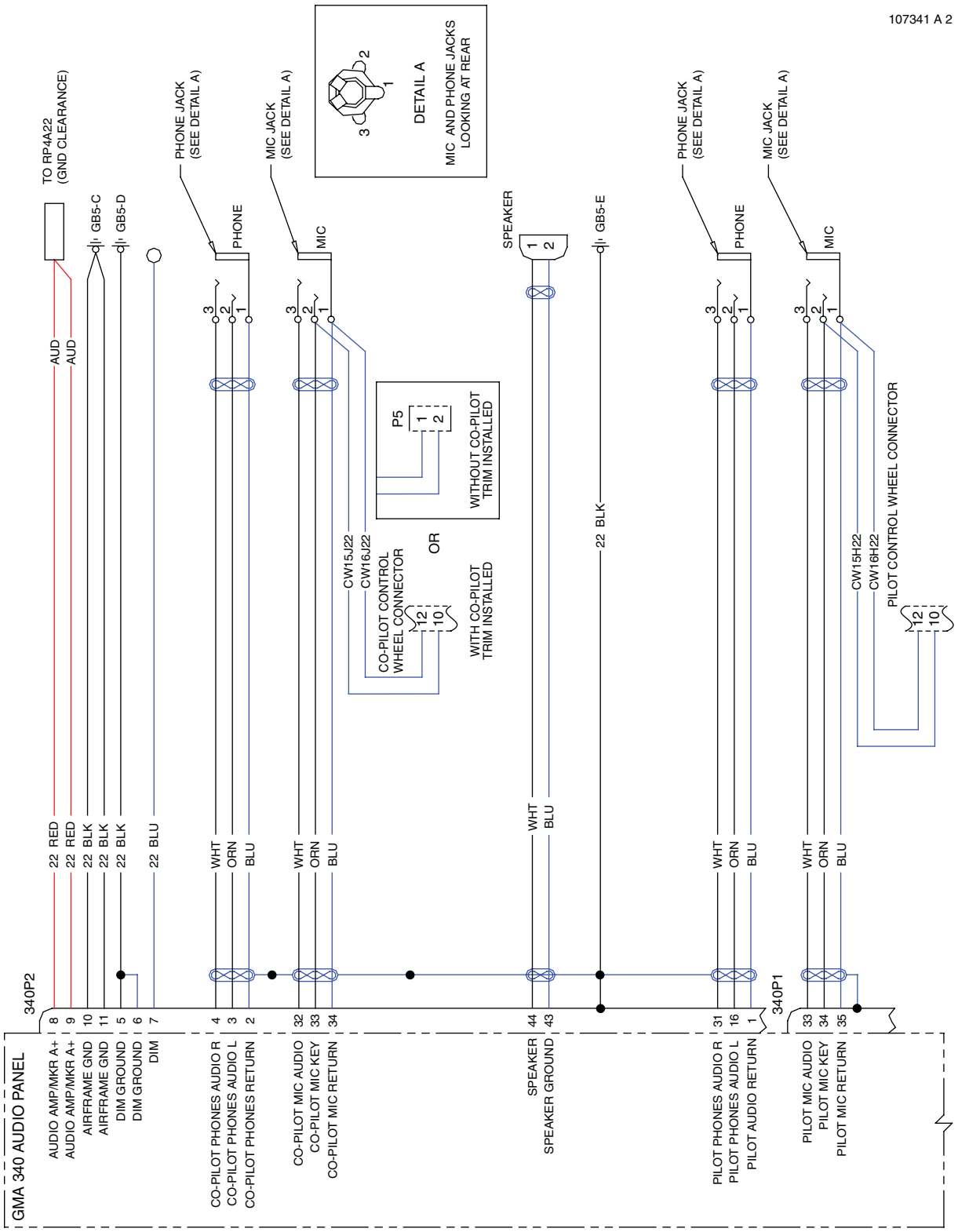
Audio Panel
Figure 1

Effectivity
Seneca V
with Avidyne Entegra



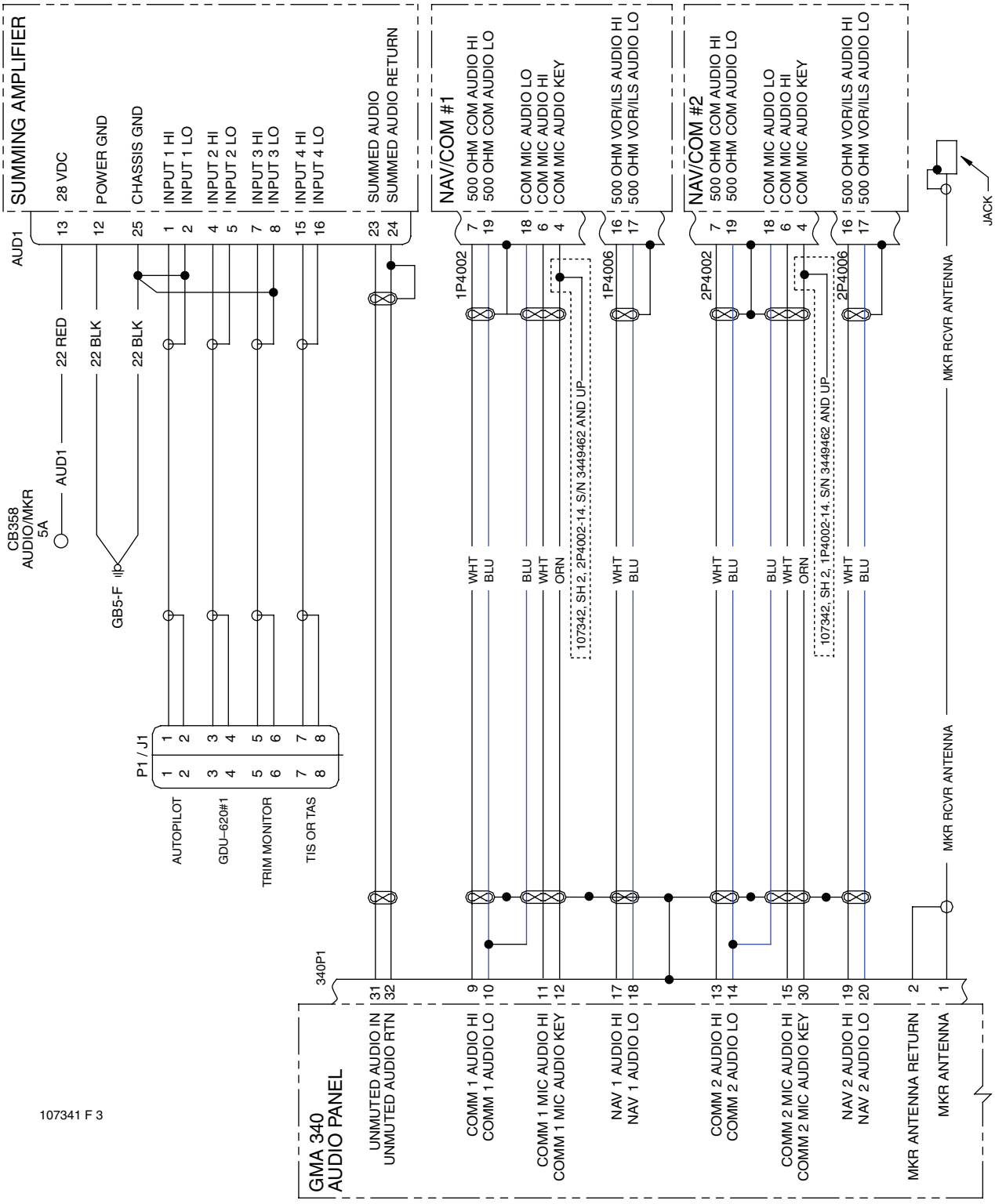
Effectivity
Seneca V
with Avidyne Entegra

Passenger Interphone
Figure 2



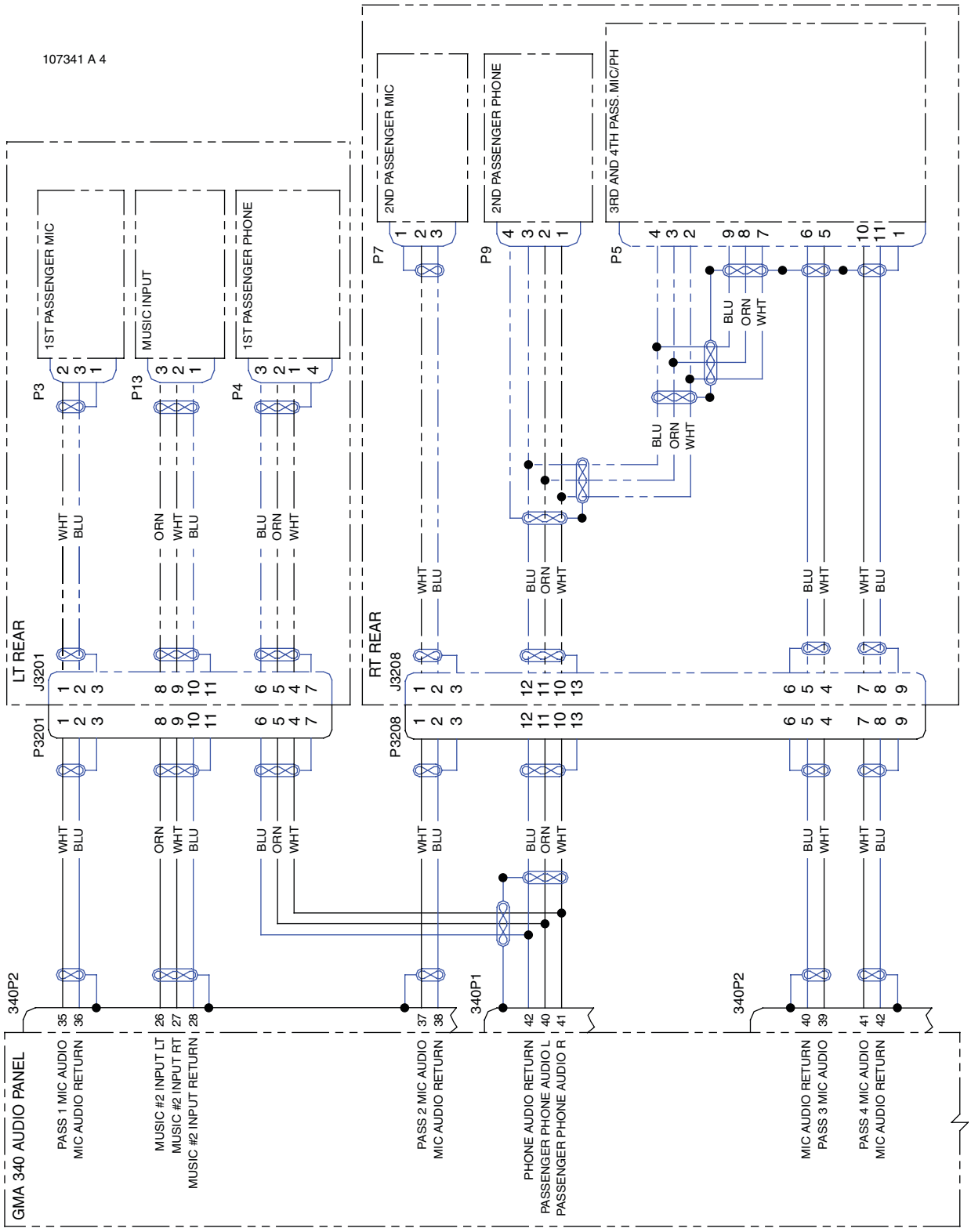
Audio Panel
Figure 3 (Sheet 1 of 3)

Effectivity
Seneca V
with Garmin G600



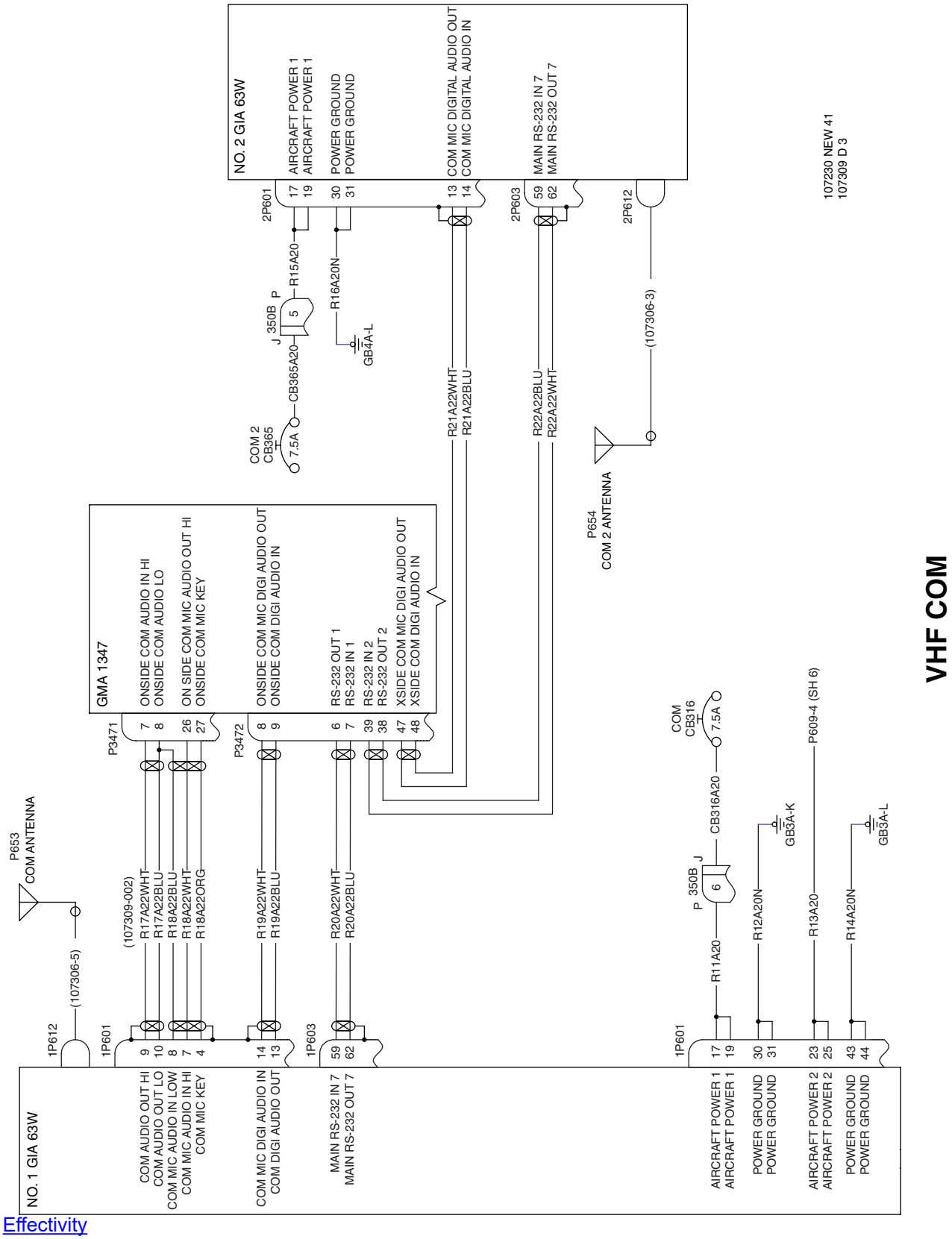
Effectivity
Seneca V
with Garmin G600

Audio Panel
Figure 3 (Sheet 2 of 3)



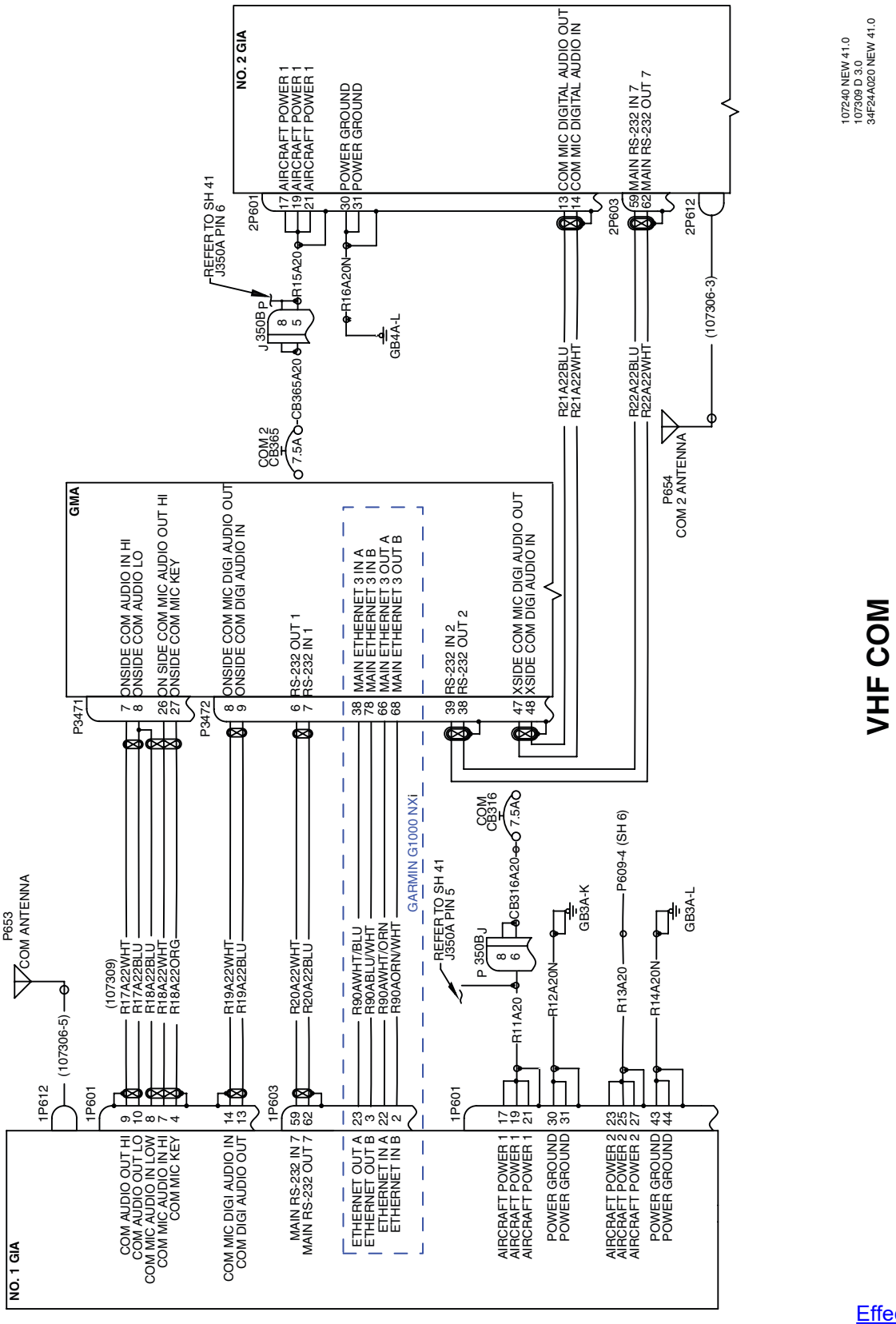
Audio Panel
Figure 3 (Sheet 3 of 3)

Effectivity
Seneca V
with Garmin G600



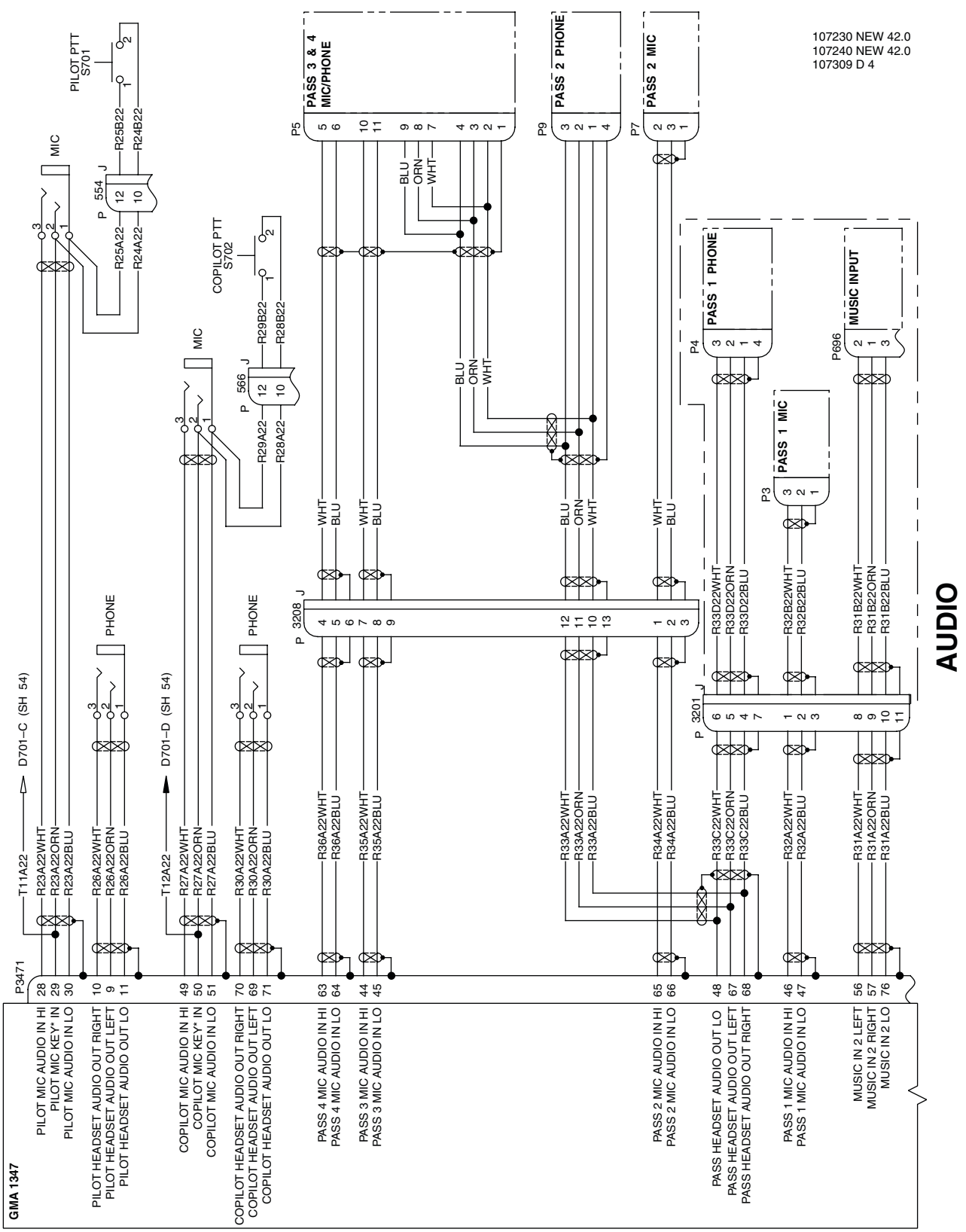
Effectivity
Seneca V
with Garmin G1000

Audio Panel
Figure 4 (Sheet 1 of 7)



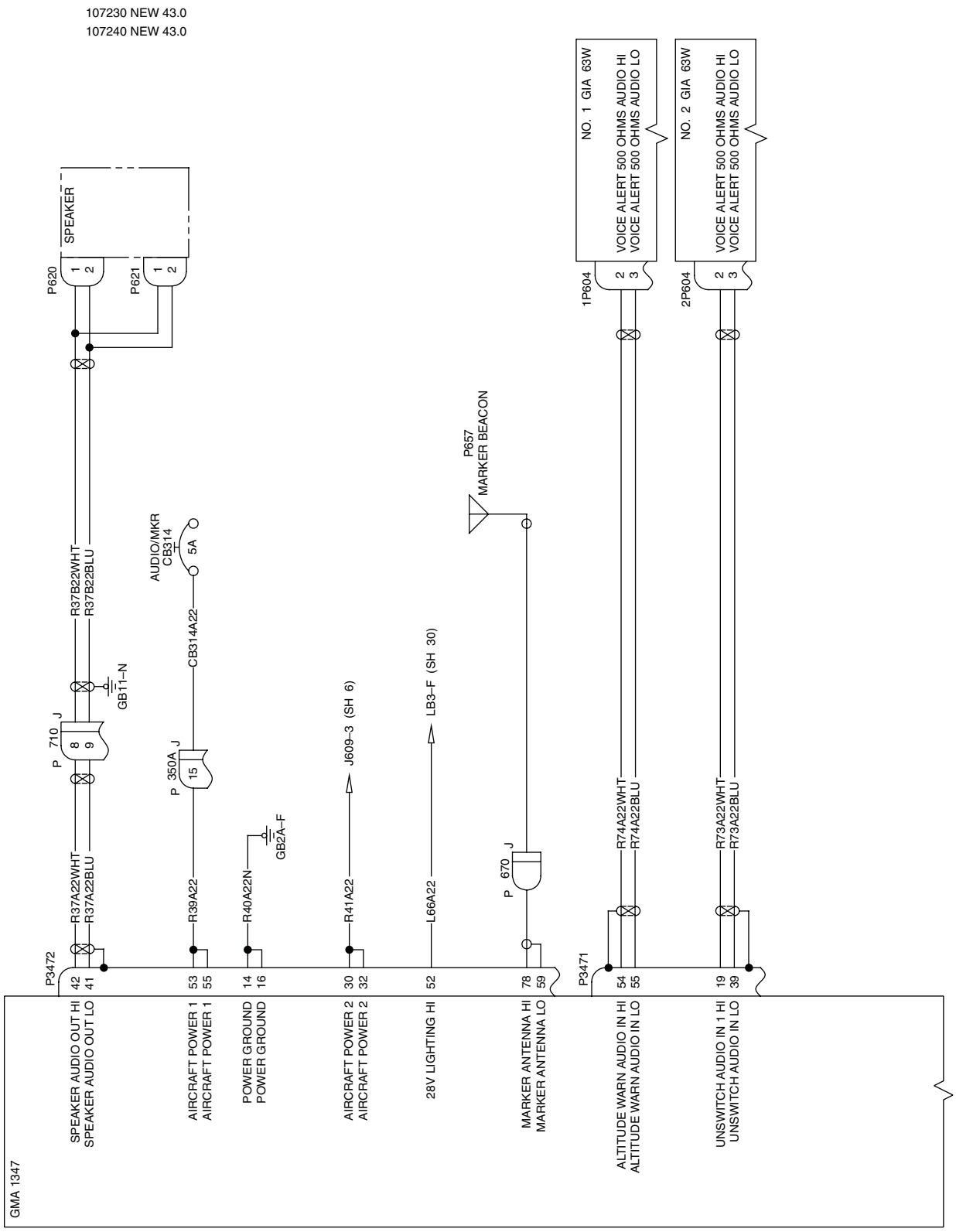
Audio Panel
Figure 4 (Sheet 2 of 7)

Effectivity
Seneca V
with Garmin GFC 700 / G1000 NXi



Effectivity
Seneca V
with Garmin G1000 / GFC 700

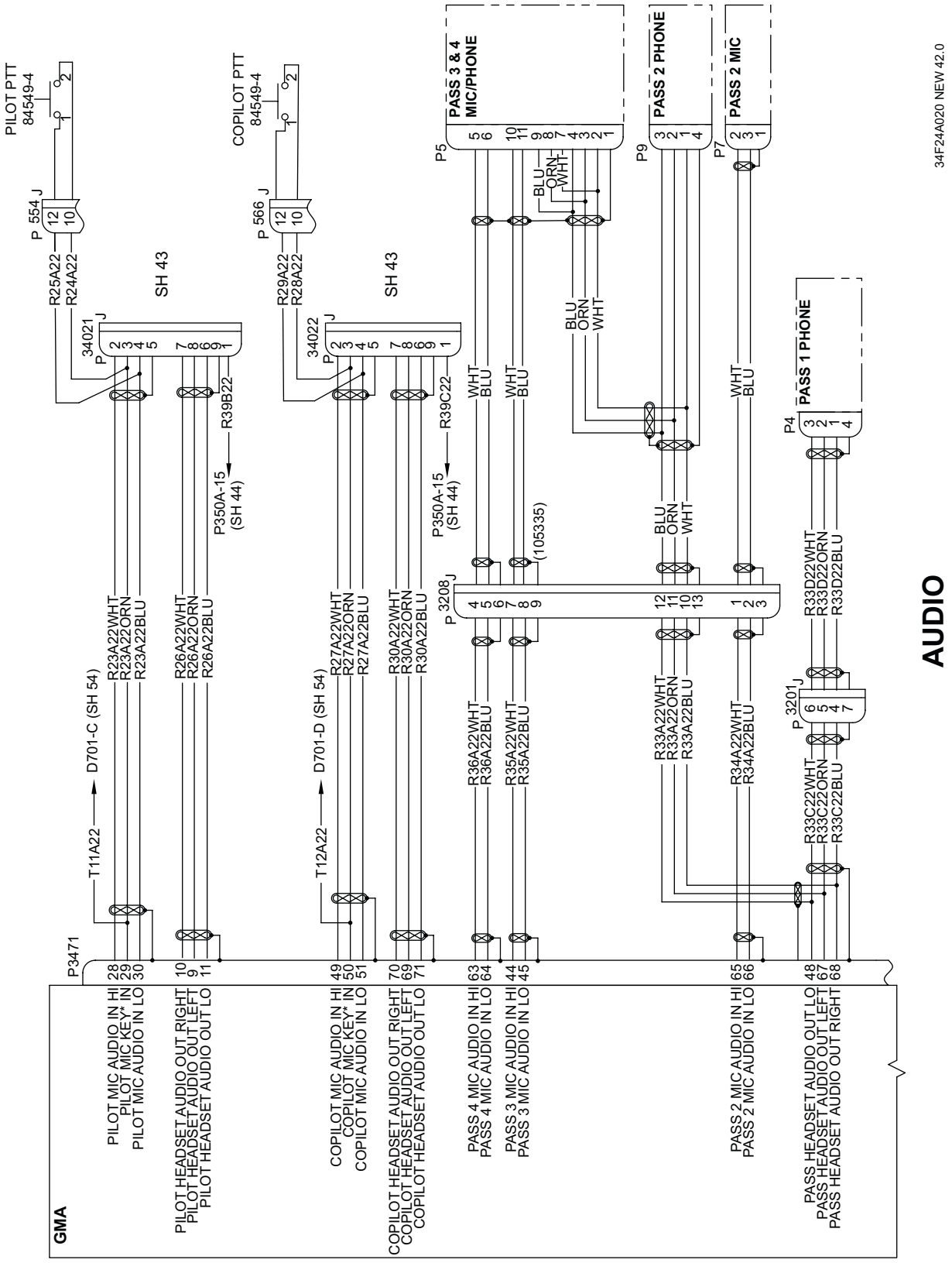
Audio Panel
Figure 4 (Sheet 3 of 7)



AUDIO

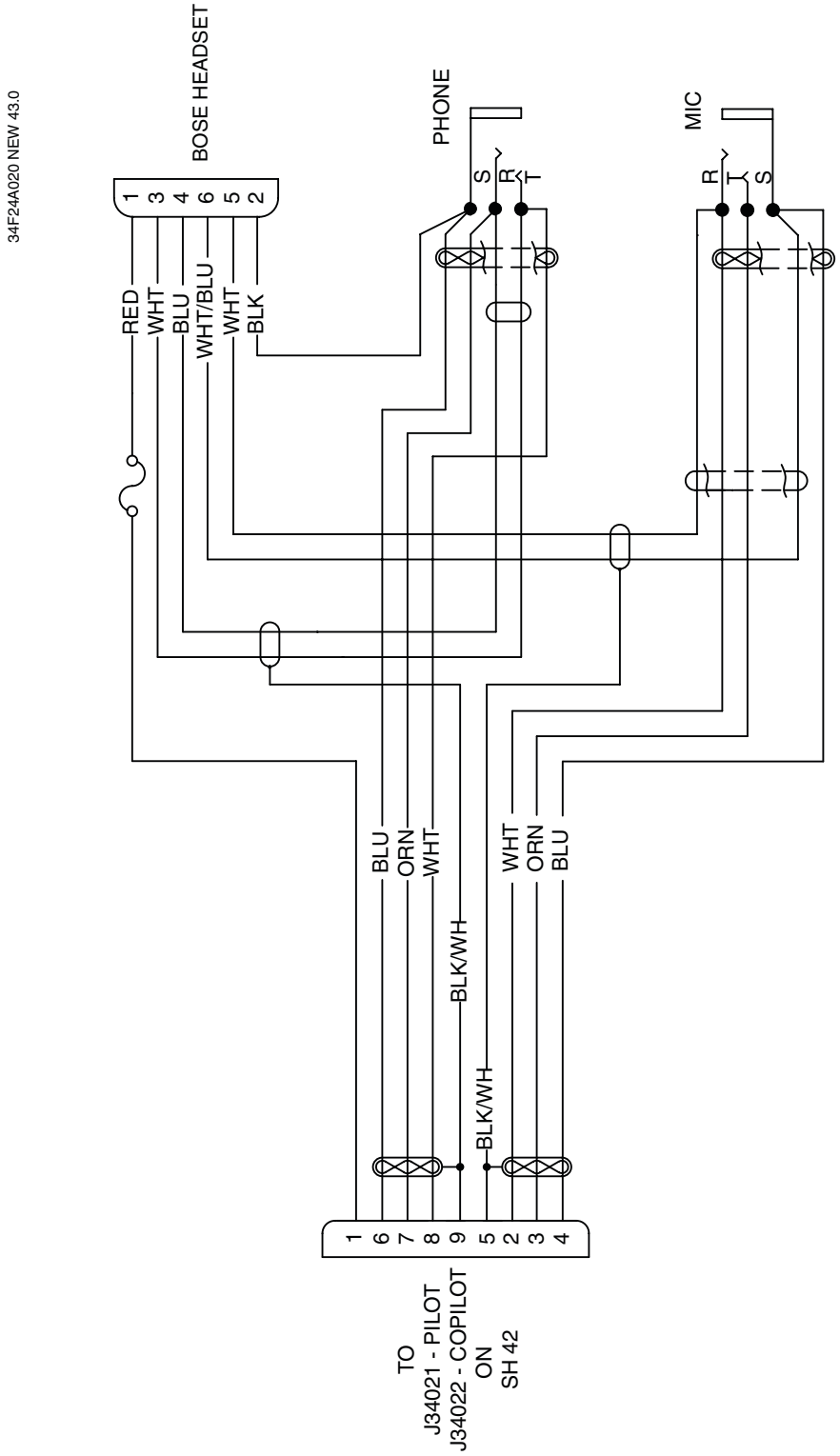
Audio Panel
Figure 4 (Sheet 4 of 7)

Effectivity
Seneca V
with Garmin G1000 / GFC 700



Effectivity
Seneca V
with Garmin G1000 NXi

Audio Panel
Figure 4 (Sheet 5 of 7)

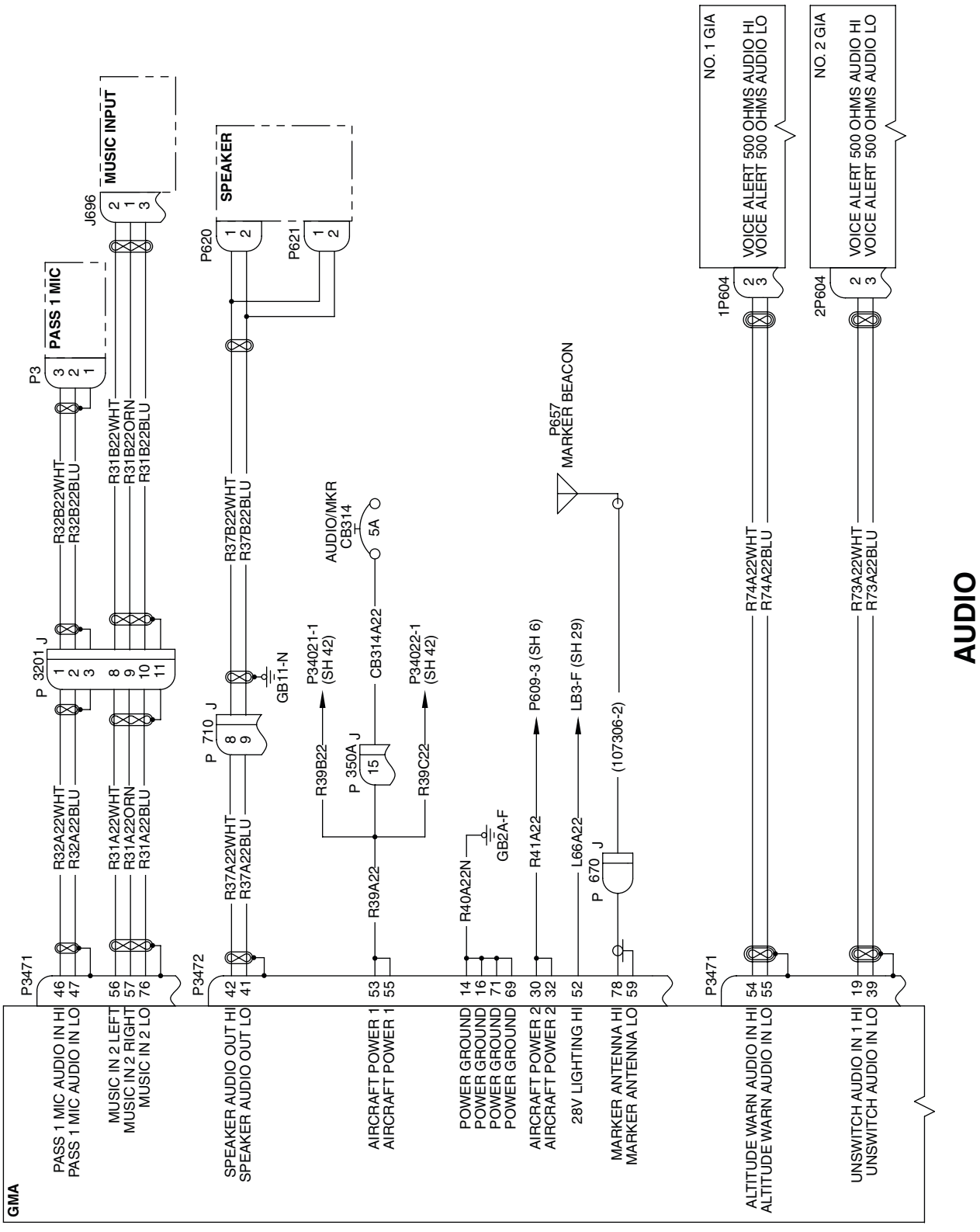


AUDIO

Audio Panel
Figure 4 (Sheet 6 of 7)

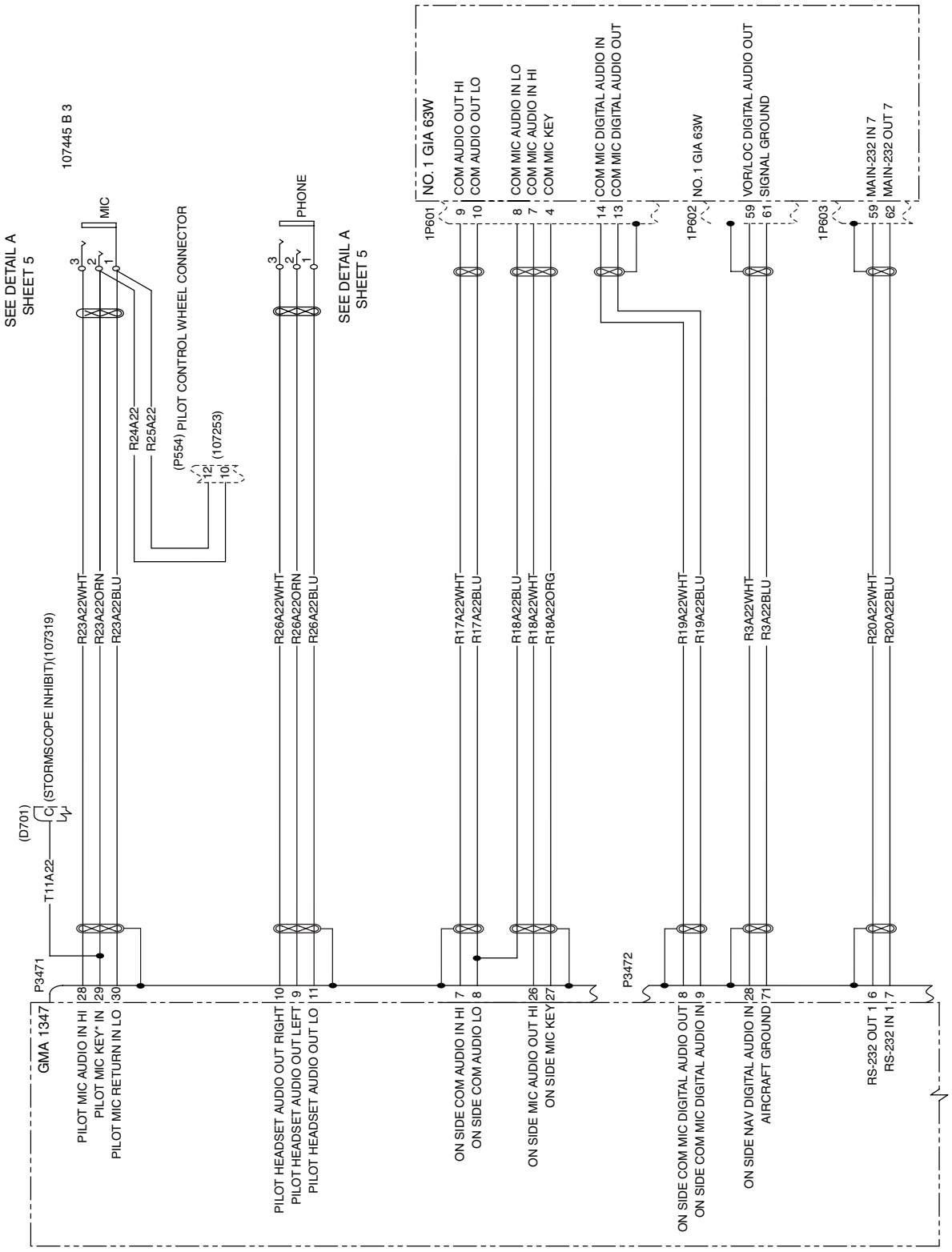
Effectivity
Seneca V
with Garmin G1000 NXi

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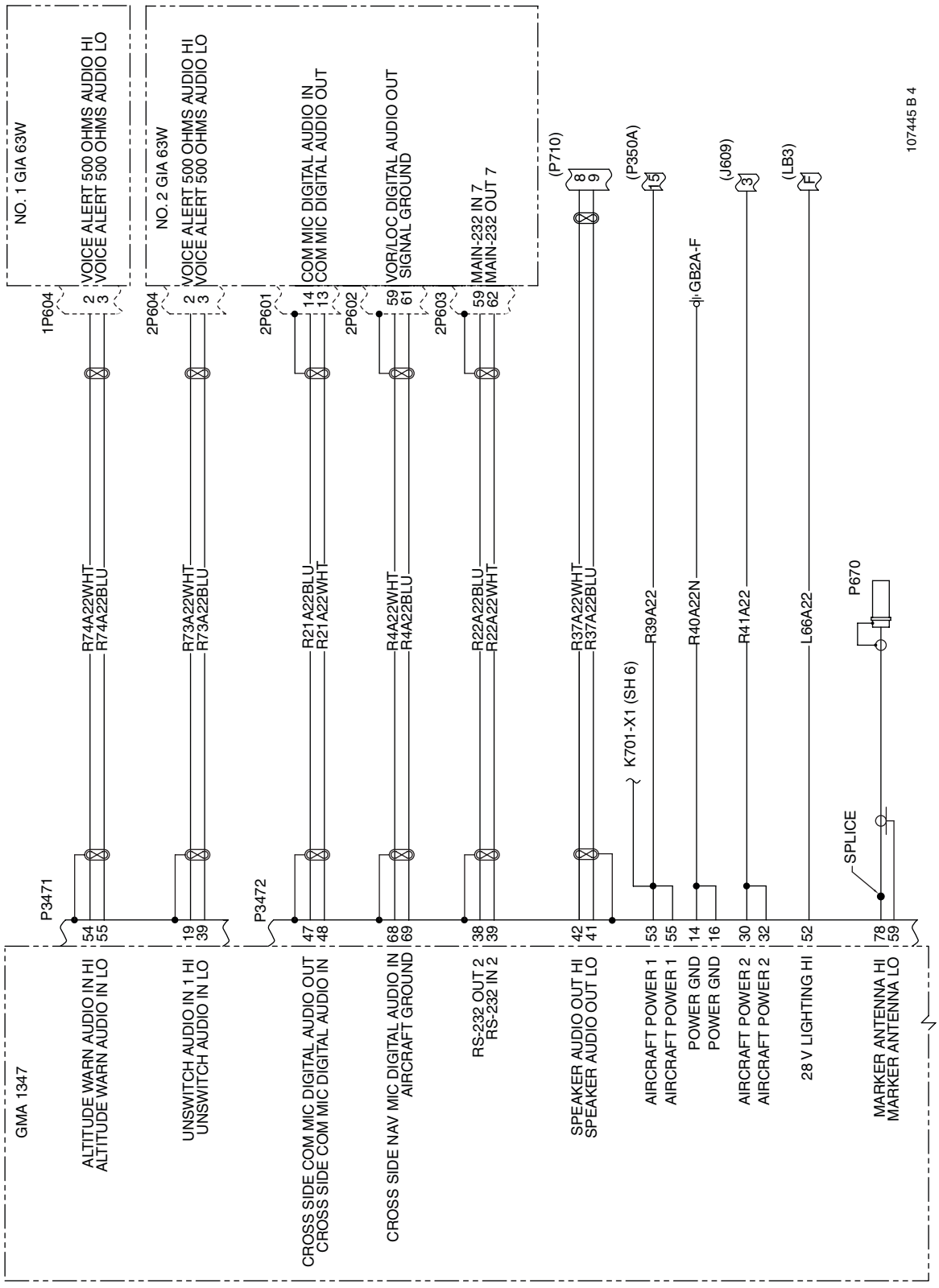
Effectivity
Seneca V
with Garmin G1000 NXi

Audio Panel
Figure 4 (Sheet 7 of 7)



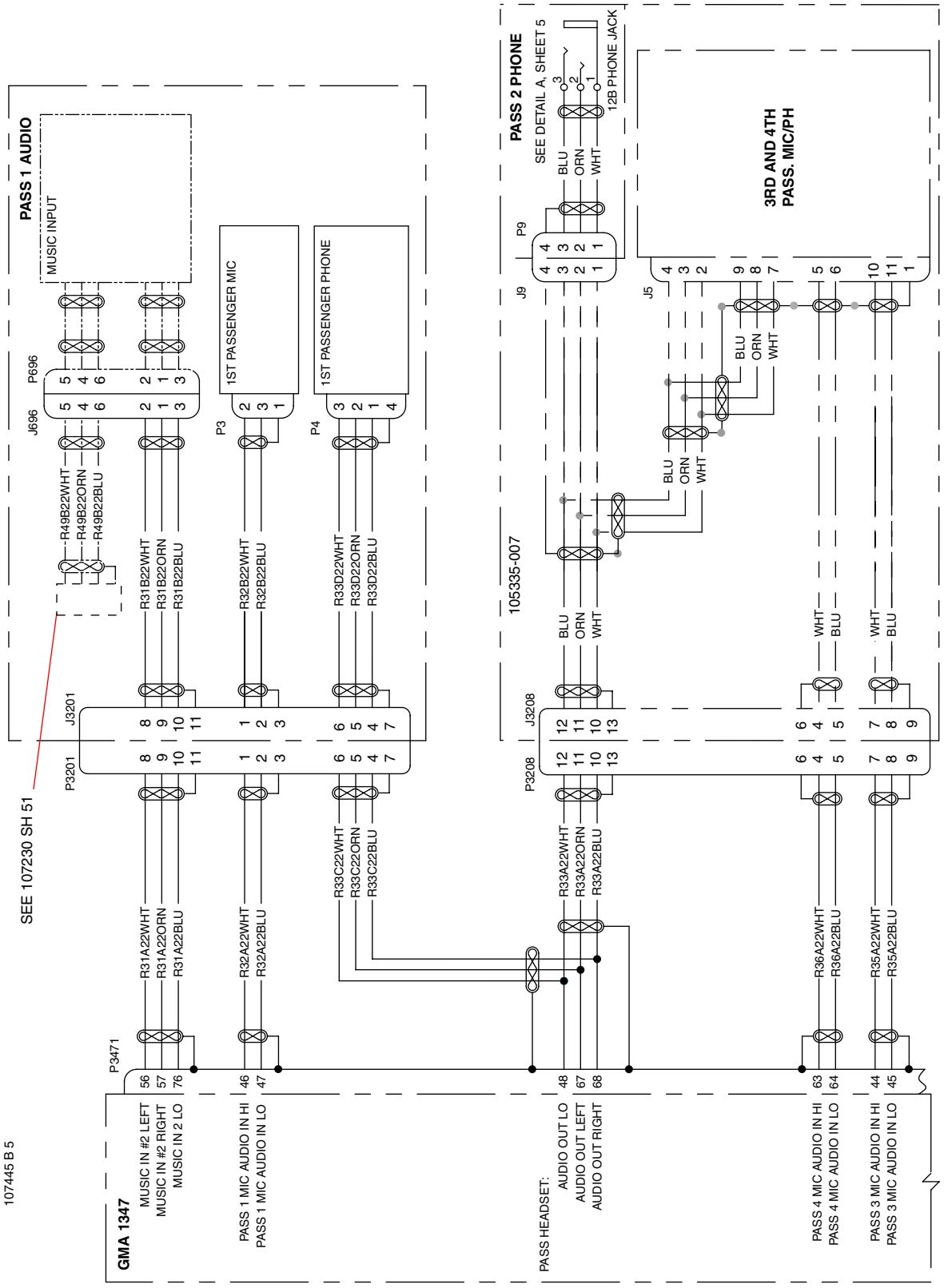
Passenger Interphone Aft Seat PTT
Figure 5 (Sheet 1 of 5)

Effectivity
Seneca V - Optional
3449479 and up



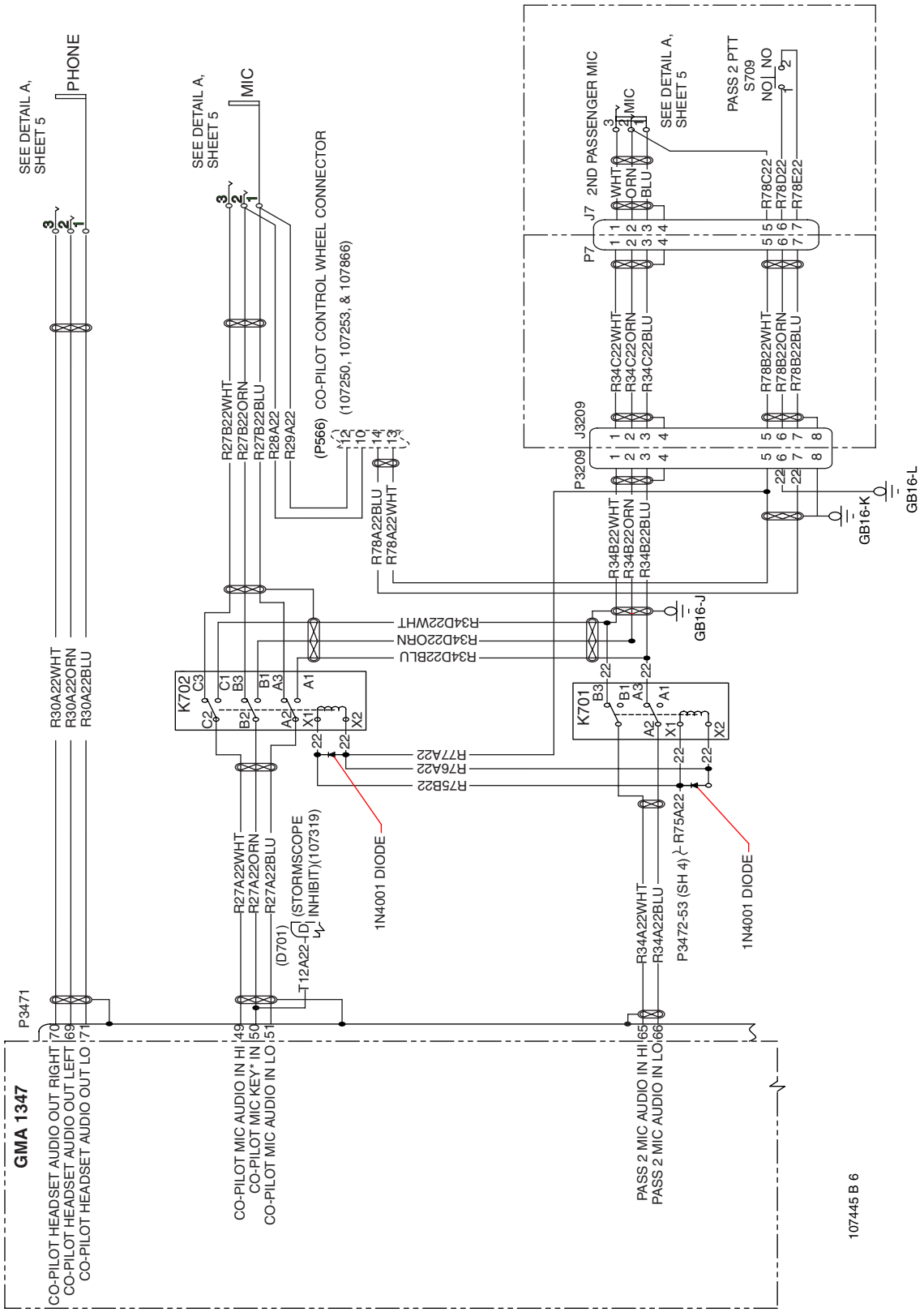
Effectivity
Seneca V - Optional
3449479 and up

Passenger Interphone Aft Seat PTT
Figure 5 (Sheet 2 of 5)



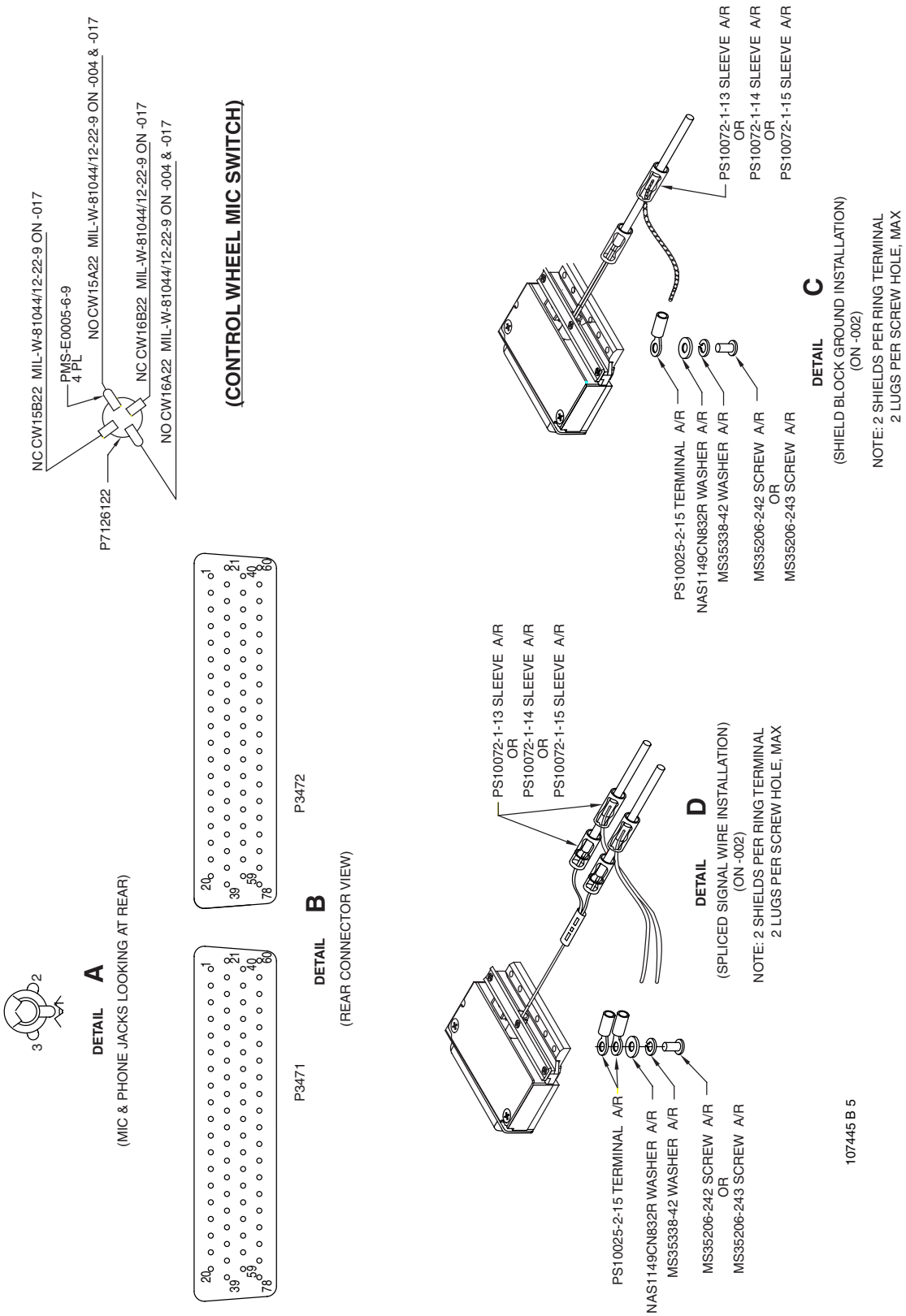
Passenger Interphone Aft Seat PTT
Figure 5 (Sheet 3 of 5)

Effectivity
Seneca V - Optional
3449479 and up



Effectivity
Seneca V - Optional
3449479 and up

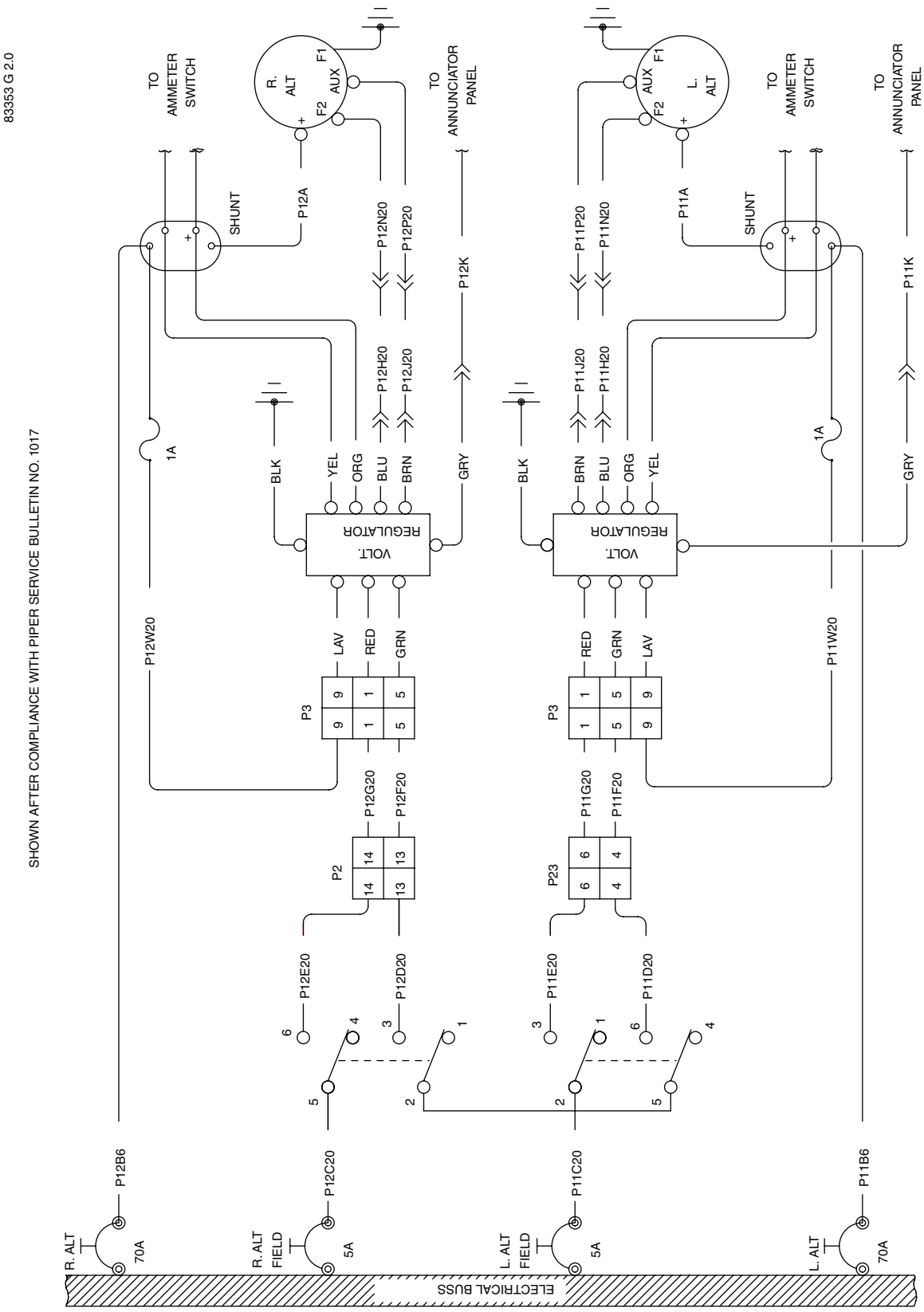
Passenger Interphone Aft Seat PTT
Figure 5 (Sheet 4 of 5)



Passenger Interphone Aft Seat PTT
Figure 5 (Sheet 5 of 5)

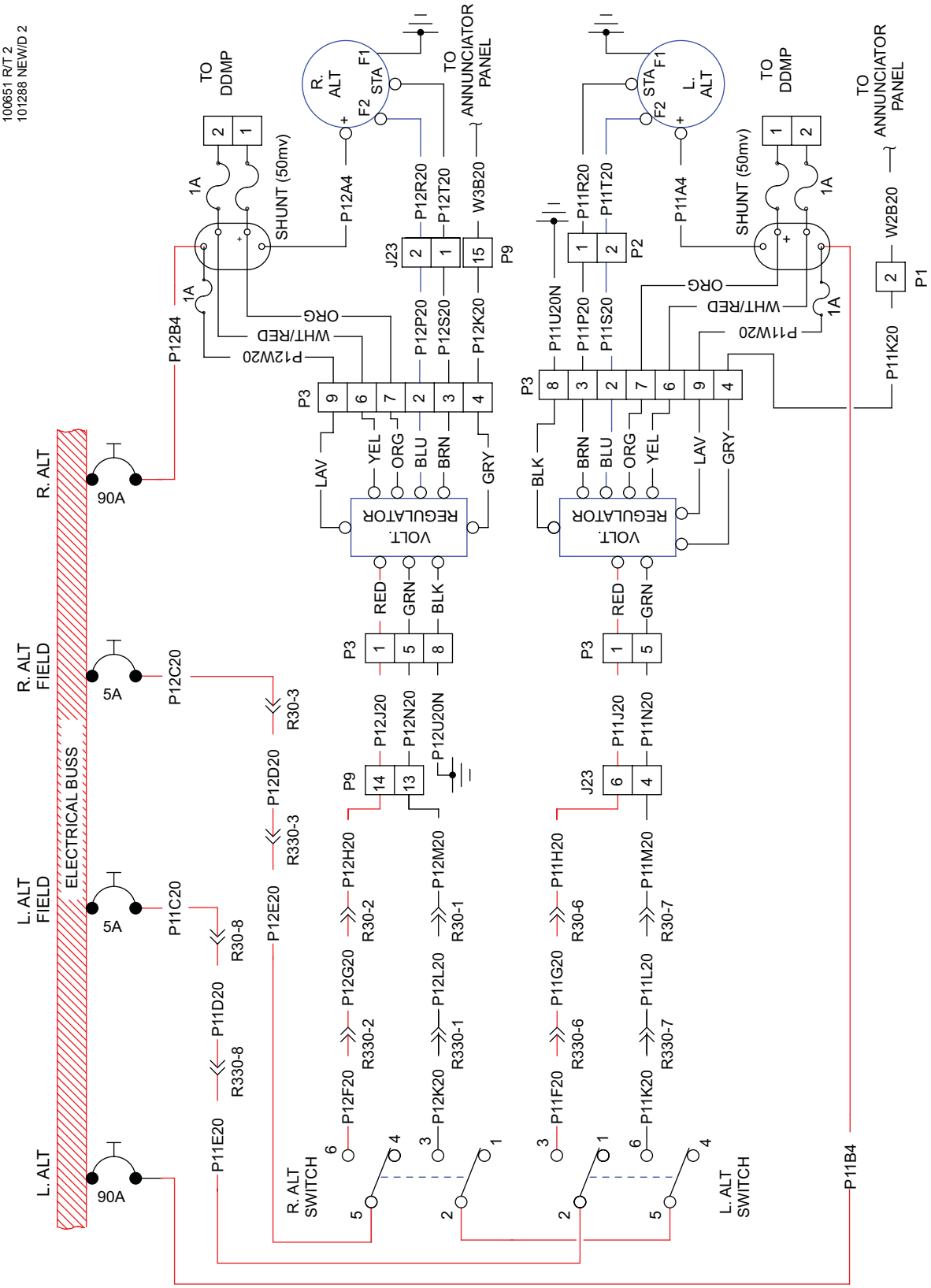
Effectivity
Seneca V - Optional
3449479 and up

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Alternator Power
Figure 1 (Sheet 1 of 4)

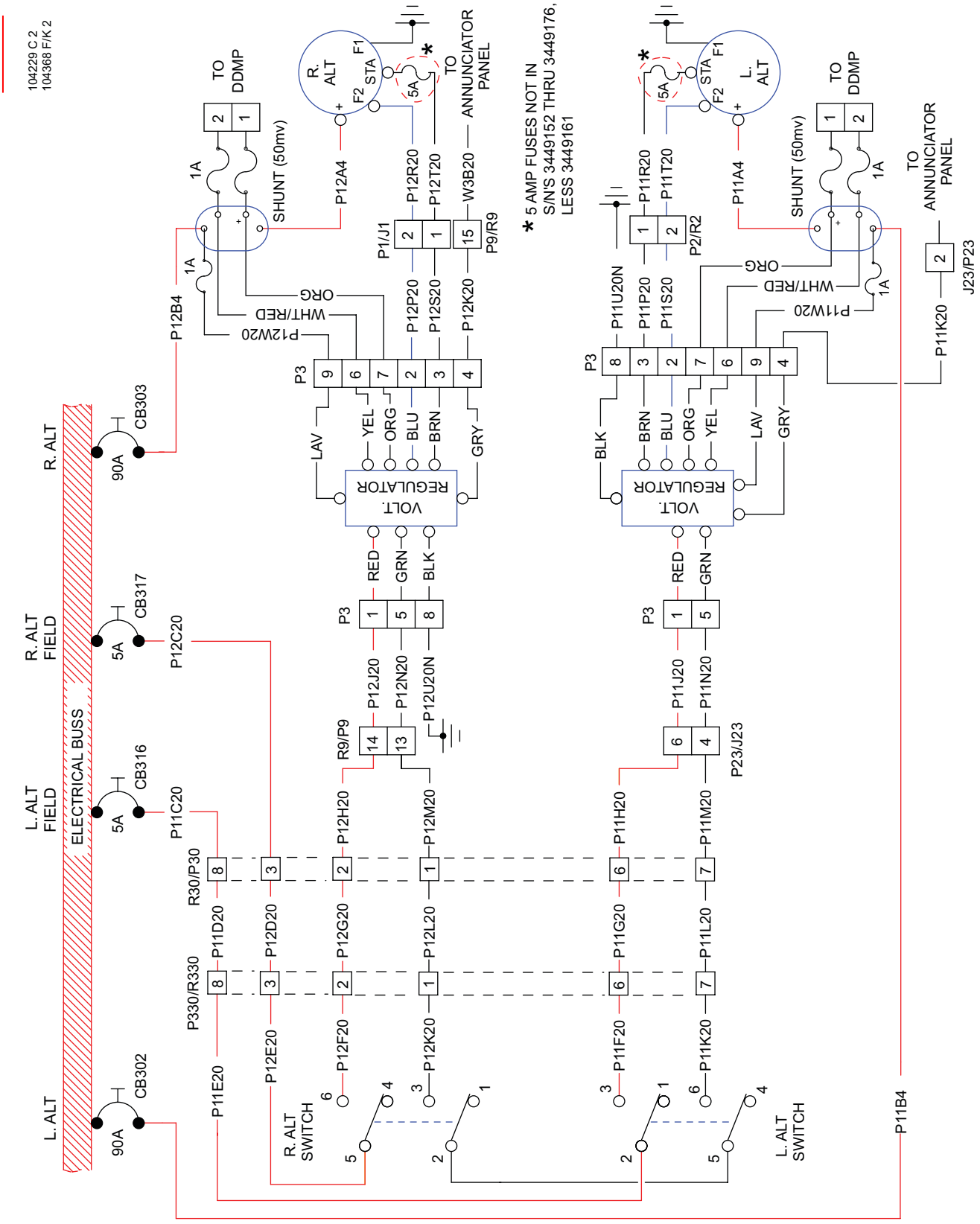
Effectivity
Seneca IV



FOR S/N'S 3449001 THRU 3449034, SHOWN AFTER COMPLIANCE WITH PIPER SERVICE BULLETIN NO. 1017.

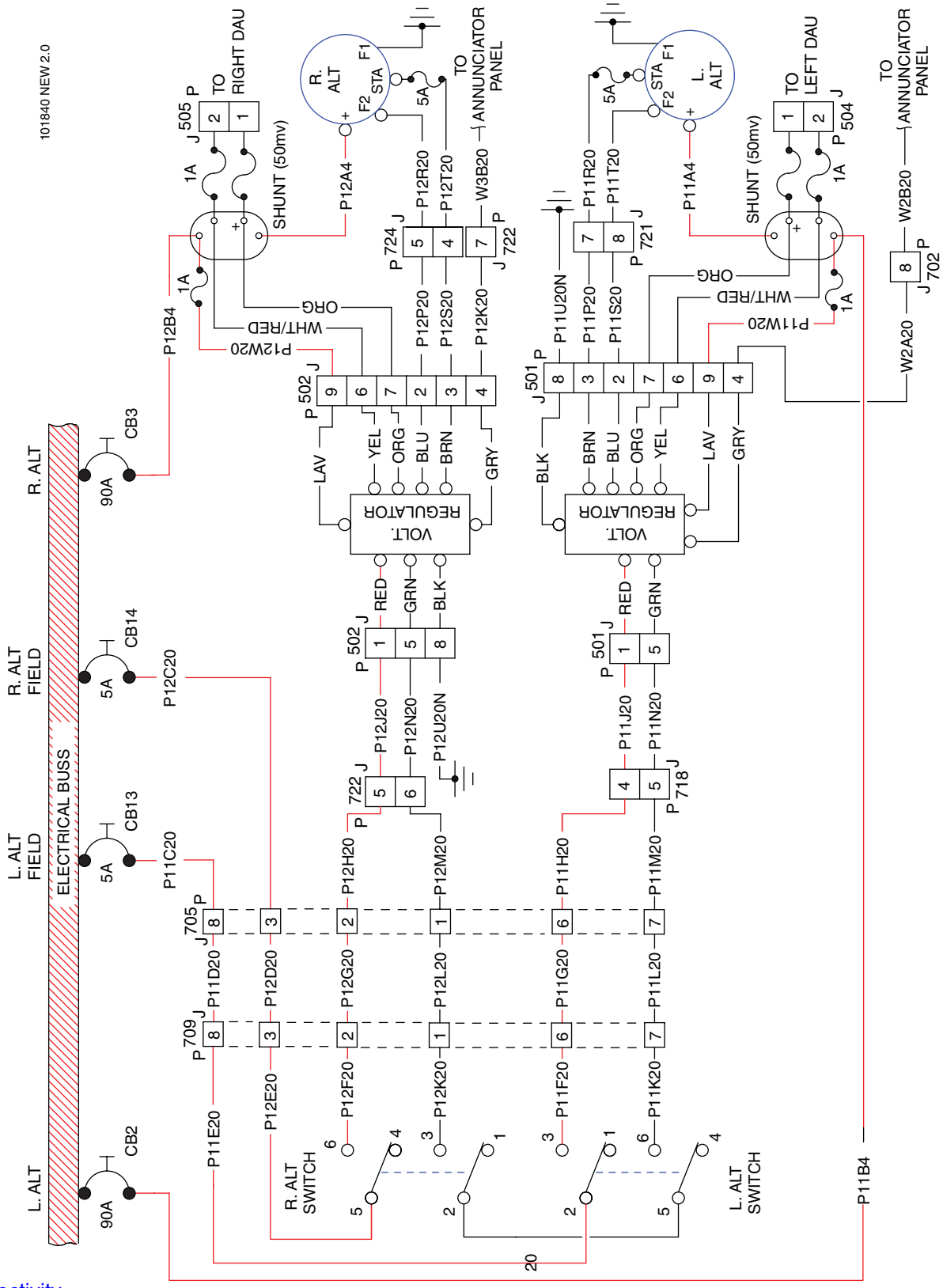
Effectivity
3449001 thru 3449151

Alternator Power
Figure 1 (Sheet 2 of 4)



Alternator Power
Figure 1 (Sheet 3 of 4)

Effectivity
3449152 & up

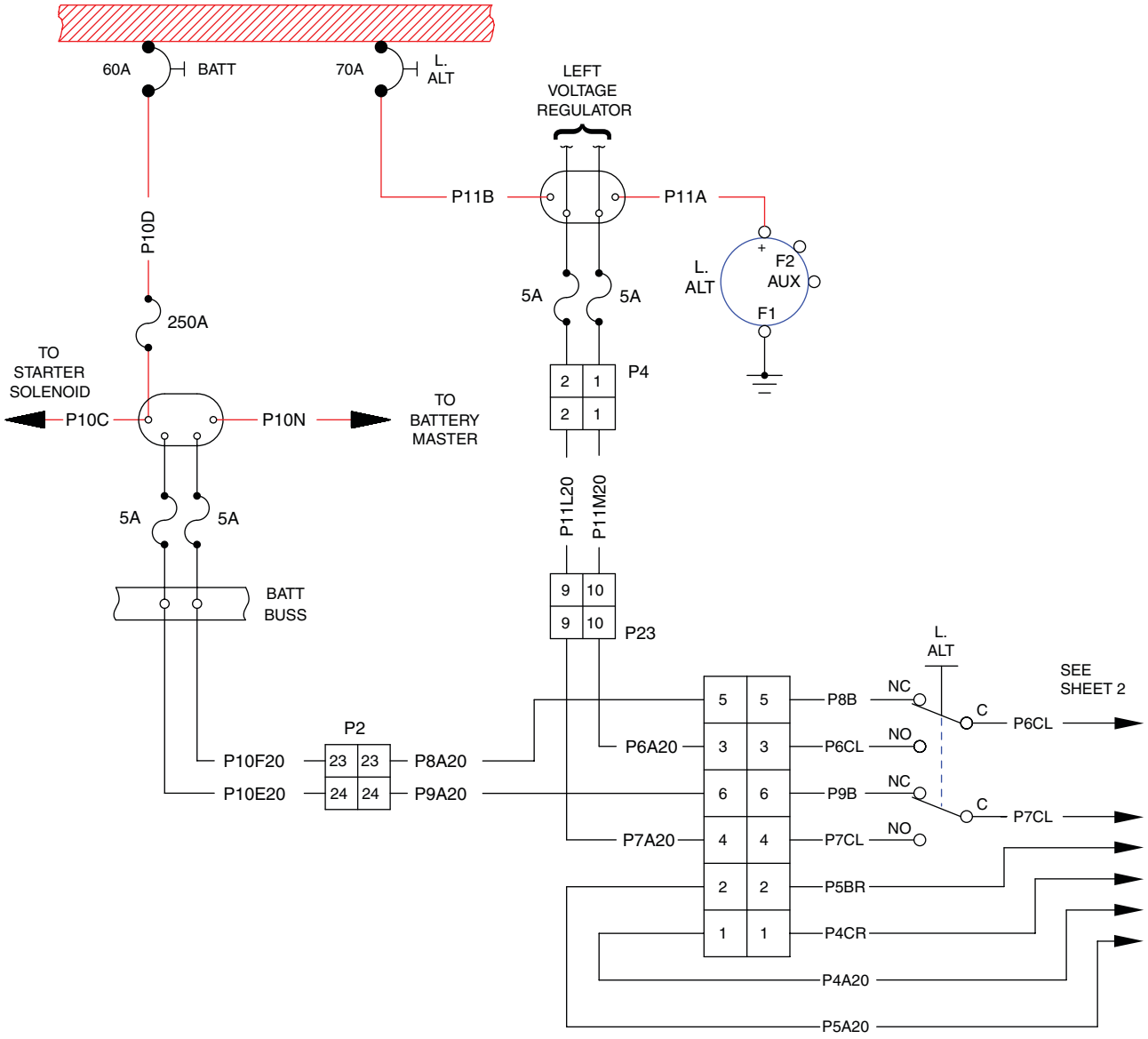


Alternator Power
Figure 1 (Sheet 4 of 4)

Effectivity
Seneca V
with Avidyne Entegra

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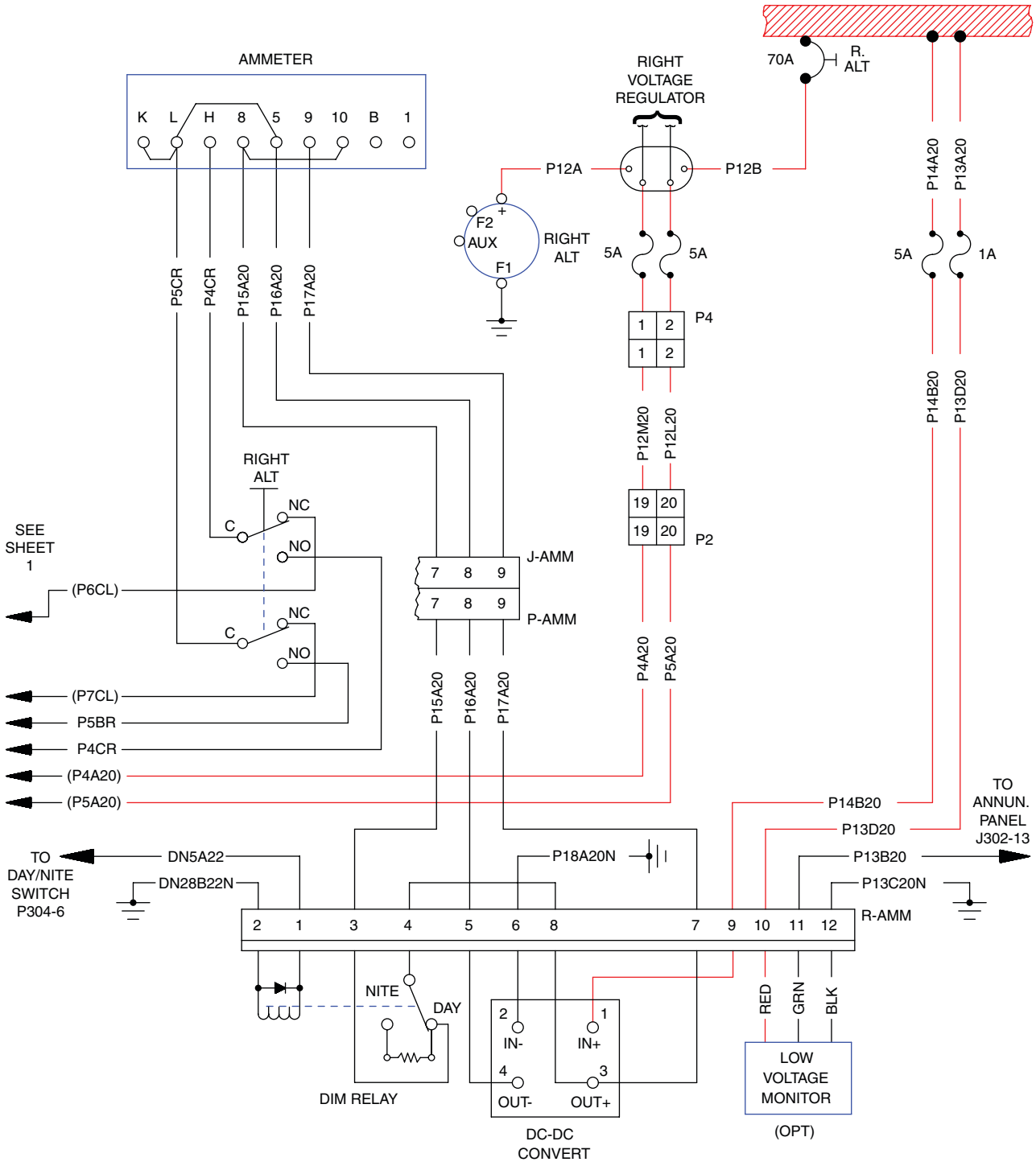
83353 F/G 3.0



SHOWN AFTER COMPLIANCE WITH PIPER SERVICE BULLETIN NO. 1014.

Ammeter / Low Volt Monitor
Figure 2 (Sheet 1 of 2)

83353 E/G 3.1

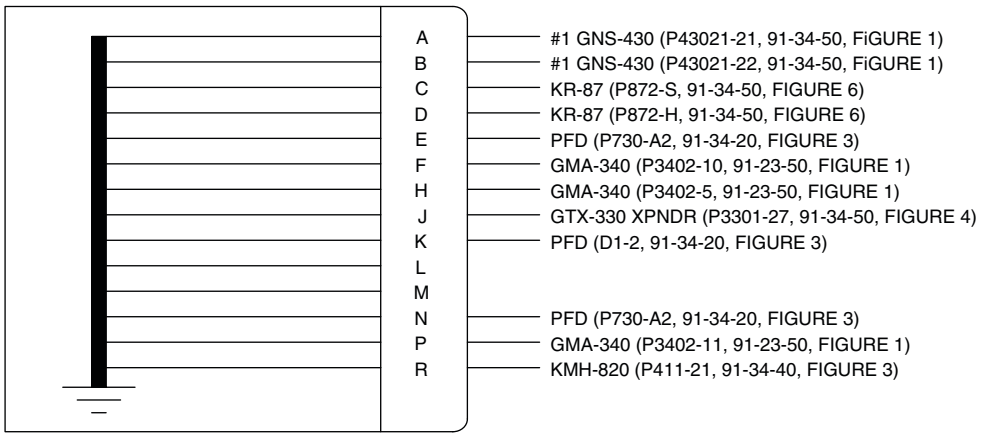


Ammeter / Low Volt Monitor
Figure 2 (Sheet 2 of 2)

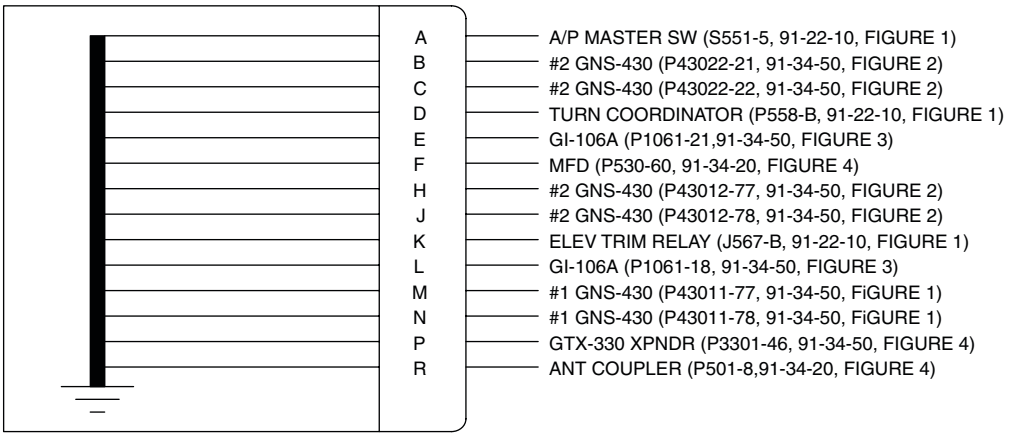
Effectivity
Seneca IV

101813 C 26.0

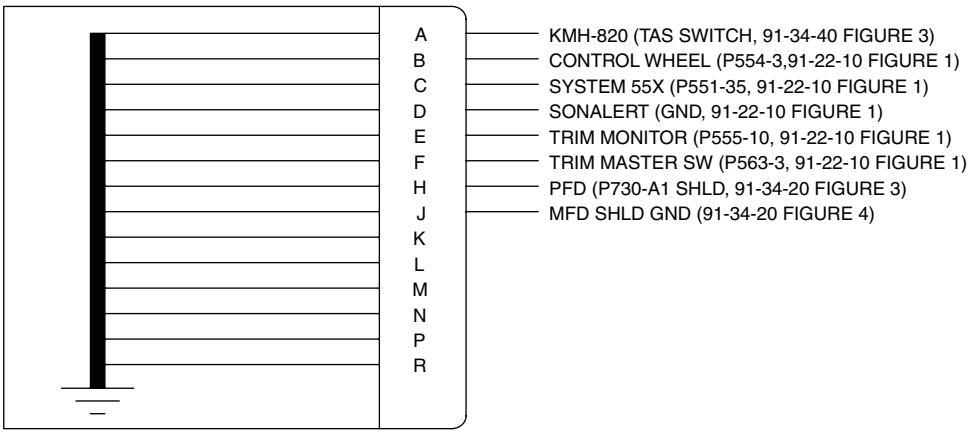
GB1A



GB2A

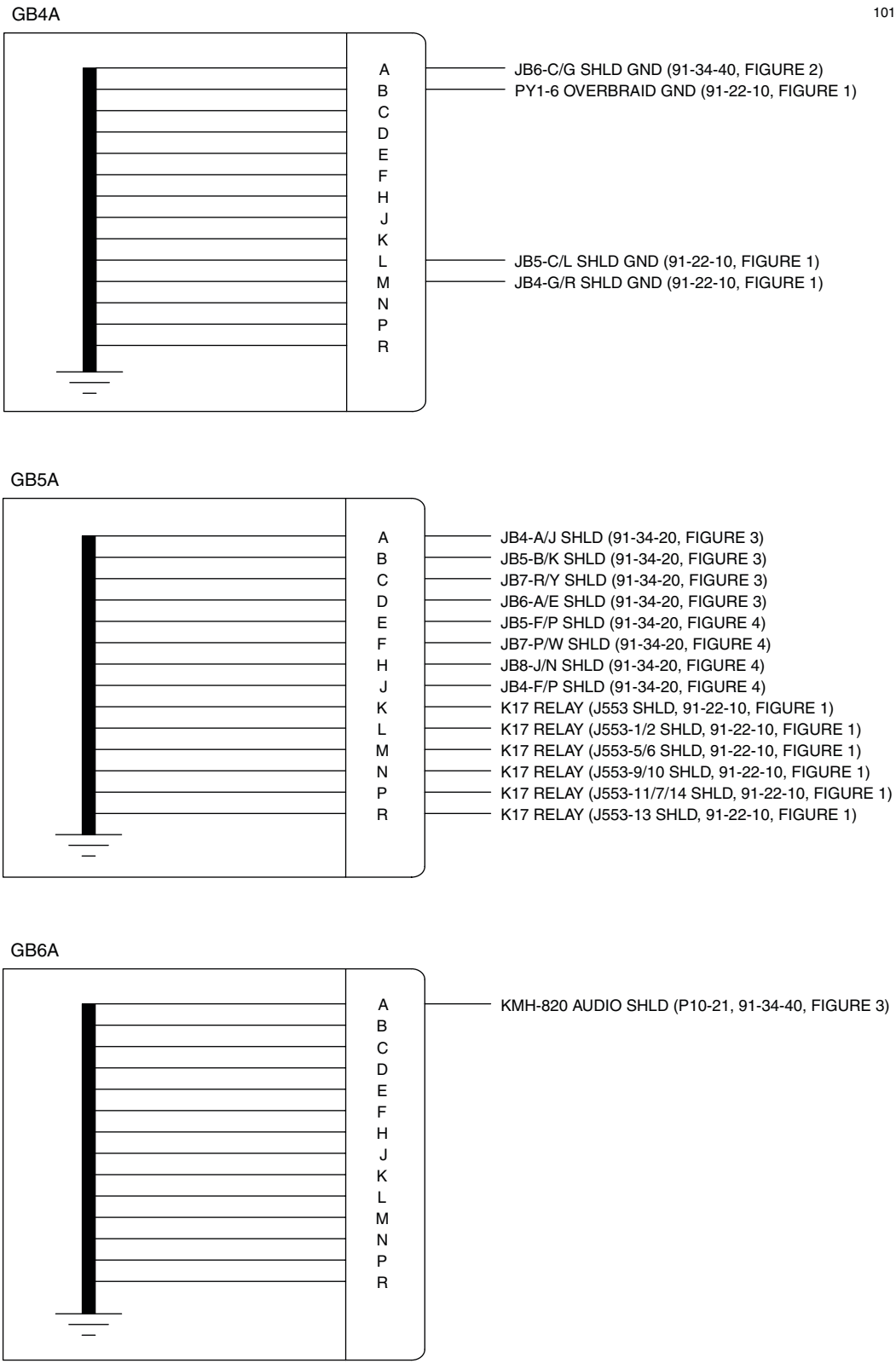


GB3A



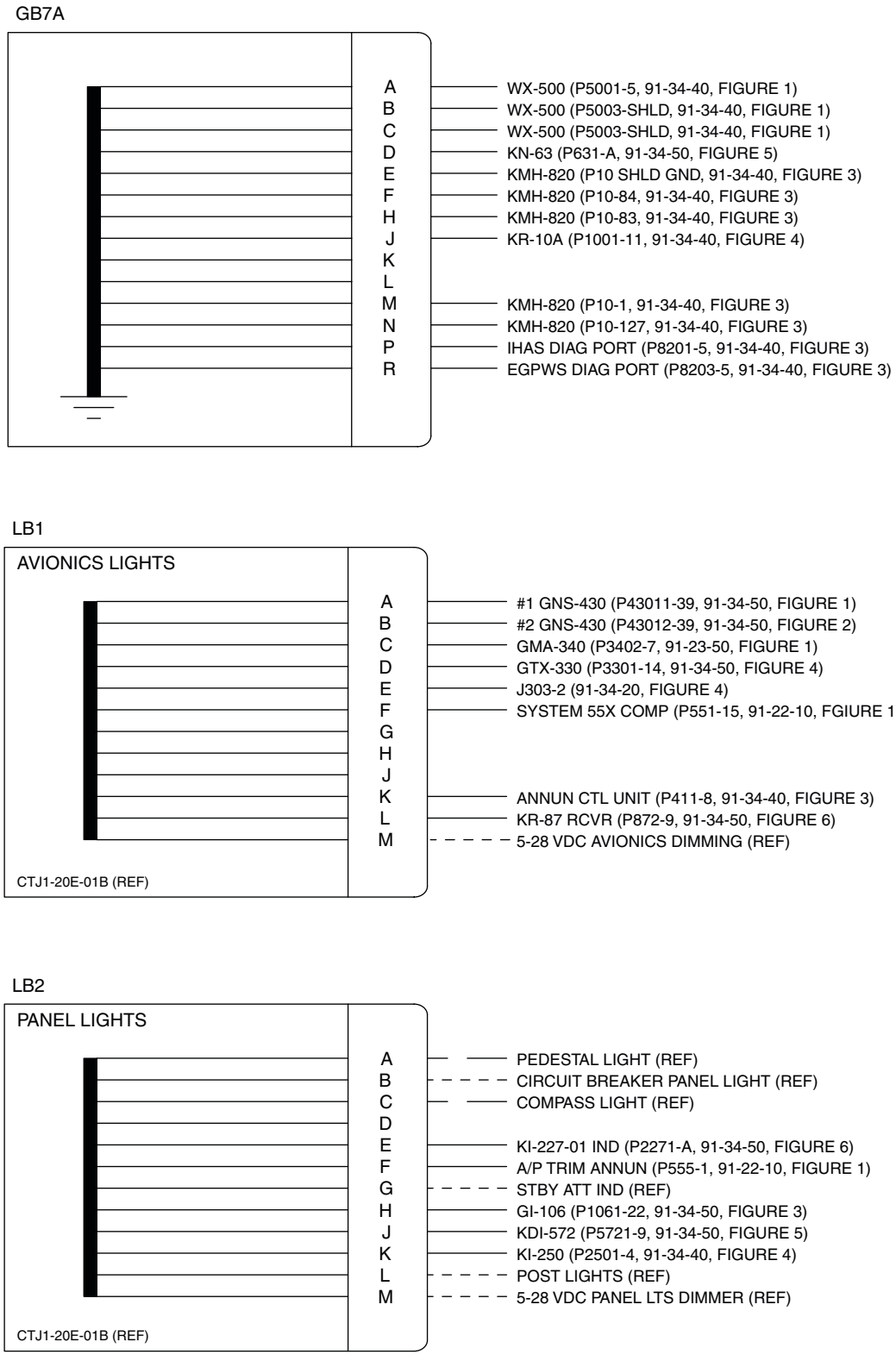
Effectivity
Seneca V
with Avidyne Entegra

Terminal Blocks and Grounds
Figure 3 (Sheet 1 of 7)



Terminal Blocks and Grounds
Figure 3 (Sheet 2 of 7)

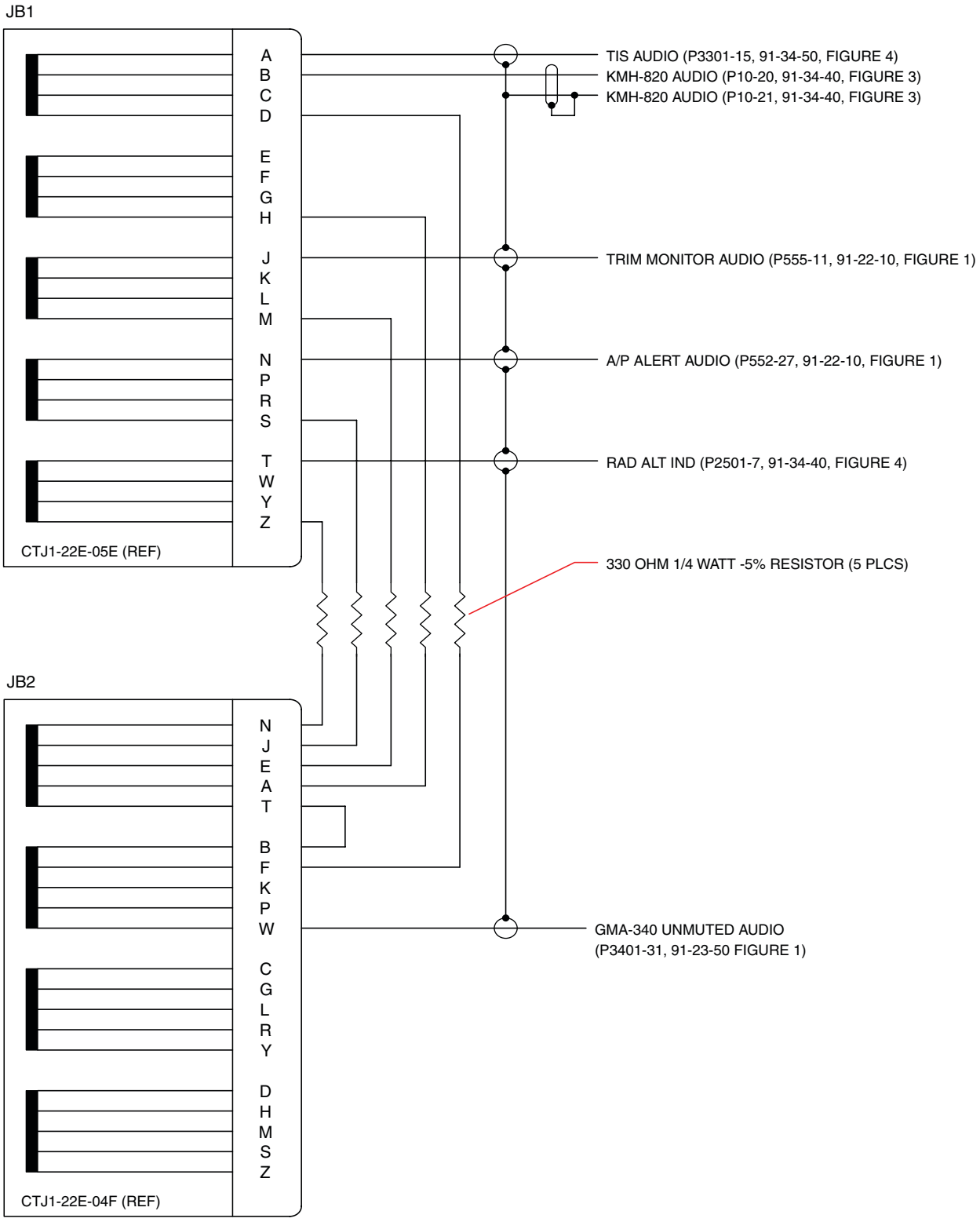
Effectivity
Seneca V
with Avidyne Entegra



Effectivity
Seneca V
with Avidyne Entegra

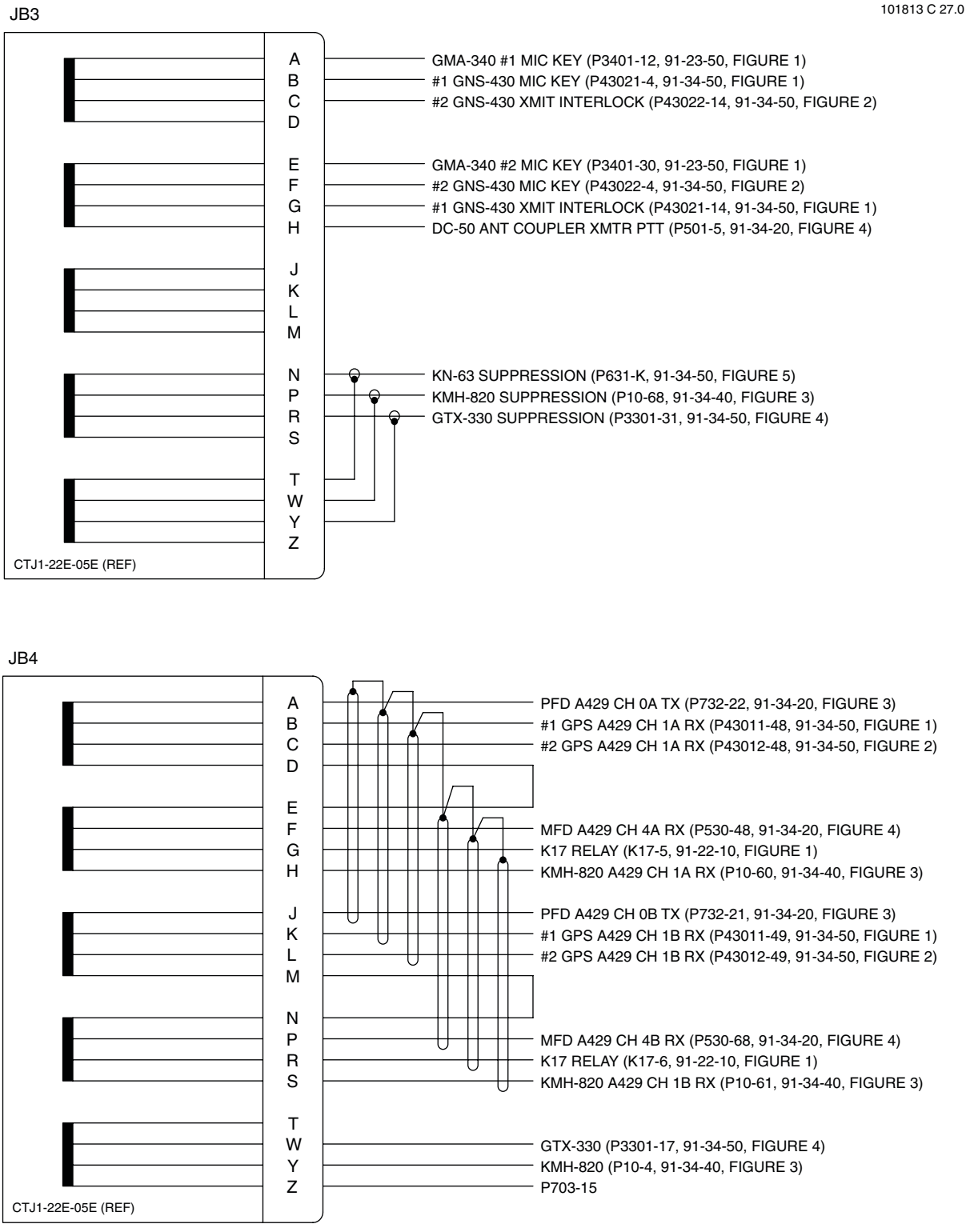
Terminal Blocks and Grounds
Figure 3 (Sheet 3 of 7)

101813 C 26.0



Terminal Blocks and Grounds
Figure 3 (Sheet 4 of 7)

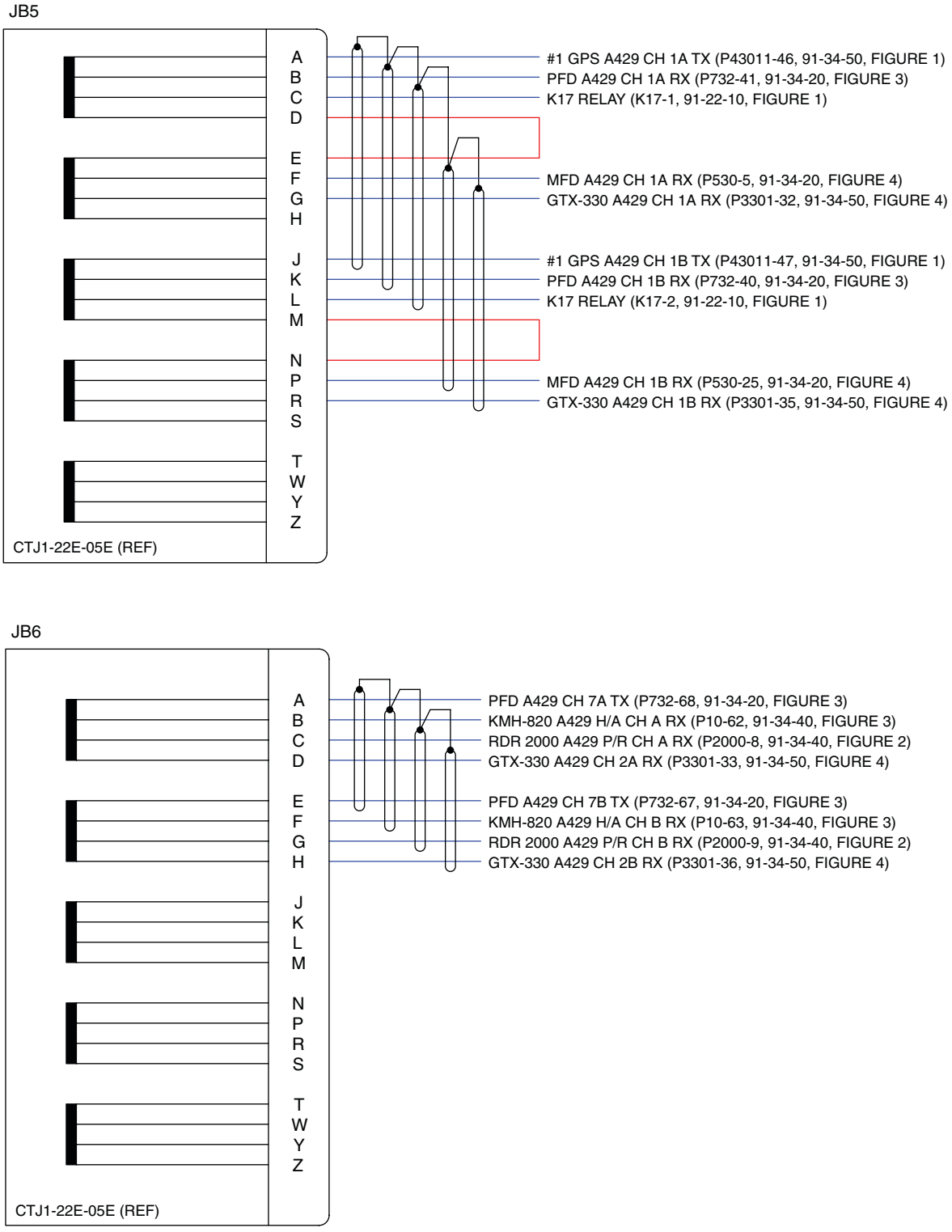
Effectivity
Seneca V
with Avidyne Entegra



Effectivity
Seneca V
with Avidyne Entegra

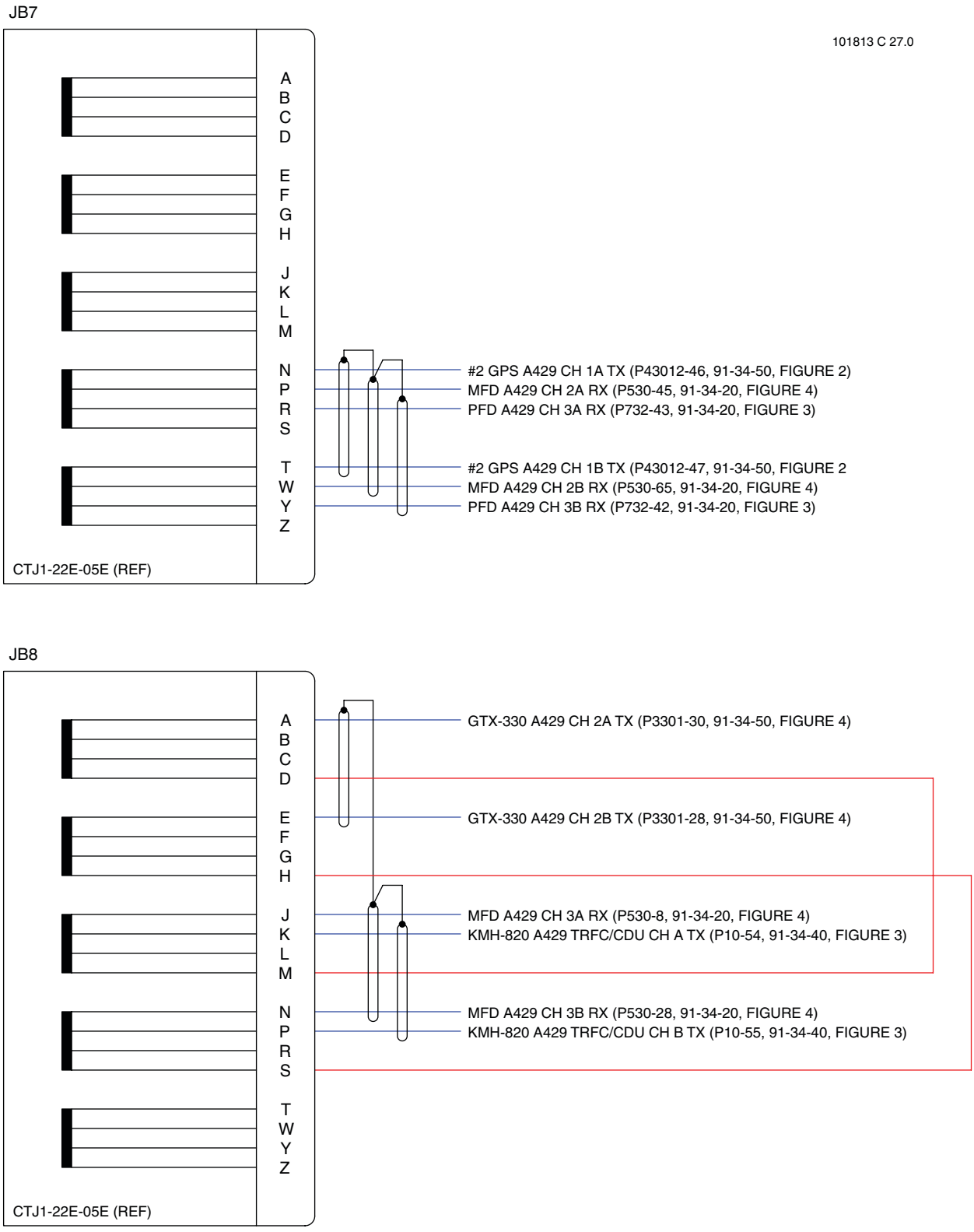
Terminal Blocks and Grounds
Figure 3 (Sheet 5 of 7)

101813 C 27.0



Terminal Blocks and Grounds
Figure 3 (Sheet 6 of 7)

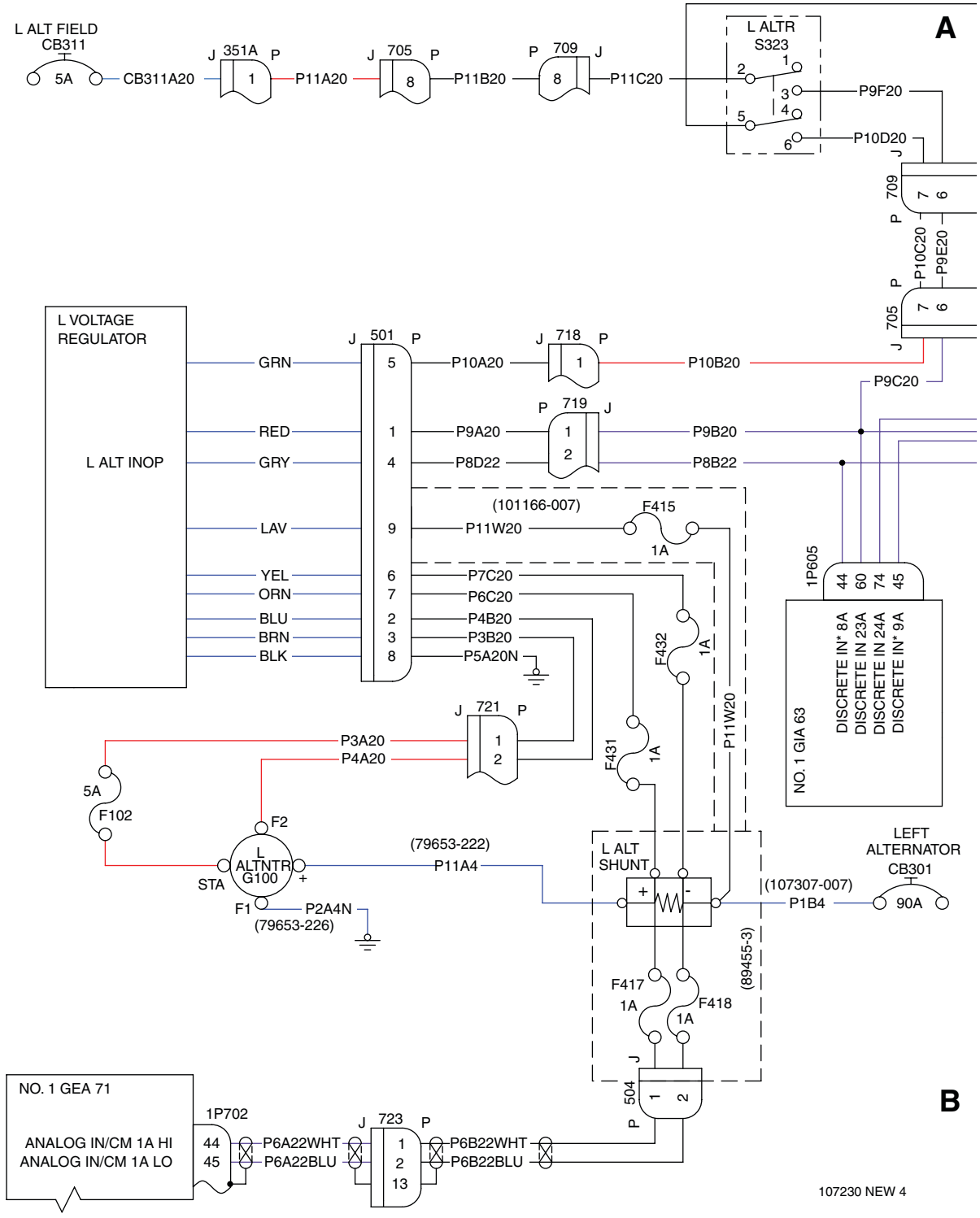
Effectivity
Seneca V
with Avidyne Entegra



Effectivity
Seneca V
with Avidyne Entegra

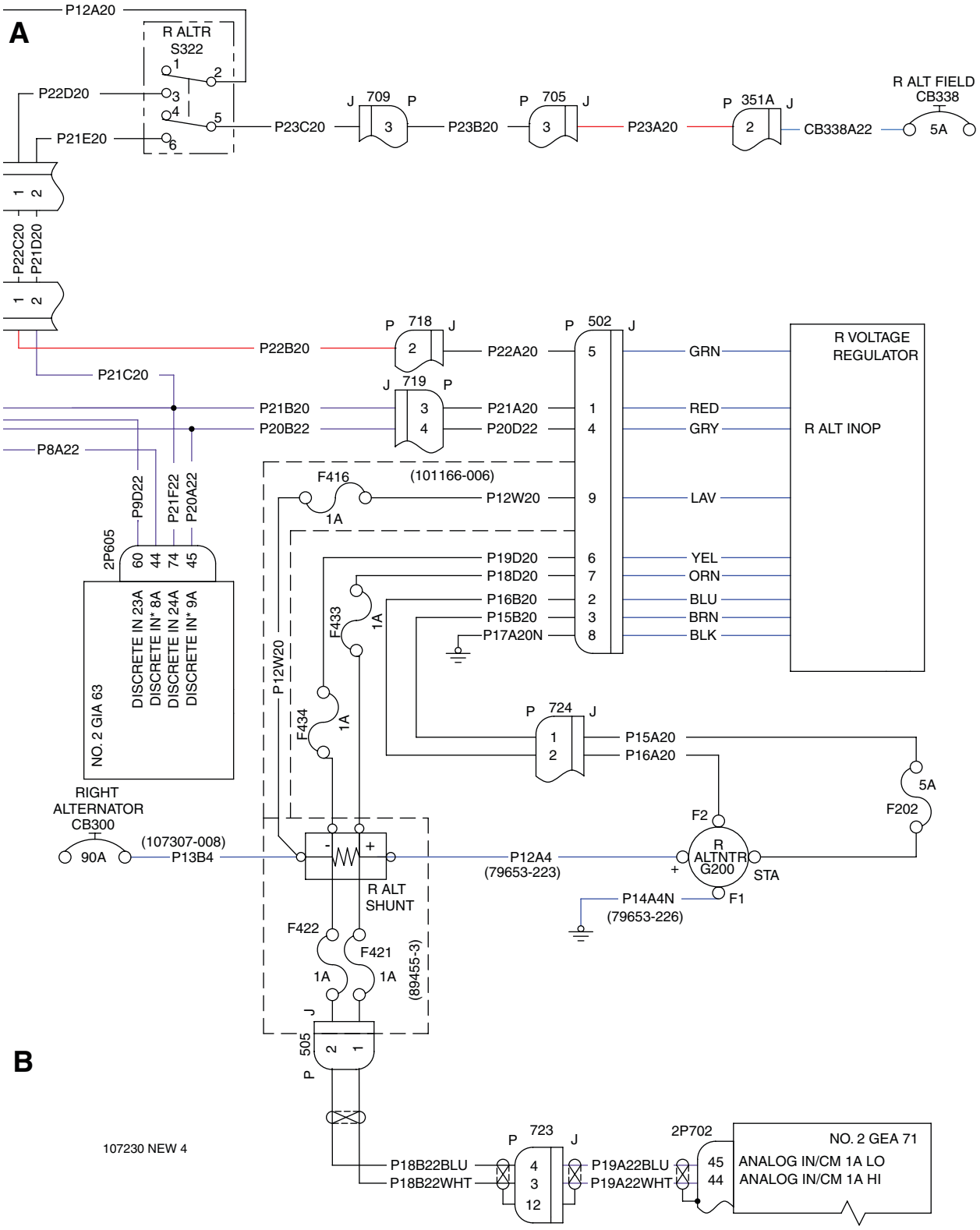
Terminal Blocks and Grounds
Figure 3 (Sheet 7 of 7)

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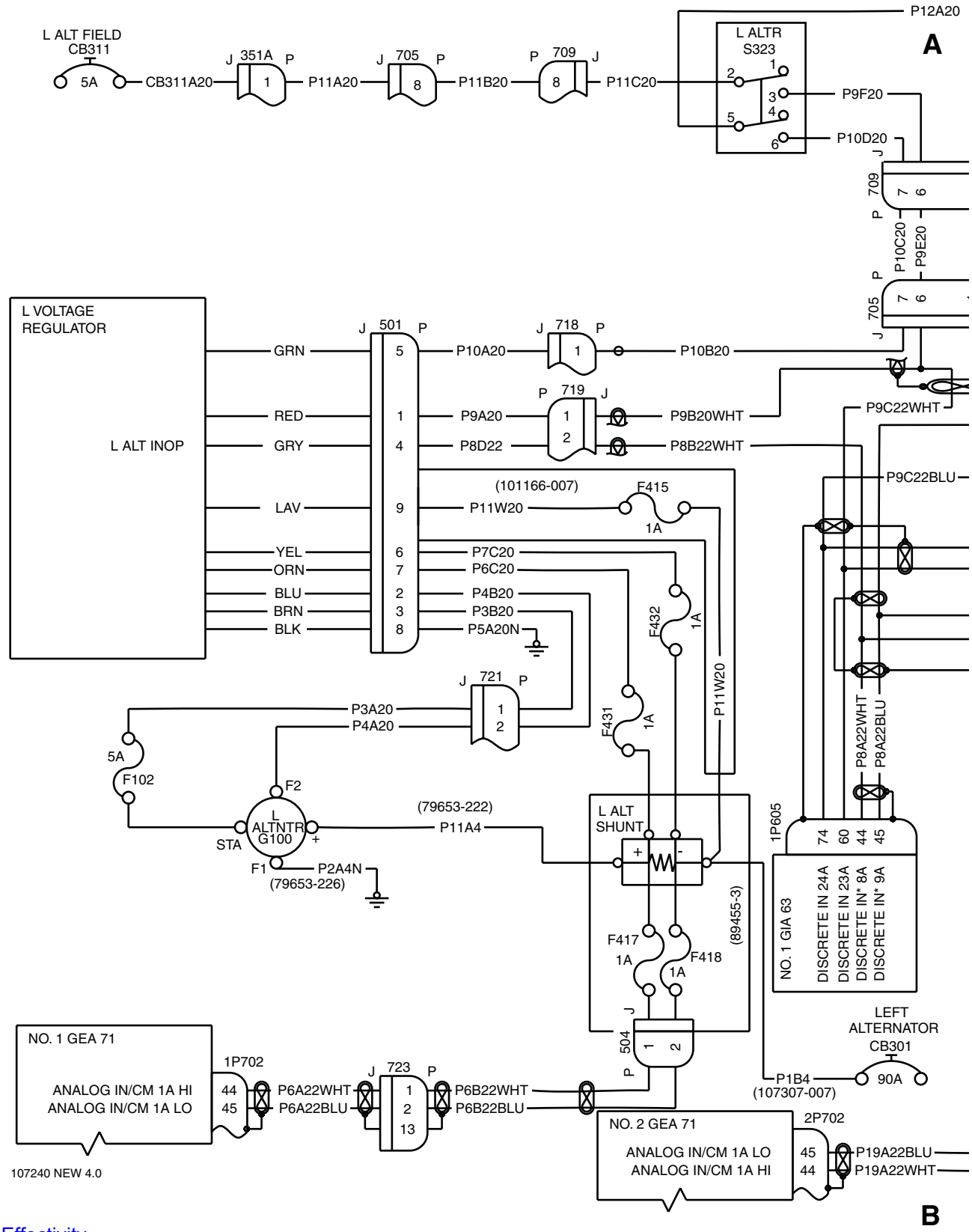
Effectivity
Seneca V
with Garmin G1000

Electrical Generation
Figure 4 (Sheet 1 of 6)



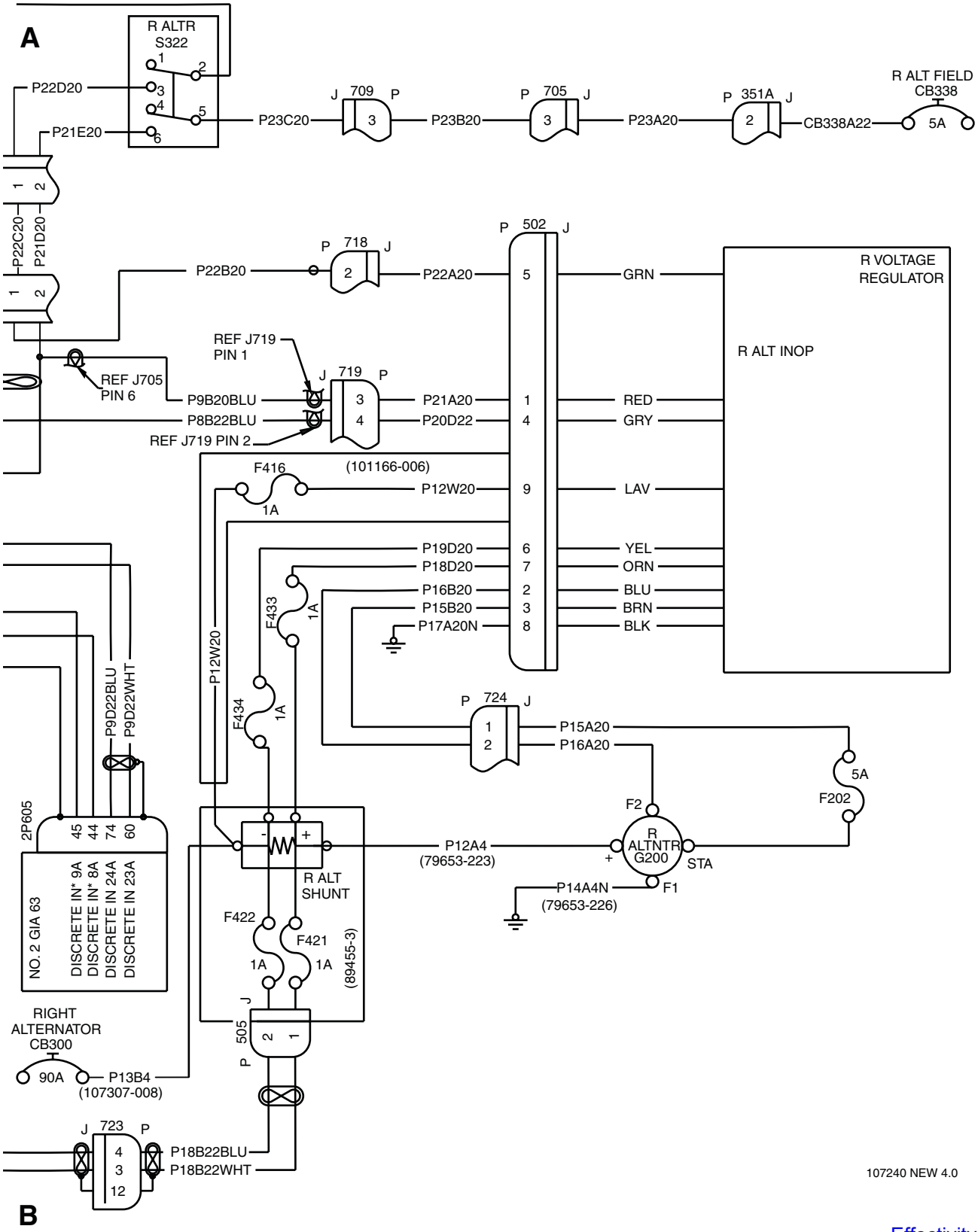
Electrical Generation
Figure 4 (Sheet 2 of 6)

Effectivity
Seneca V
with Garmin G1000



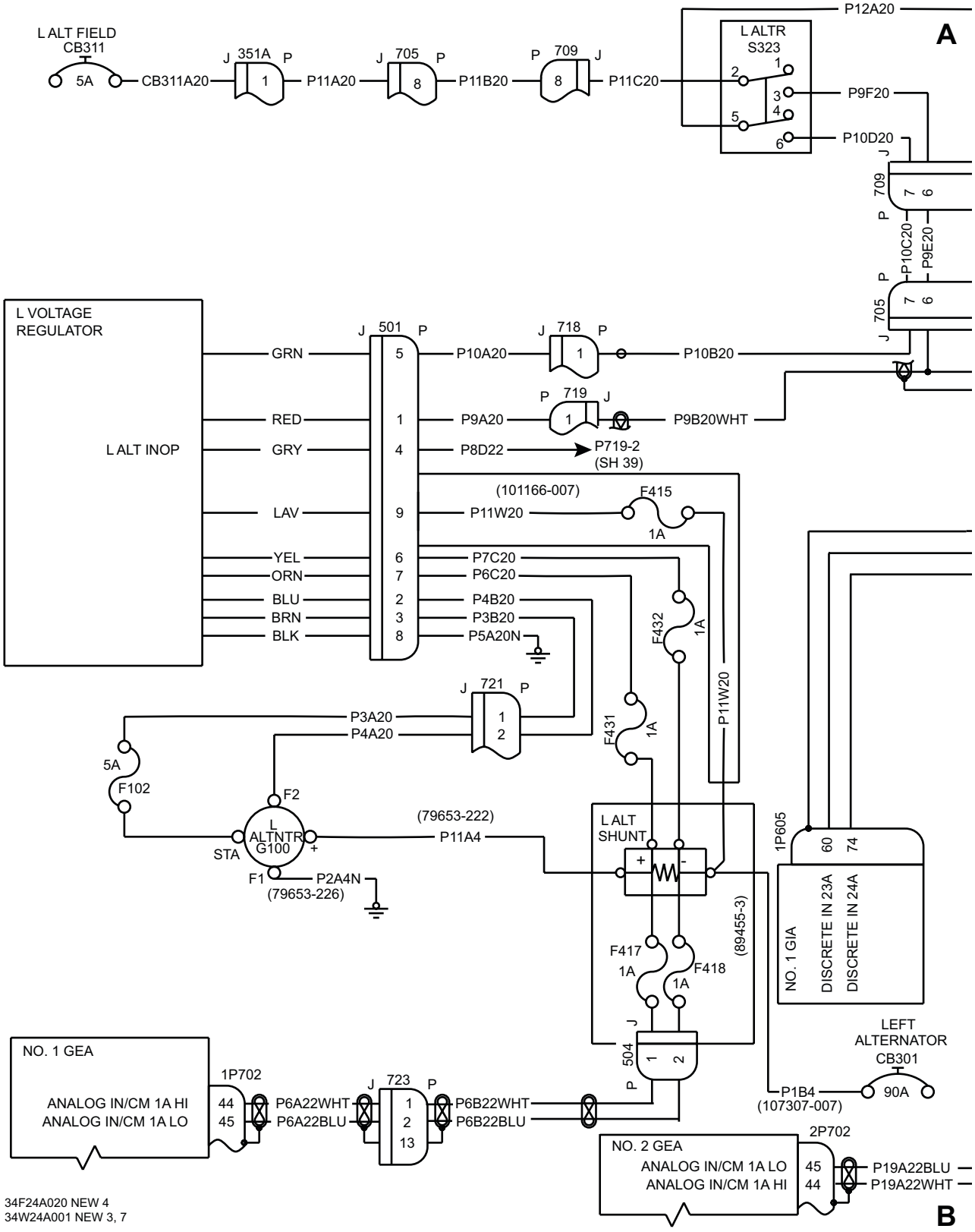
Effectivity
Seneca V
with Garmin GFC 700

Electrical Generation
Figure 4 (Sheet 3 of 6)



Electrical Generation
Figure 4 (Sheet 4 of 6)

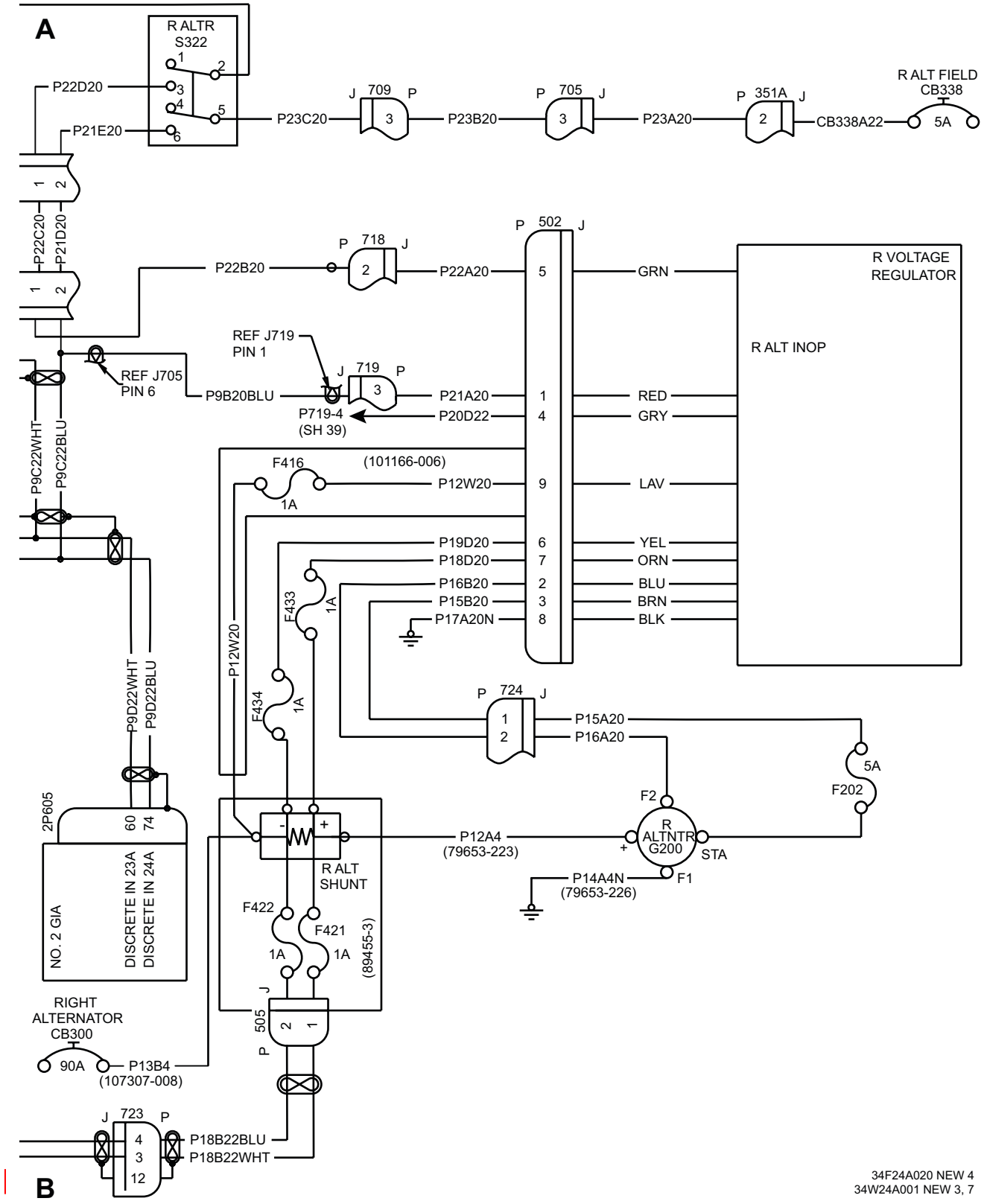
Effectivity
Seneca V
with Garmin GFC 700



34F24A020 NEW 4
34W24A001 NEW 3, 7

Effectivity
Seneca V
with Garmin G1000 NXi

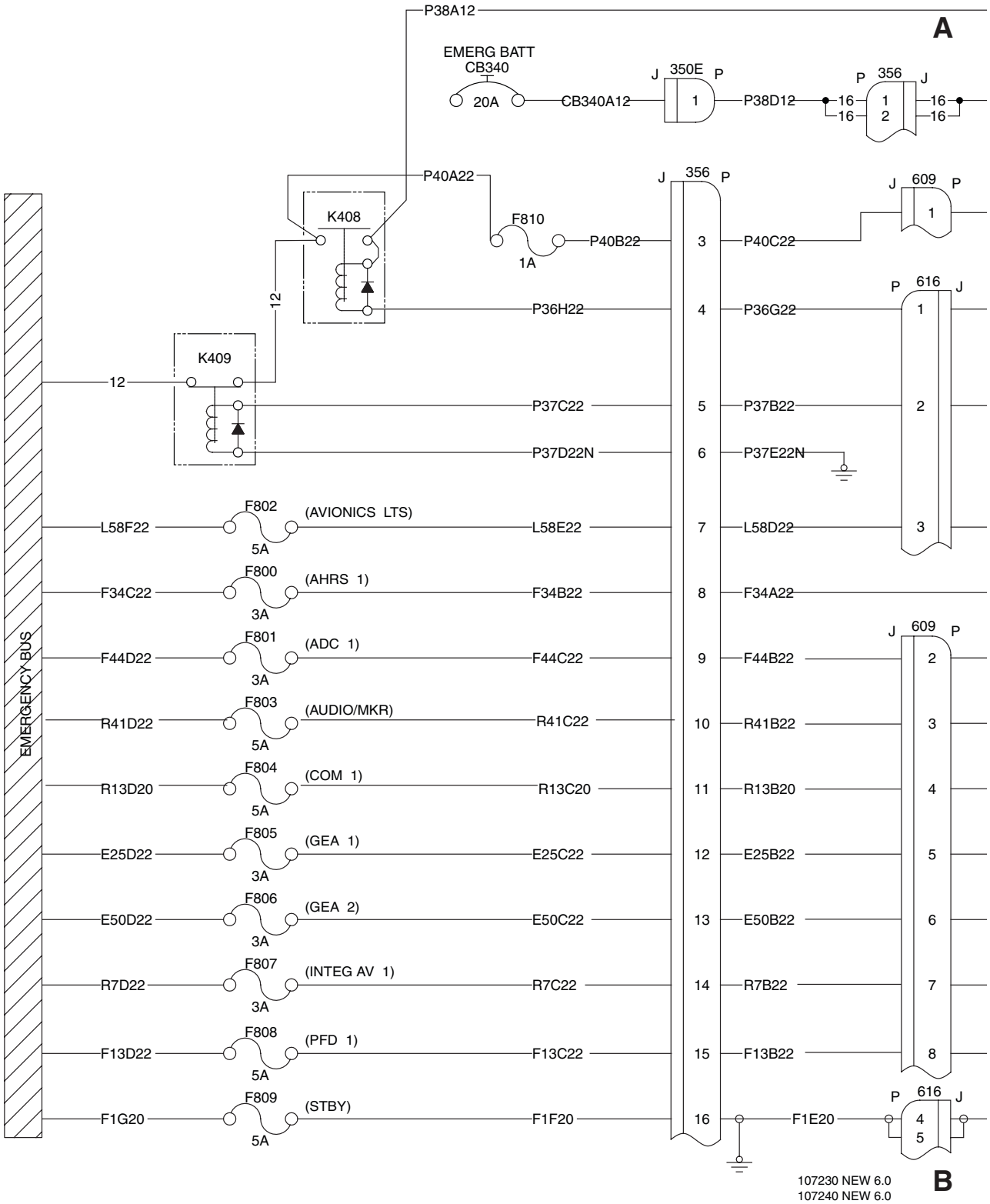
Electrical Generation
Figure 4 (Sheet 5 of 6)



34F24A020 NEW 4
34W24A001 NEW 3, 7

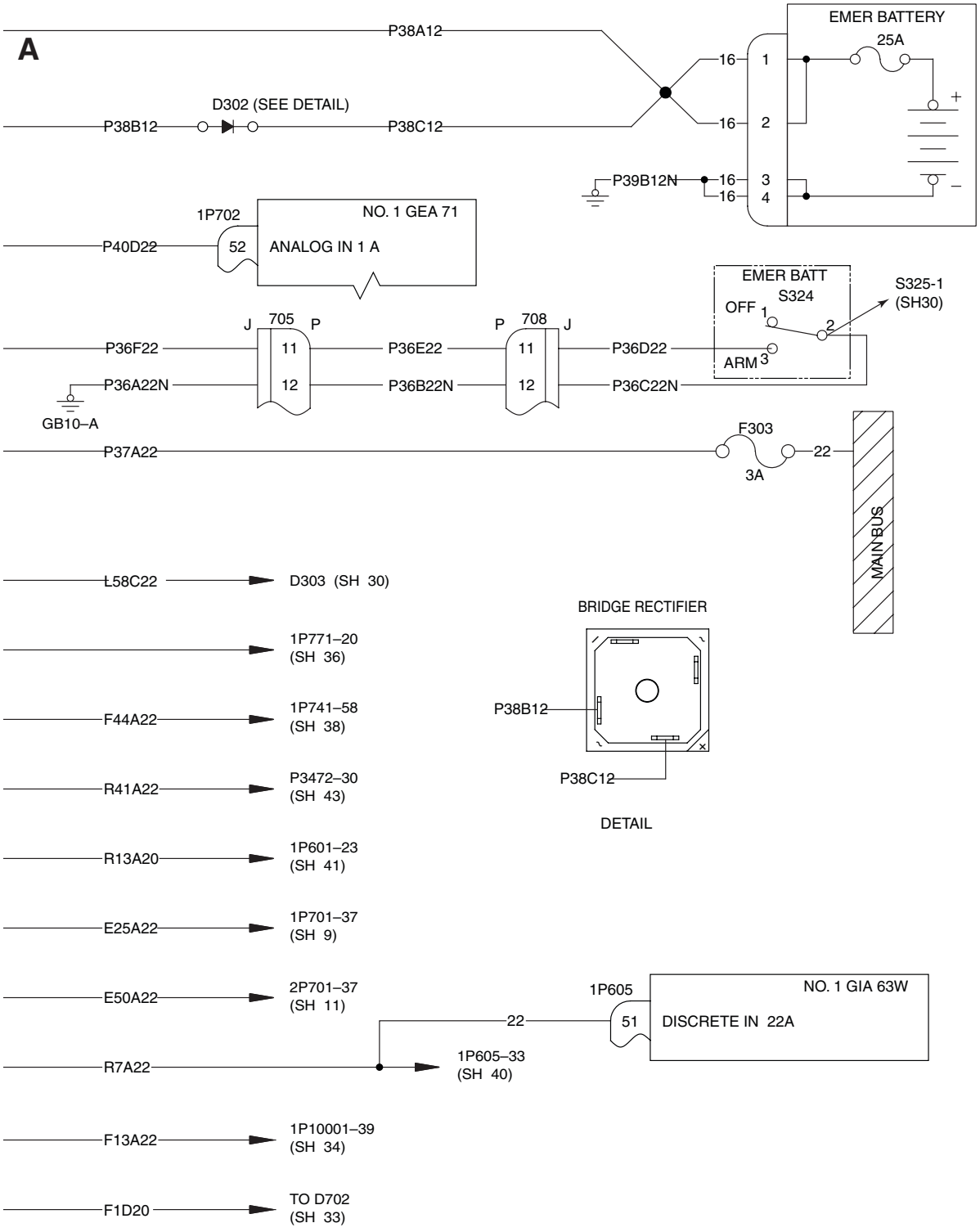
Electrical Generation
Figure 4 (Sheet 6 of 6)

Effectivity
Seneca V
with Garmin G1000 NXi



Effectivity
Seneca V
with Garmin G1000

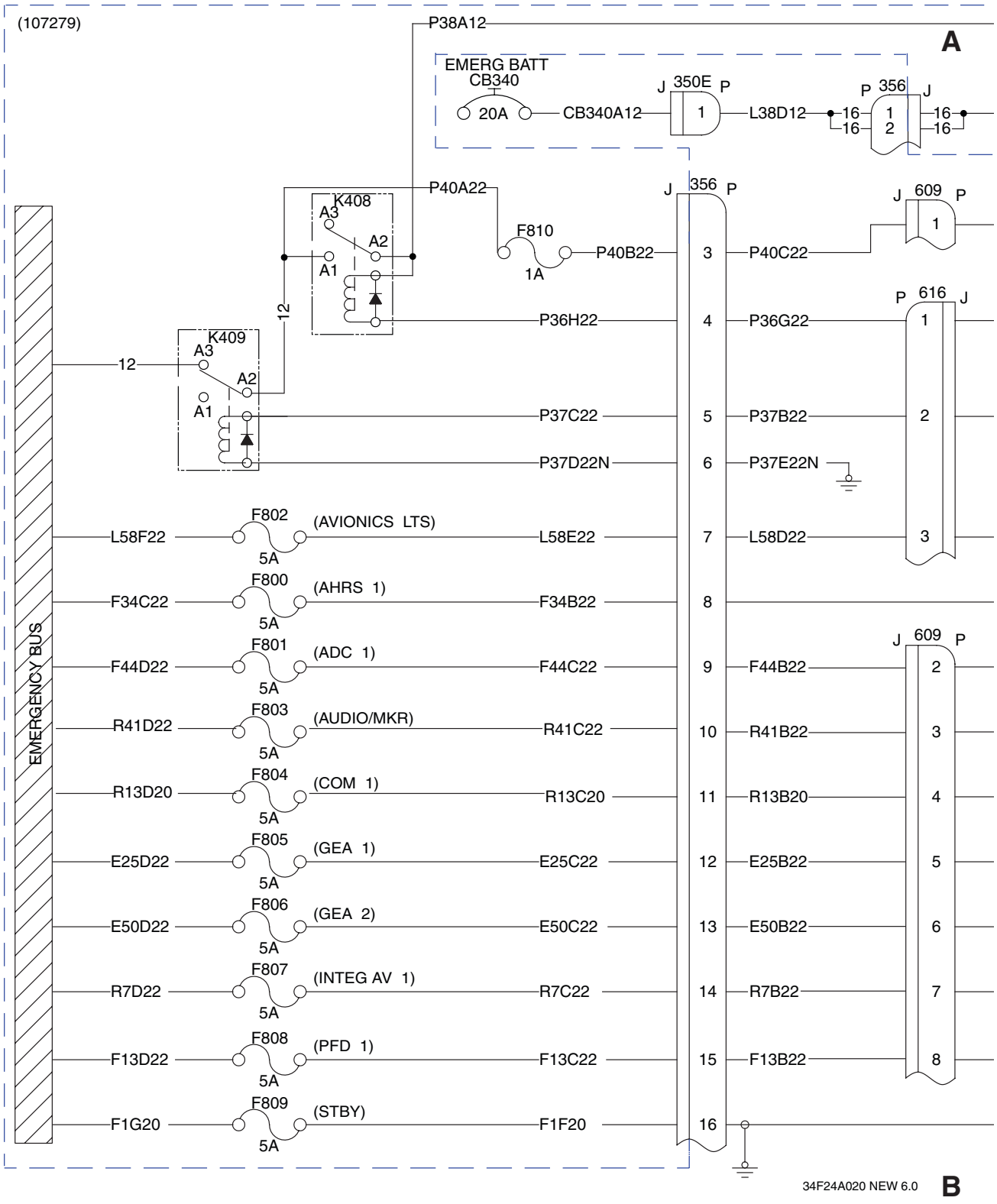
Emergency Power
Figure 5 (Sheet 1 of 4)



B 107230 NEW 6.0
107240 NEW 6.0

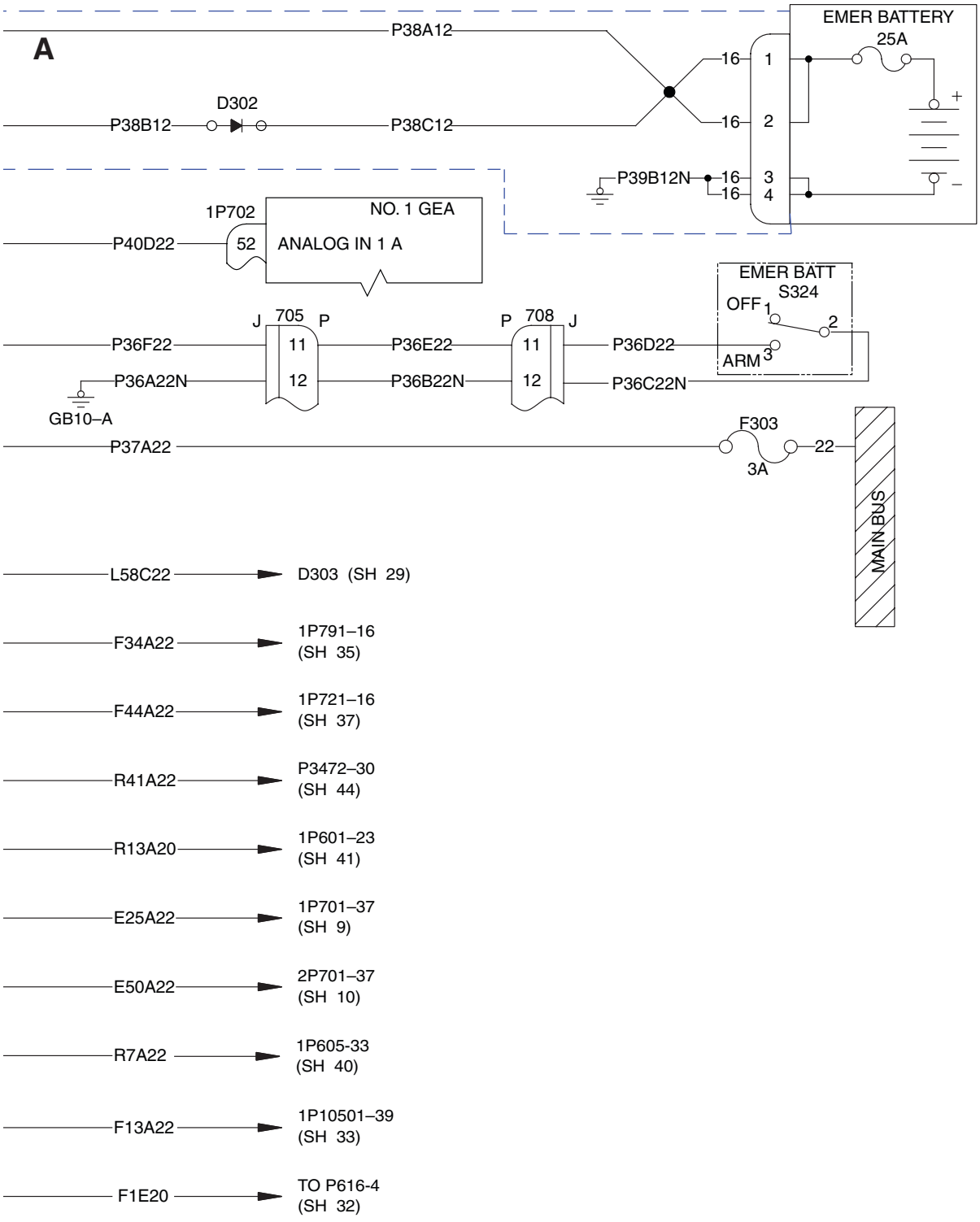
Emergency Power
Figure 5 (Sheet 2 of 4)

Effectivity
Seneca V
with Garmin G1000



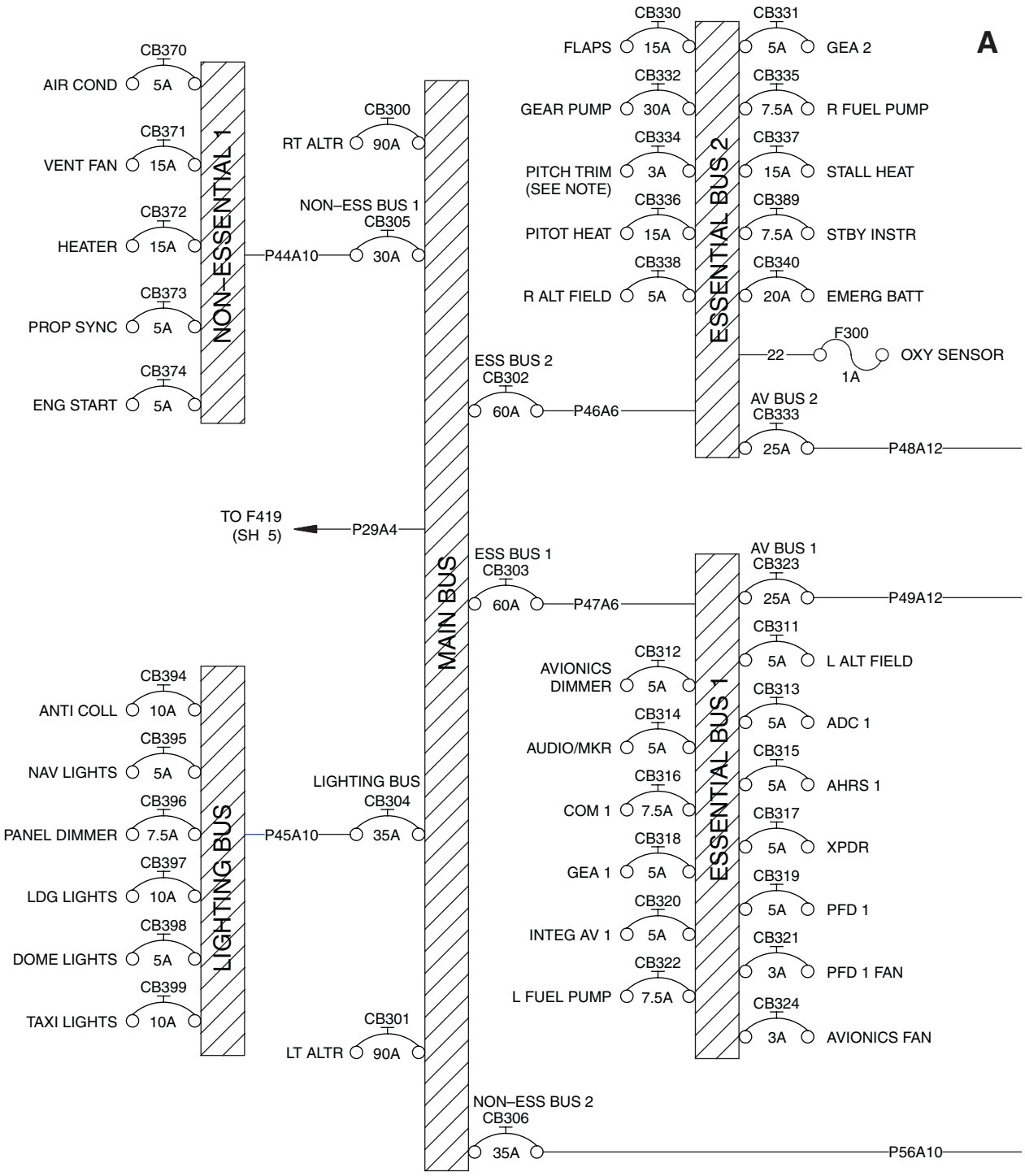
Effectivity
Seneca V
with Garmin G1000 NXi

Emergency Power
Figure 5 (Sheet 3 of 4)



Emergency Power
Figure 5 (Sheet 4 of 4)

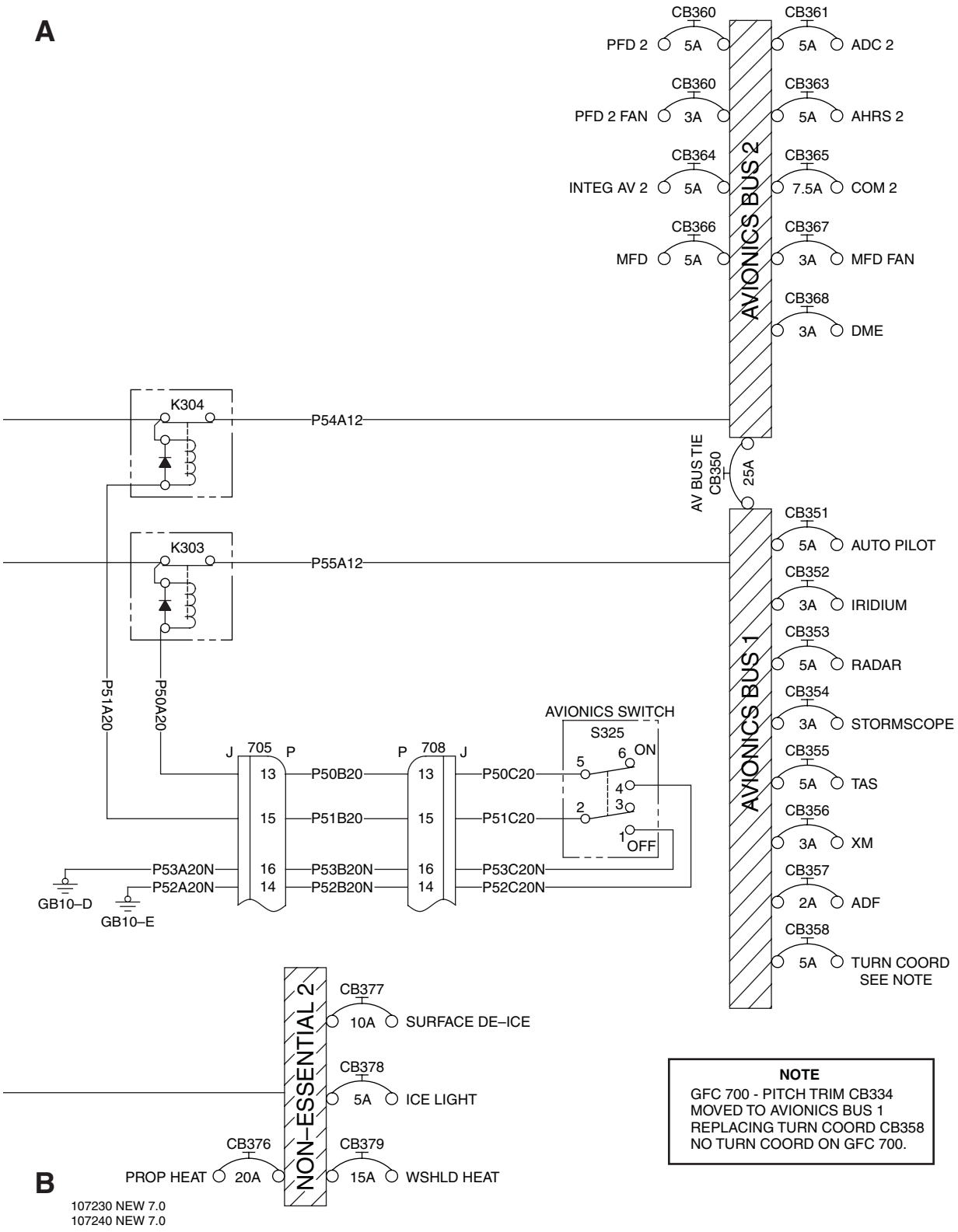
Effectivity
Seneca V
with Garmin G1000 NXi



107230 NEW 7.0
107240 NEW 7.0

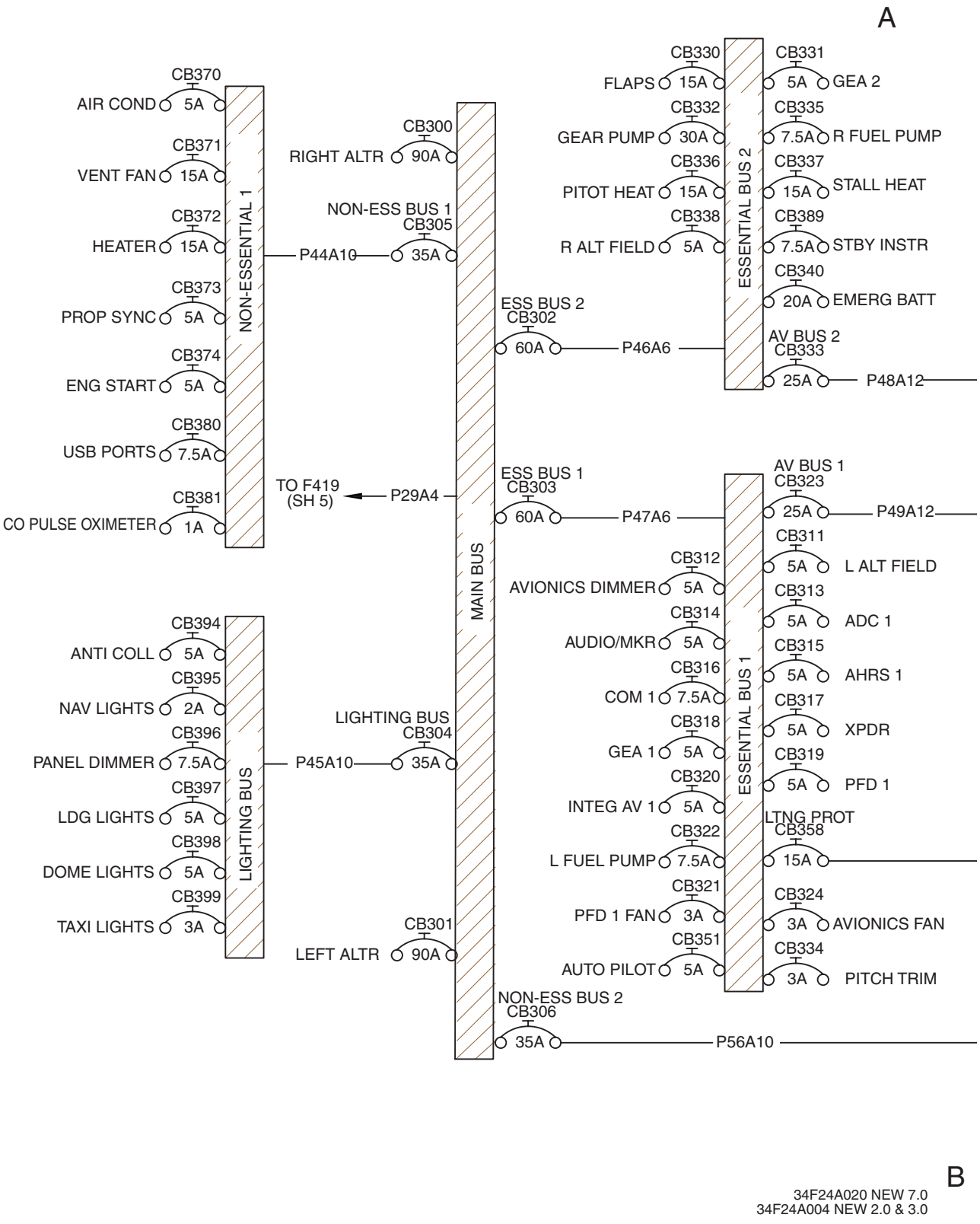
Effectivity
Seneca V
with Garmin G1000

Power Distribution
Figure 6 (Sheet 1 of 4)



Power Distribution
Figure 6 (Sheet 2 of 4)

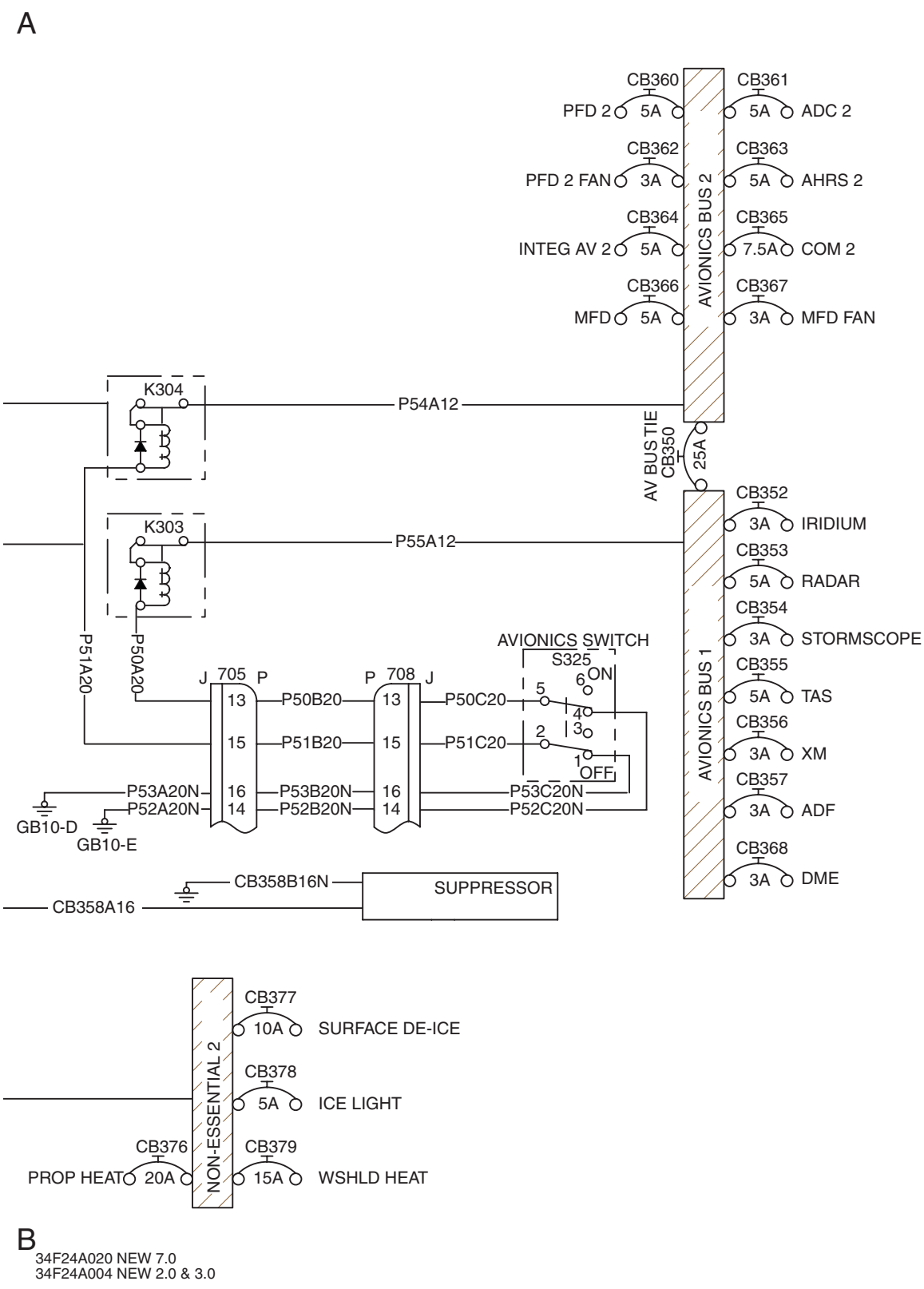
Effectivity
Seneca V
with Garmin G1000



34F24A020 NEW 7.0
34F24A004 NEW 2.0 & 3.0

Effectivity
Seneca V
with Garmin G1000 NXi

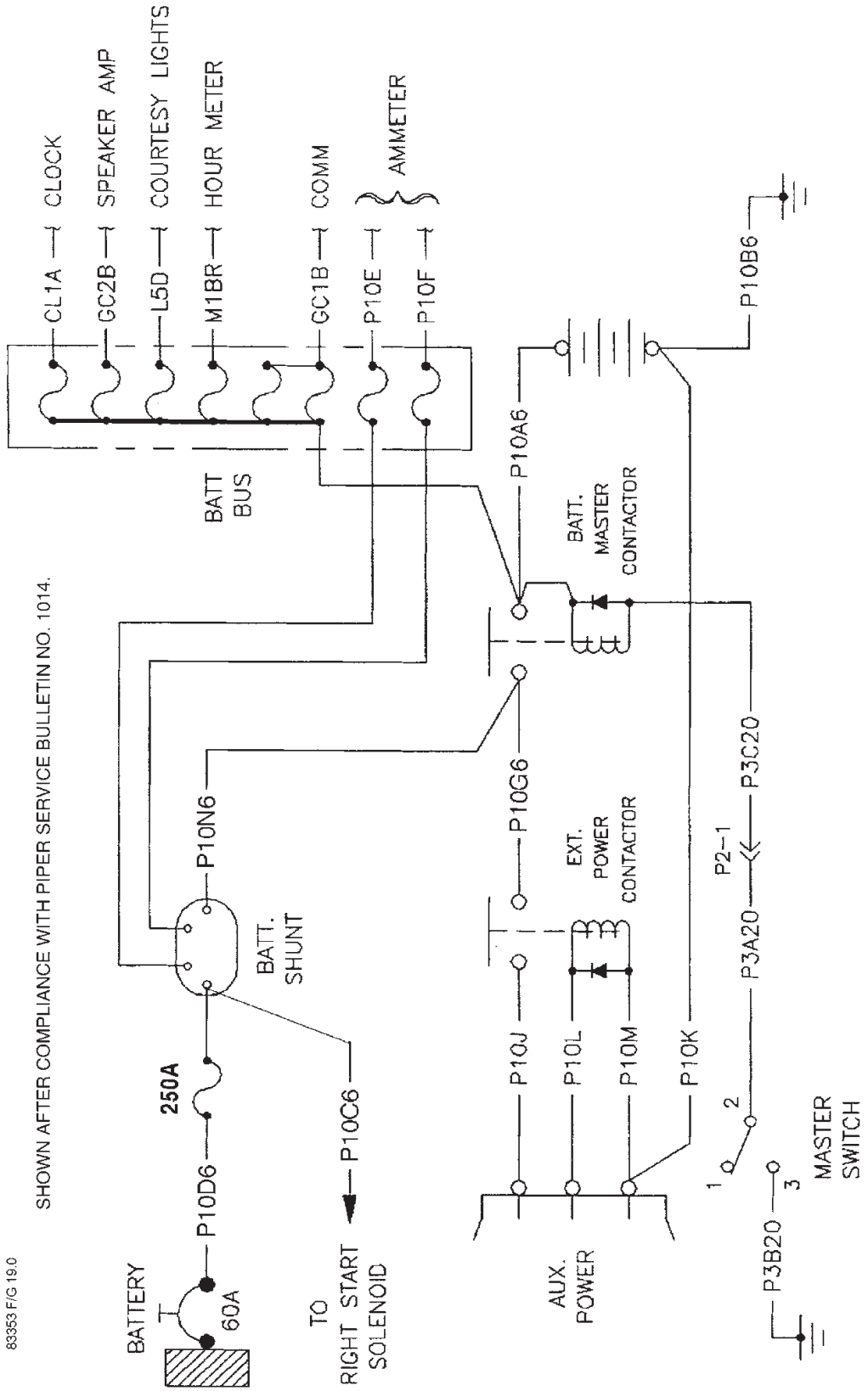
Power Distribution
Figure 6 (Sheet 3 of 4)



Power Distribution
Figure 6 (Sheet 4 of 4)

Effectivity
Seneca V
with Garmin G1000 NXi

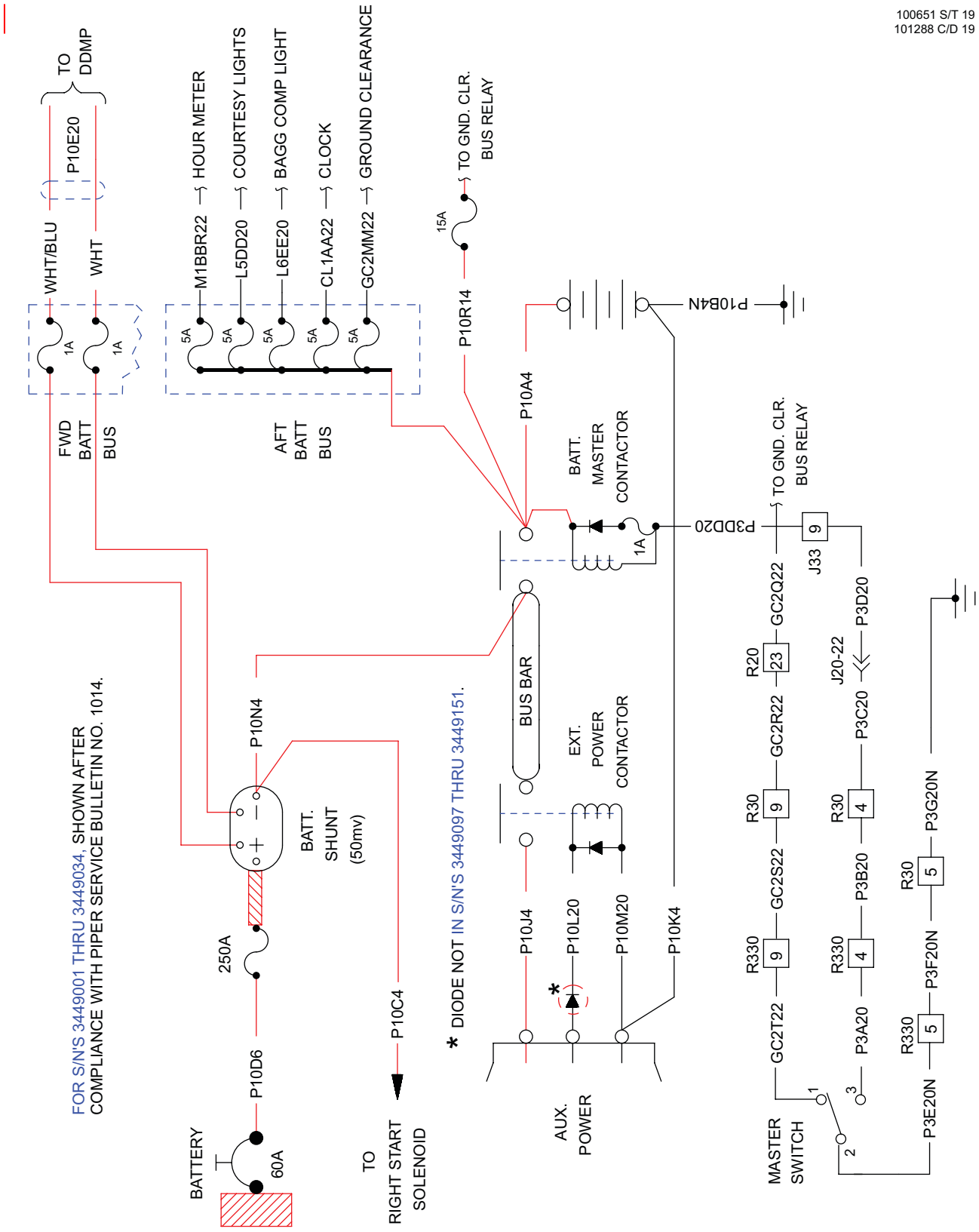
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Battery Master / External Power
Figure 1 (Sheet 1 of 4)

Effectivity
Seneca IV

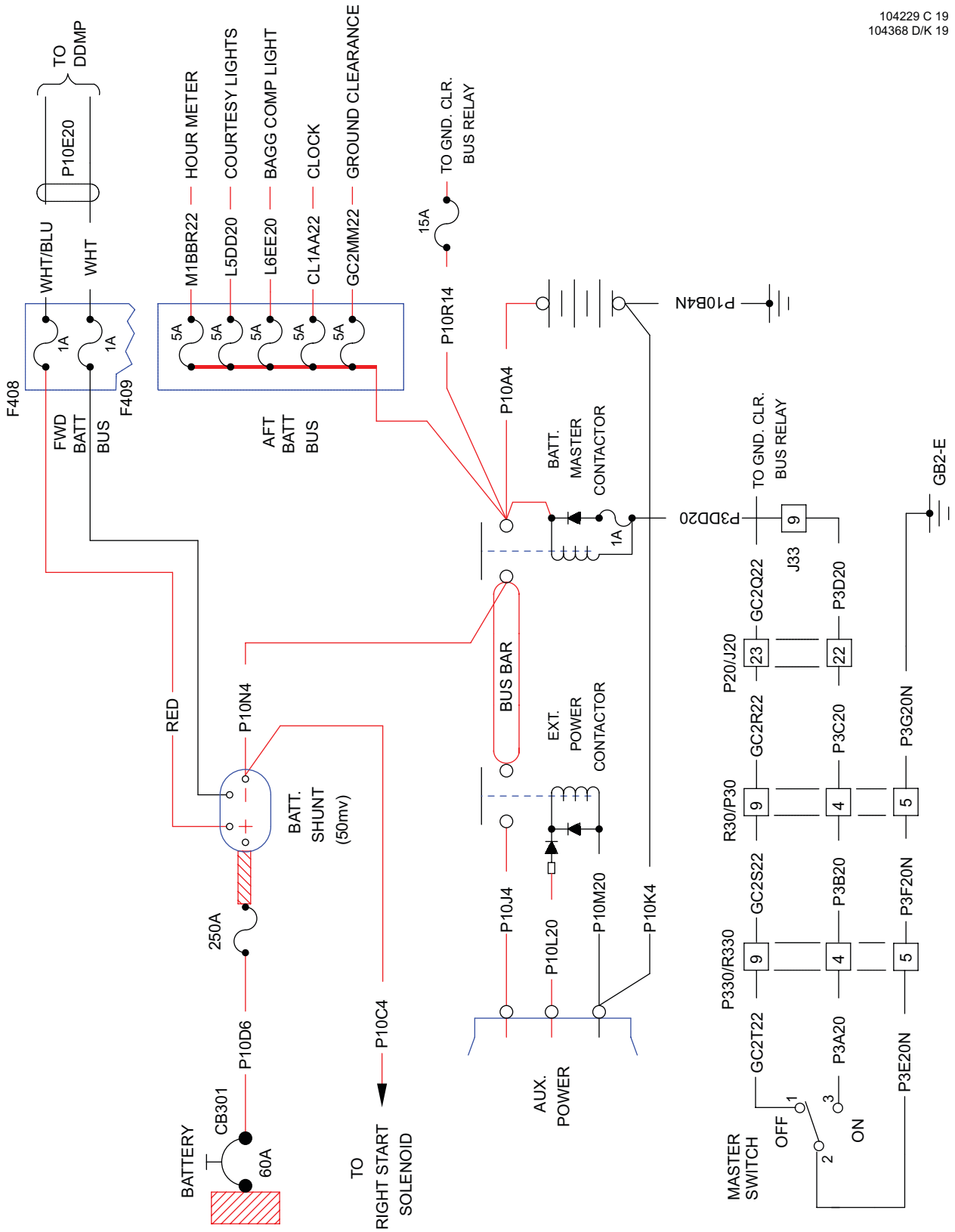
100651 S/T 19
101288 C/D 19



Effectivity
3449001 thru 3449151

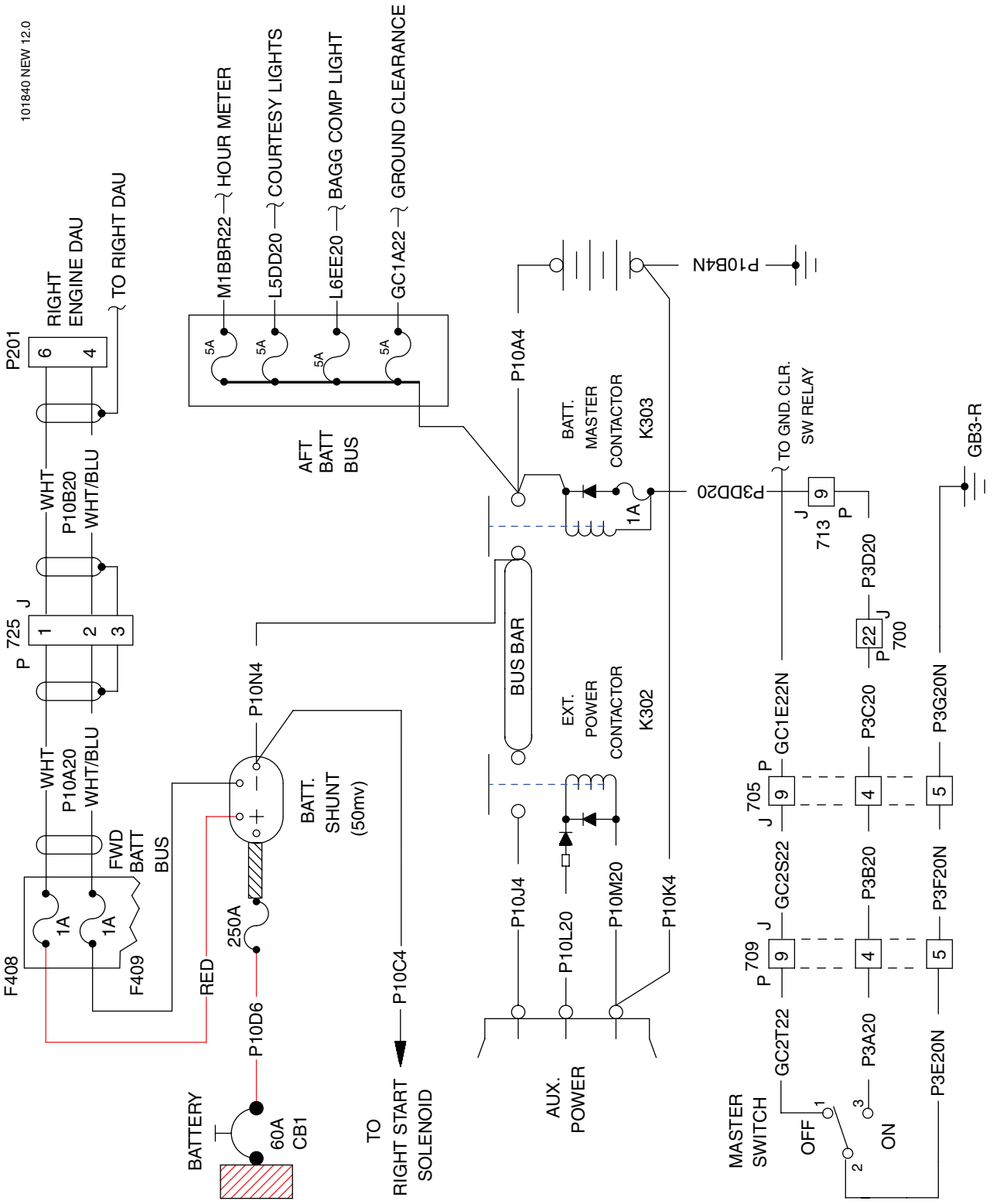
Battery Master / External Power
Figure 1 (Sheet 2 of 4)

104229 C 19
104368 D/K 19



Battery Master / External Power
Figure 1 (Sheet 3 of 4)

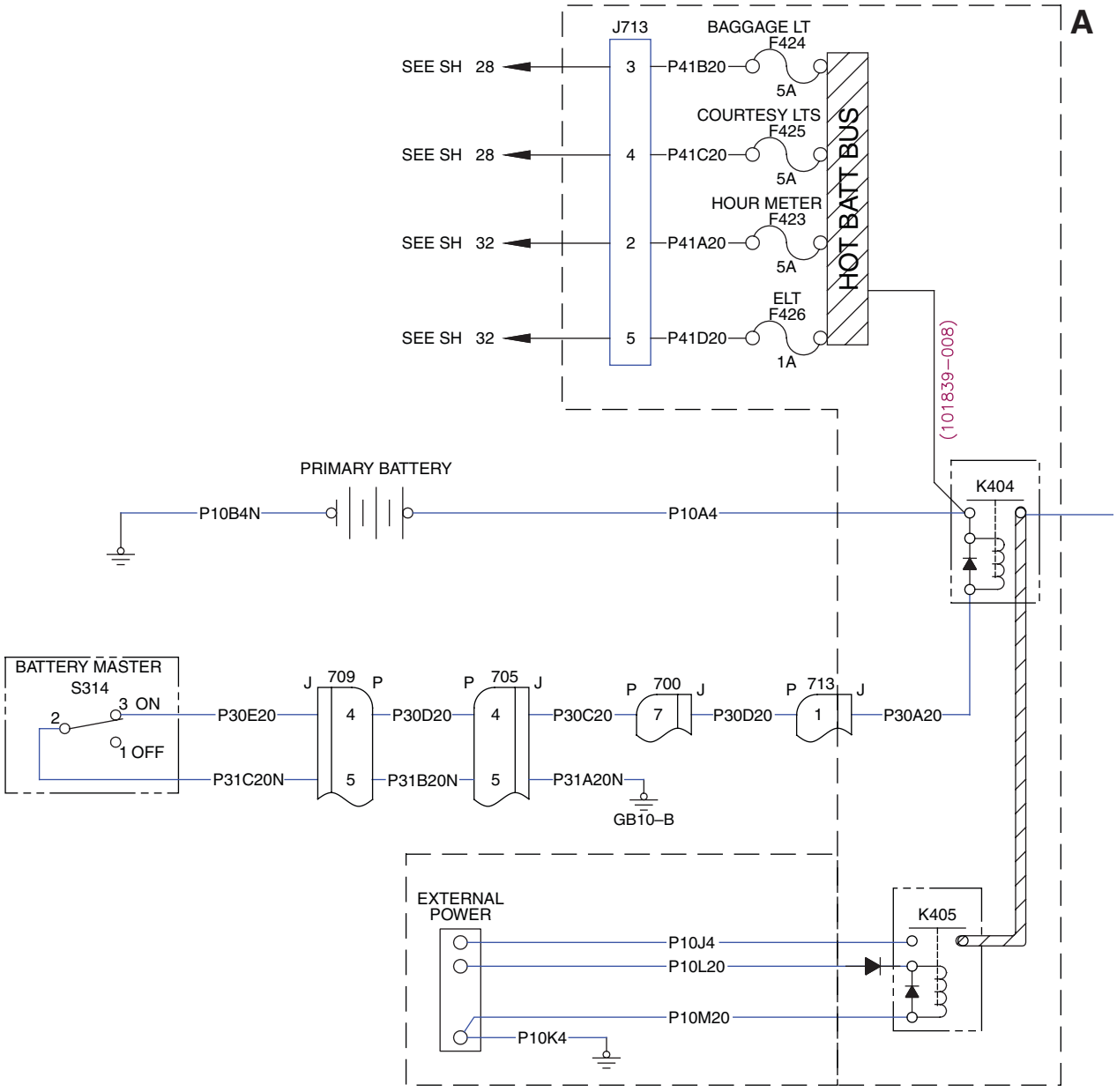
Effectivity
3449152 & up



[Effectivity
Seneca V
with Avidyne Entegra]

Battery Master / External Power
Figure 1 (Sheet 4 of 4)

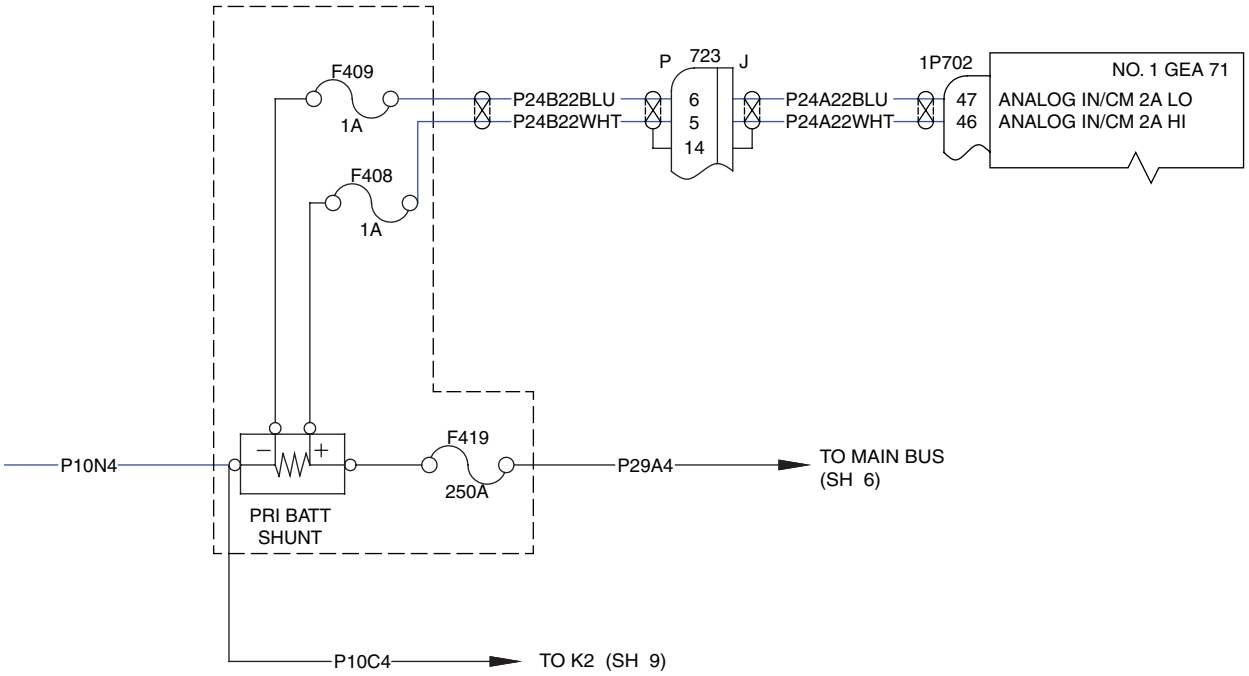
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Effectivity
Seneca V
with Garmin G1000

Primary Power
Figure 2 (Sheet 1 of 4)

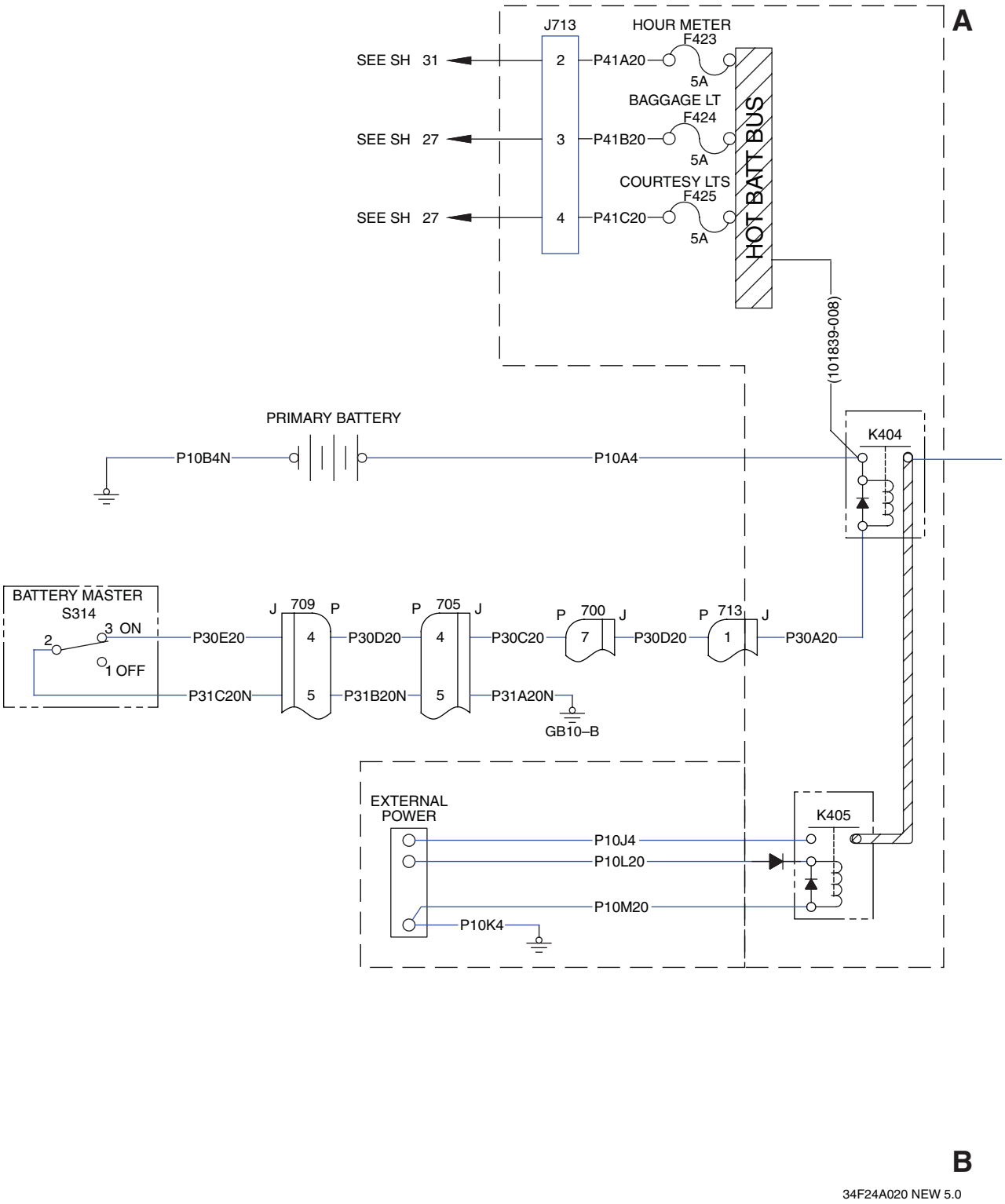
A



B

107230 NEW 5.0
107240 NEW 5.0

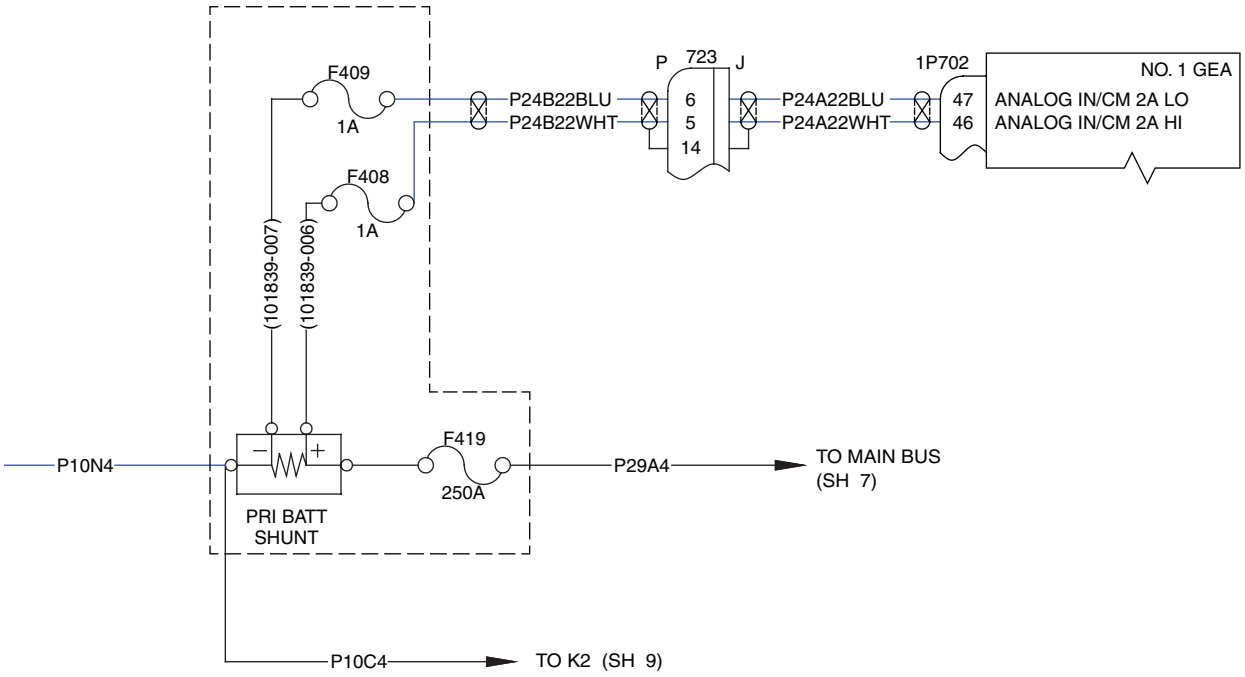
Primary Power
Figure 2 (Sheet 2 of 4)



Effectivity
Seneca V
with Garmin G1000 NXi

Primary Power
Figure 2 (Sheet 3 of 4)

A



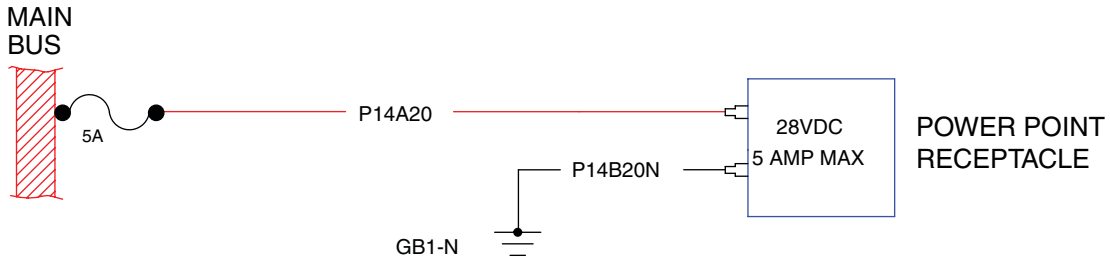
B

34F24A020 NEW 5.0

Primary Power
Figure 2 (Sheet 4 of 4)

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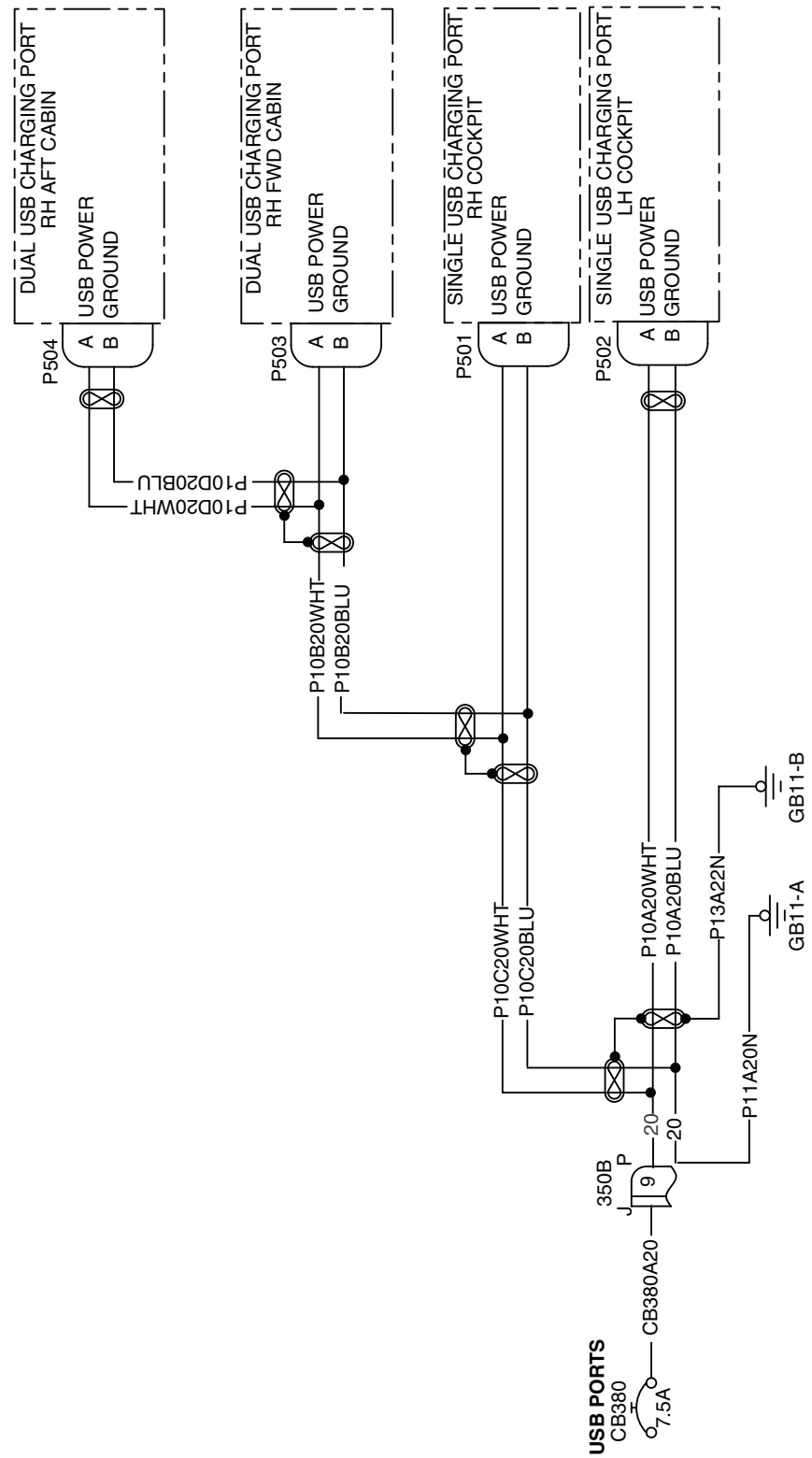
104229 NEW/C
104368 NEW/K 12.0



Power Point
Figure 1

Effectivity
3449152 & up

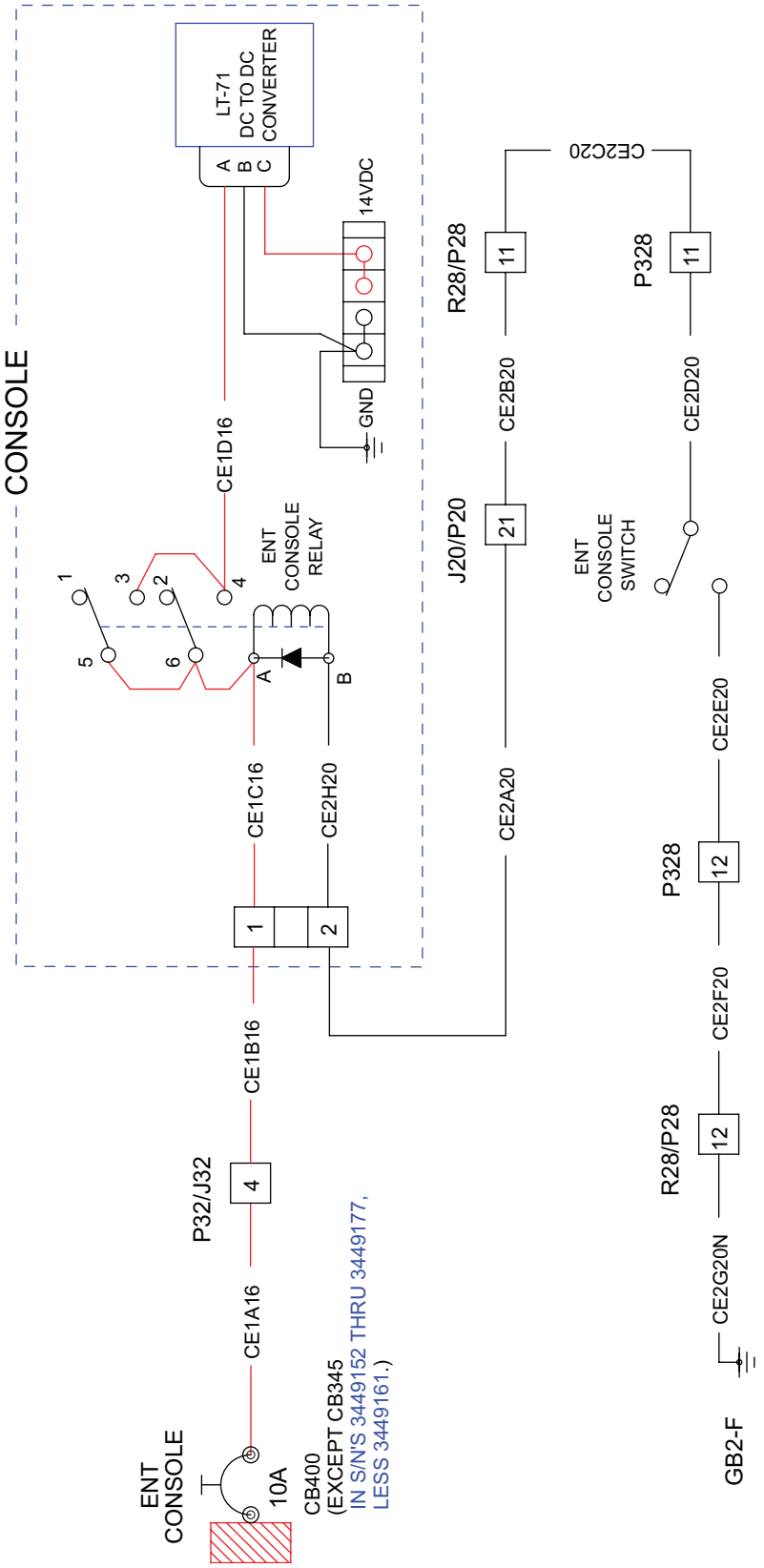
34F24A020 NEW 24.0



Effectivity
Seneca V
with Garmin G1000 NXI

USB Charging Ports
Figure 2

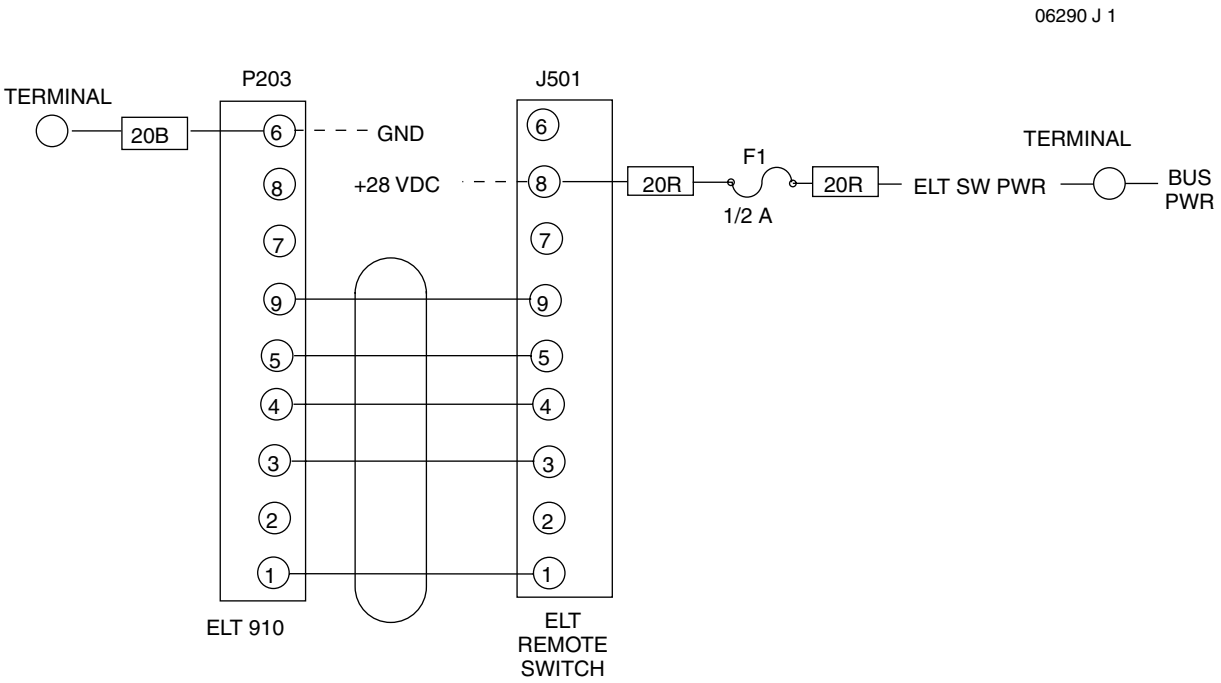
100651 H/T 44
101288 NEW/D 44
104229 C 44
104368 NEW/K 44



Entertainment Console
Figure 1

Effectivity
Seneca V

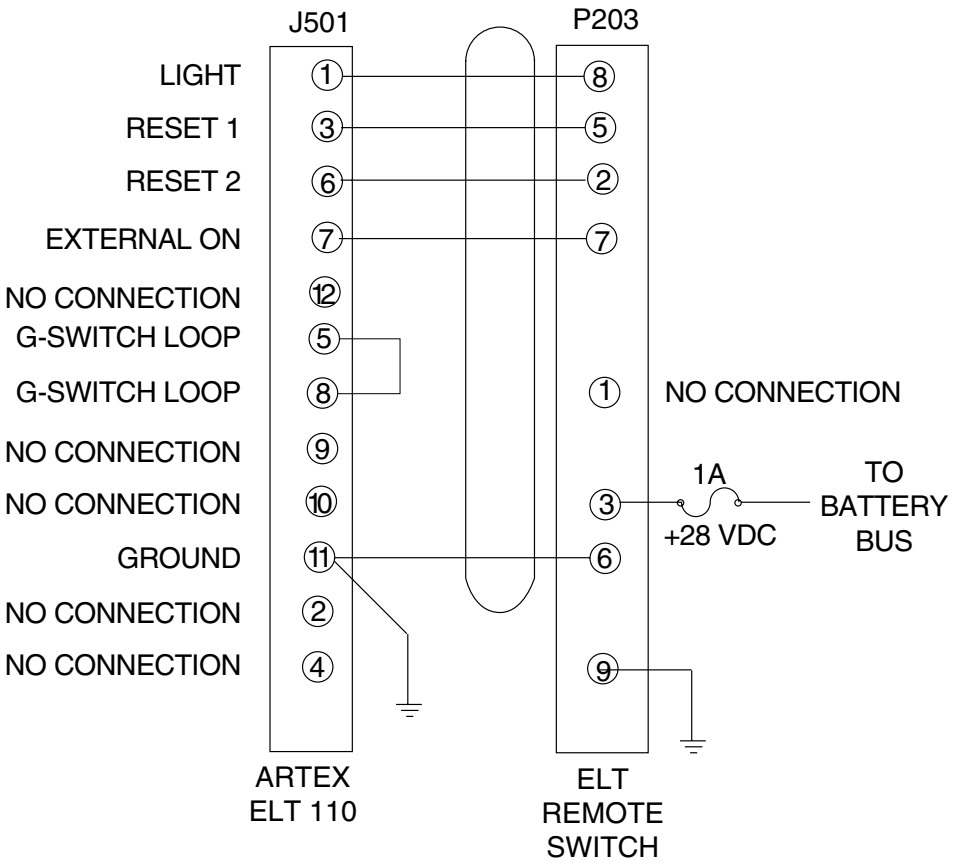
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Narco 910 ELT
Figure 1

Effectivity
Seneca IV

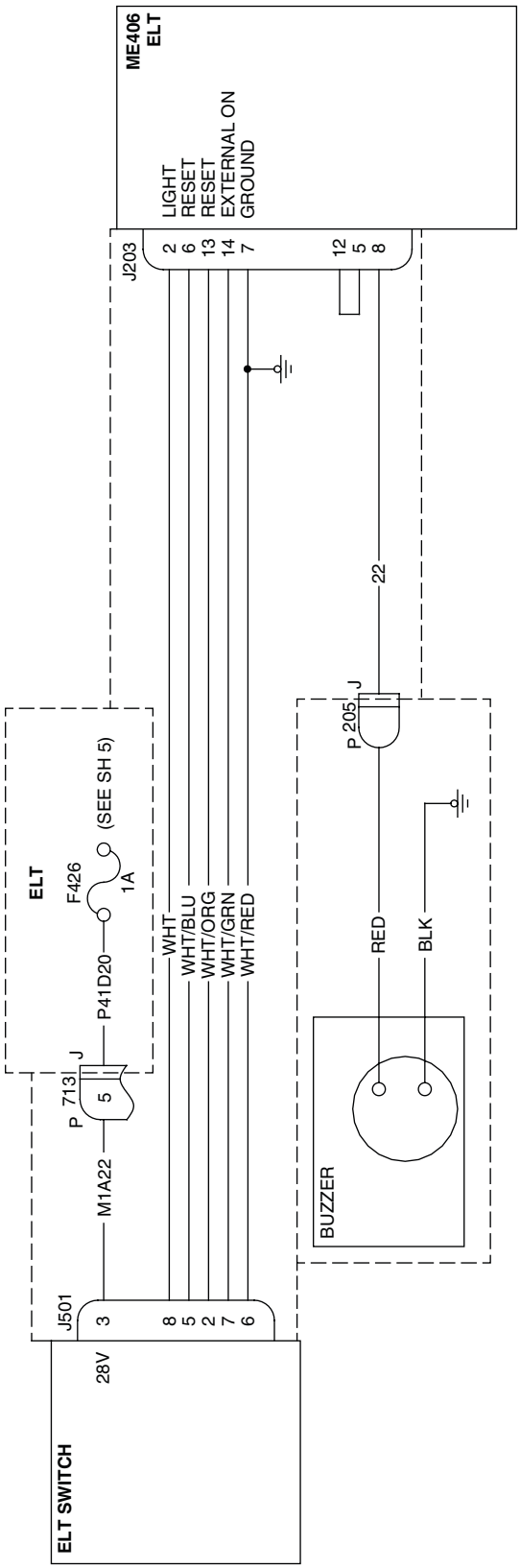
06311 T 2



Effectivity
Seneca IV, and Seneca V
3449001 thru 3449358

Artex ELT 110
Figure 2

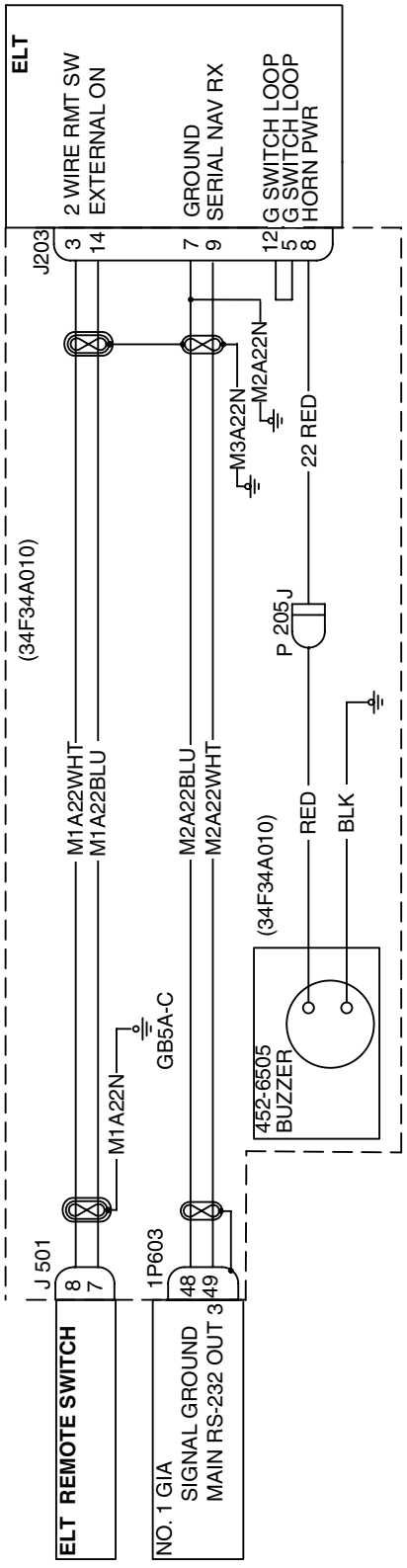
107230 NEW 32.0
107240 NEW 32.0



ME406 ELT
Figure 3

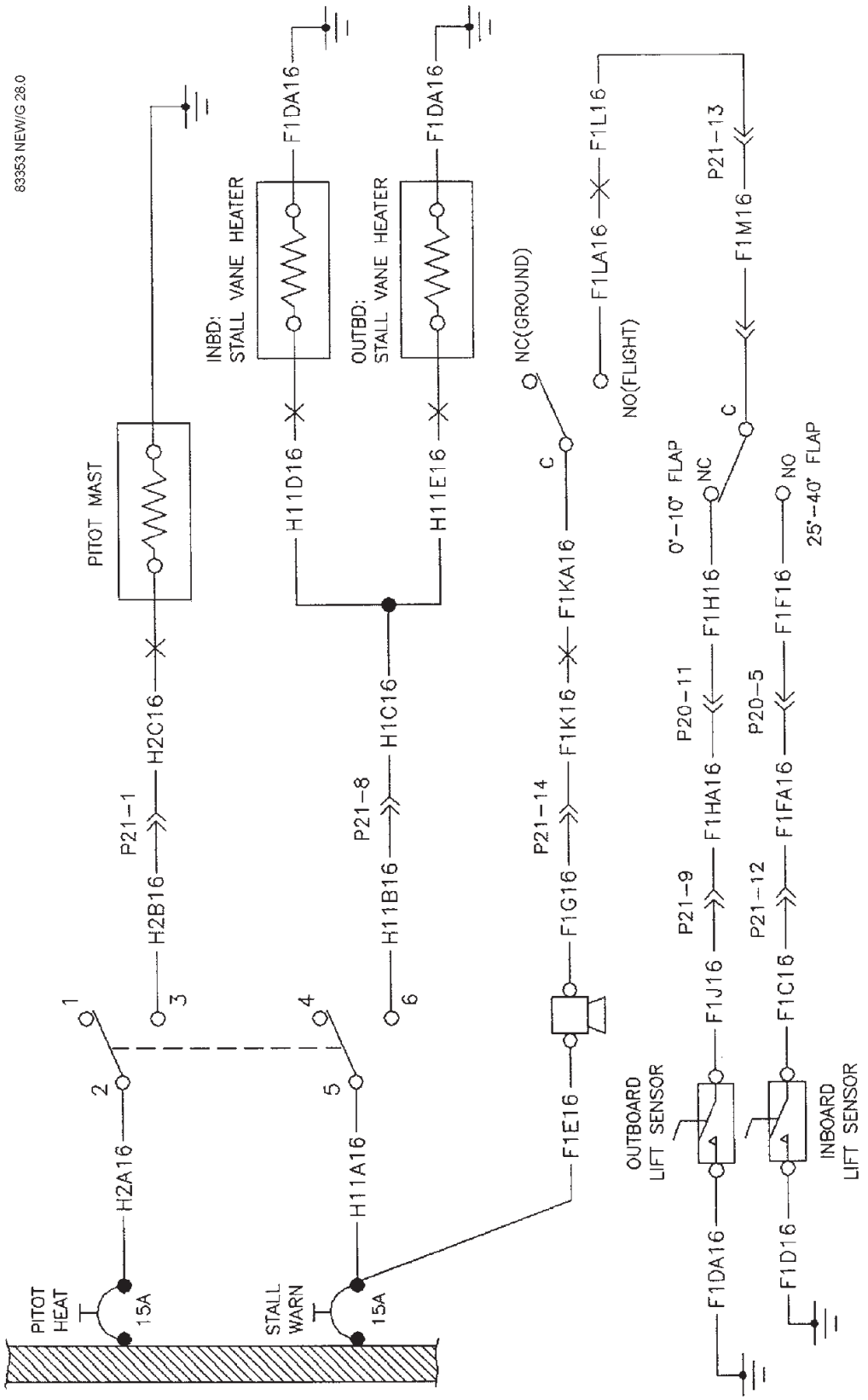
Effectivity
Seneca V
with Garmin G1000

34F24A020 NEW 31.0
34F34A010 NEW 2.0



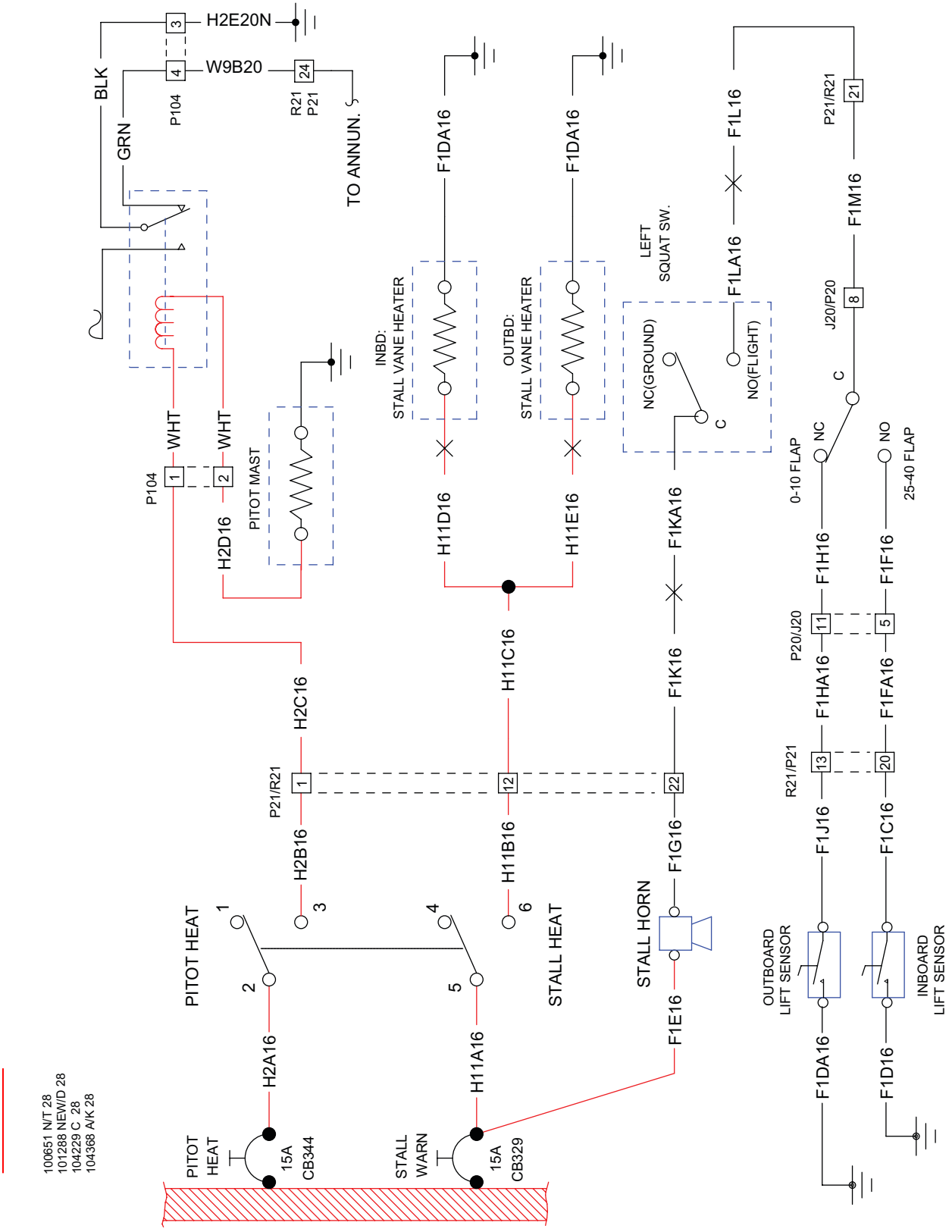
ELT 1000
Figure 4

Effectivity
Seneca V
with Garmin G1000 NXi



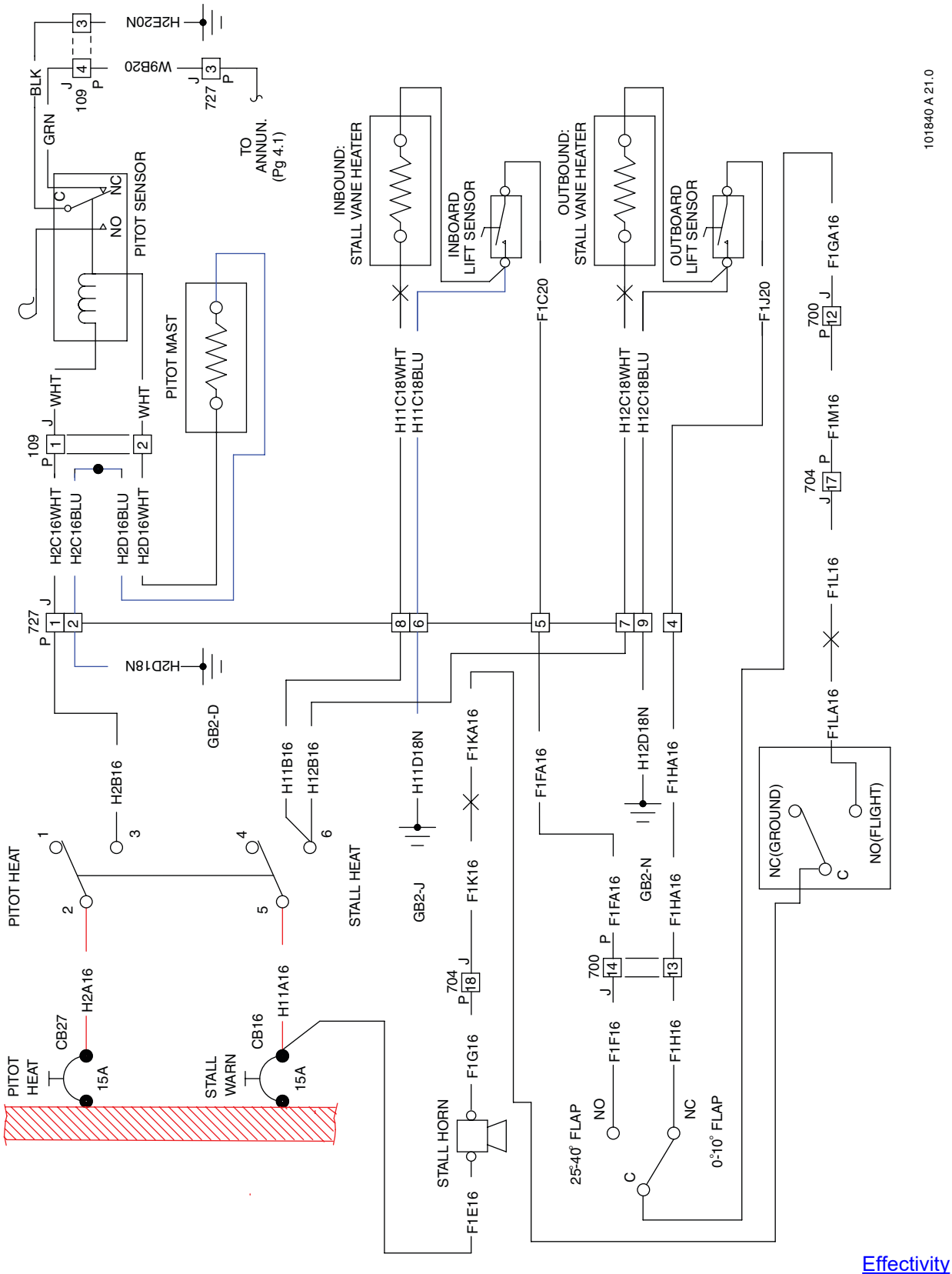
Pitot Heat and Stall Warning
Figure 1 (Sheet 1 of 3)

Effectivity
Seneca IV



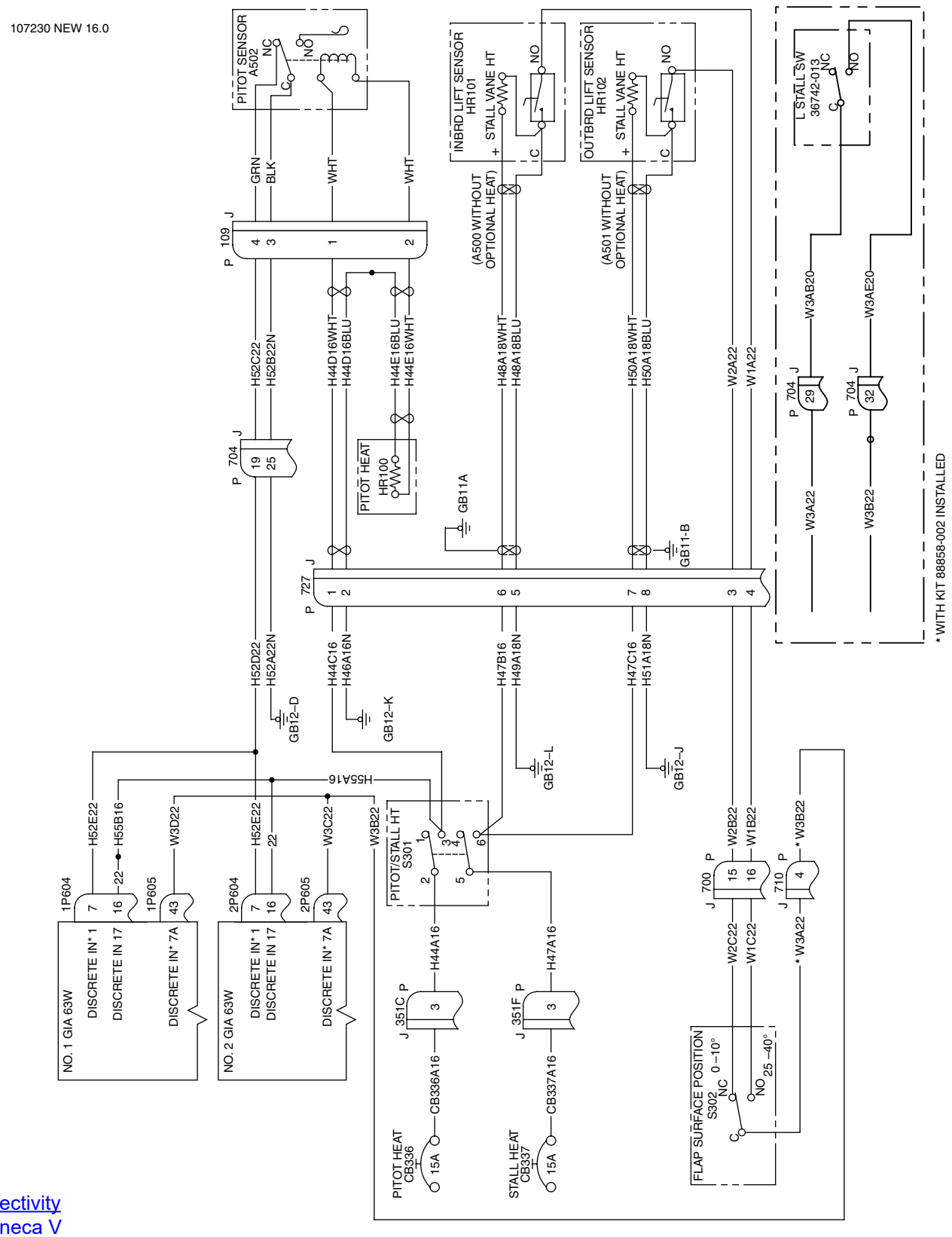
Pitot Heat and Stall Warning
Figure 1 (Sheet 2 of 3)

Effectivity
Seneca V



Pitot Heat and Stall Warning
Figure 1 (Sheet 3 of 3)

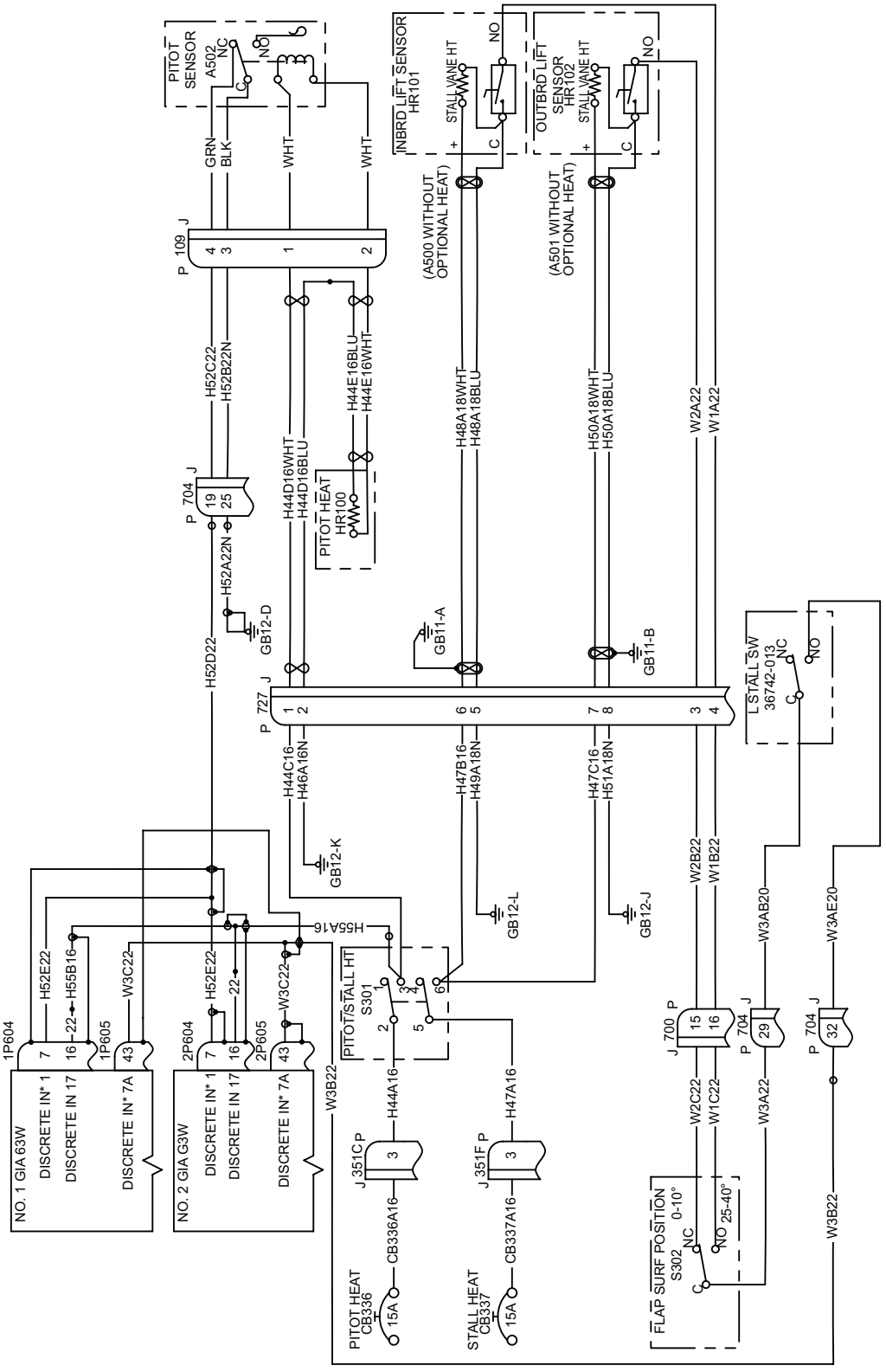
Effectivity
Seneca V
with Avidyne Entegra



Effectivity
Seneca V
S/N 3449467 thru 3449491
with Garmin G1000

Pitot Heat and Stall Warning
Figure 2 (Sheet 1 of 3)

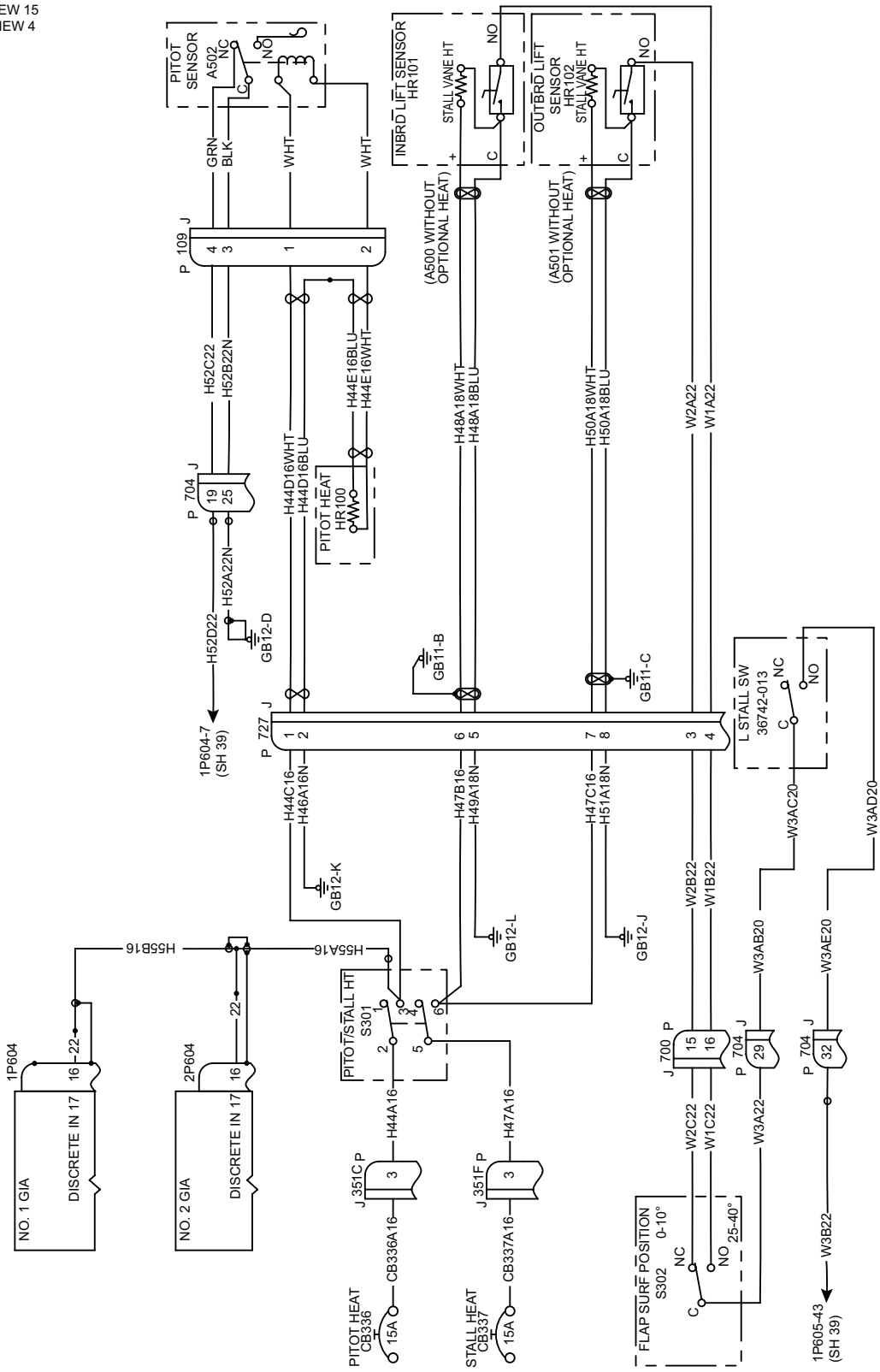
107240 NEW 16.0



Pitot Heat and Stall Warning
Figure 2 (Sheet 2 of 3)

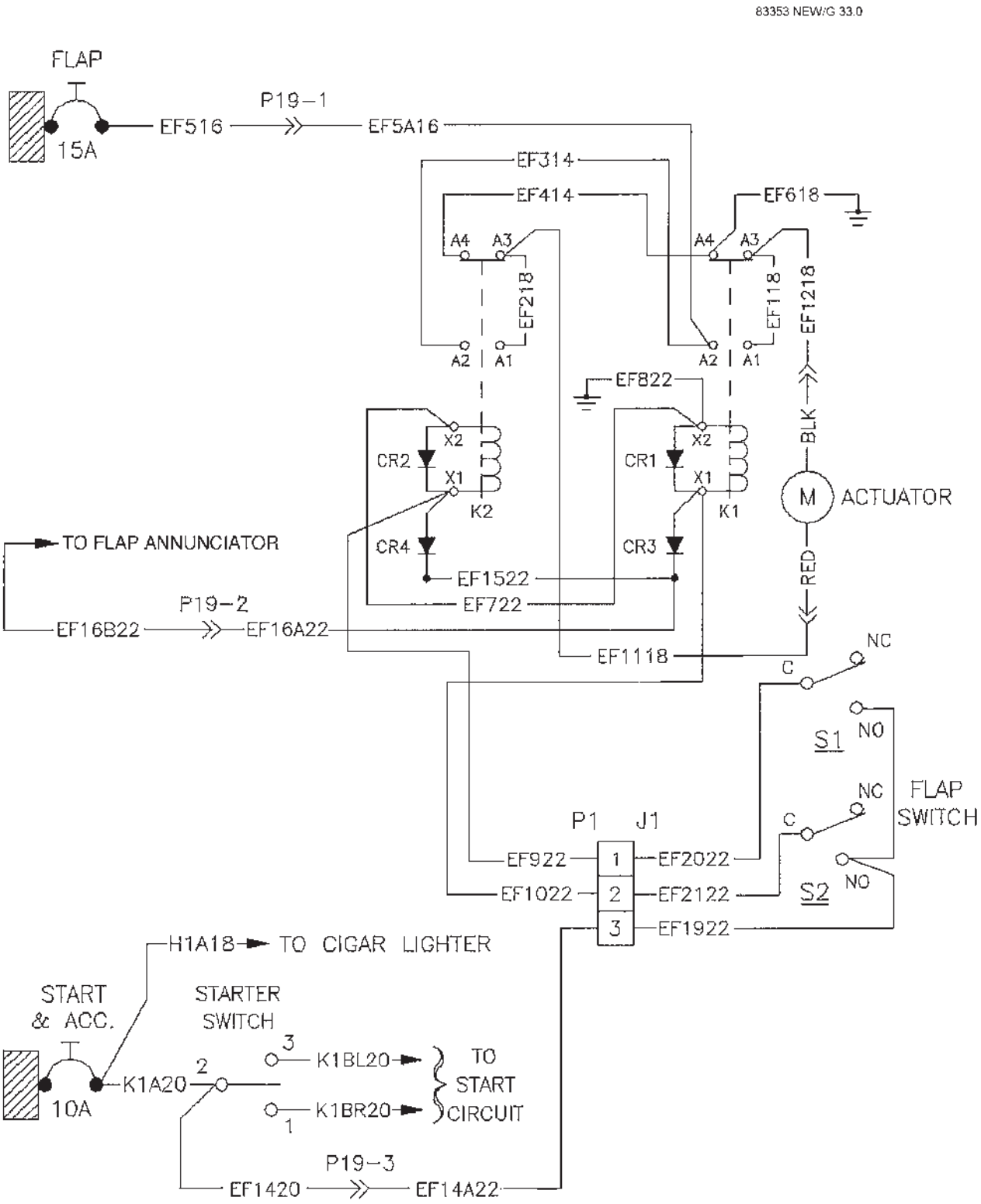
Effectivity
Seneca V
3449459, 3449492-3449508
with Garmin G1000 / GFC 700

34F24A020 NEW 15
34W24A001 NEW 4



Effectivity
Seneca V
with Garmin G1000 NXi

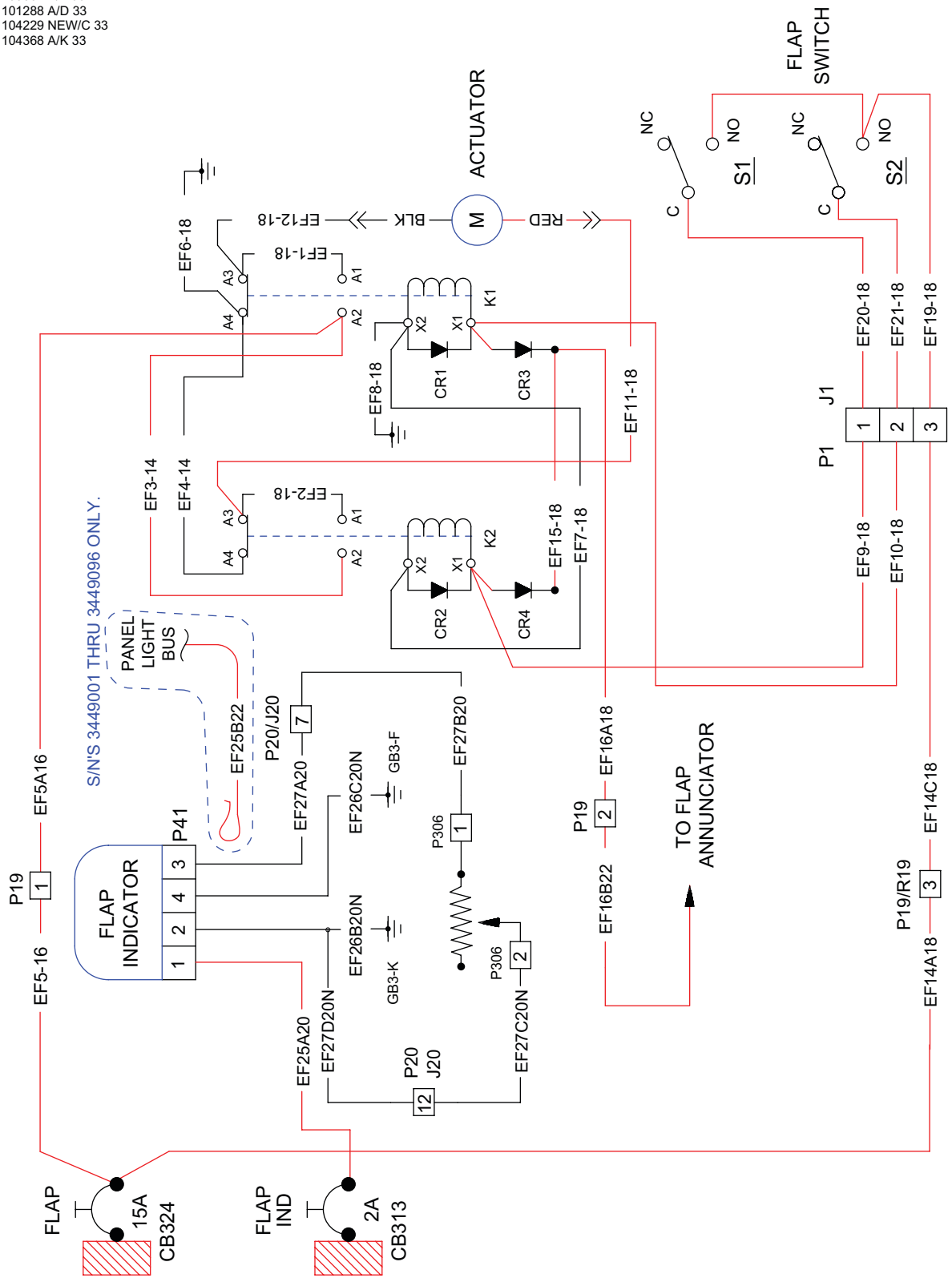
Pitot Heat and Stall Warning
Figure 2 (Sheet 3 of 3)



Electric Flaps
Figure 1 (Sheet 1 of 3)

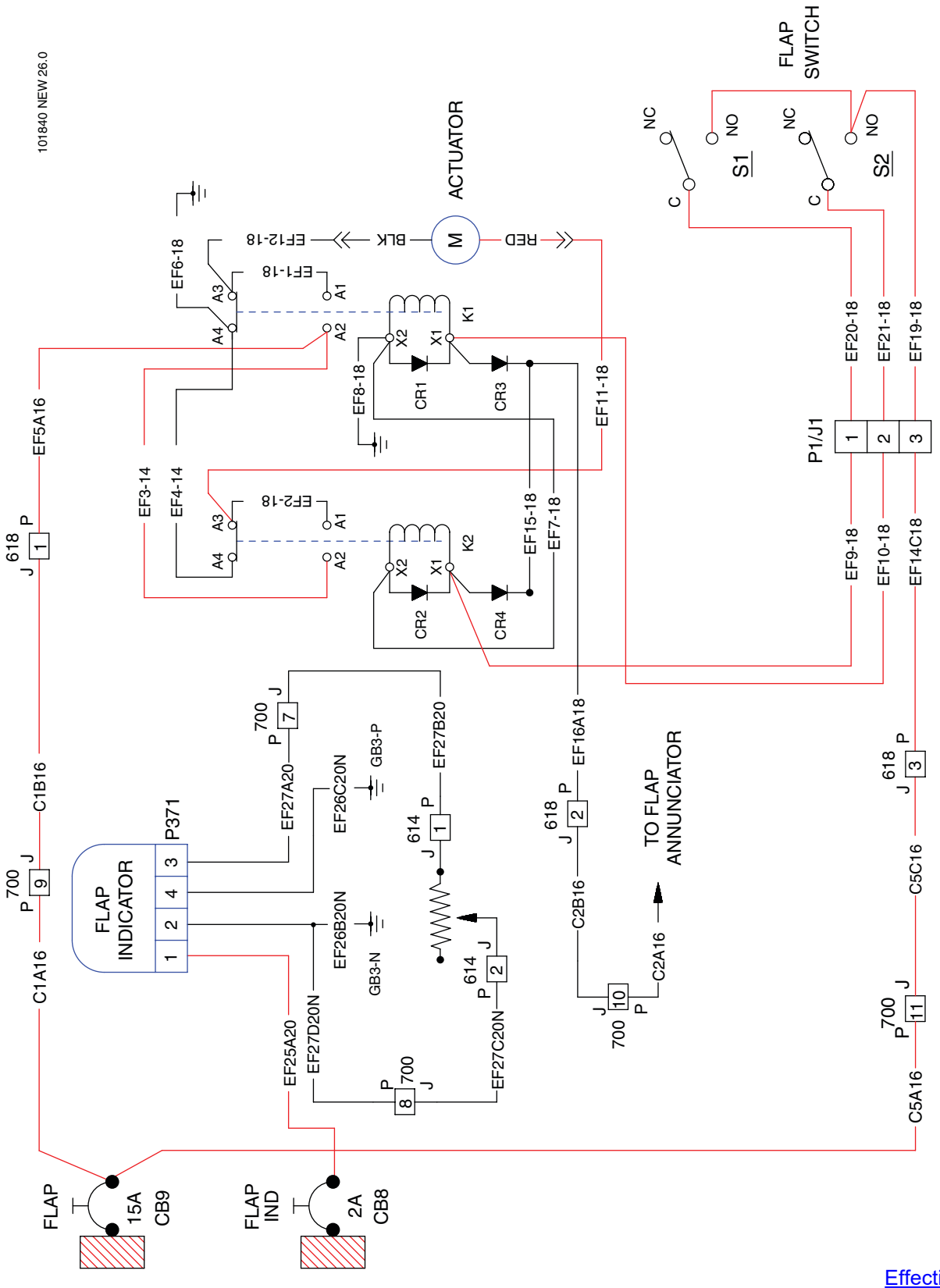
Effectivity
Seneca IV

100651 P/T 33
101288 A/D 33
104229 NEW/C 33
104368 A/K 33



Effectivity
Seneca V

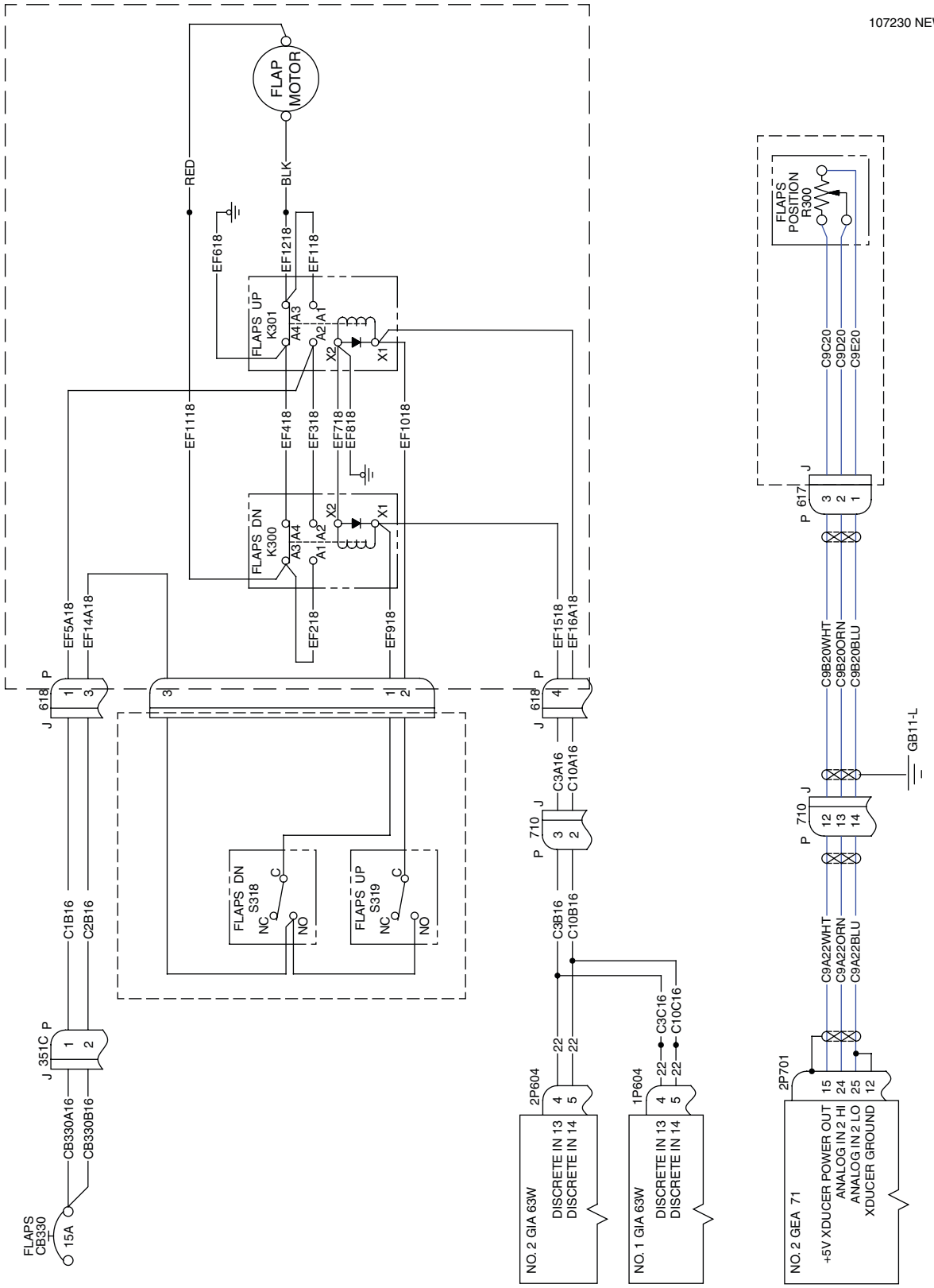
Electric Flaps
Figure 1 (Sheet 2 of 3)



Electric Flaps
Figure 1 (Sheet 3 of 3)

Effectivity
Seneca V
with Avidyne Entegra

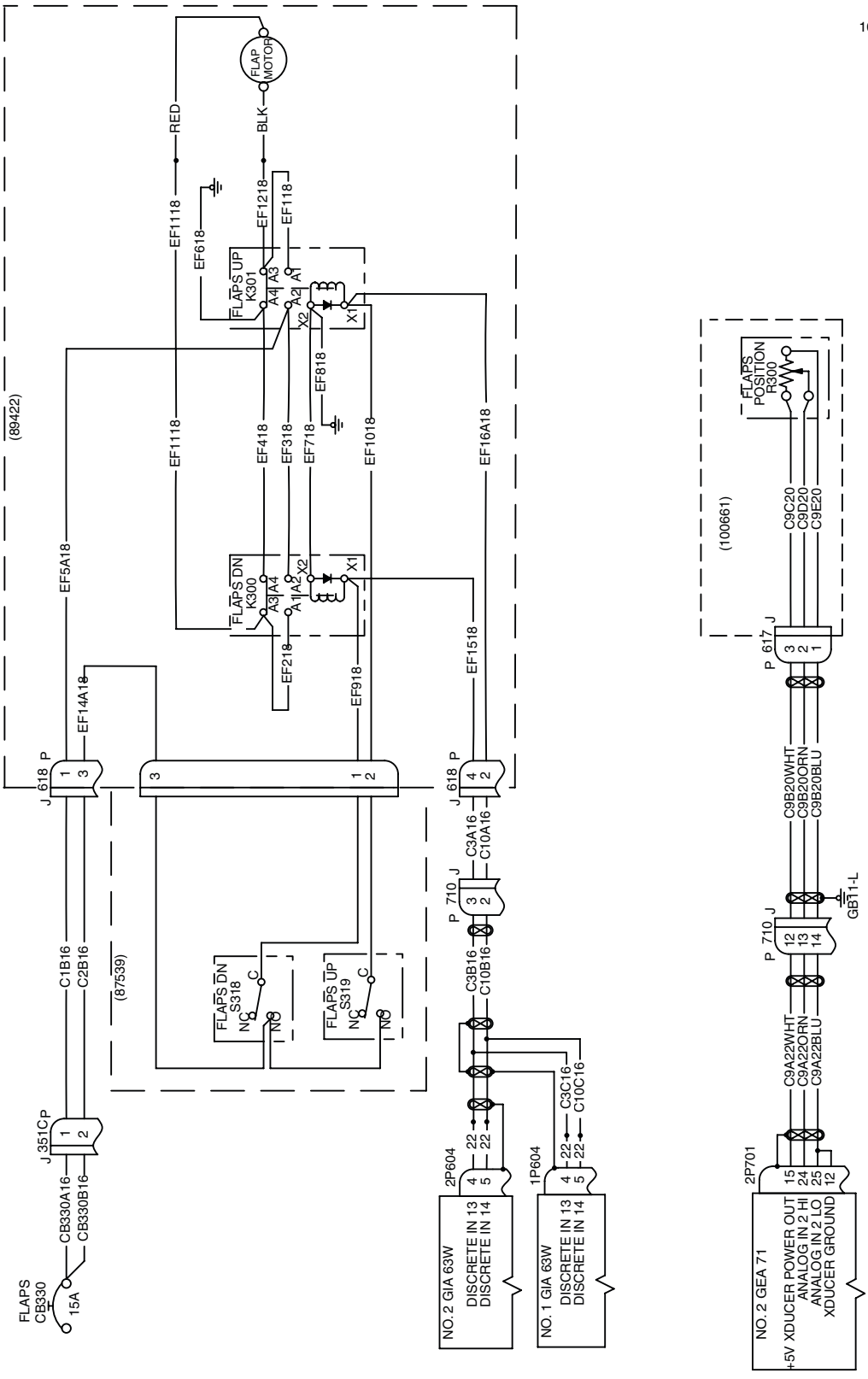
107230 NEW 17.0



Effectivity
Seneca V
with Garmin G1000

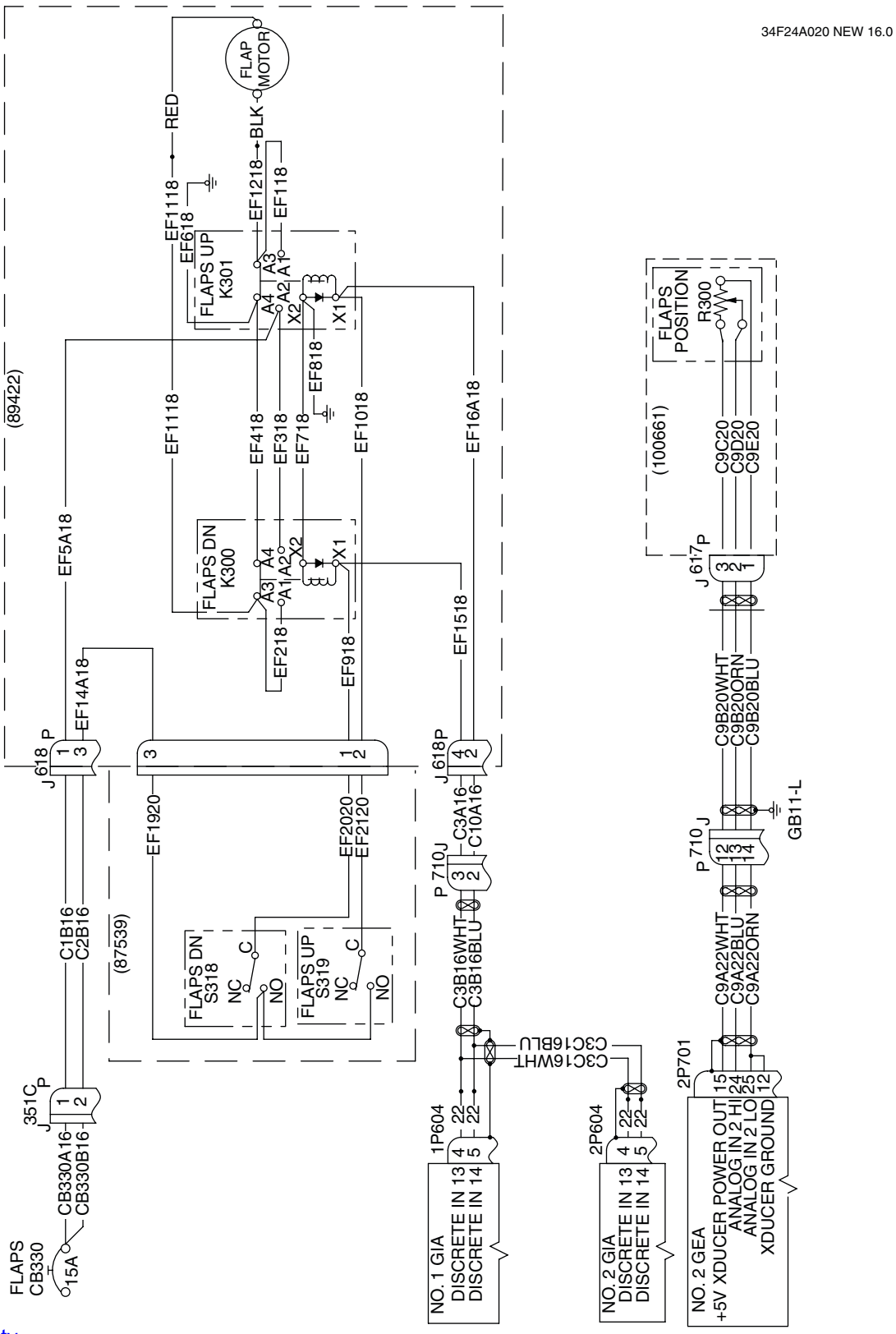
Electric Flaps
Figure 2 (Sheet 1 of 3)

107240 NEW 17.0



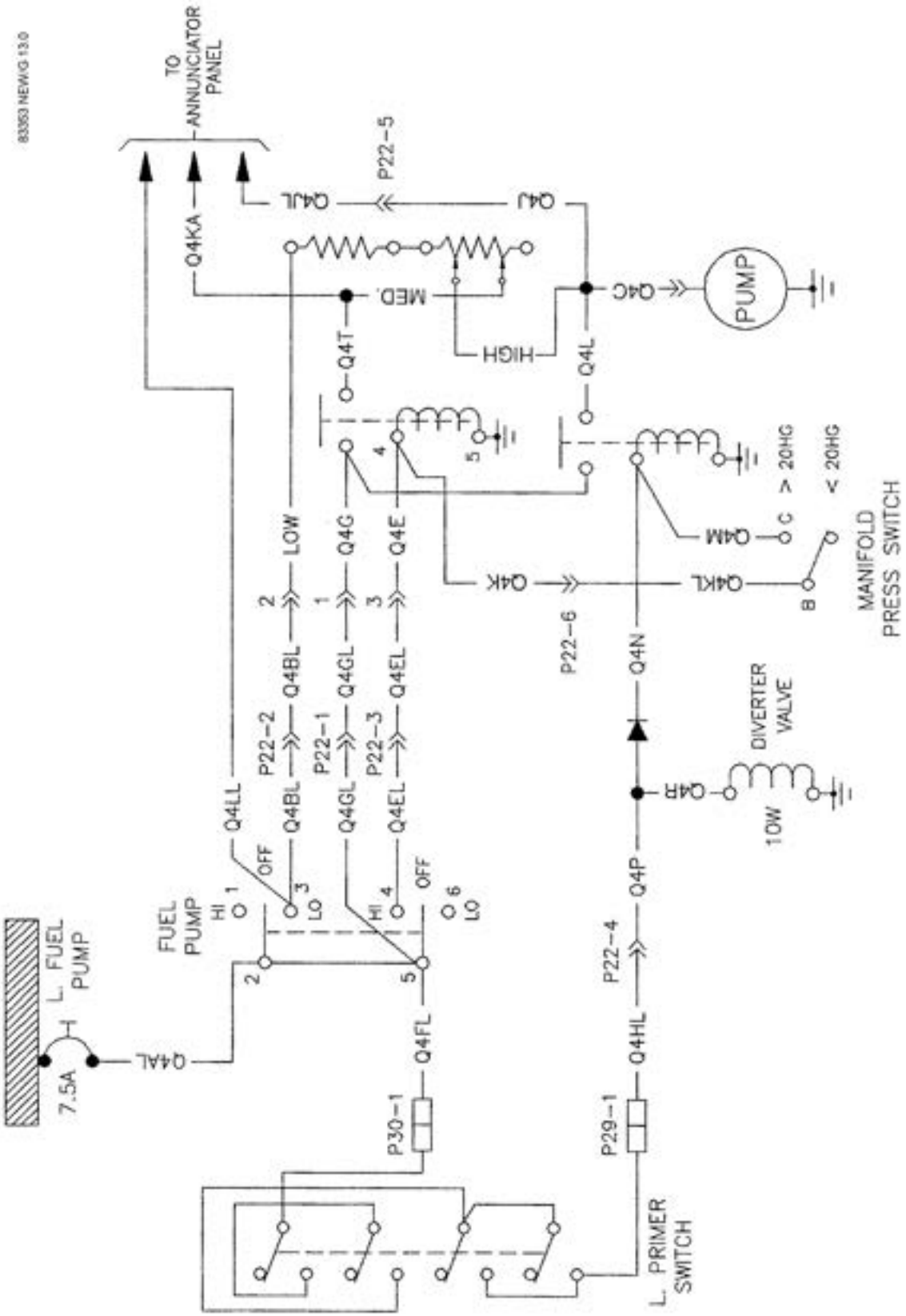
Electric Flaps
Figure 2 (Sheet 2 of 3)

Effectivity
Seneca V
with Garmin G1000 / GFC 700



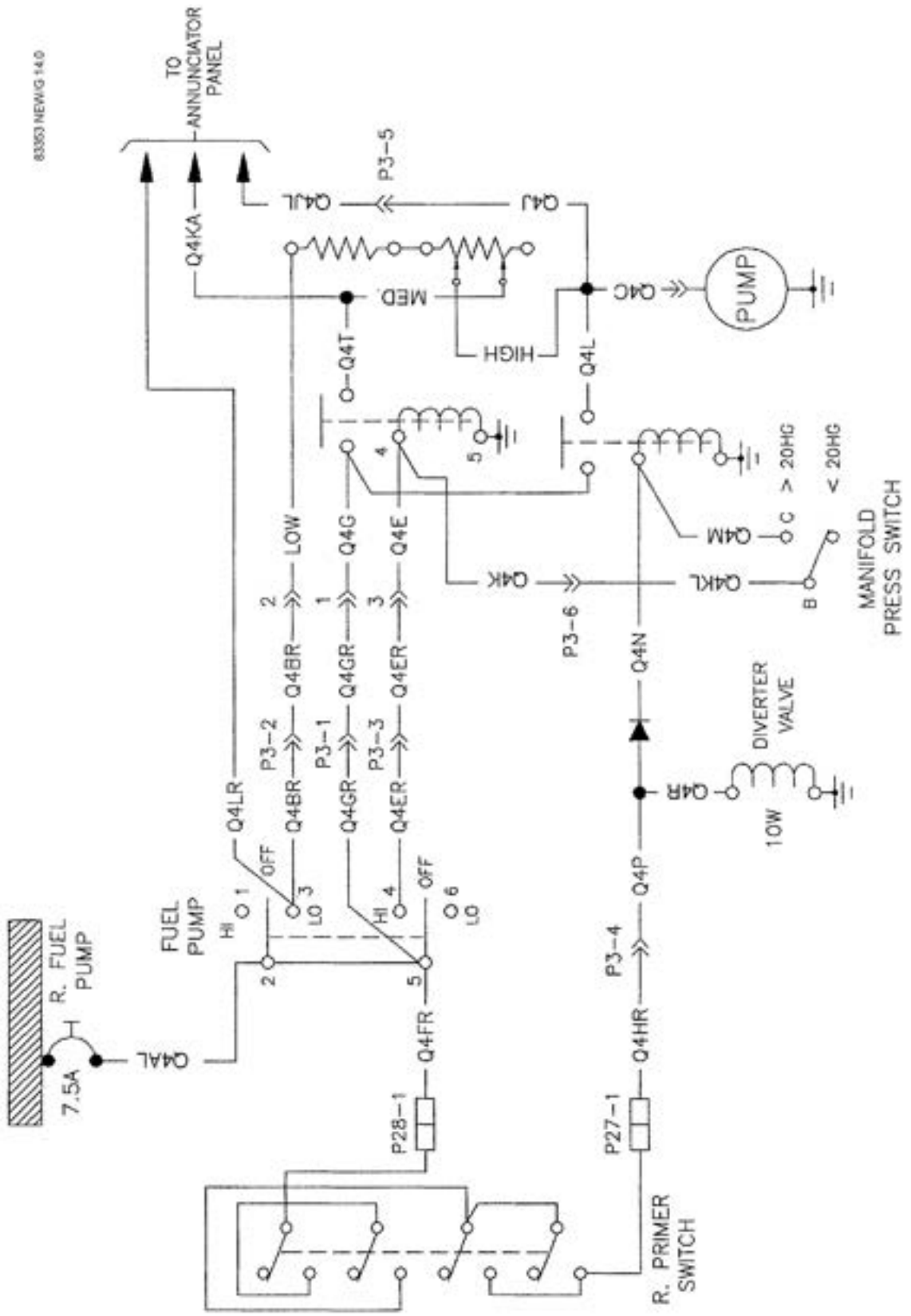
Effectivity
Seneca V
with Garmin G1000 NXi

Electric Flaps
Figure 2 (Sheet 3 of 3)



Fuel Pumps
Figure 1 (Sheet 1 of 7)

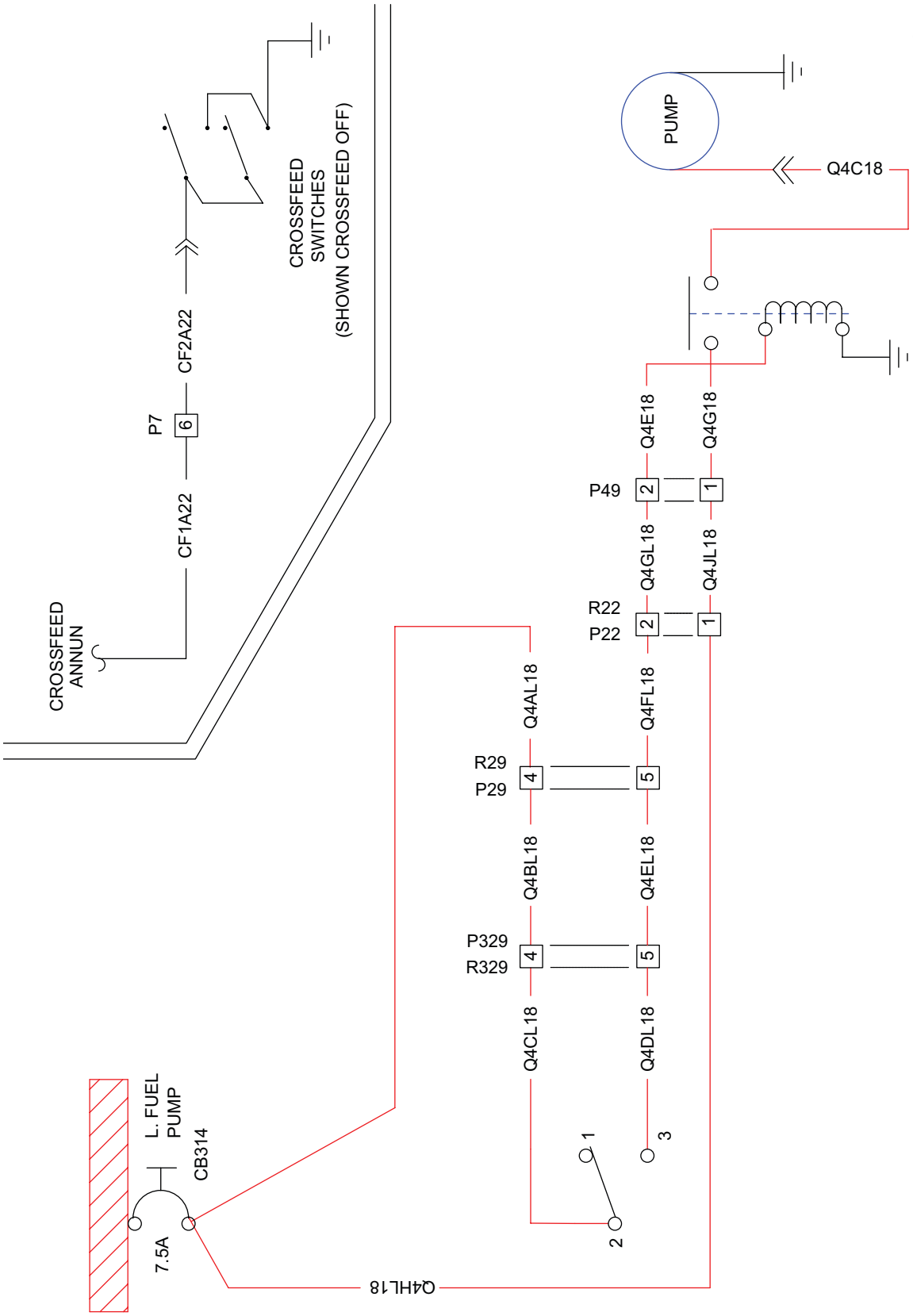
Effectivity
Left - Seneca IV



Fuel Pumps
Figure 1 (Sheet 2 of 7)

Effectivity
Seneca IV - Right

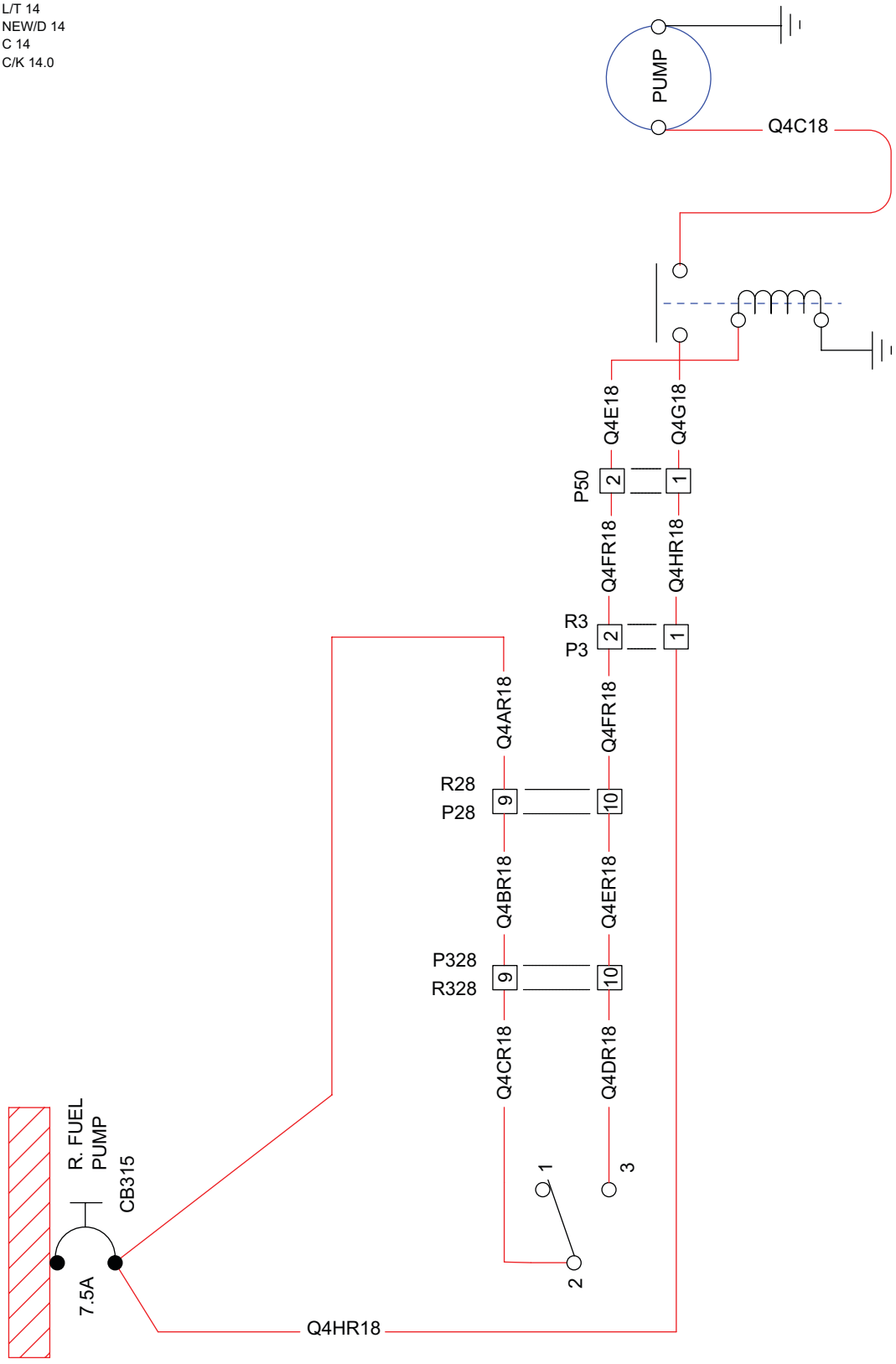
100651 N/T 13
101288 NEW/D 13
104229 C 13
104368 A/K 13.0



Fuel Pumps
Figure 1 (Sheet 3 of 7)

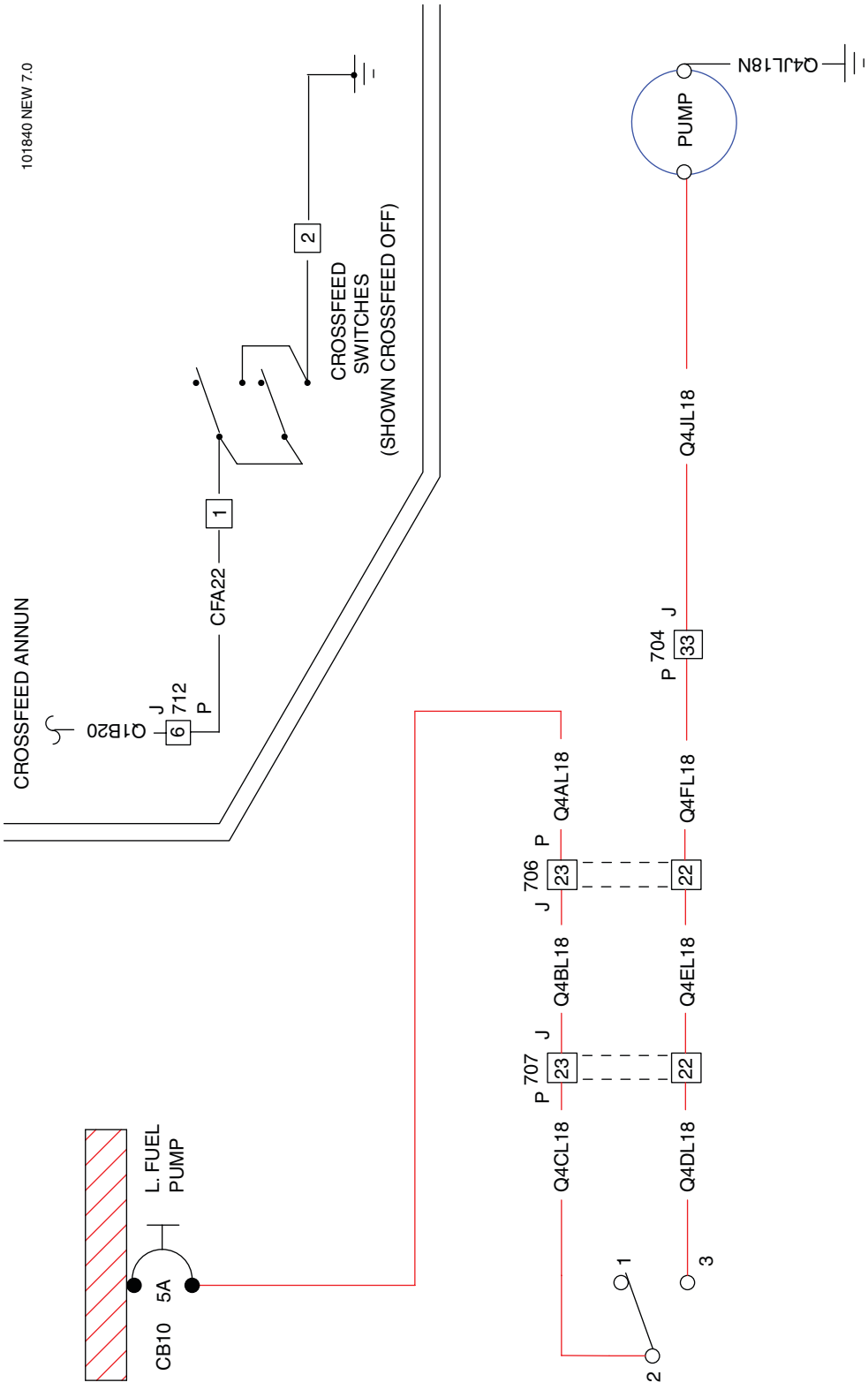
Effectivity
Left - Seneca V

100651 L/T 14
101288 NEW/D 14
104229 C 14
104368 C/K 14.0



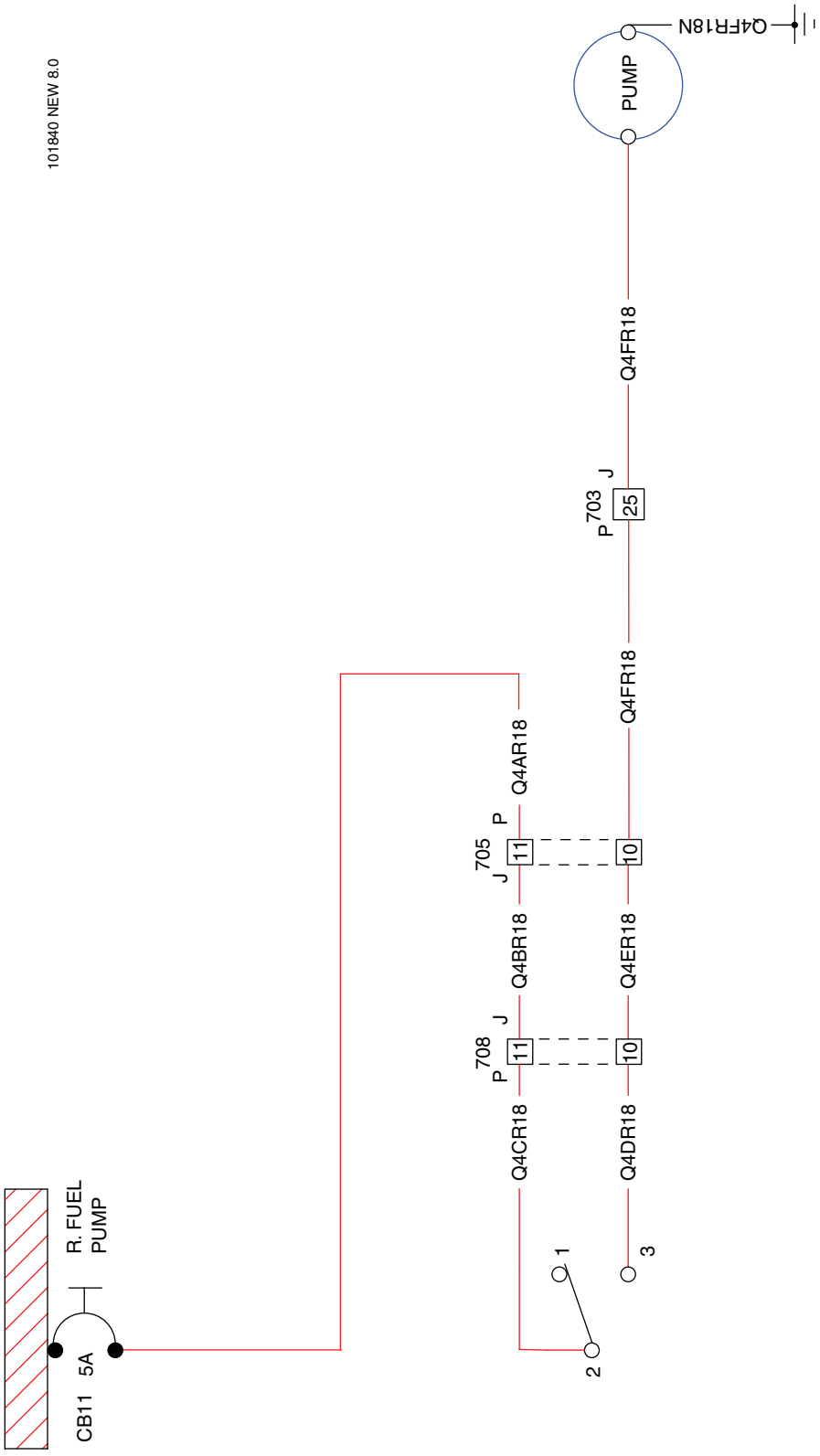
Effectivity
Seneca V - Right

Fuel Pumps
Figure 1 (Sheet 4 of 7)



Fuel Pumps
Figure 1 (Sheet 5 of 7)

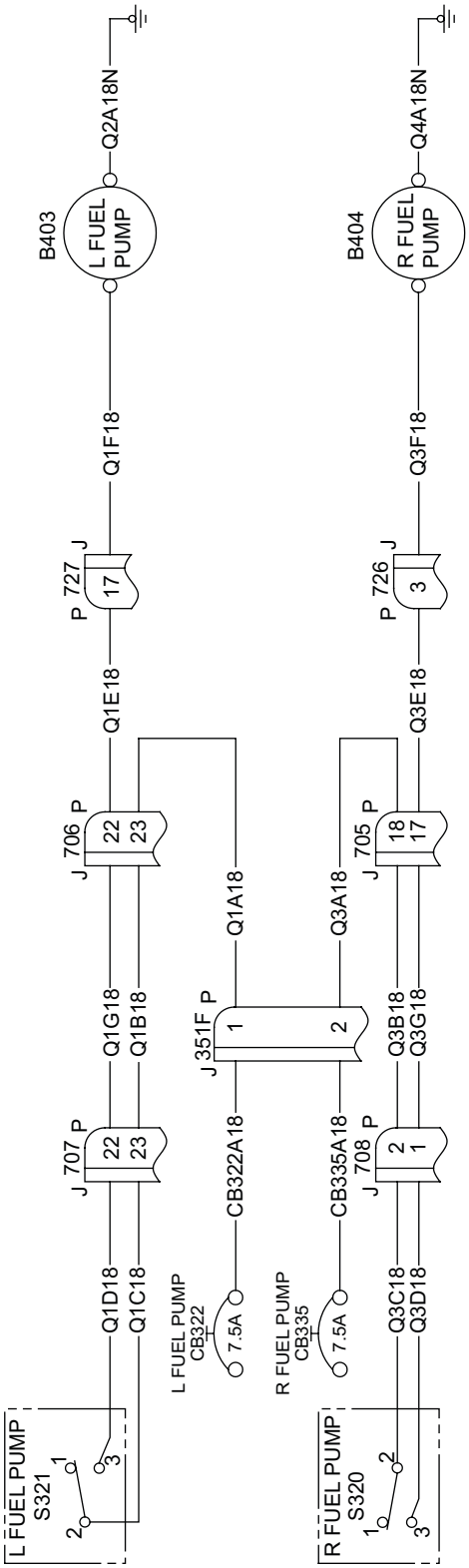
Effectivity
Left - Seneca V
with Avidyne Entegra



Effectivity
Seneca V - Right
with Avidyne Entegra

Fuel Pumps
Figure 1 (Sheet 6 of 7)

107230 NEW 15
107240 NEW 15
34F24A020 NEW 14
34W24A001 NEW 3, 7

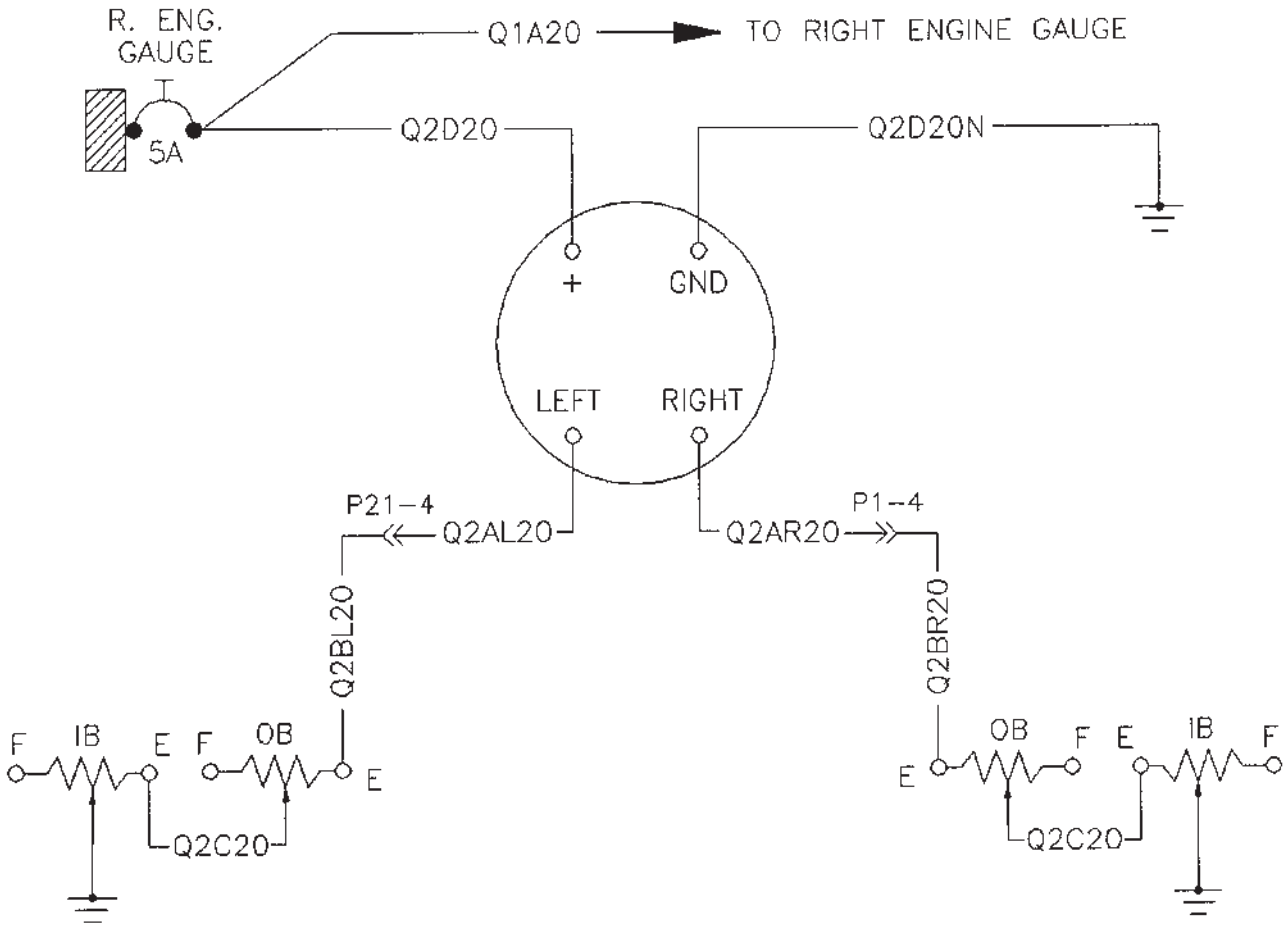


Fuel Pumps
Figure 1 (Sheet 7 of 7)

Effectivity
Seneca V
with Garmin G1000 / G1000 NXi

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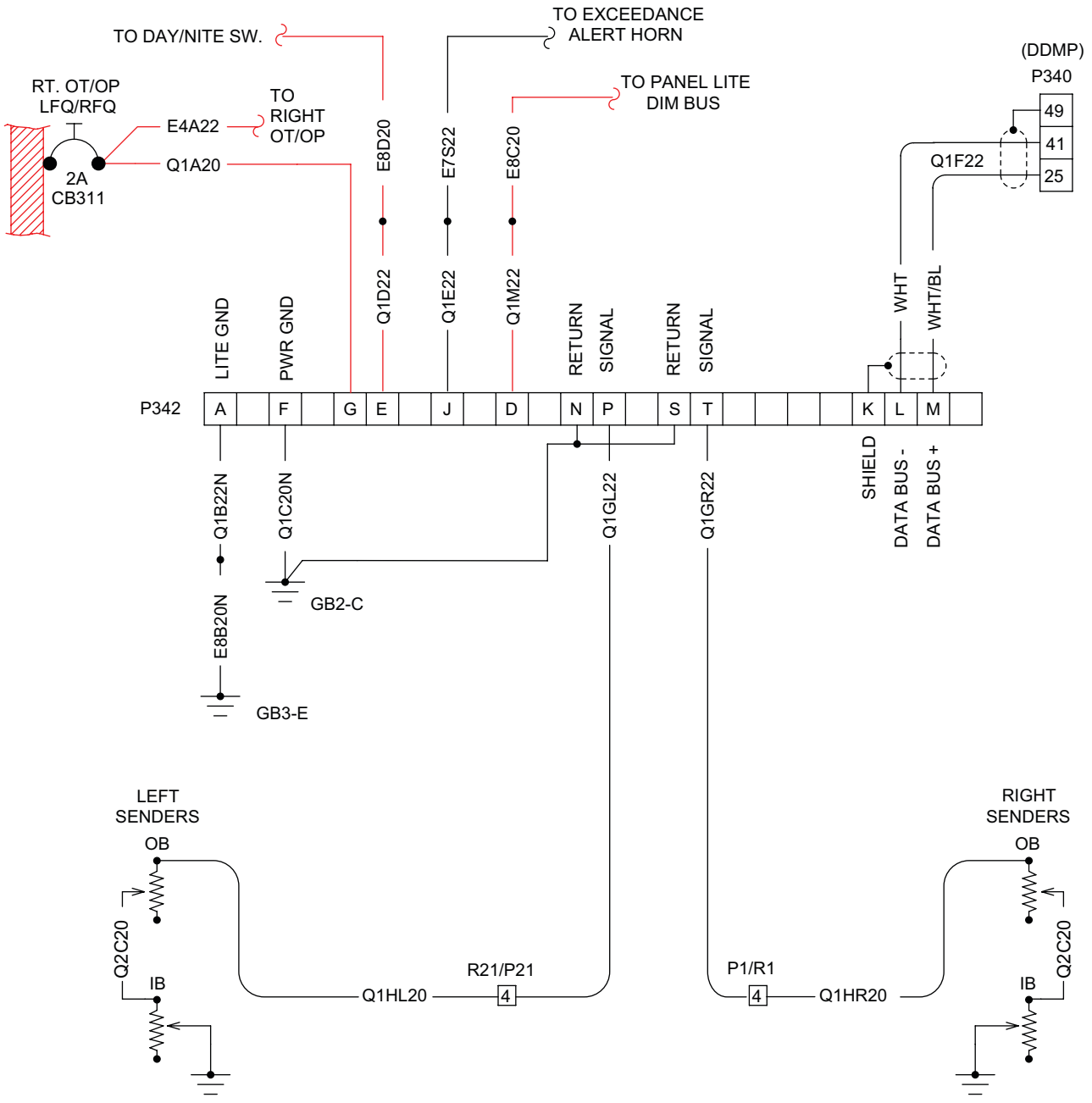
83353 NEW/G 7.0



Fuel Quantity
Figure 1 (Sheet 1 of 2)

Effectivity
Seneca IV

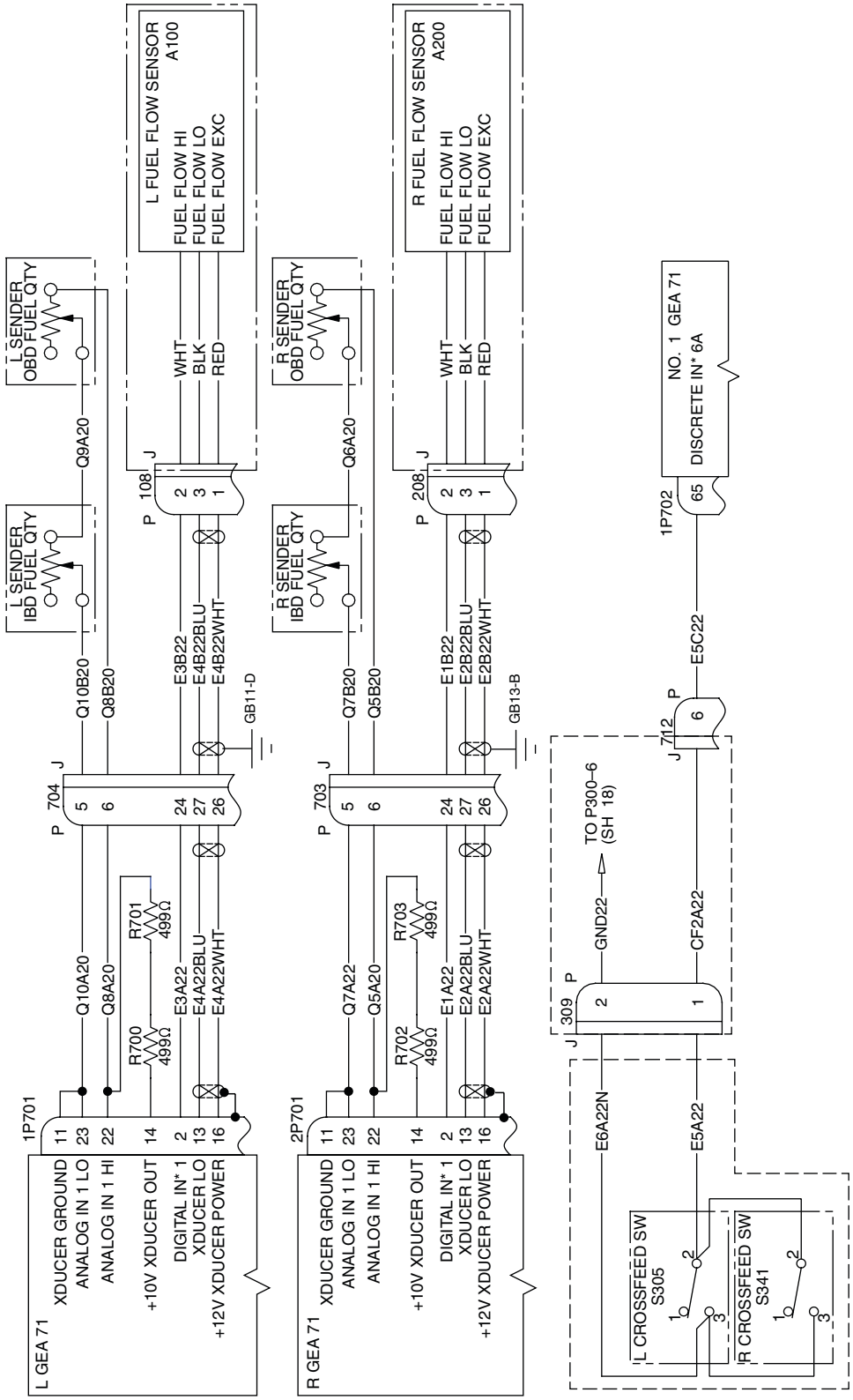
100651 B/K 7
101288 NEW/D 7
104229 NEW/C 7
104368 NEW/K 7.0



Effectivity
Seneca V
with Standard, Meggitt EFIS
or Garmin G600

Fuel Quantity
Figure 1 (Sheet 2 of 2)

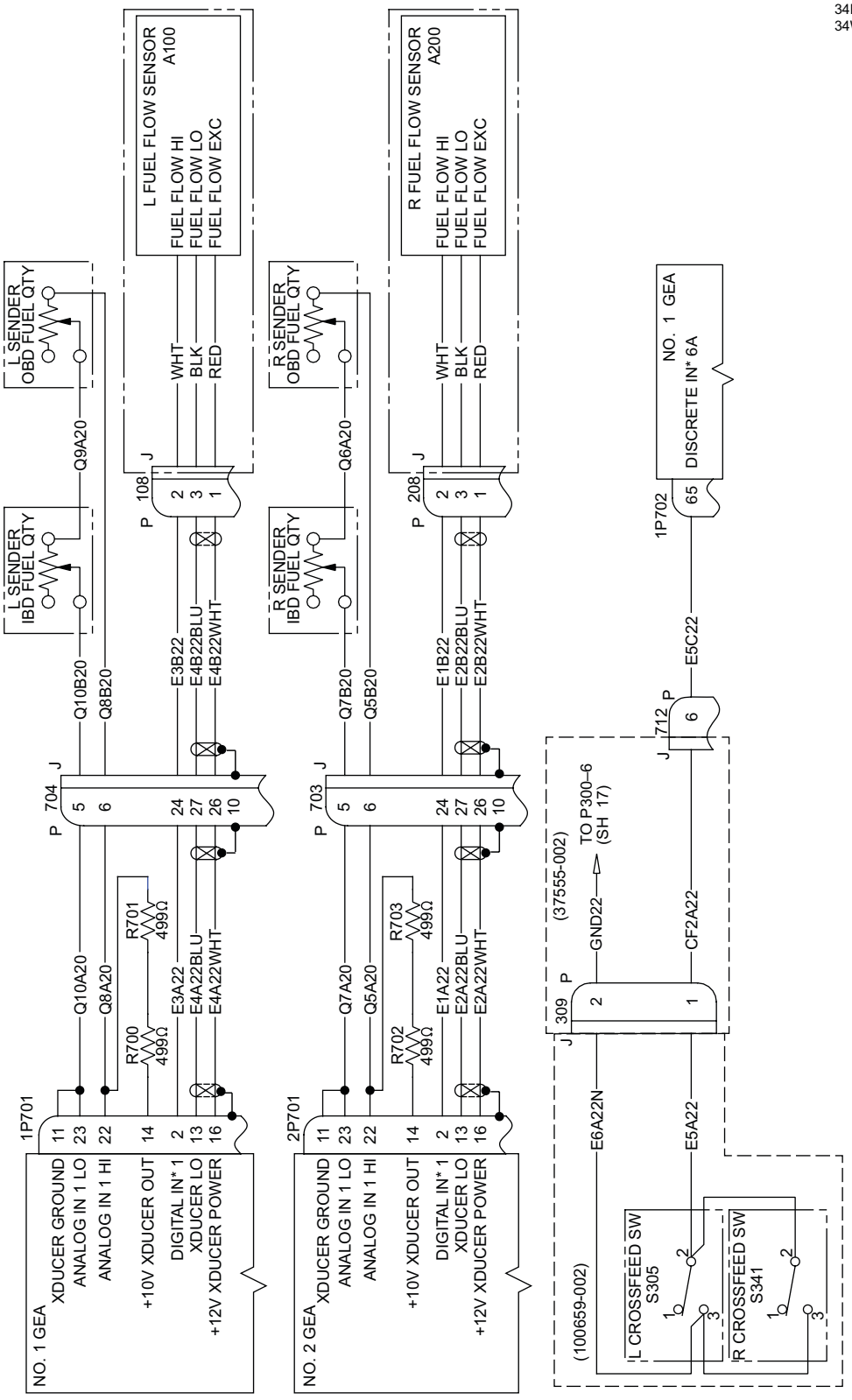
107230 NEW 15.0
107240 NEW 15.0



Fuel Indicating
Figure 2 (Sheet 1 of 2)

Effectivity
Seneca V
with Garmin G1000

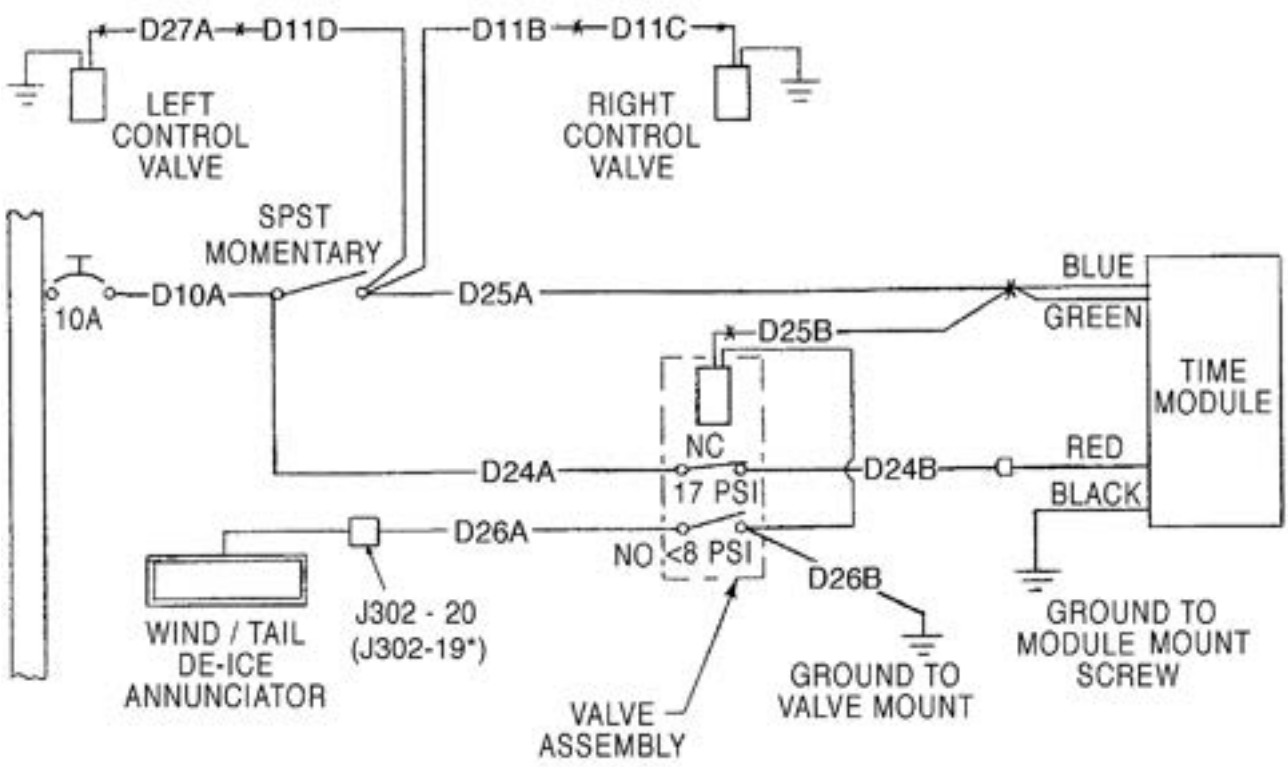
34F24A020 NEW 14
34W24A001 NEW 6, 9



Effectivity
Seneca V
with Garmin G1000 NXi

Fuel Indicating
Figure 2 (Sheet 2 of 2)

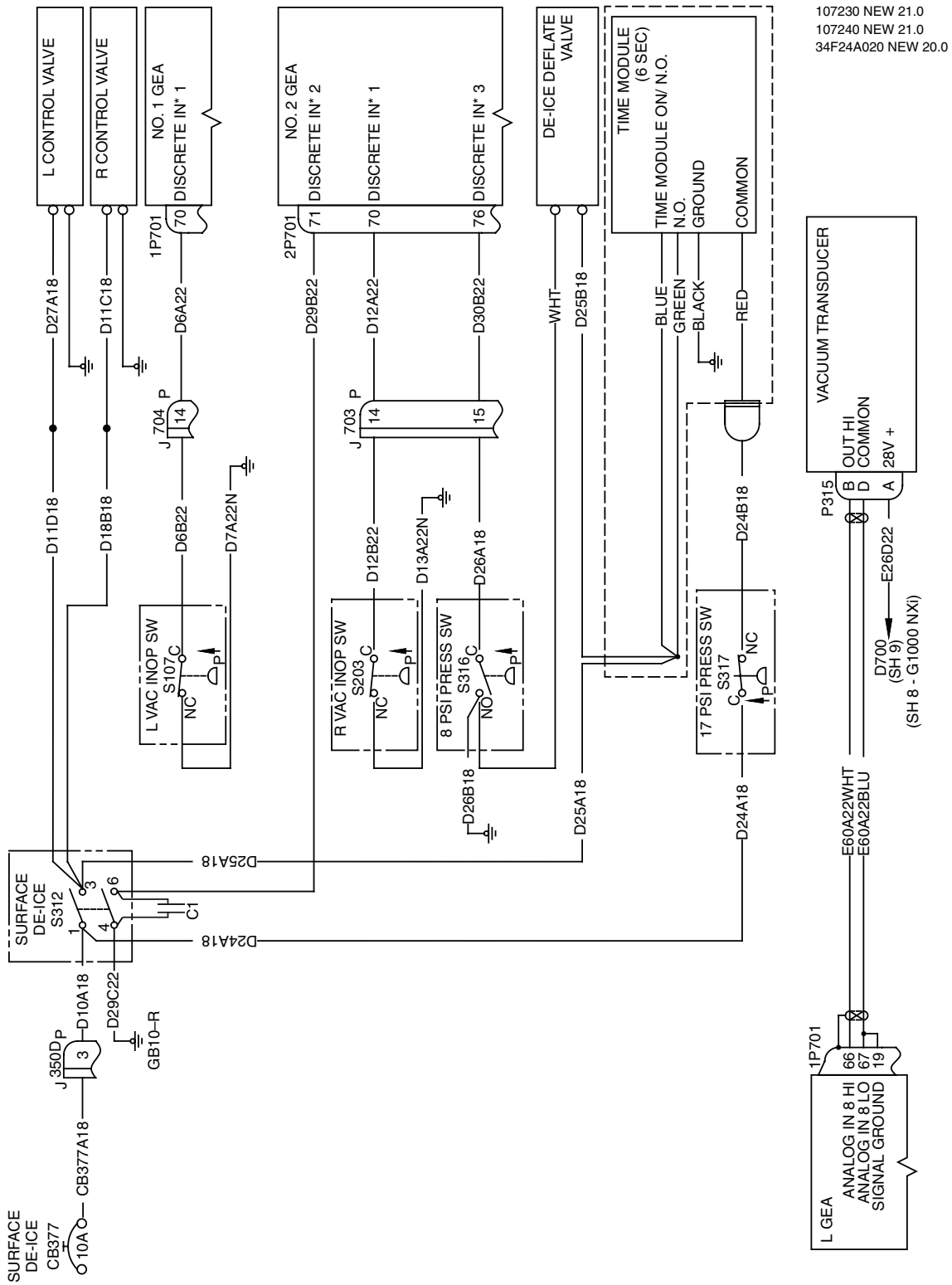
85186 WAD 3
101772 D 2



* WITH AVIDYNE ENTEGRA

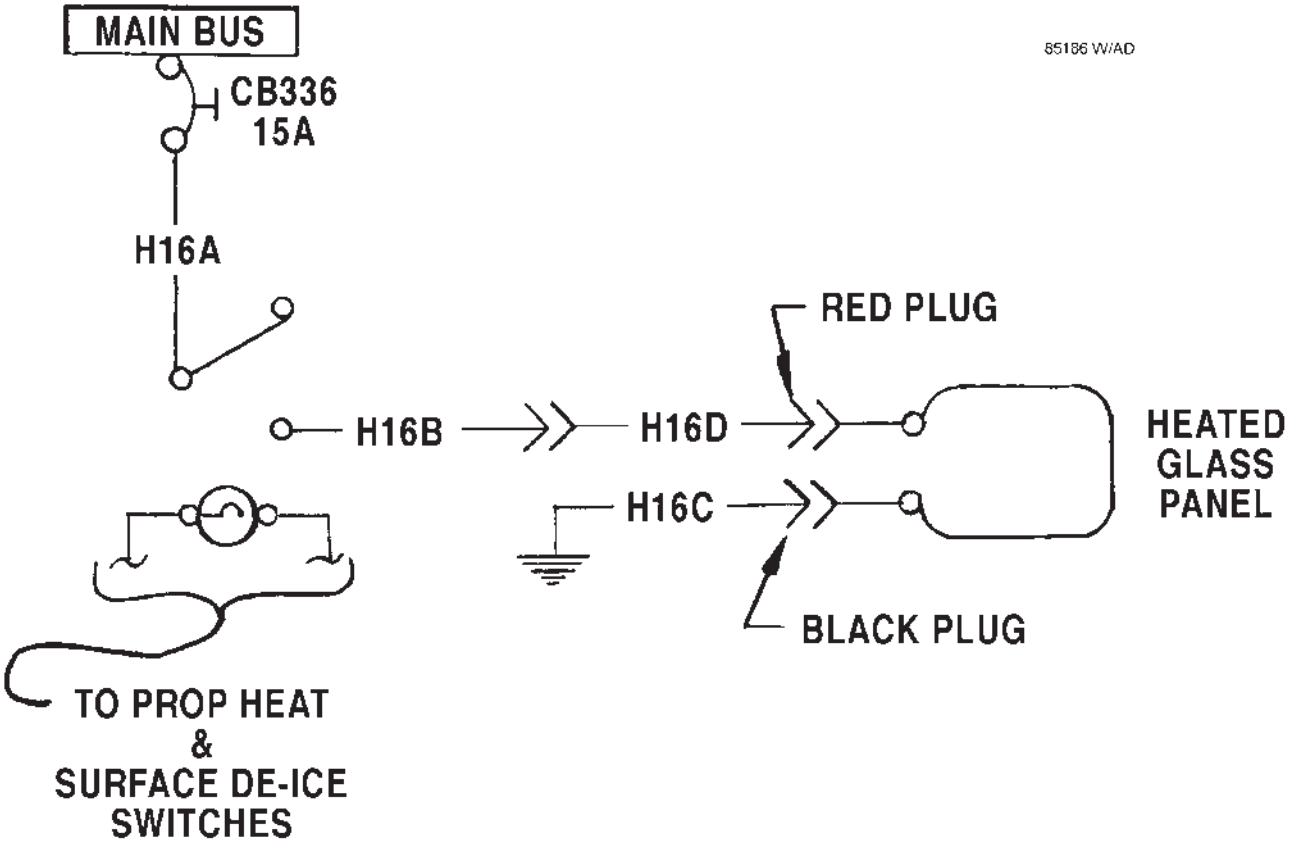
NOTE - 18 GAUGE WIRE THROUGHOUT

Surface Deice
Figure 1 (Sheet 1 of 2)



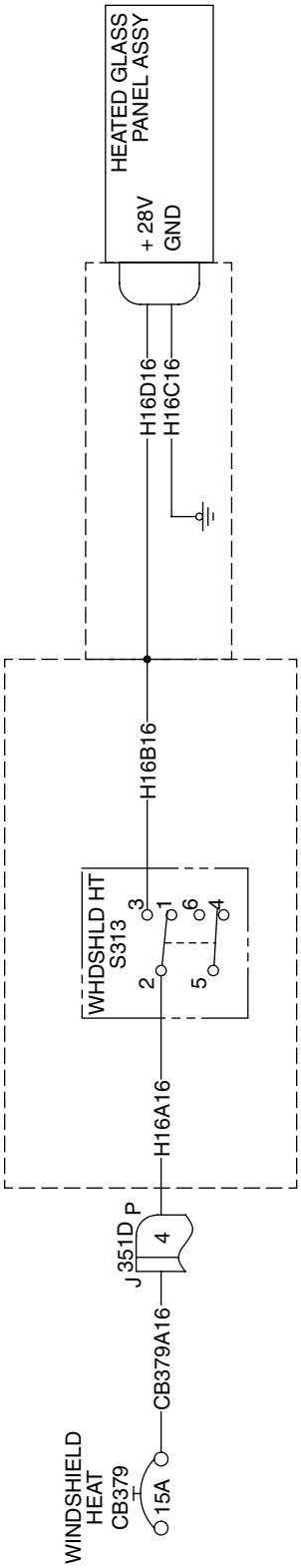
Effectivity
Seneca V
with Garmin G1000 / G1000 NXi

Surface Deice
Figure 1 (Sheet 2 of 2)



Windshield Deice
Figure 1 (Sheet 1 of 2)

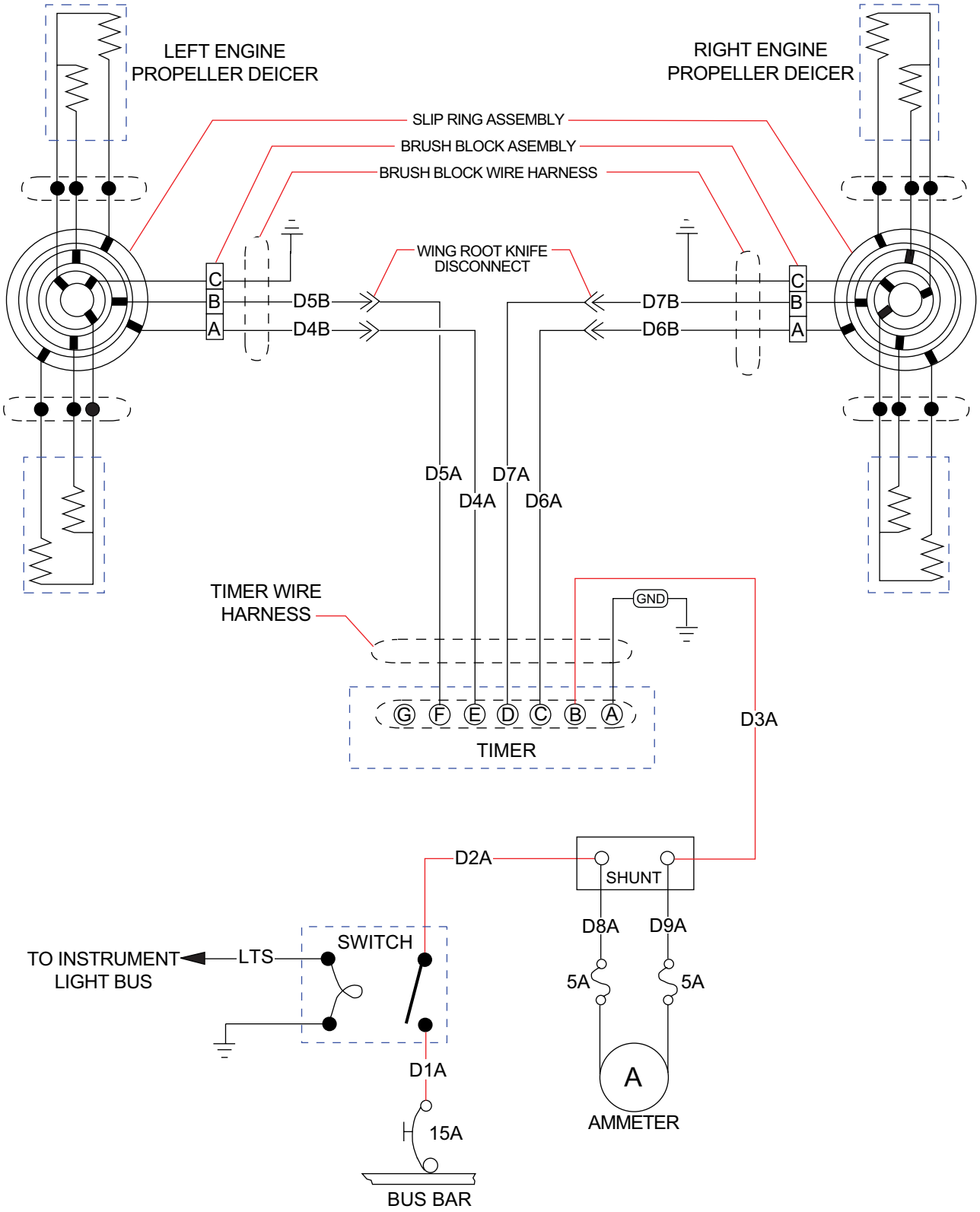
107230 NEW 22.0
107240 NEW 22.0
34F24A020 NEW 21.0



Effectivity
Seneca V
with Garmin G1000 / G1000 NXi

Windshield Deice
Figure 1 (Sheet 2 of 2)

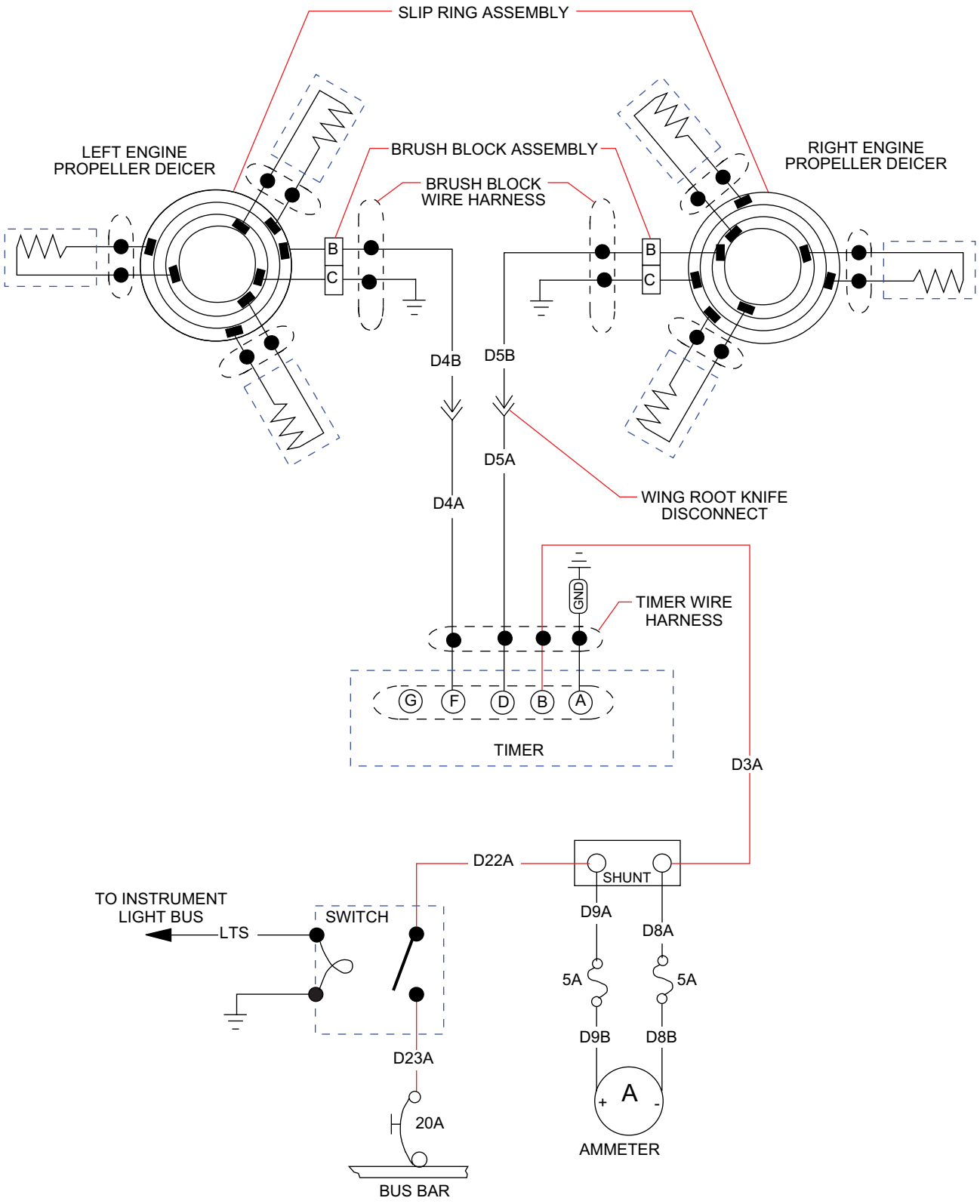
89426 E/J 2



Propeller Deice
Figure 1 (Sheet 1 of 4)

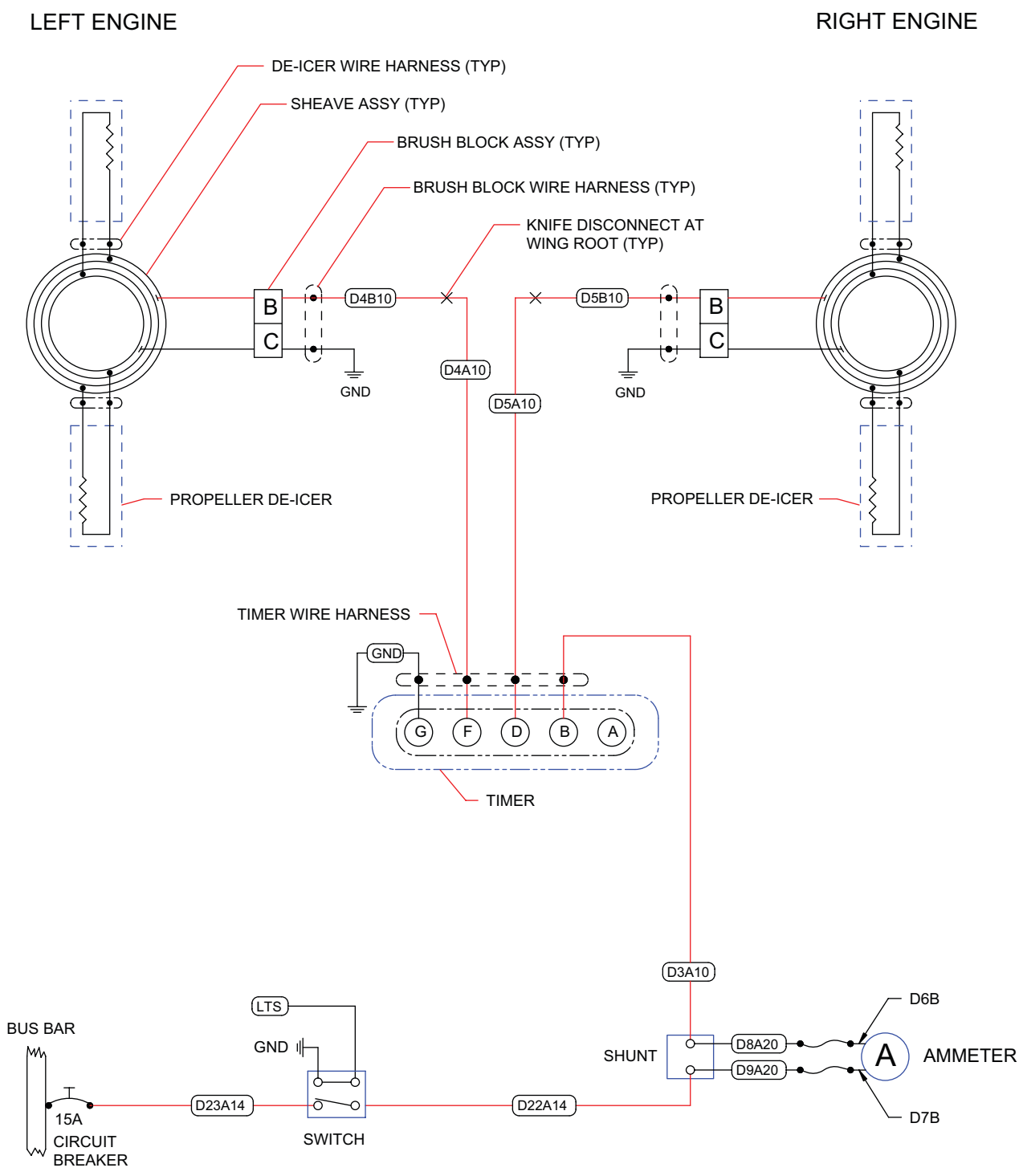
Effectivity
Seneca IV - Two-blade

89426 E/J 4



Effectivity
Seneca IV - Three-blade

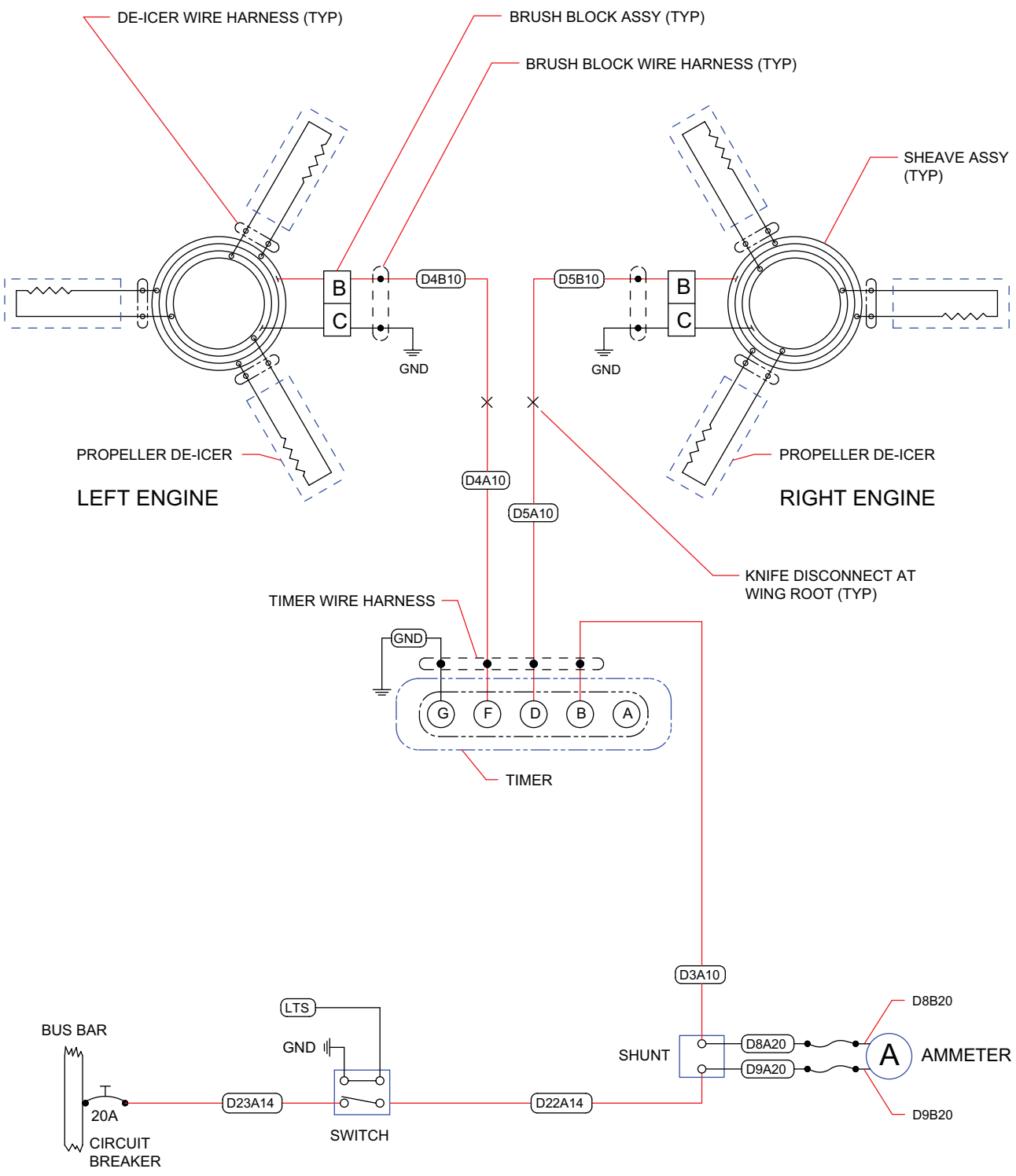
Propeller Deice
Figure 1 (Sheet 2 of 4)



Propeller Deice
Figure 1 (Sheet 3 of 4)

Effectivity
Seneca V - Two-blade

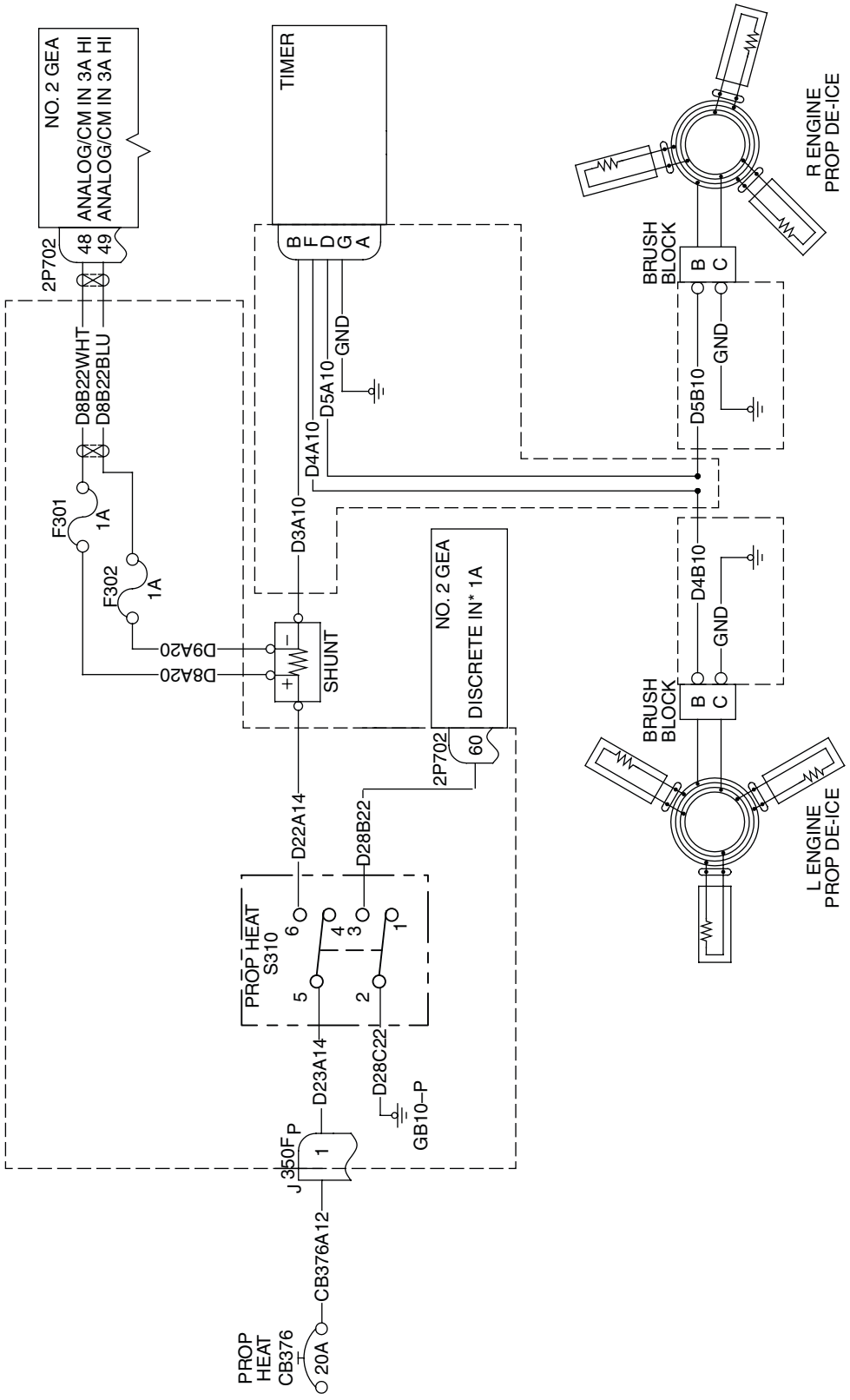
100735 E/J 5



Effectivity
Seneca V - Three-blade

Propeller Deice
Figure 1 (Sheet 4 of 4)

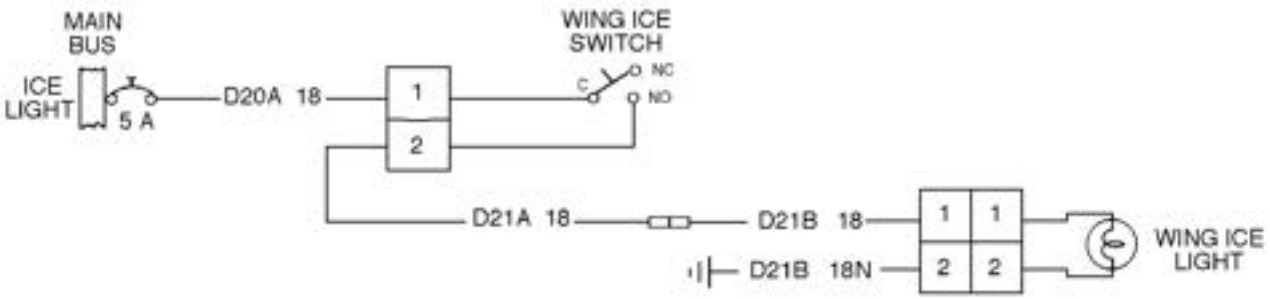
107230 NEW 20.0
107240 NEW 20.0
34F24A020 NEW 19.0



Propeller Deice
Figure 2

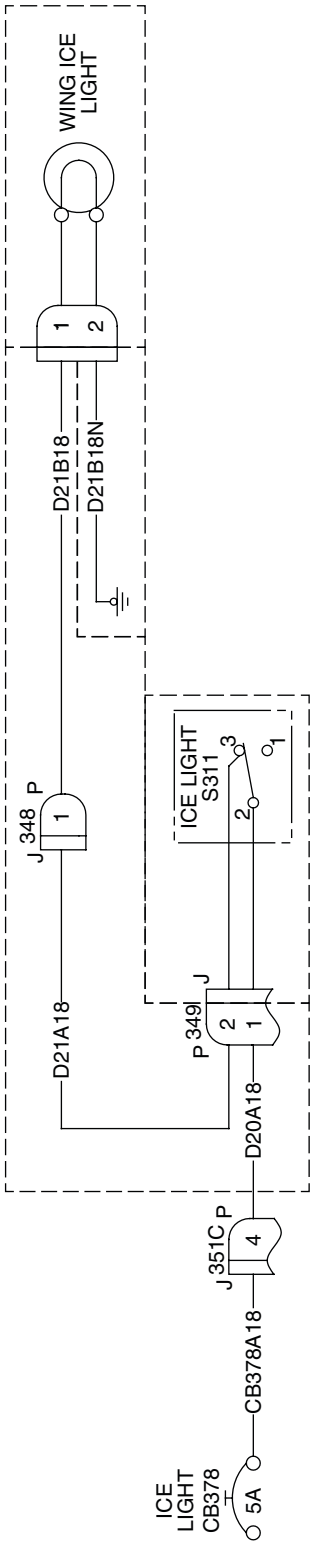
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85186 W/AD 2
87291 N/Y 1, 2



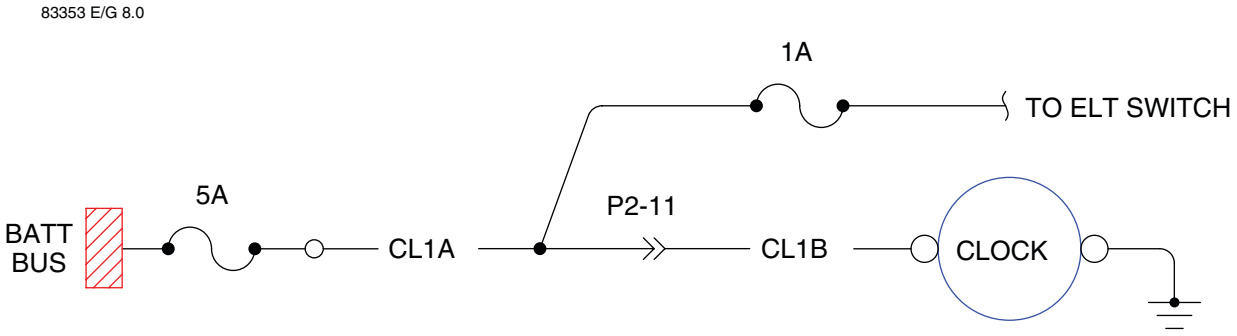
Wing Ice Light (Typical)
Figure 1 (Sheet 1 of 2)

107230 NEW 22.0
107240 NEW 22.0
34F24A020 NEW 21.0



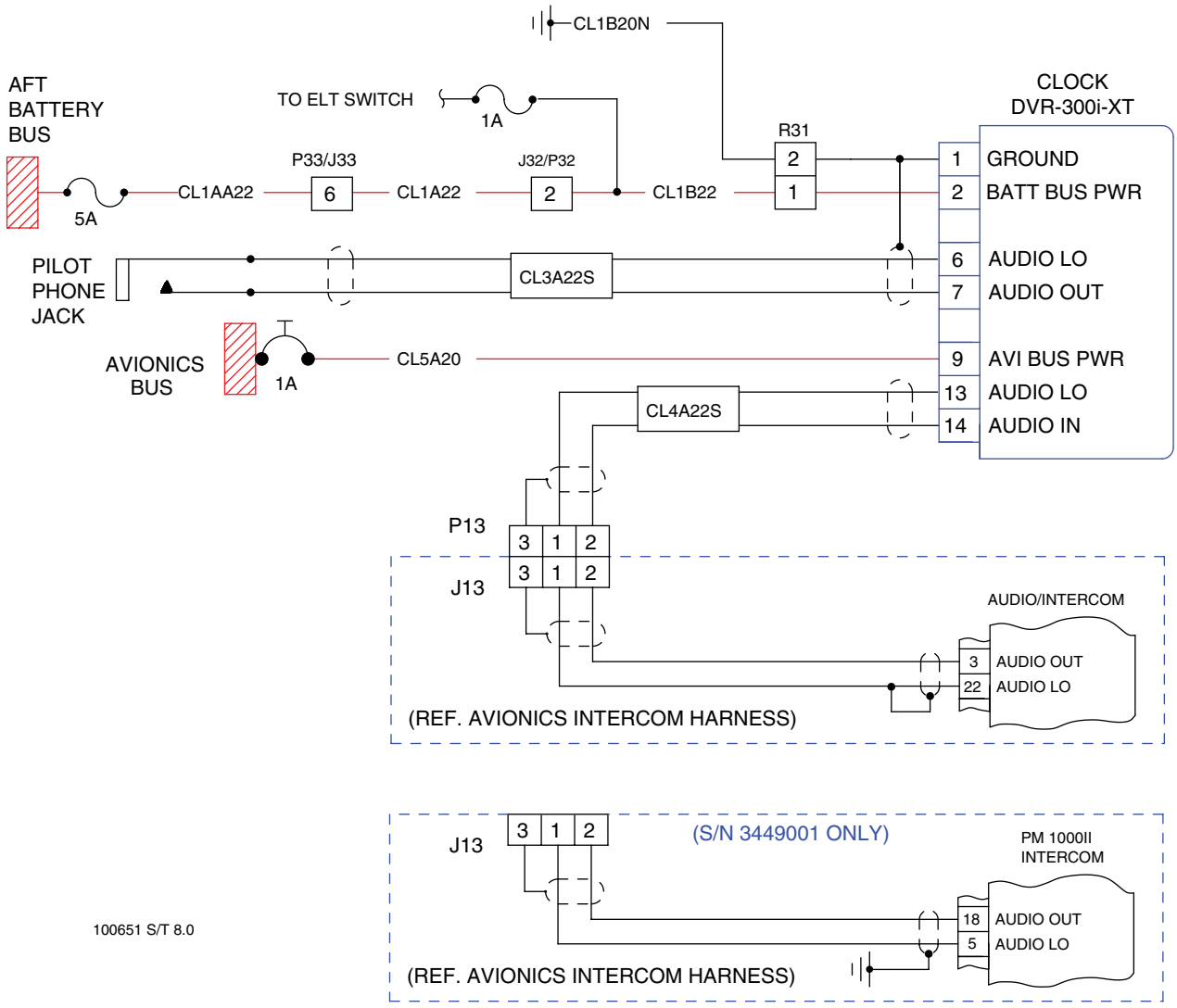
Effectivity
Seneca V
with Garmin G1000 / G1000 NXi

Wing Ice Light (Typical)
Figure 1 (Sheet 2 of 2)



Clock
Figure 1

Effectivity
Seneca IV

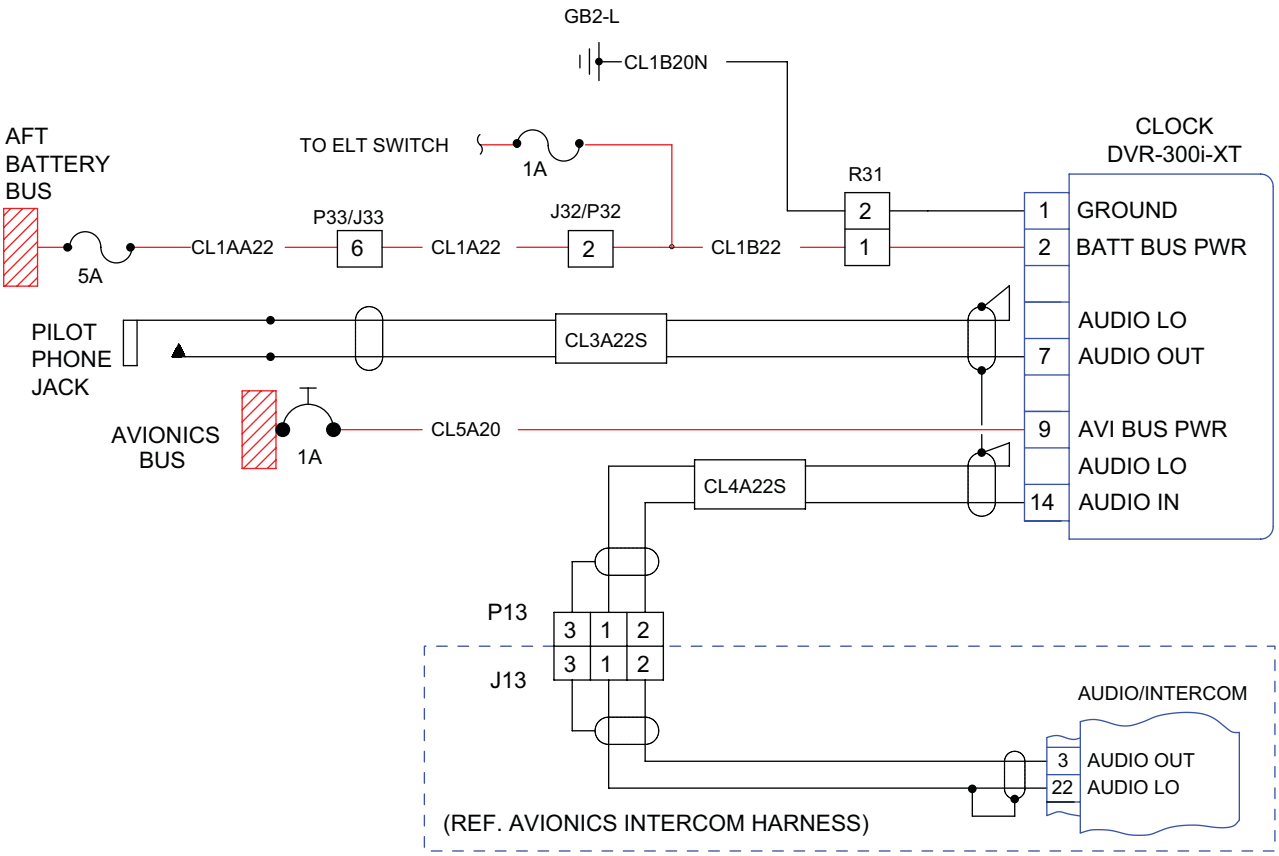


100651 S/T 8.0

Effectivity
3449001 thru 3449096

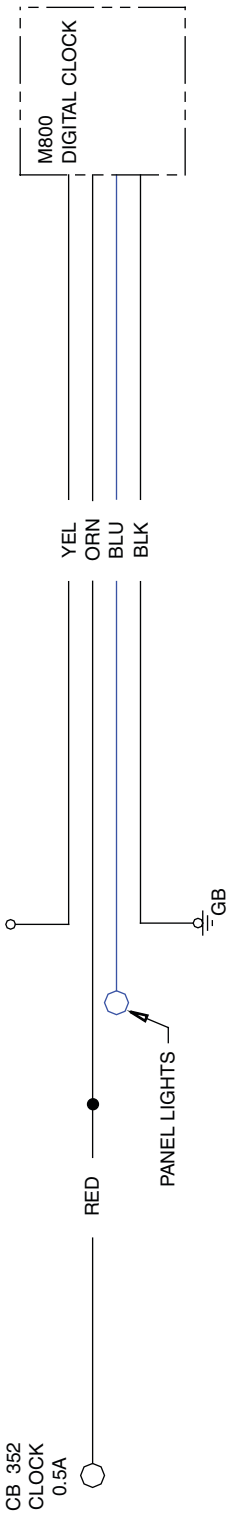
Digital Voice Recorder Clock
Figure 2 (Sheet 1 of 2)

101288 C/D 8
104229 NEW/C 8
104368 NEW/K 8.0



Digital Voice Recorder Clock
Figure 2 (Sheet 2 of 2)

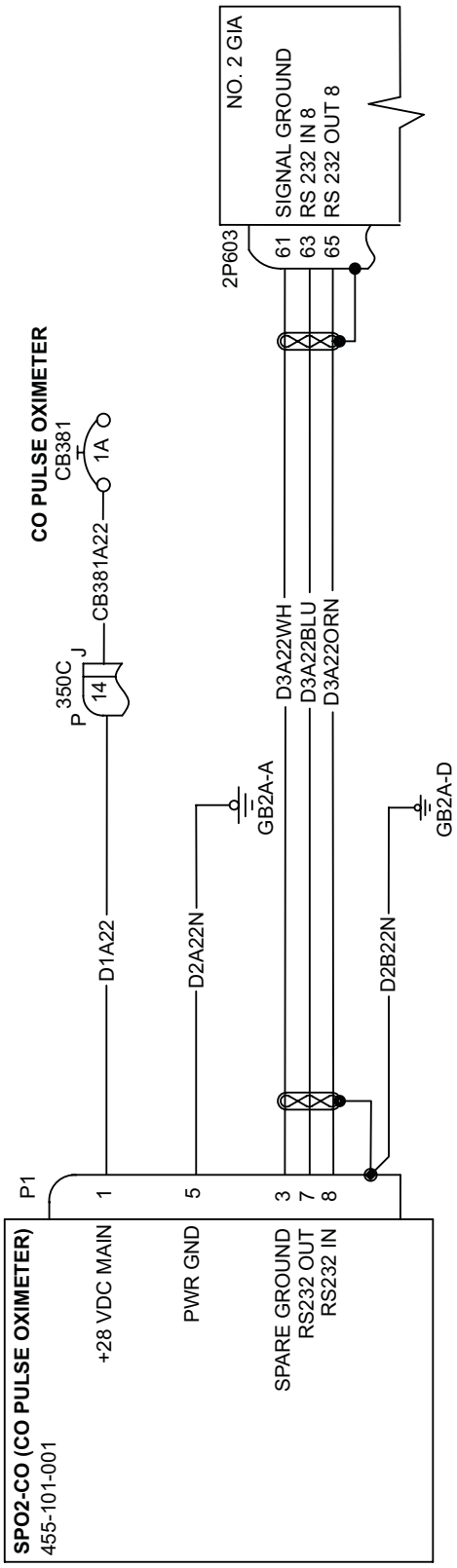
Effectivity
3449097 and up



Effectivity
Seneca V
with Garmin G600

Digital Clock
Figure 3

34F24A020 NEW 48.0
34F24A008 A 2.0

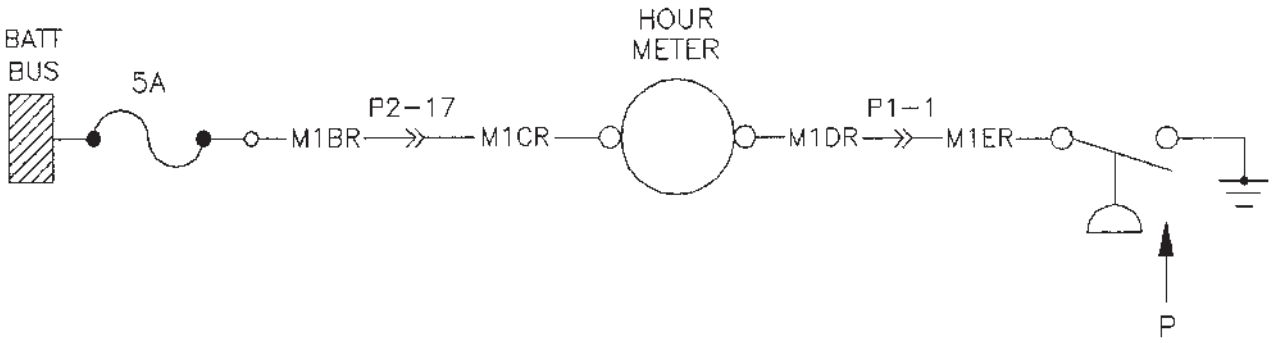


CO Pulse Oximeter
Figure 4

Effectivity
Seneca V
with Garmin G1000 NXi

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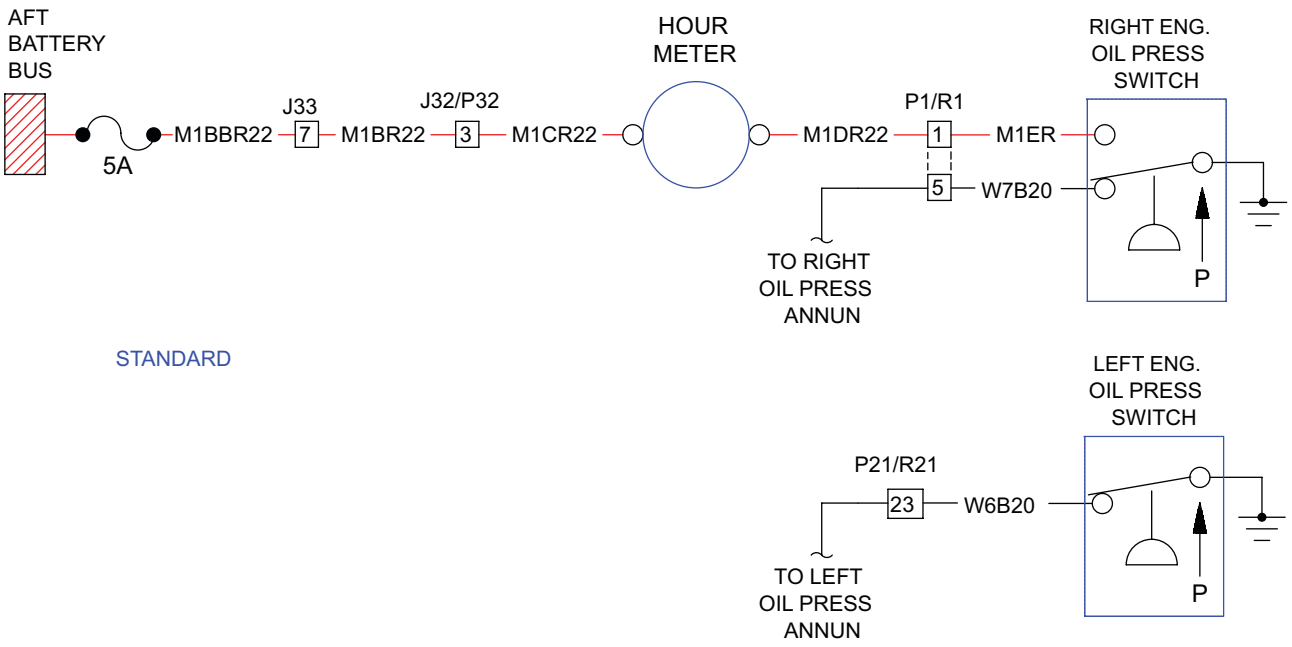
83353 NEW/G 15.0



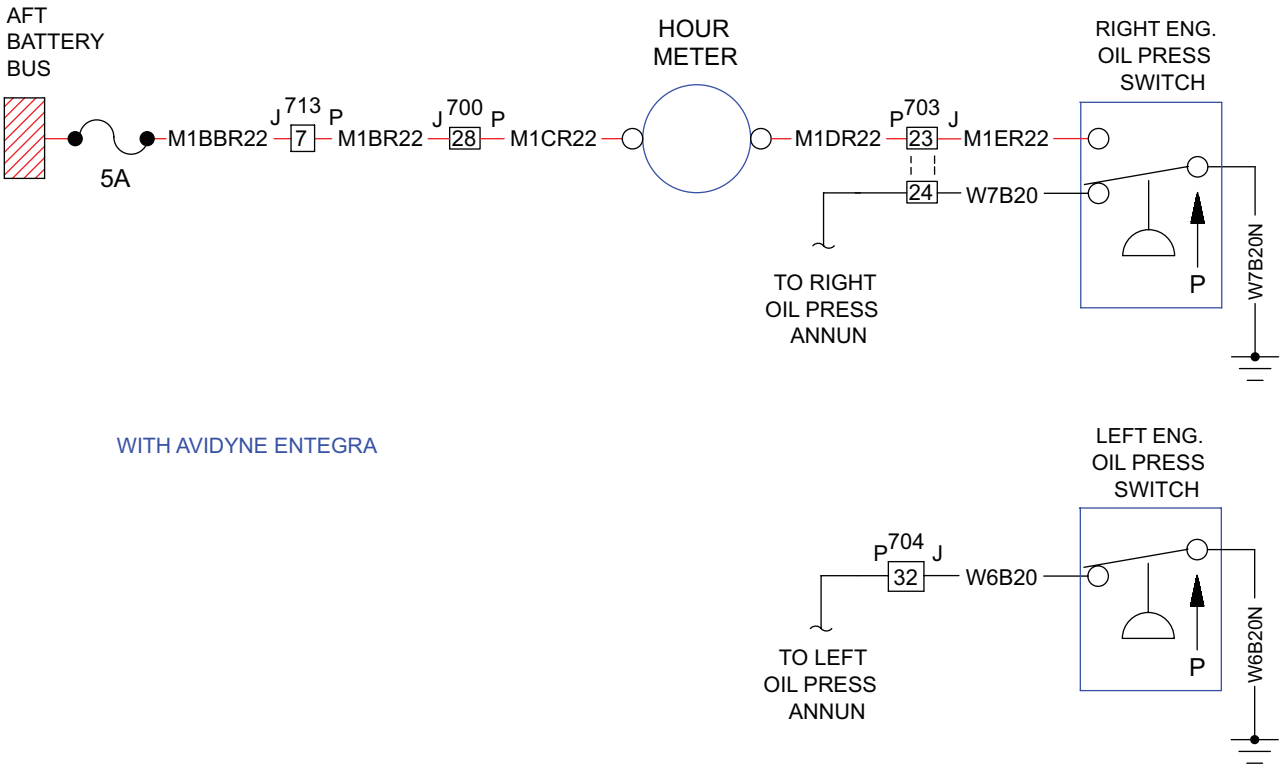
Hour Meter and Oil Pressure Switch(es)
Figure 1 (Sheet 1 of 2)

Effectivity
Seneca IV

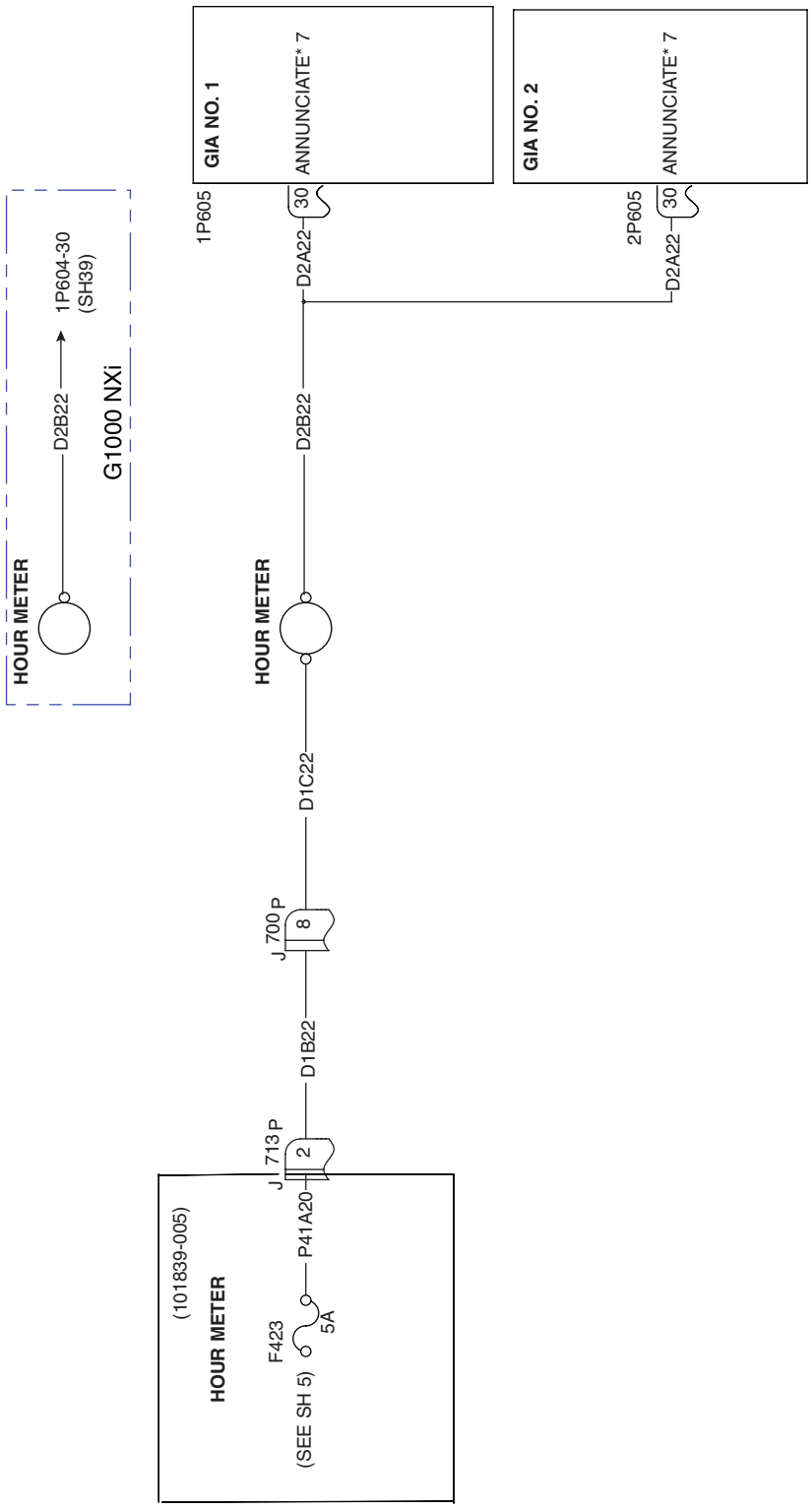
100651 K/T 15
101288 NEW/D 15
104229 NEW/C 15
104368 NEW/K 15.0



101840 NEW 9.0

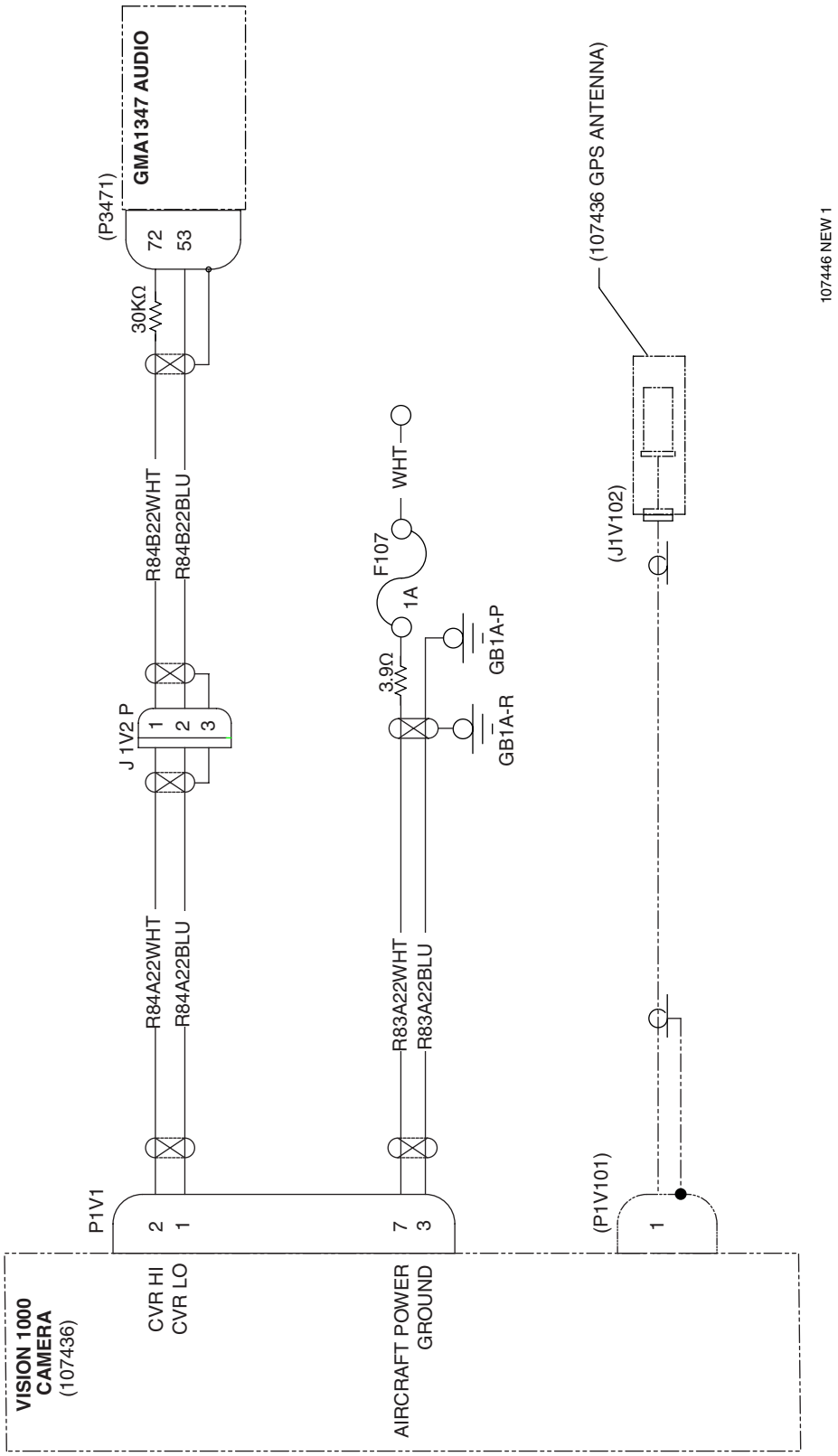


Hour Meter and Oil Pressure Switch(es)
Figure 1 (Sheet 2 of 2)



Hour Meter
Figure 2

Effectivity
Seneca V
with Garmin G1000 / G1000 NXi

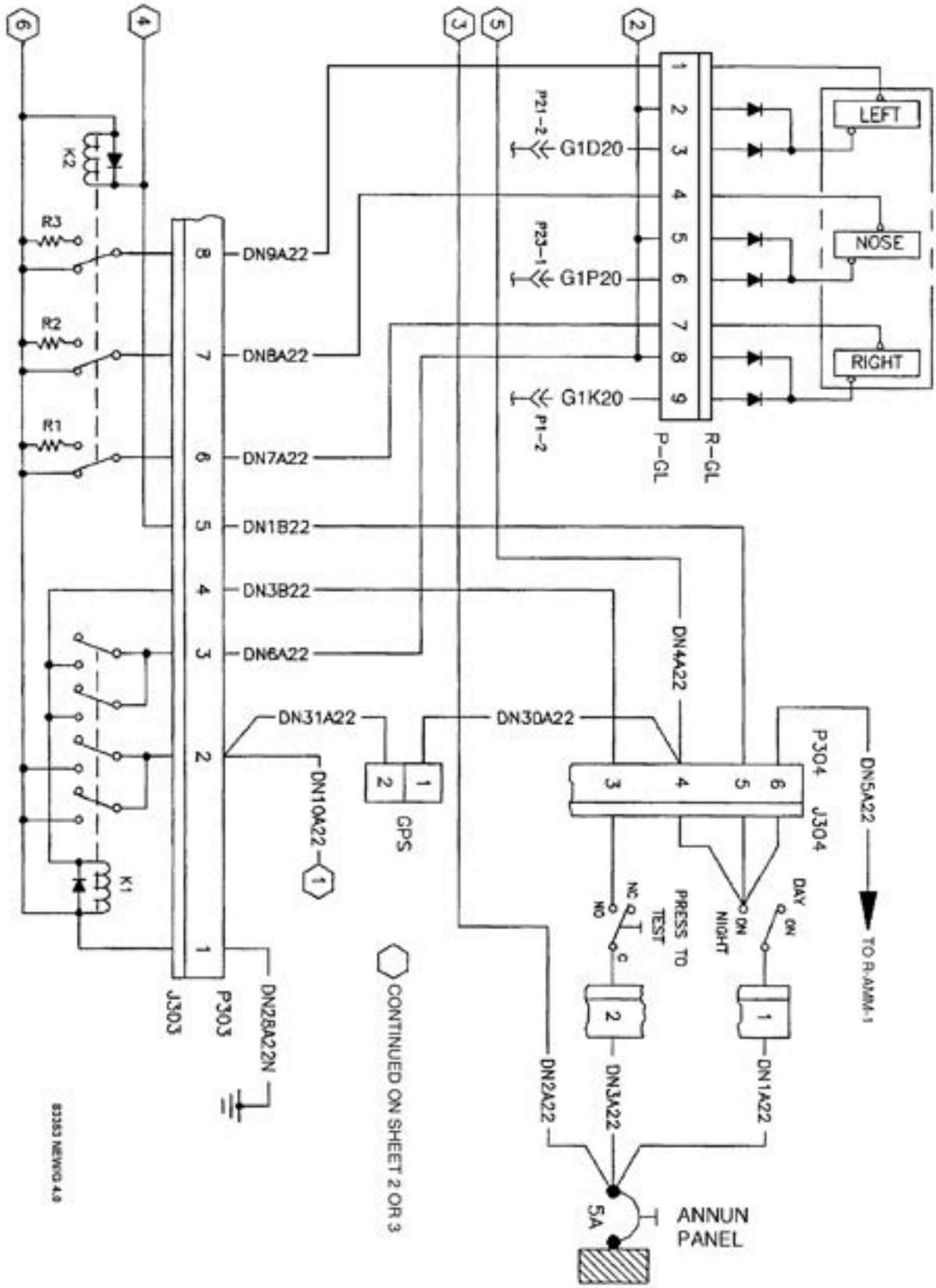


Effectivity
Seneca V - Optional
3449479 and up

Vision 1000 Flight Data Monitoring System
Figure 3

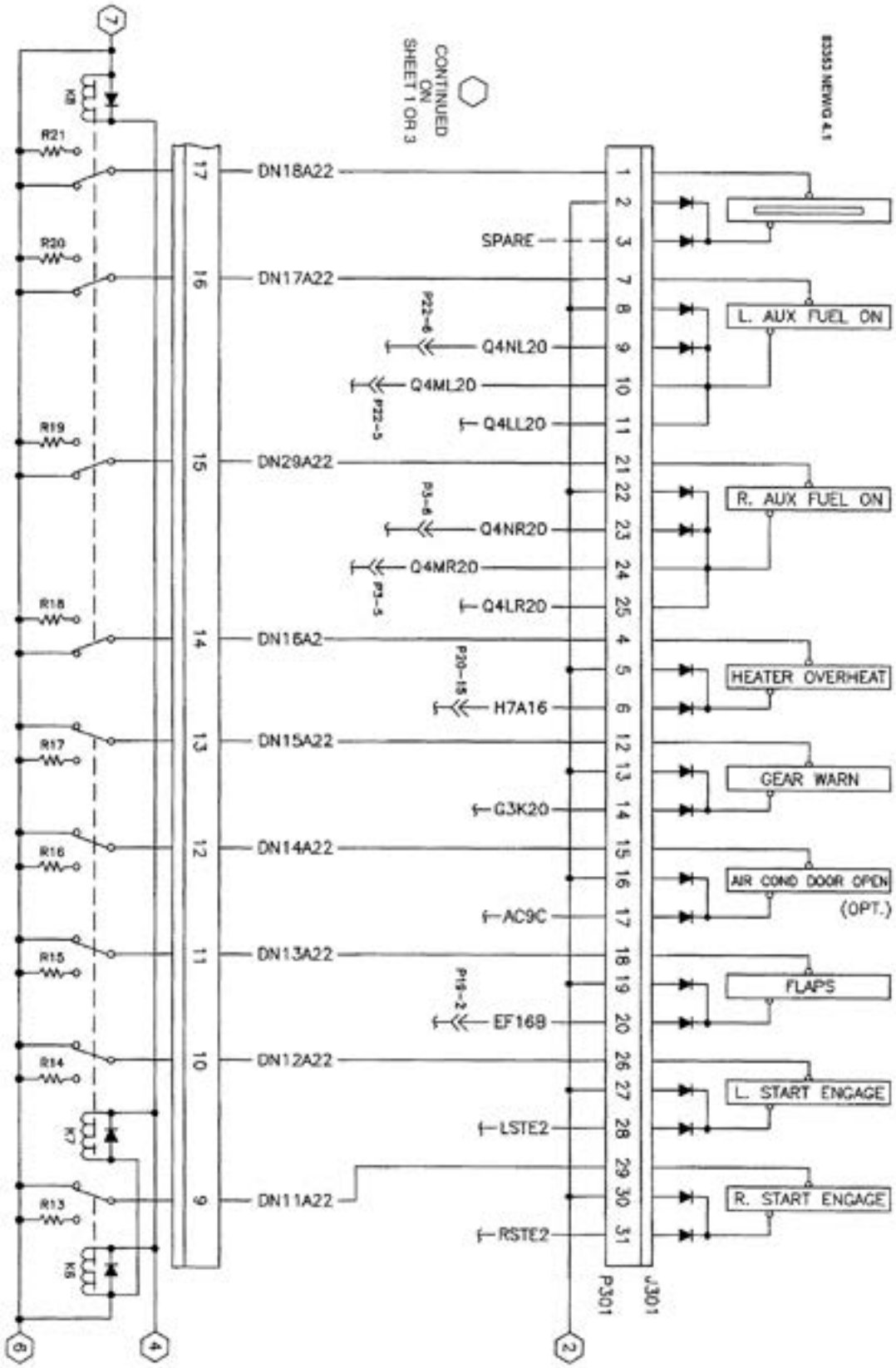
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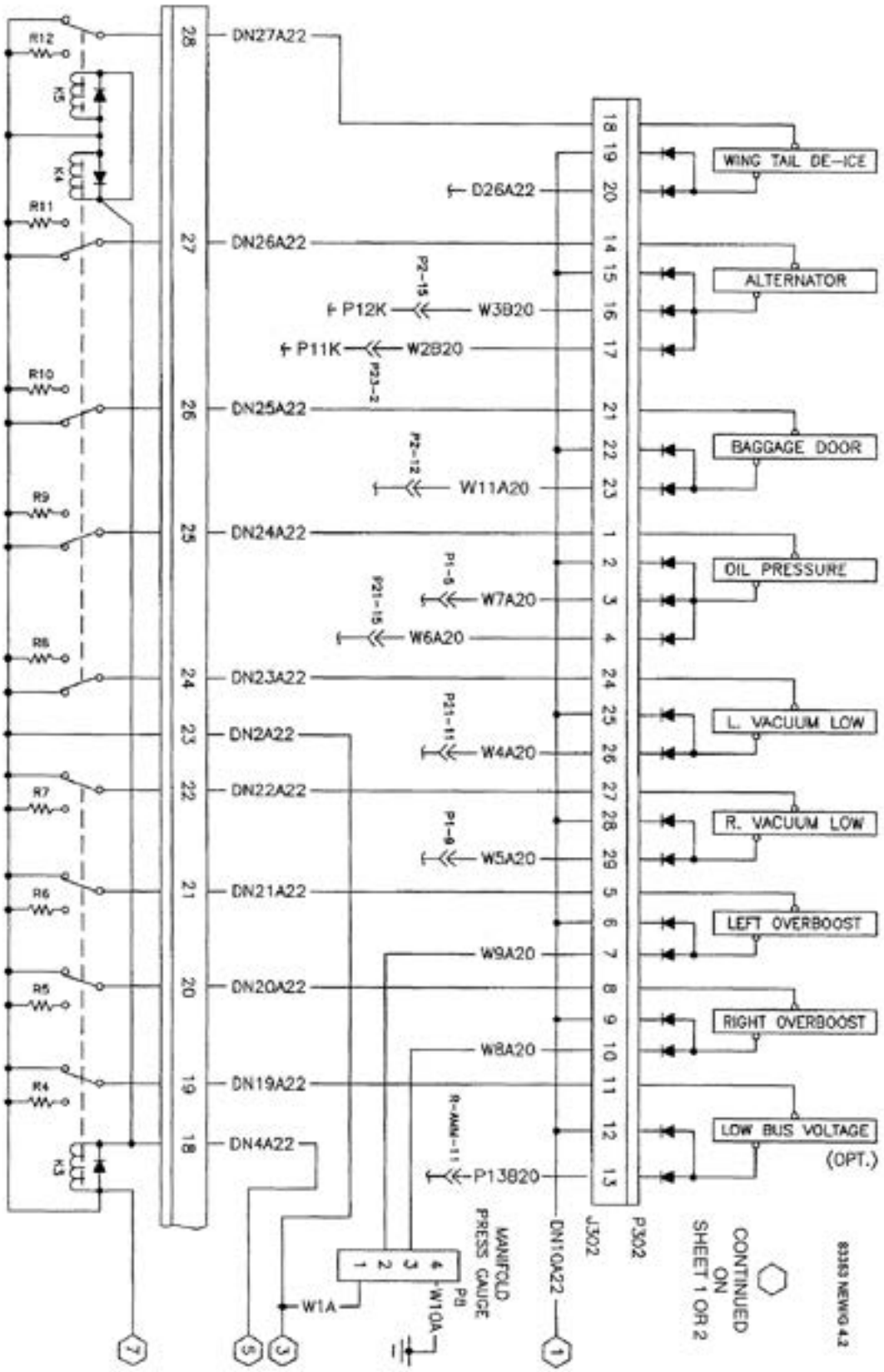


Annunciator
Figure 1 (Sheet 1 of 14)

Effectivity
Seneca IV

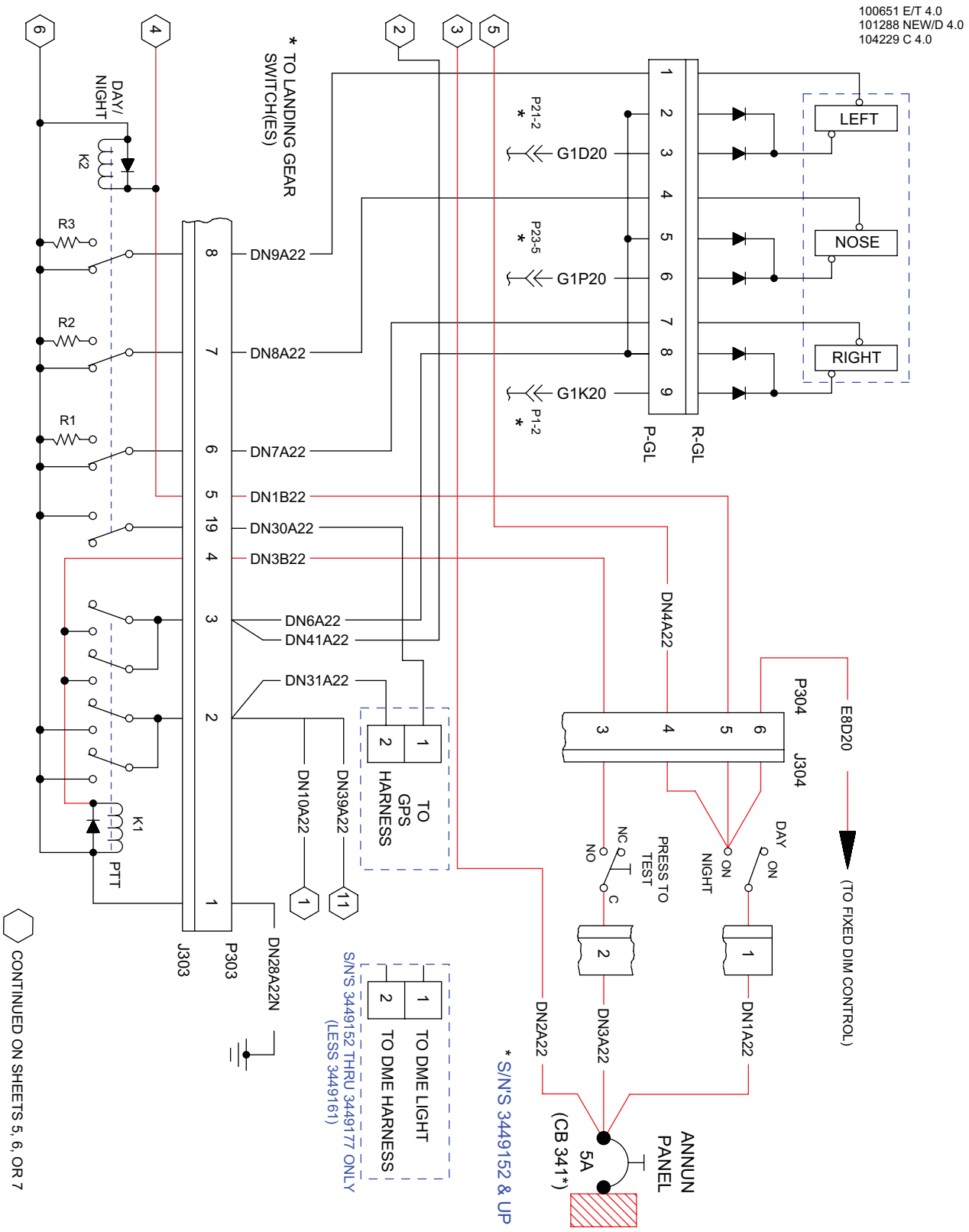


Annunciator
Figure 1 (Sheet 2 of 14)



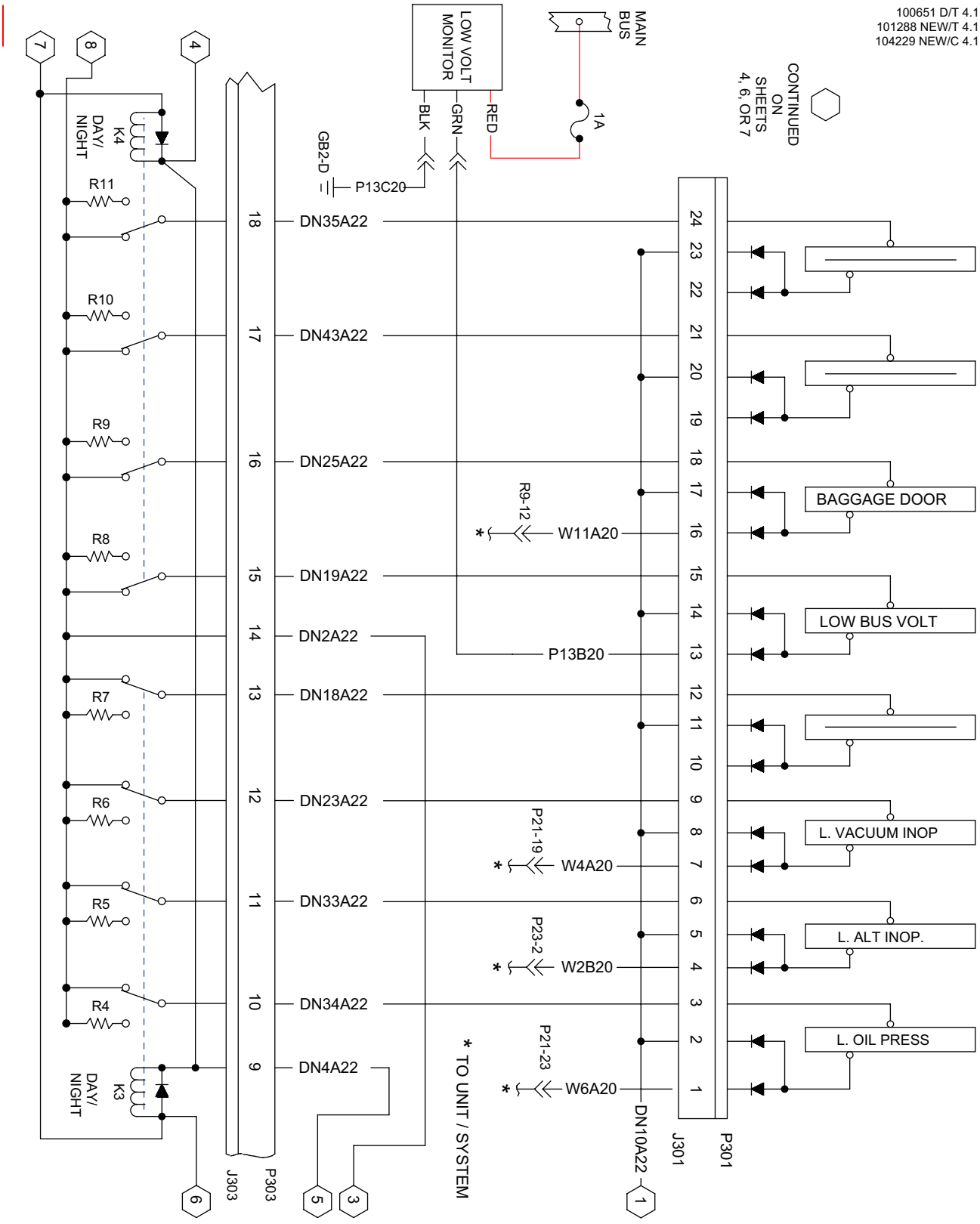
Annunciator
Figure 1 (Sheet 3 of 14)

Effectivity
Seneca IV



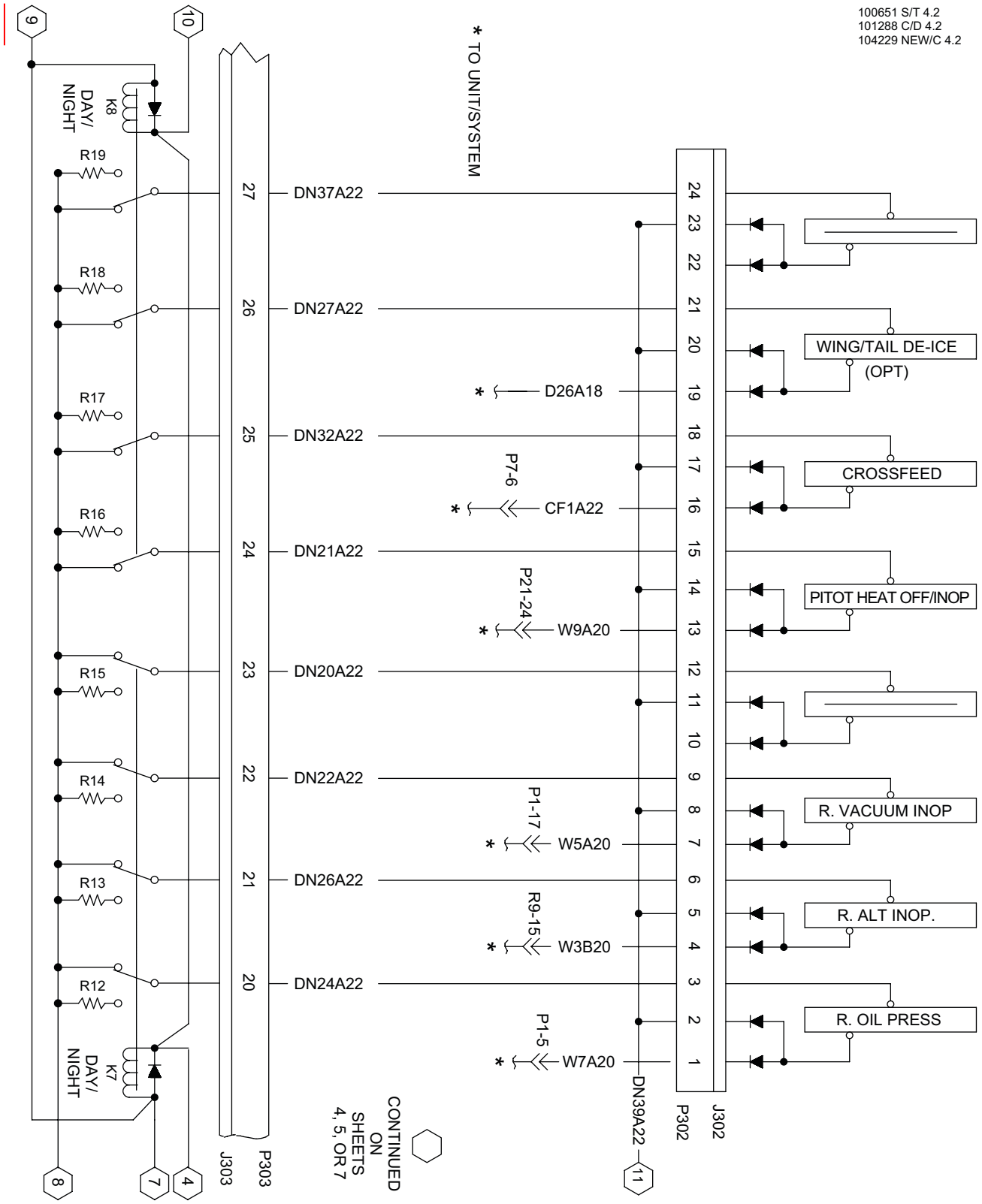
Effectivity
3449001 thru 3449177, less 3449161

Annunciator
Figure 1 (Sheet 4 of 14)



Annunciator
Figure 1 (Sheet 5 of 14)

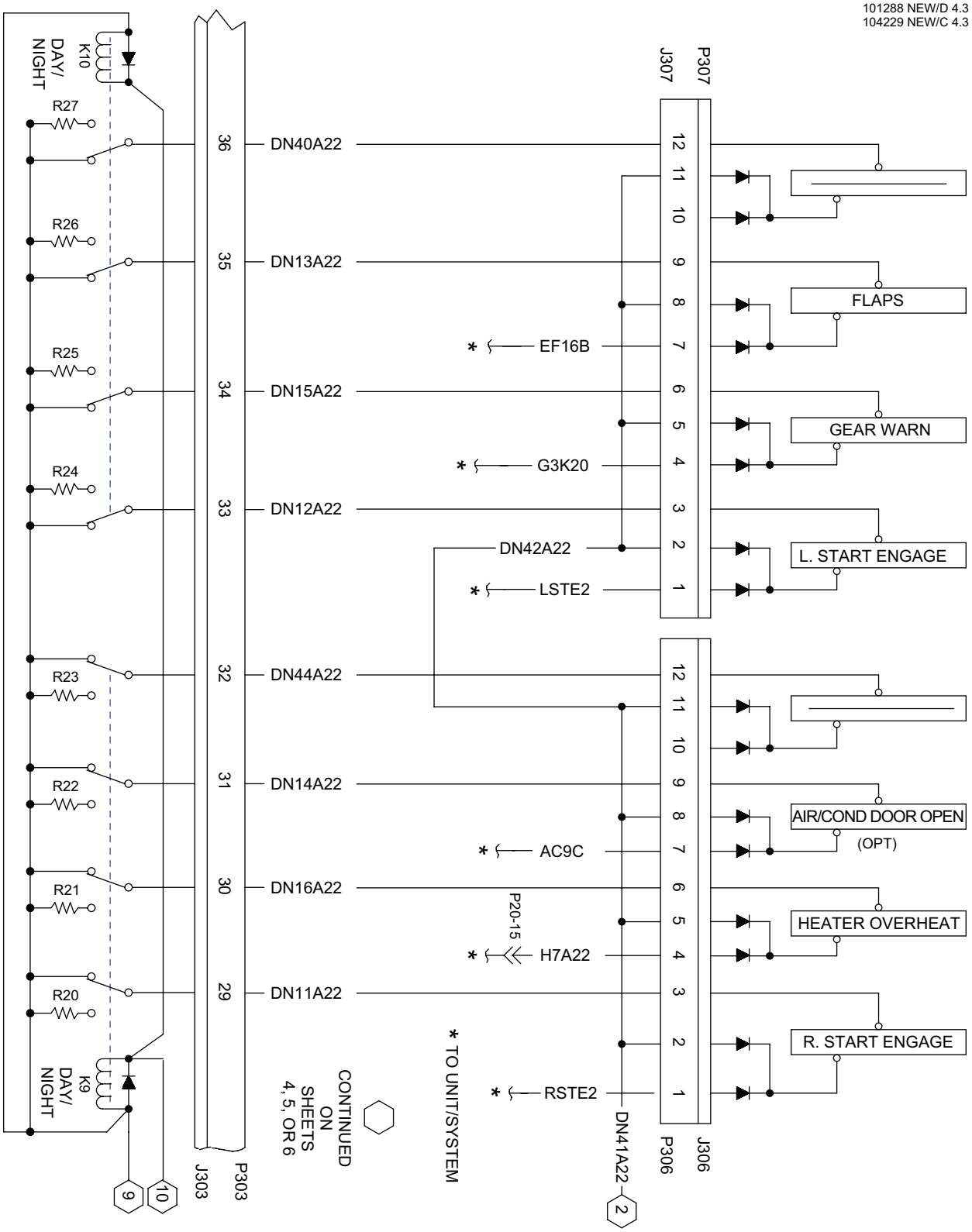
Effectivity
3449001 thru 3449177,
less 3449161



Effectivity
3449001 thru 3449177,
less 3449161

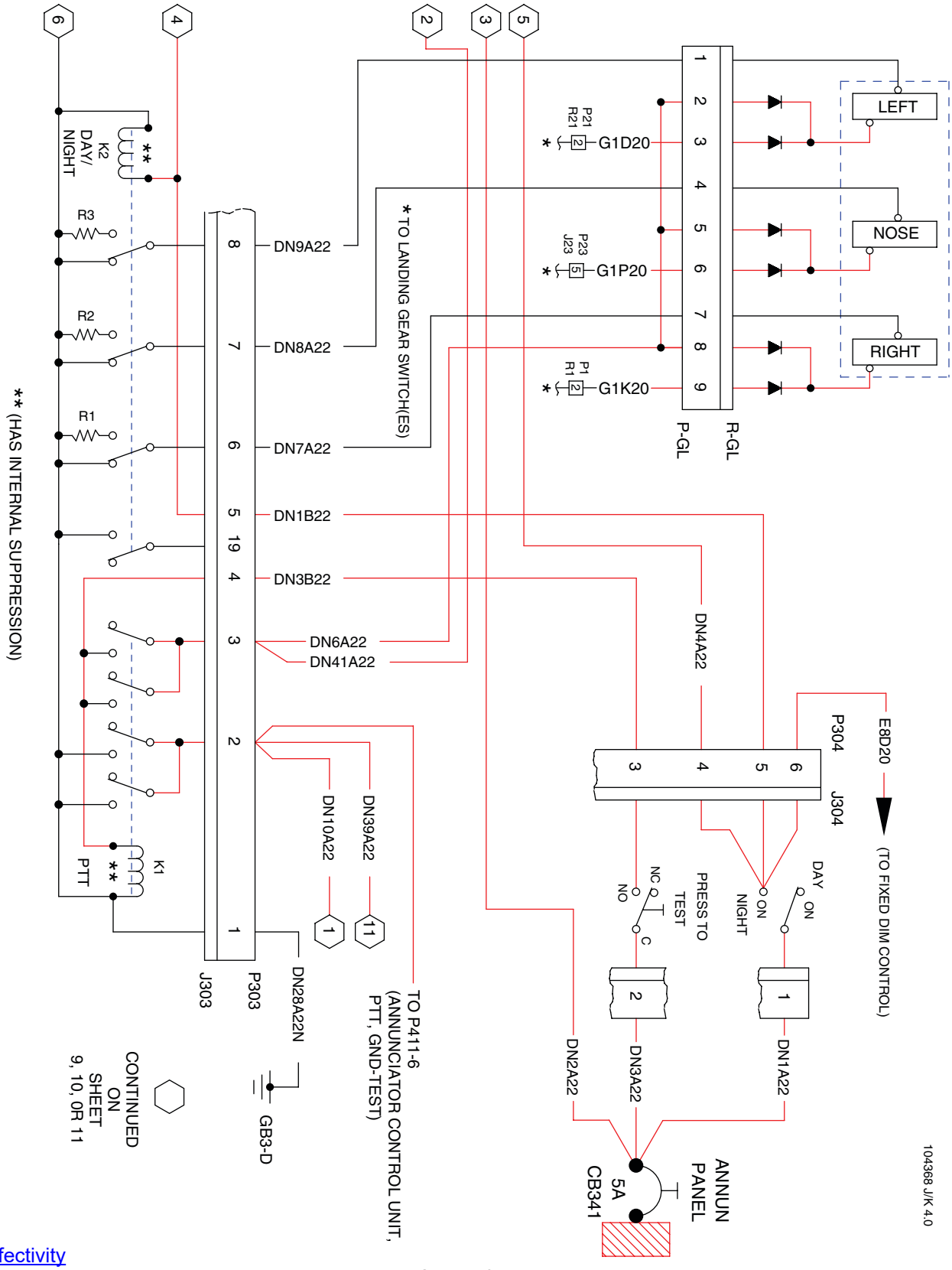
Annunciator
Figure 1 (Sheet 6 of 14)

100651 D/T 4.3
101288 NEW/D 4.3
104229 NEW/C 4.3



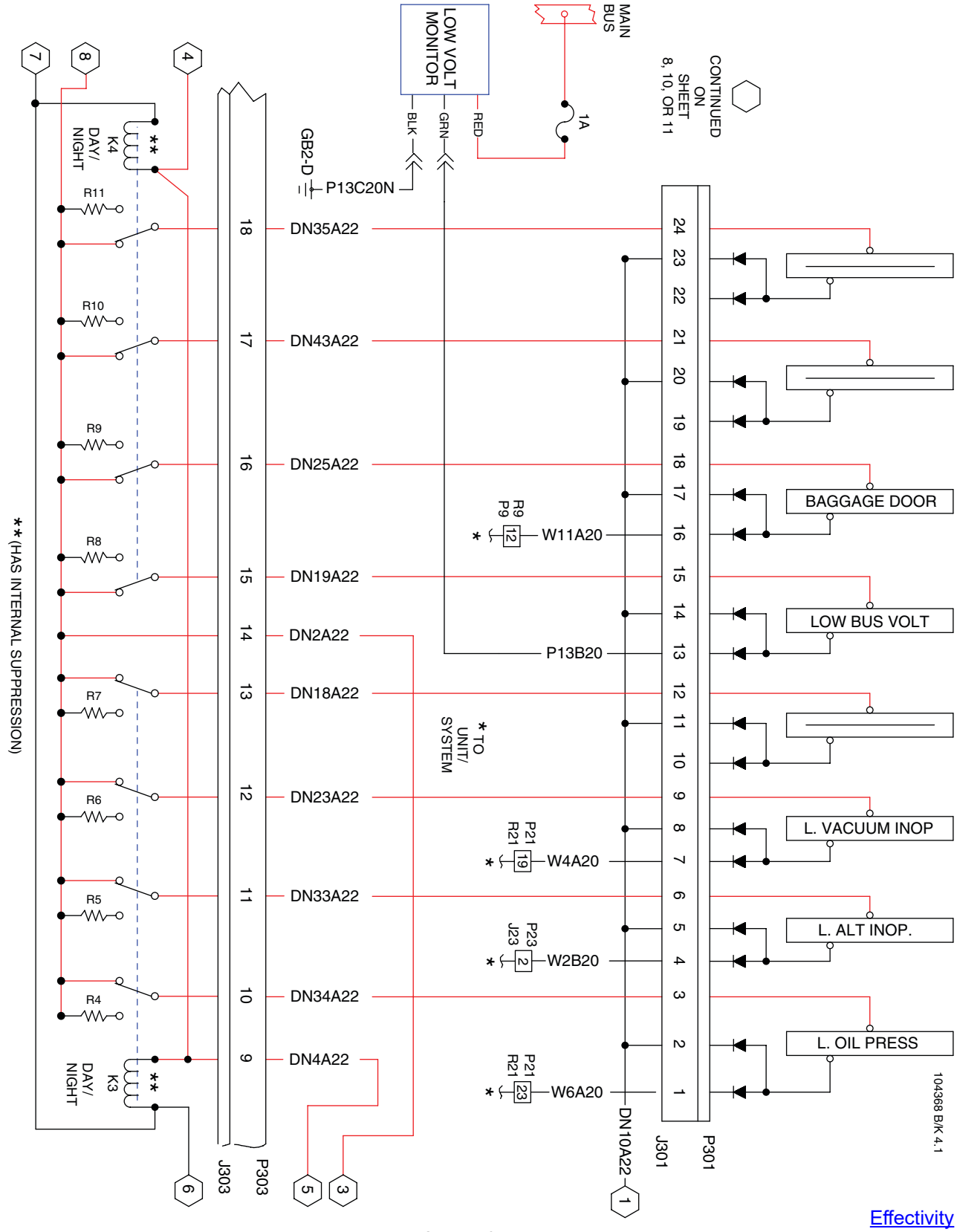
Annunciator
Figure 1 (Sheet 7 of 14)

Effectivity
3449001 thru 3449177,
less 3449161



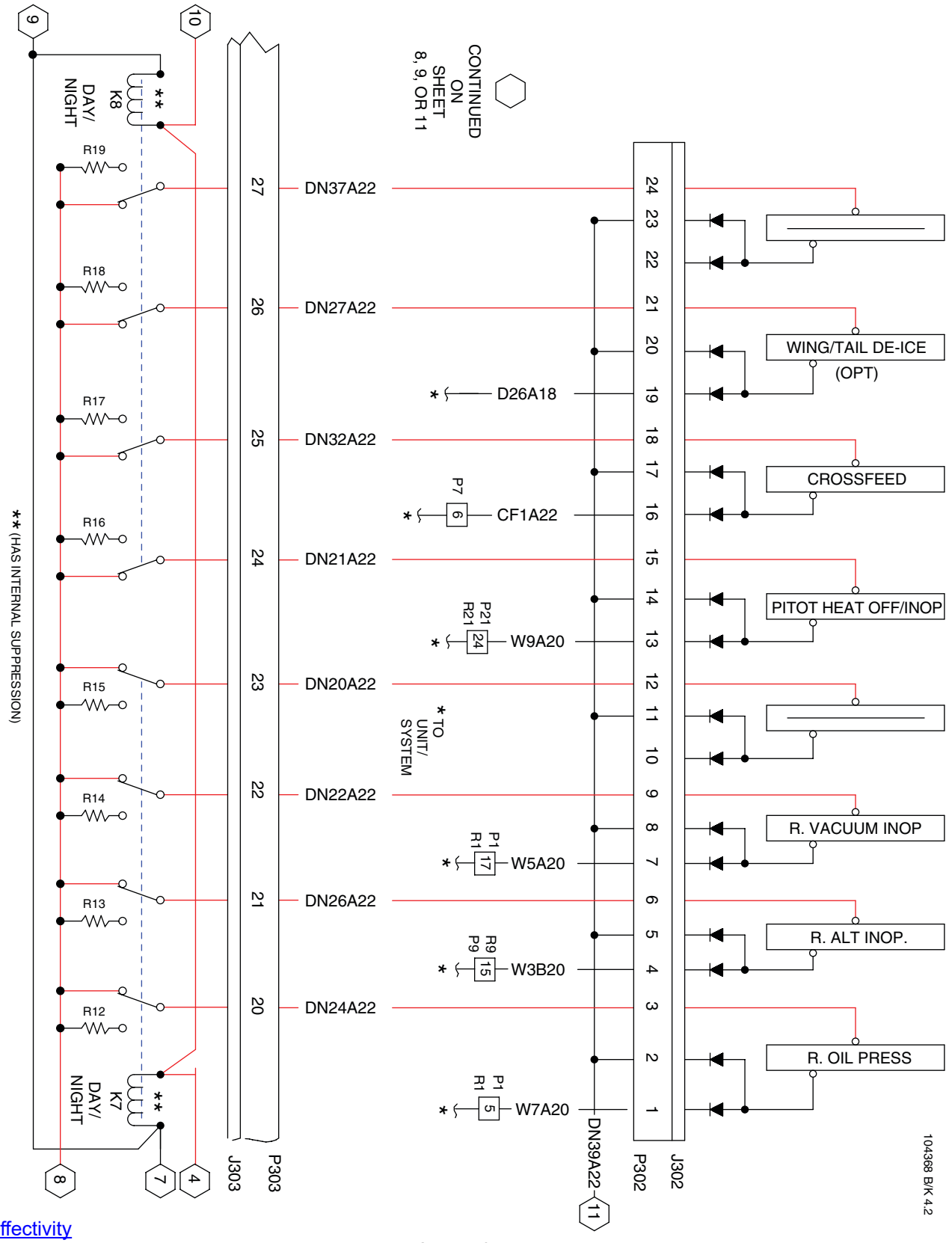
Effectivity
3449161,
3449178 and Up

Annunciator
Figure 1 (Sheet 8 of 14)



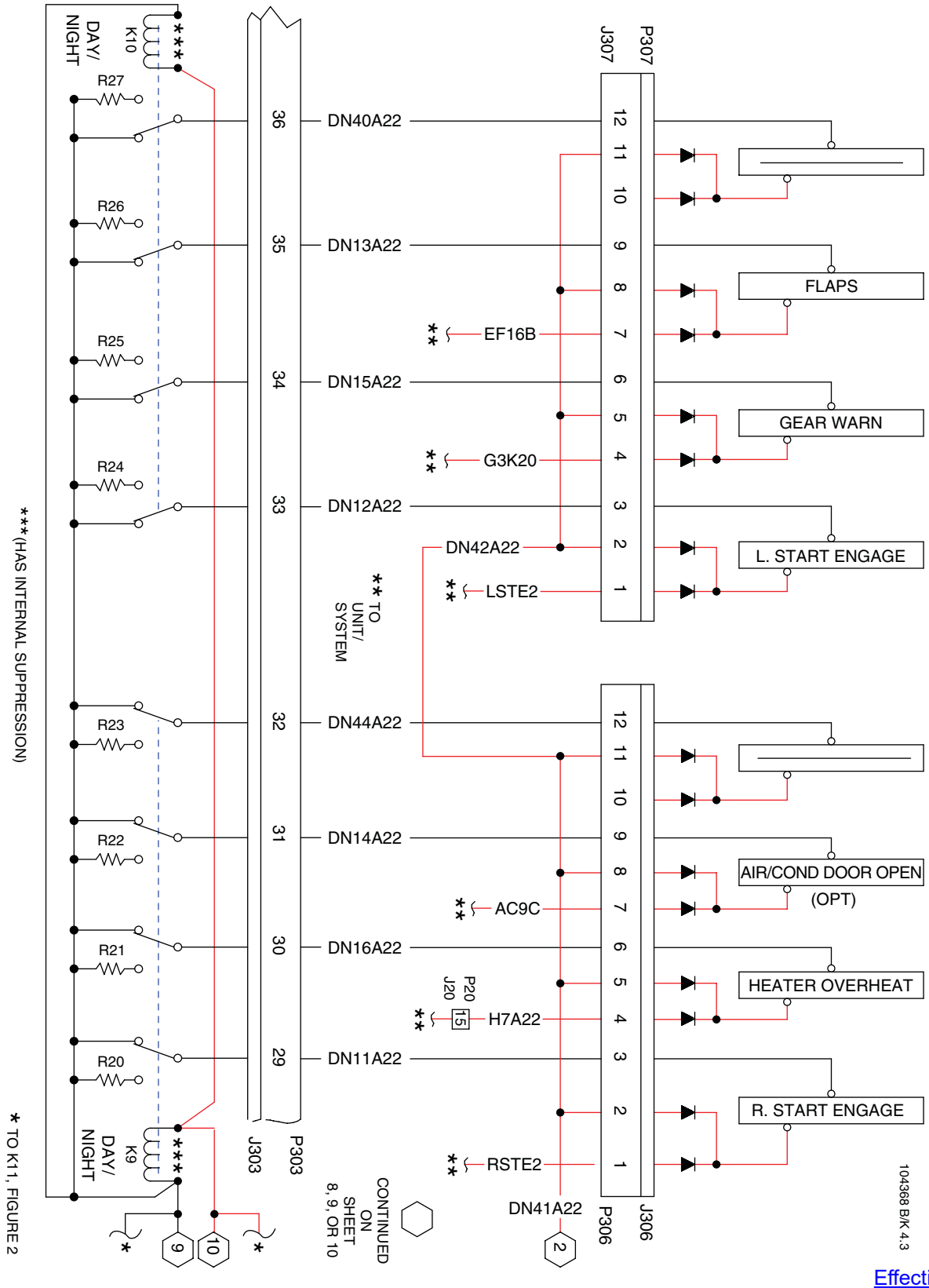
Annunciator
Figure 1 (Sheet 9 of 14)

Effectivity
3449161,
3449178 and Up



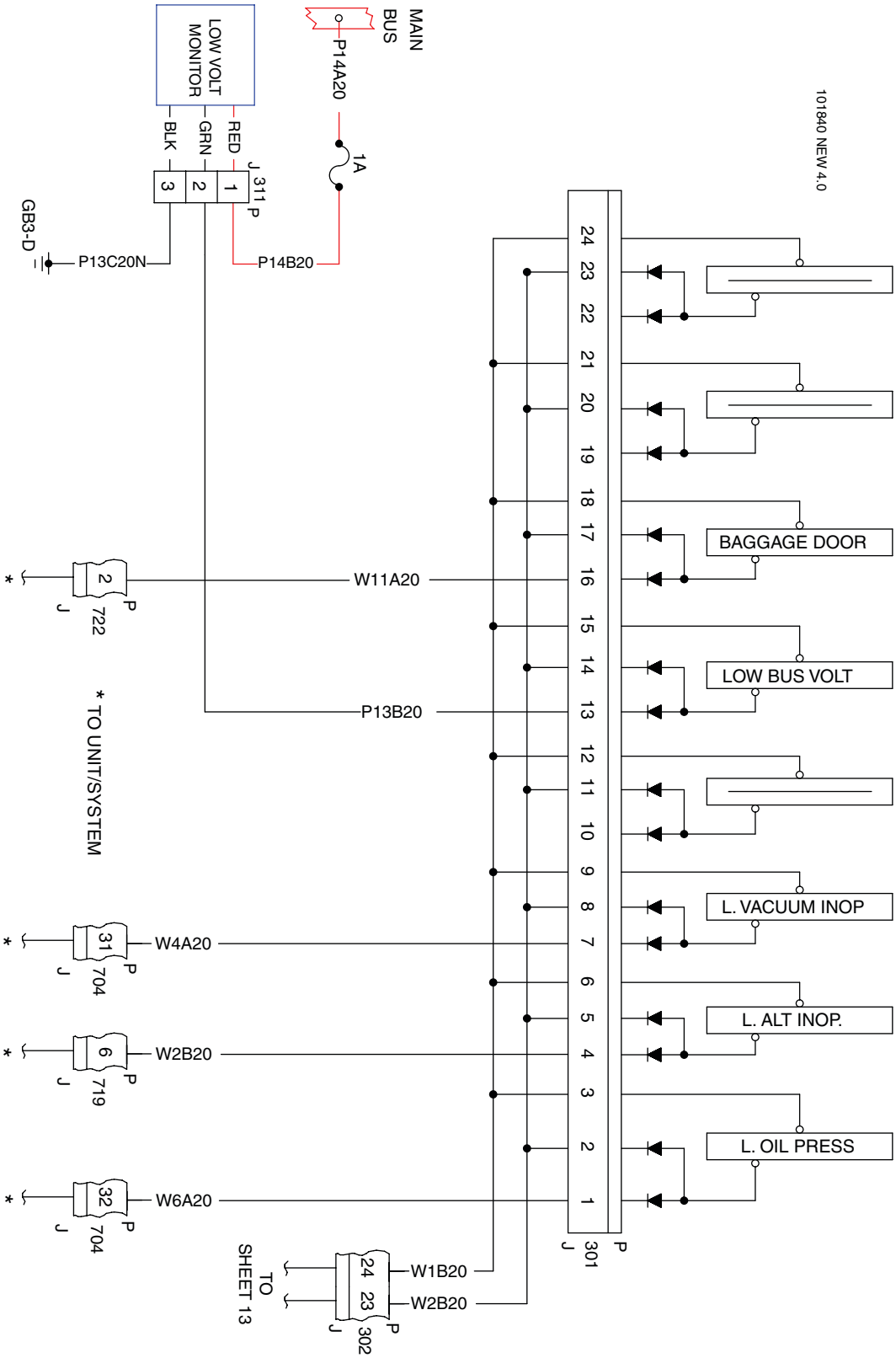
Effectivity
3449161,
3449178 and Up

Annunciator
Figure 1 (Sheet 10 of 14)



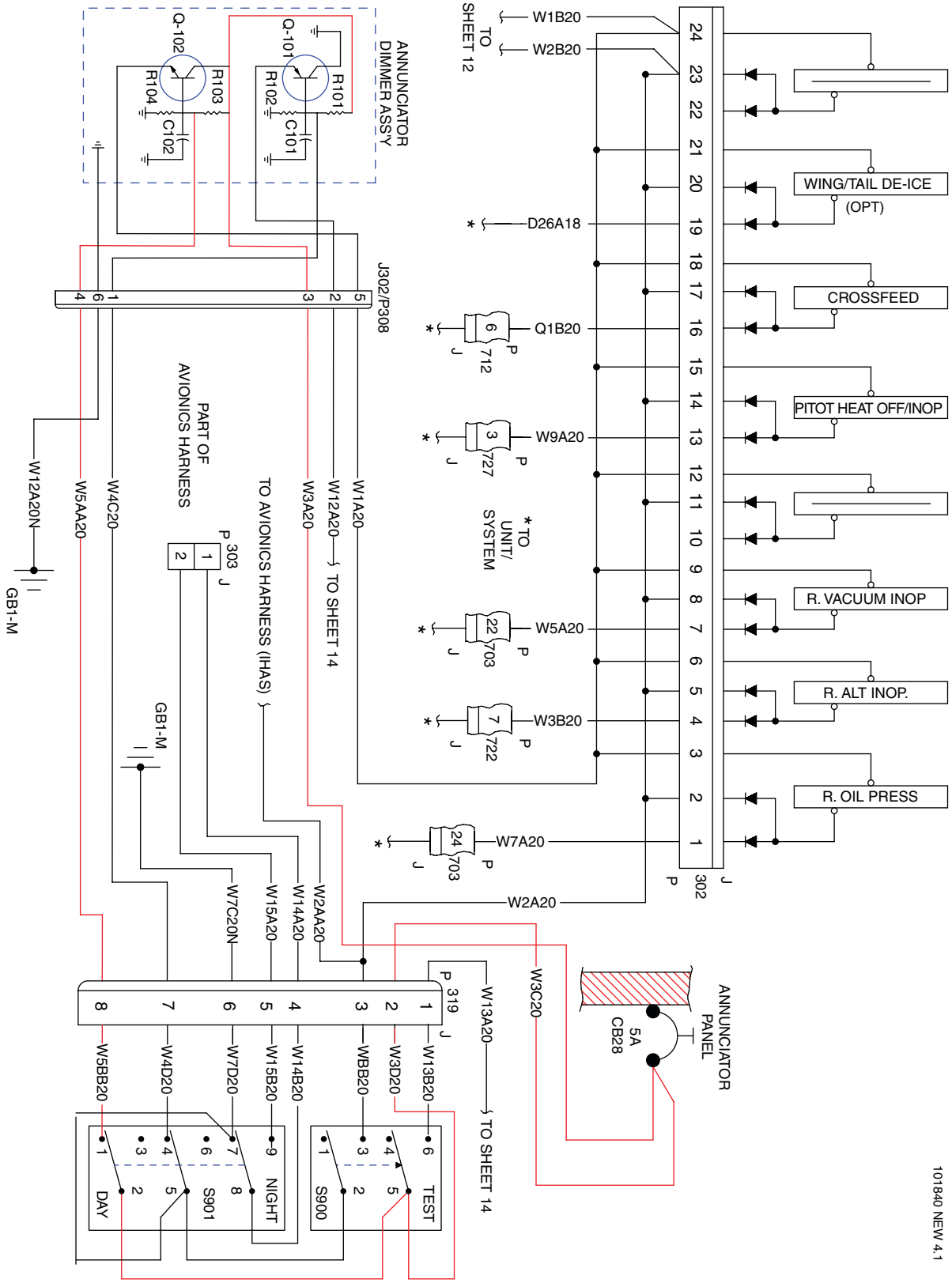
Annunciator
Figure 1 (Sheet 11 of 14)

Effectivity
3449161,
3449178 and Up



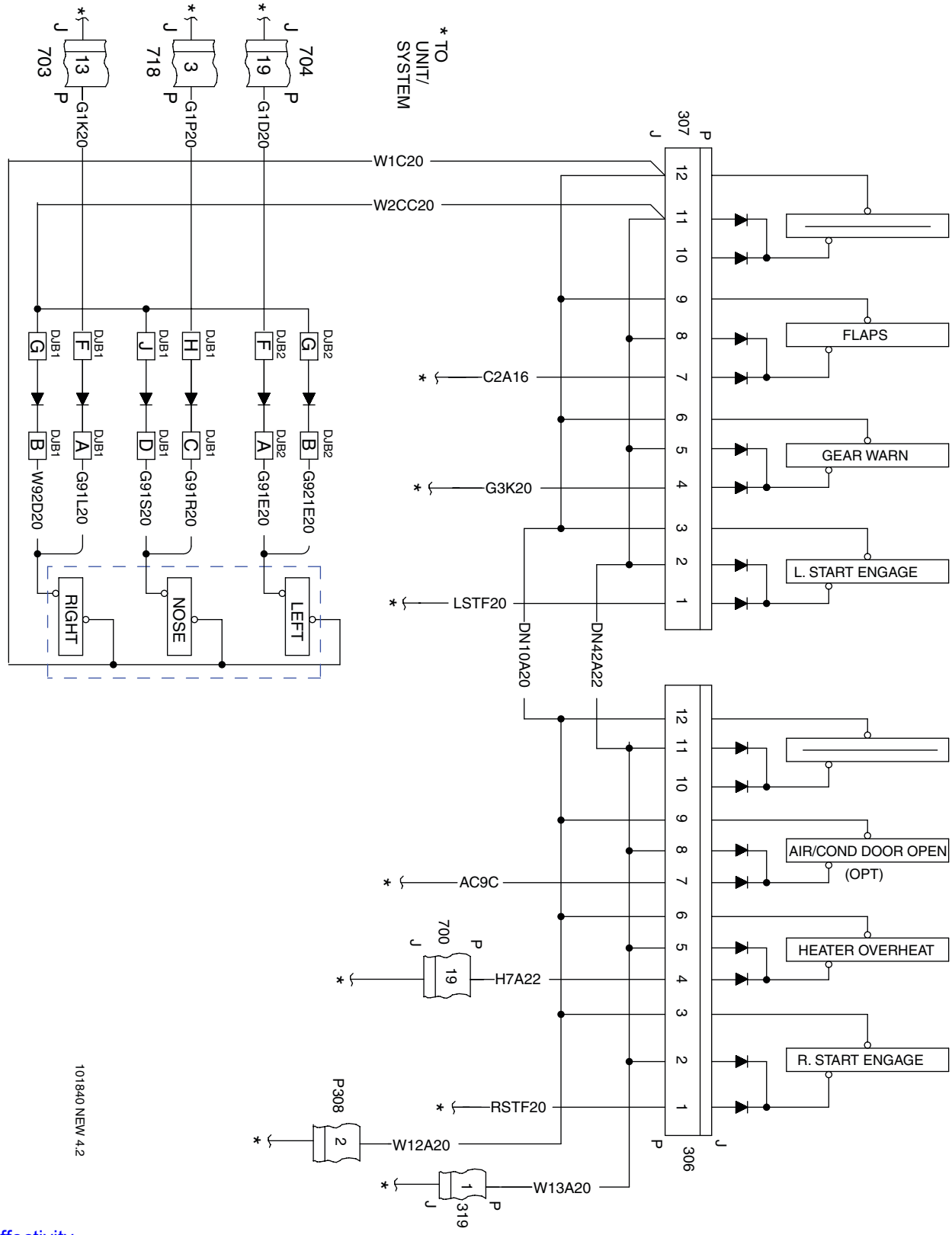
Effectivity
Seneca V
with Avidyne Entegra

Annunciator
Figure 1 (Sheet 12 of 14)



Annunciator
Figure 1 (Sheet 13 of 14)

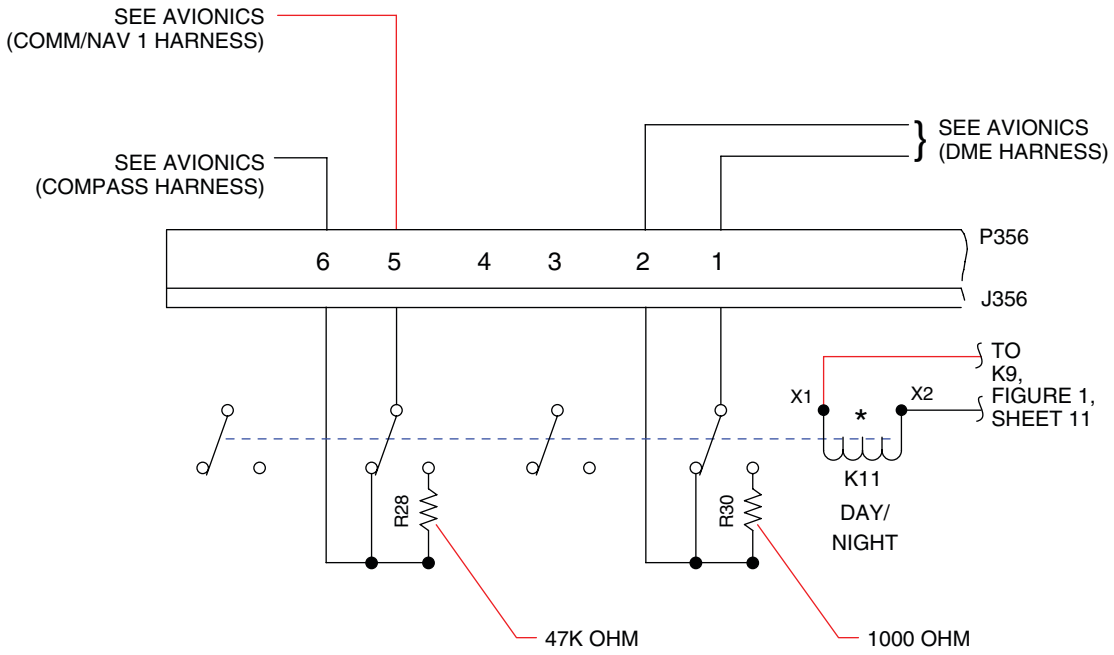
Effectivity
Seneca V
with Avidyne Entegra



Effectivity
Seneca V
with Avidyne Entegra

Annunciator
Figure 1 (Sheet 14 of 14)

104368 E/K 4.4

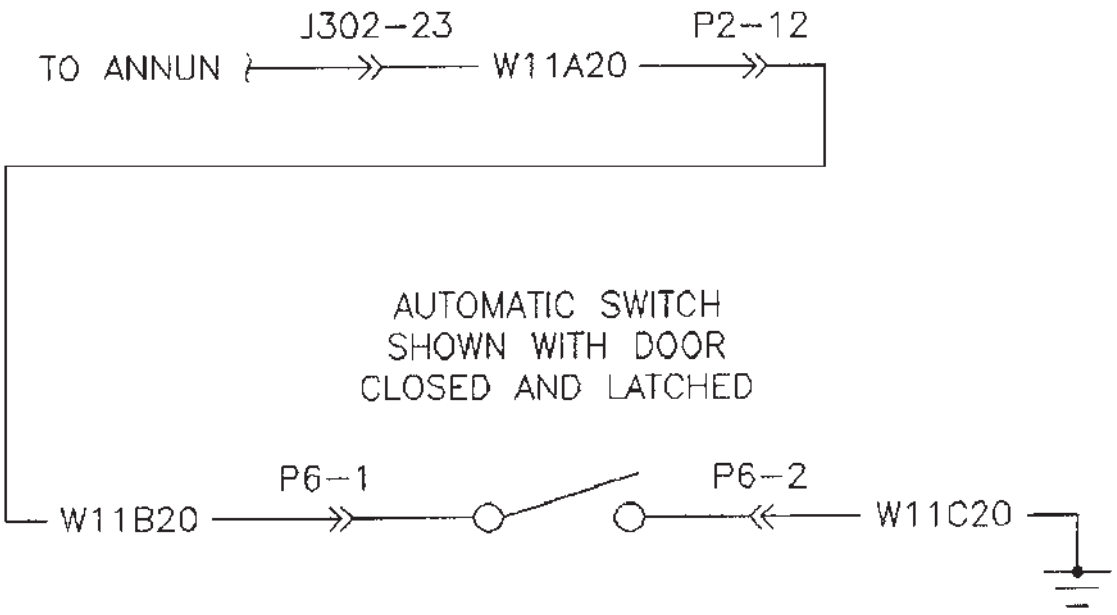


* (HAS INTERNAL SUPPRESSION)

Avionics Annunciator Dimming
Figure 2

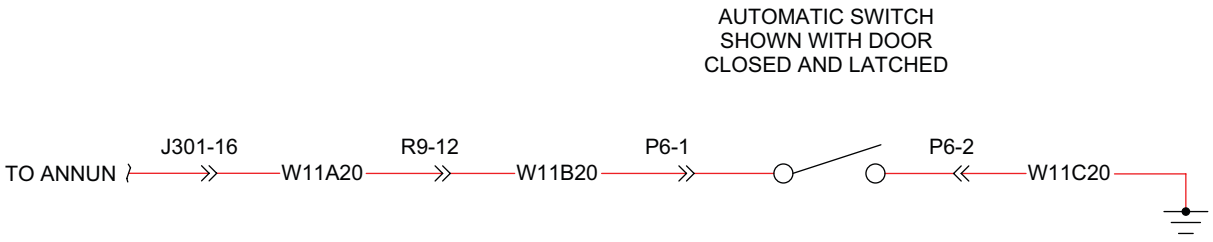
Effectivity
3449161,
3449178 and Up

83363 NEW/G 17.0



Baggage Door Annunciation
Figure 3 (Sheet 1 of 3)

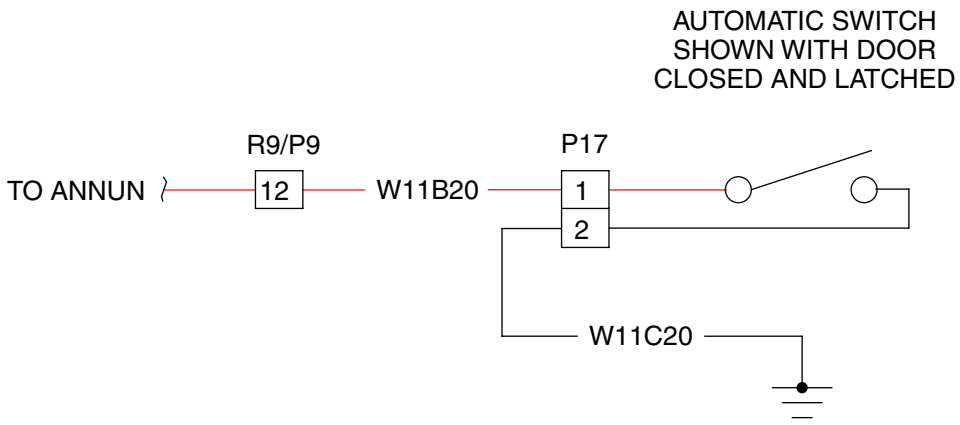
100651 D/T 17
101288 NEW/D 17.0



Baggage Door Annunciation
Figure 3 (Sheet 2 of 3)

Effectivity
3449001 thru 3449151

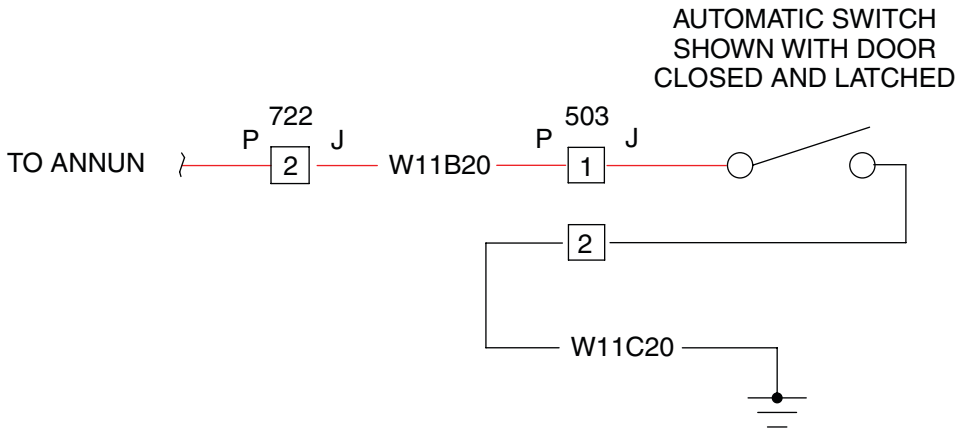
104229 NEW/C
104368 NEW/K 17.0



STANDARD

WITH AVIDYNE ENTEGRA

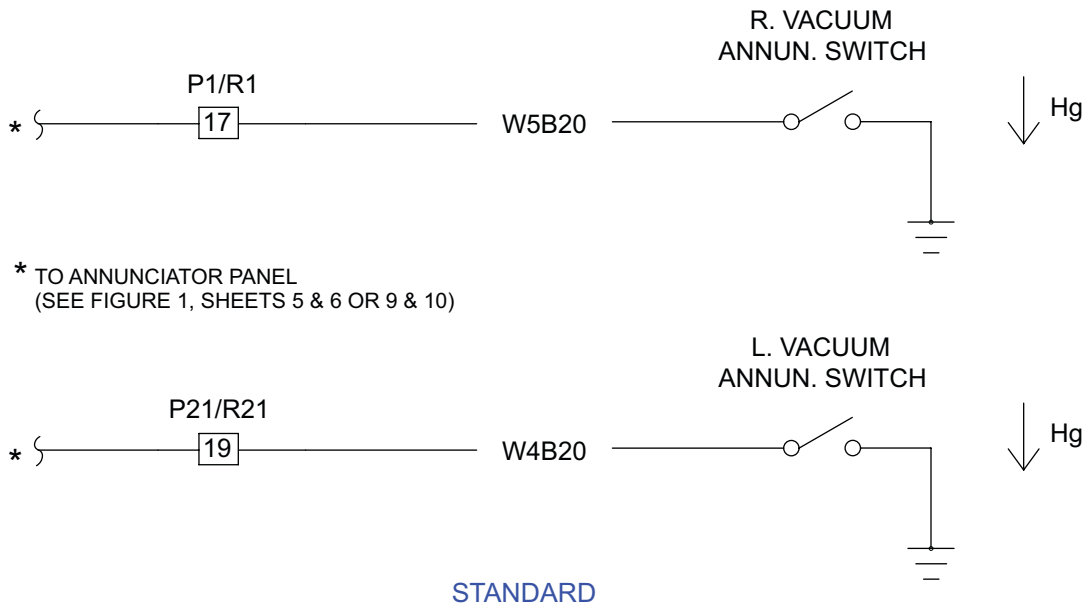
101840 NEW 11.0



Effectivity
3449152 and up

Baggage Door Annunciation
Figure 3 (Sheet 3 of 3)

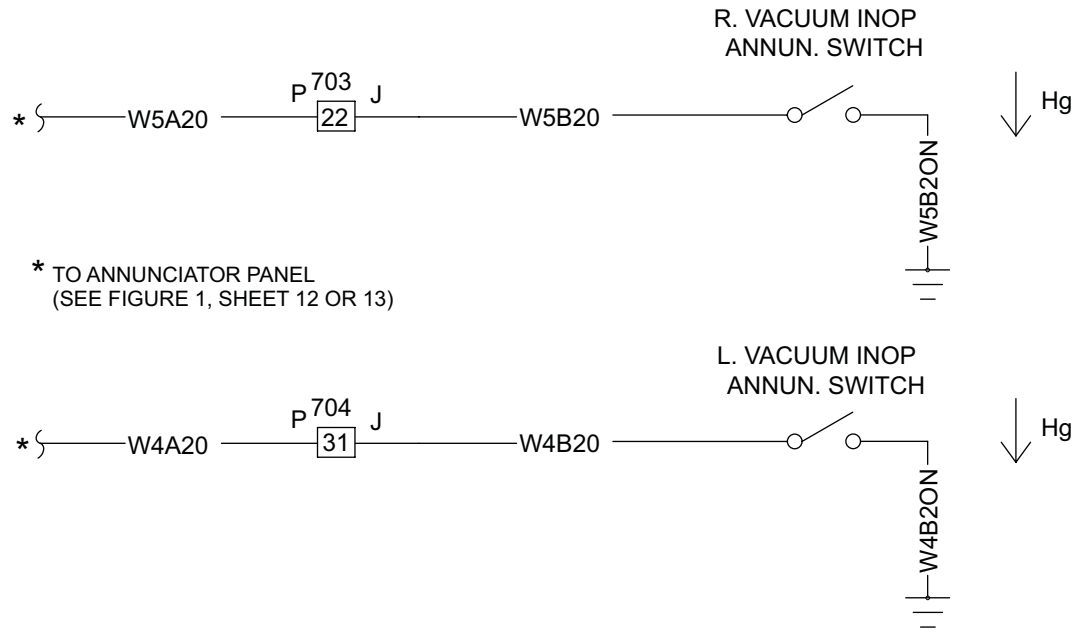
100651 J/T 3
101288 NEW/D 3
104229 NEW/C 3
104368 NEW/K 3.0



STANDARD

WITH AVIDYNE ENTEGRA

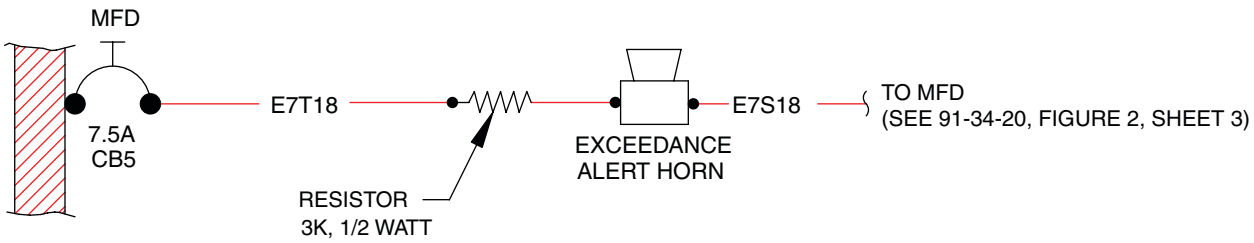
101840 NEW 3.0



Vacuum Annunciators
Figure 4

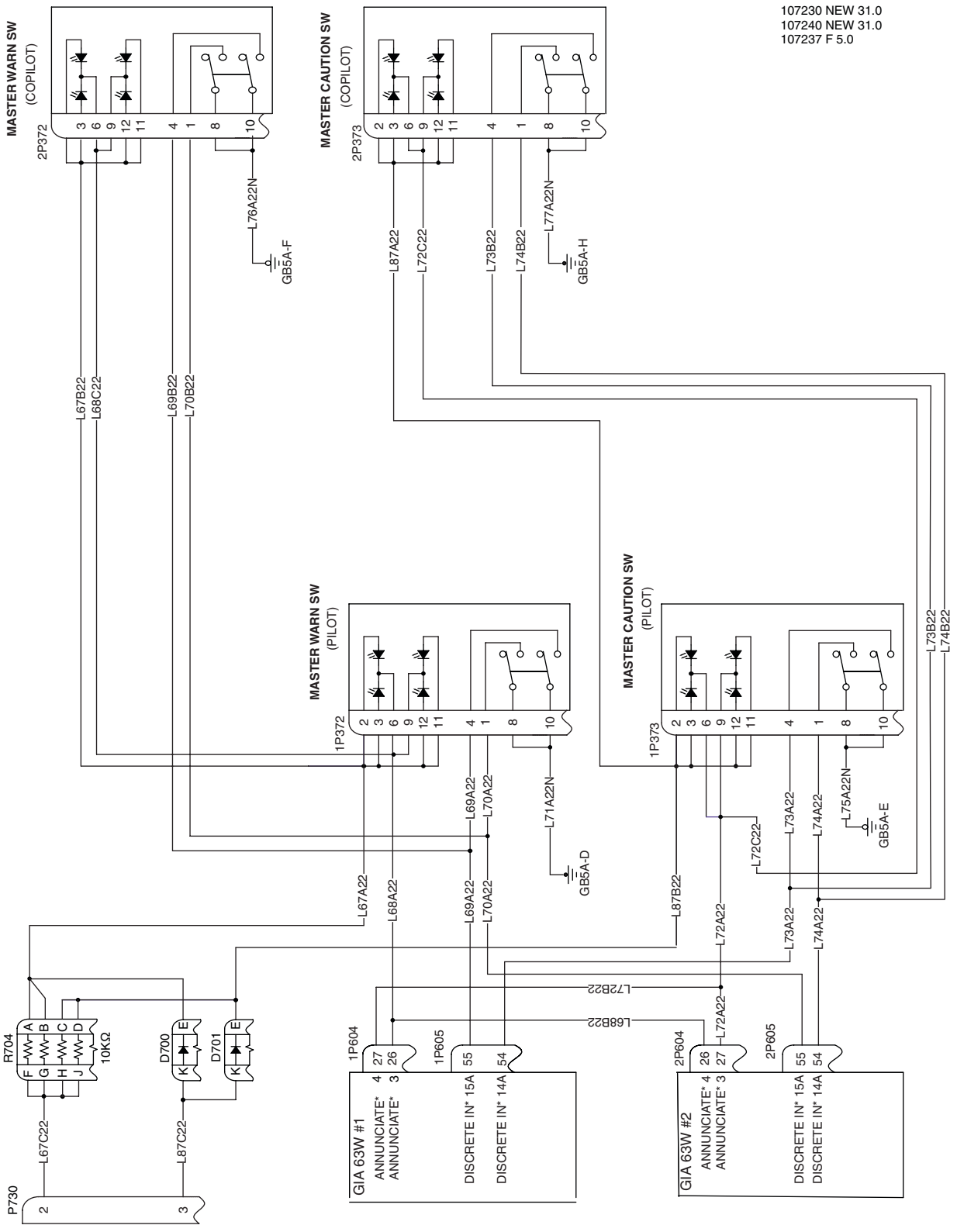
Effectivity
Seneca V

101840 NEW 4.4



Effectivity
Seneca V
with Avidyne Entegra

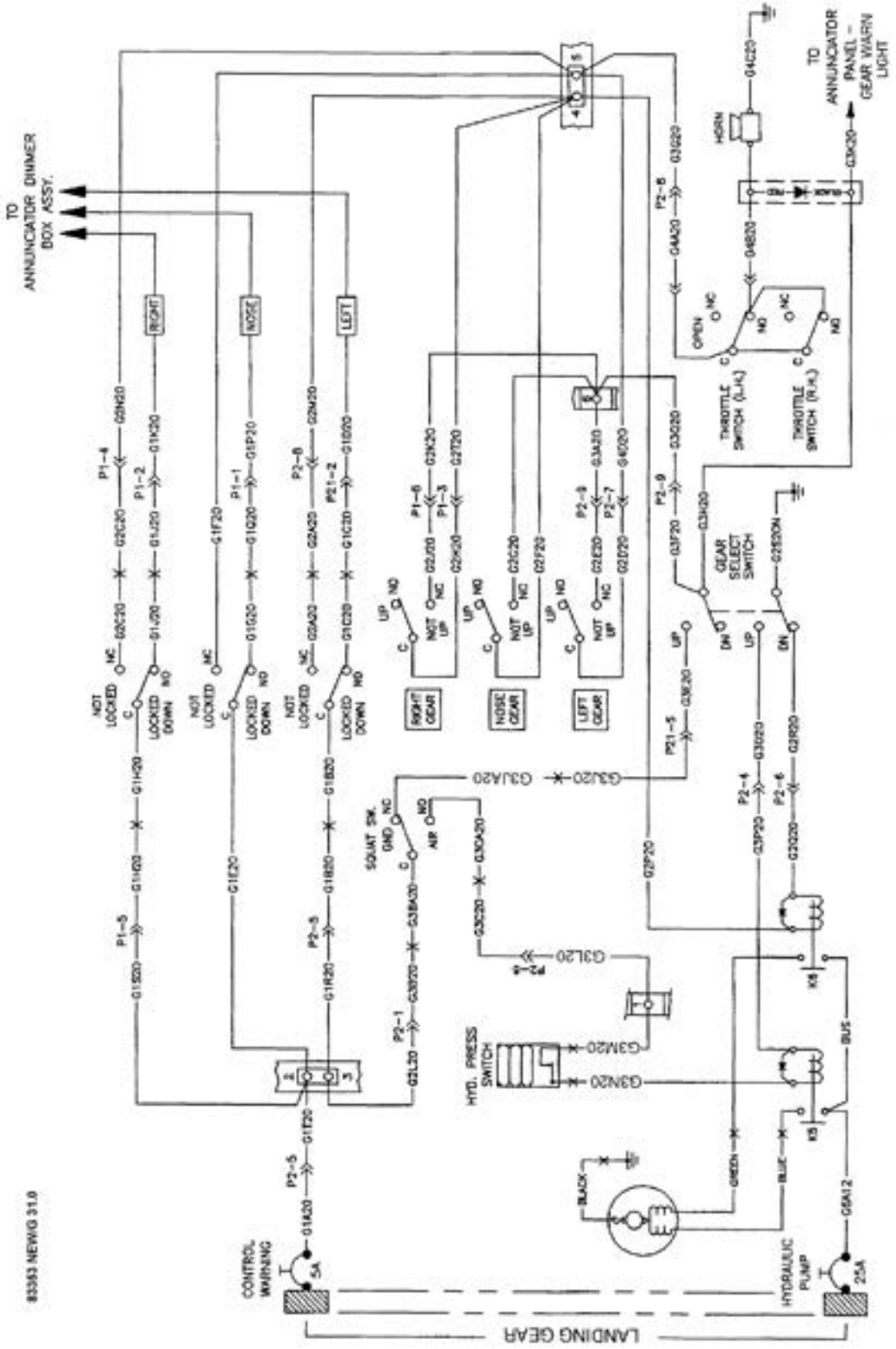
Exceedance Alert DAU Interface
Figure 5



Master Caution/Warning
Figure 6

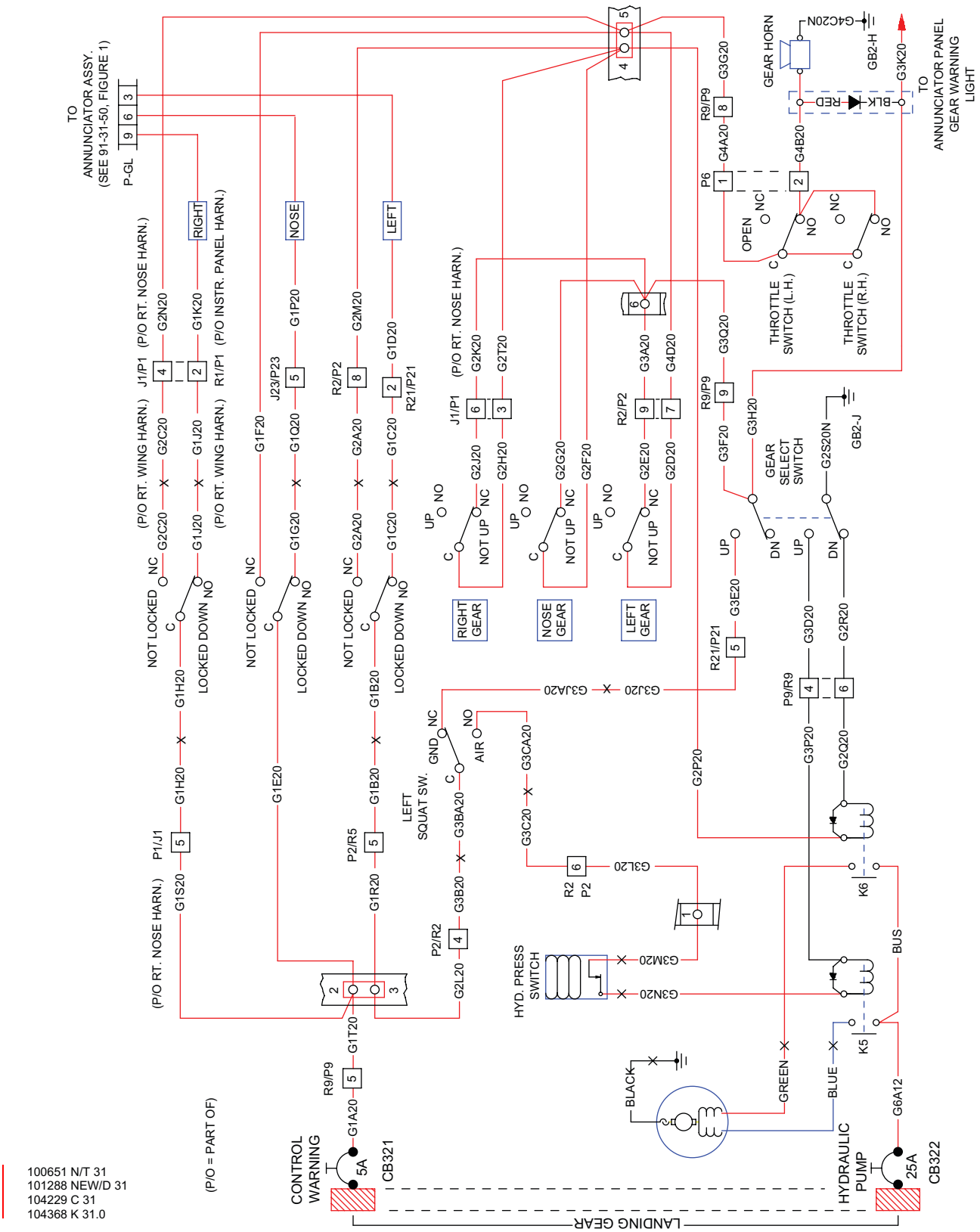
Effectivity
Seneca V
with Garmin G1000

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Landing Gear Control and Warning
Figure 1 (Sheet 1 of 4)

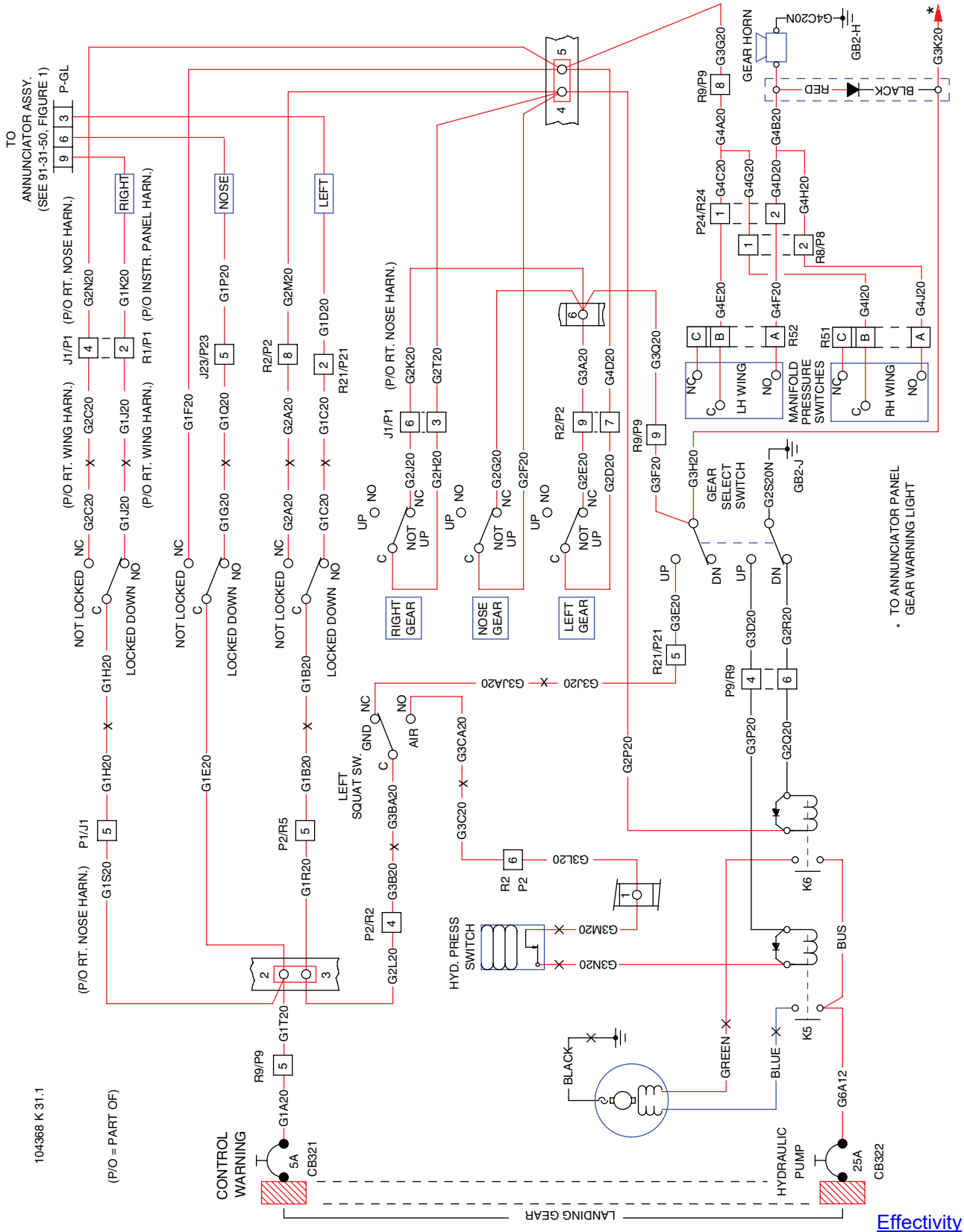
Effectivity
Seneca IV



100651 N/T 31
101288 NEW/D 31
104229 C 31
104368 K 31.0

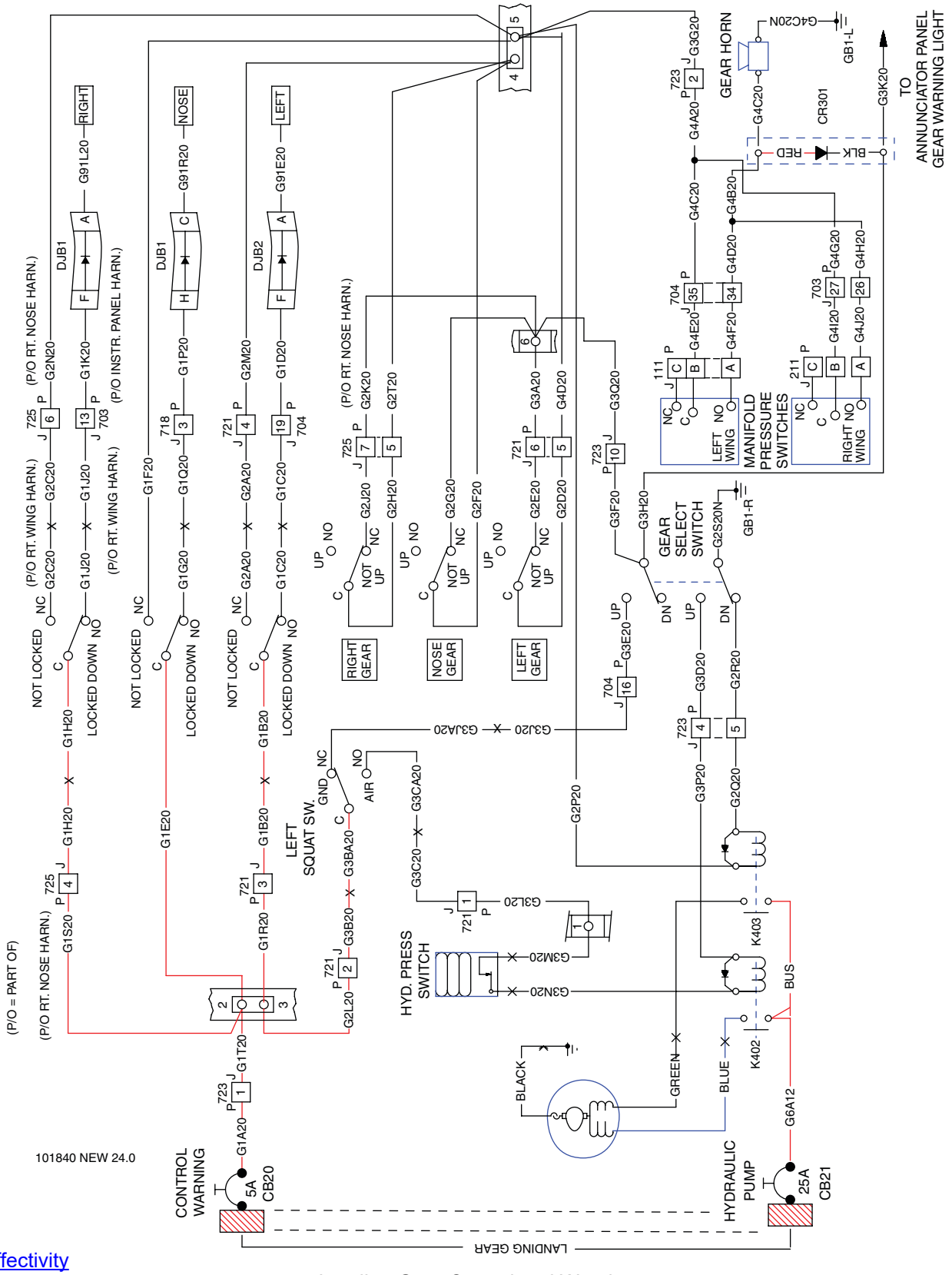
Effectivity
3449001 thru 3449300, less 3449161

Landing Gear Control and Warning
Figure 1 (Sheet 2 of 4)



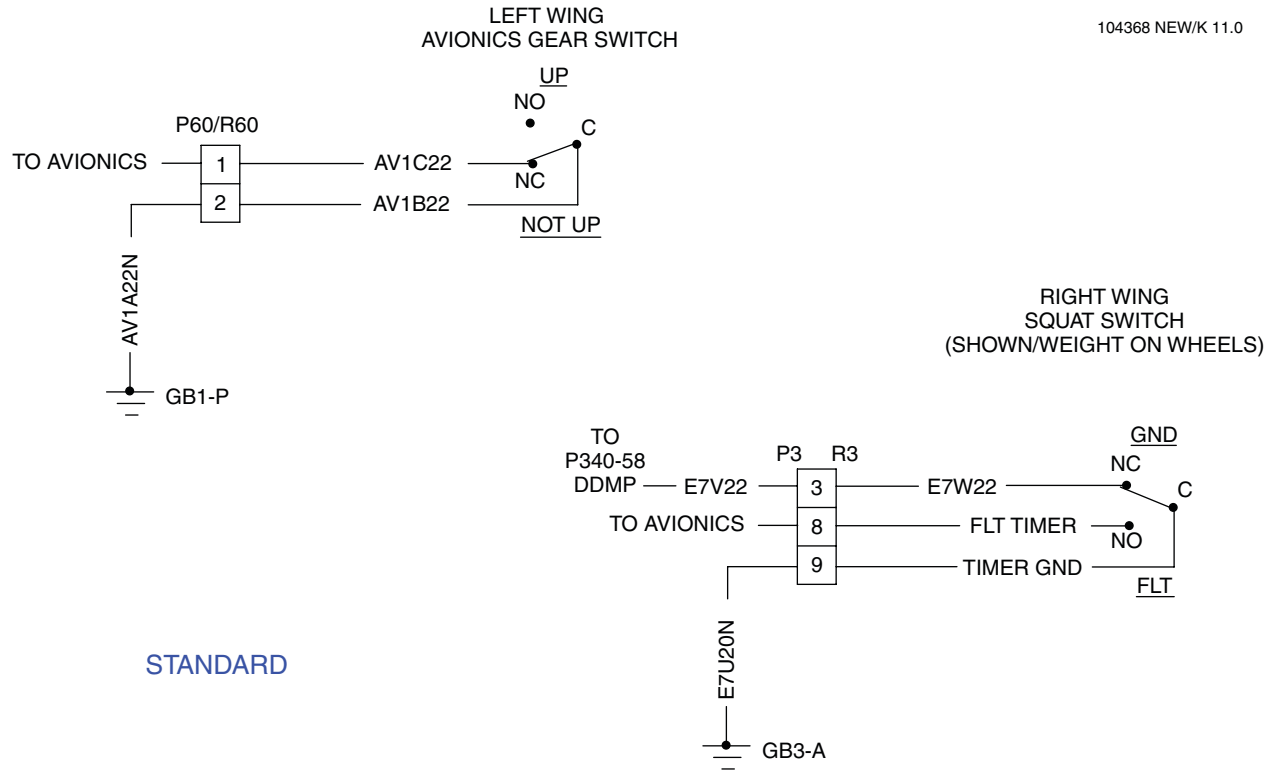
Landing Gear Control and Warning
Figure 1 (Sheet 3 of 4)

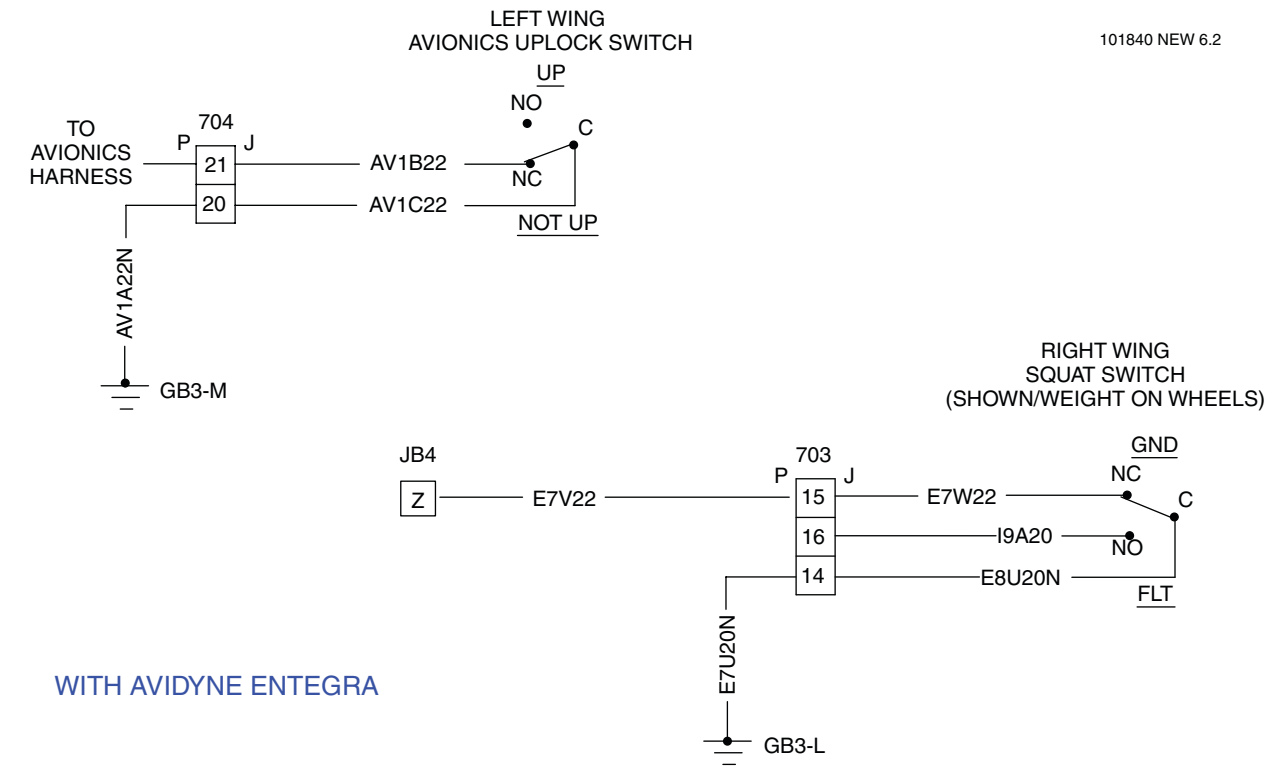
Effectivity
3449161,
3449301 and up



Effectivity
Seneca V
with Avidyne Entegra

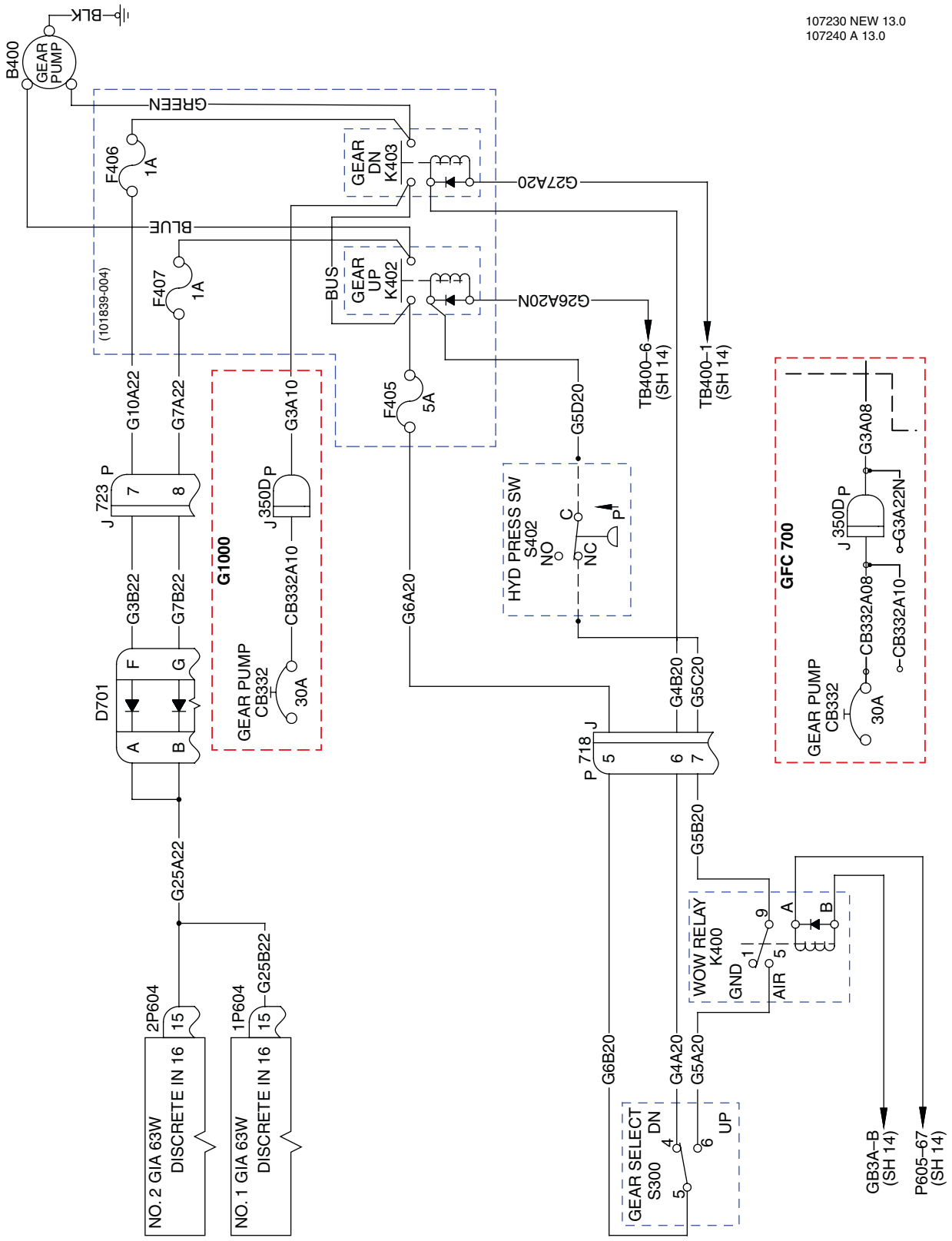
Landing Gear Control and Warning
Figure 1 (Sheet 4 of 4)





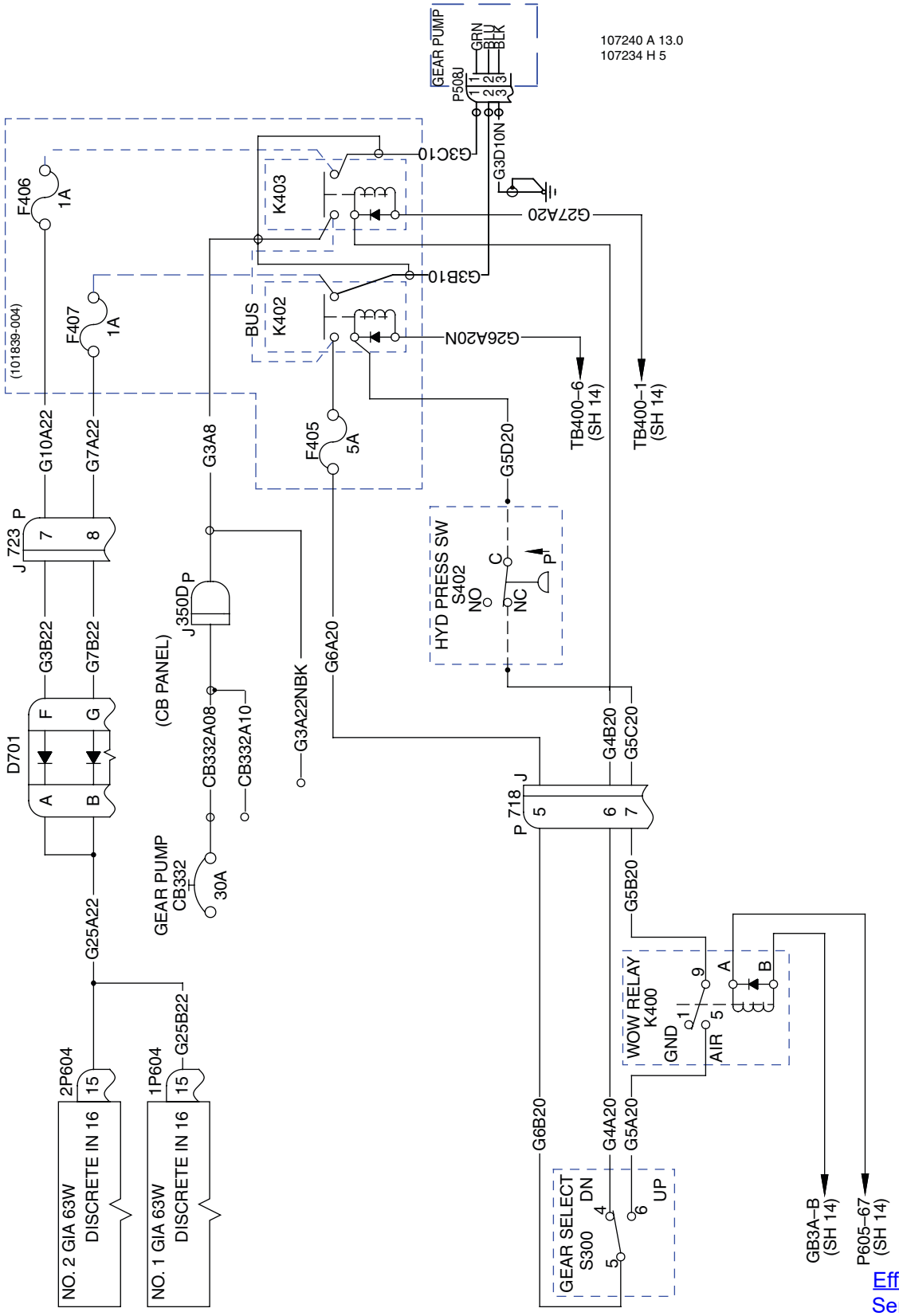
Avionics Gear Switches
Figure 2

Effectivity
3449161,
3449178 and Up



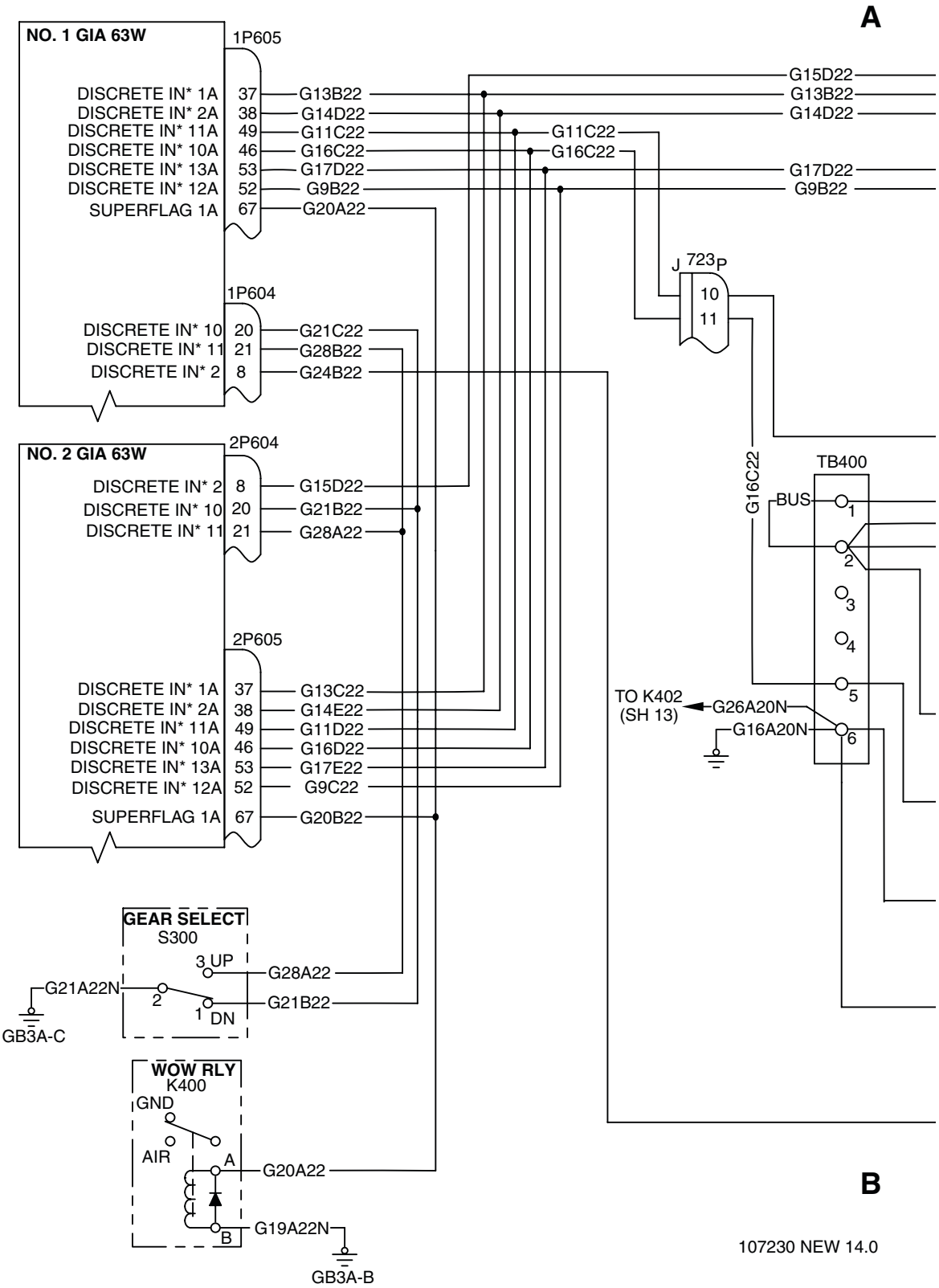
Effectivity
Seneca V
with Garmin G1000

Landing Gear Indicators
Figure 3 (Sheet 1 of 2)



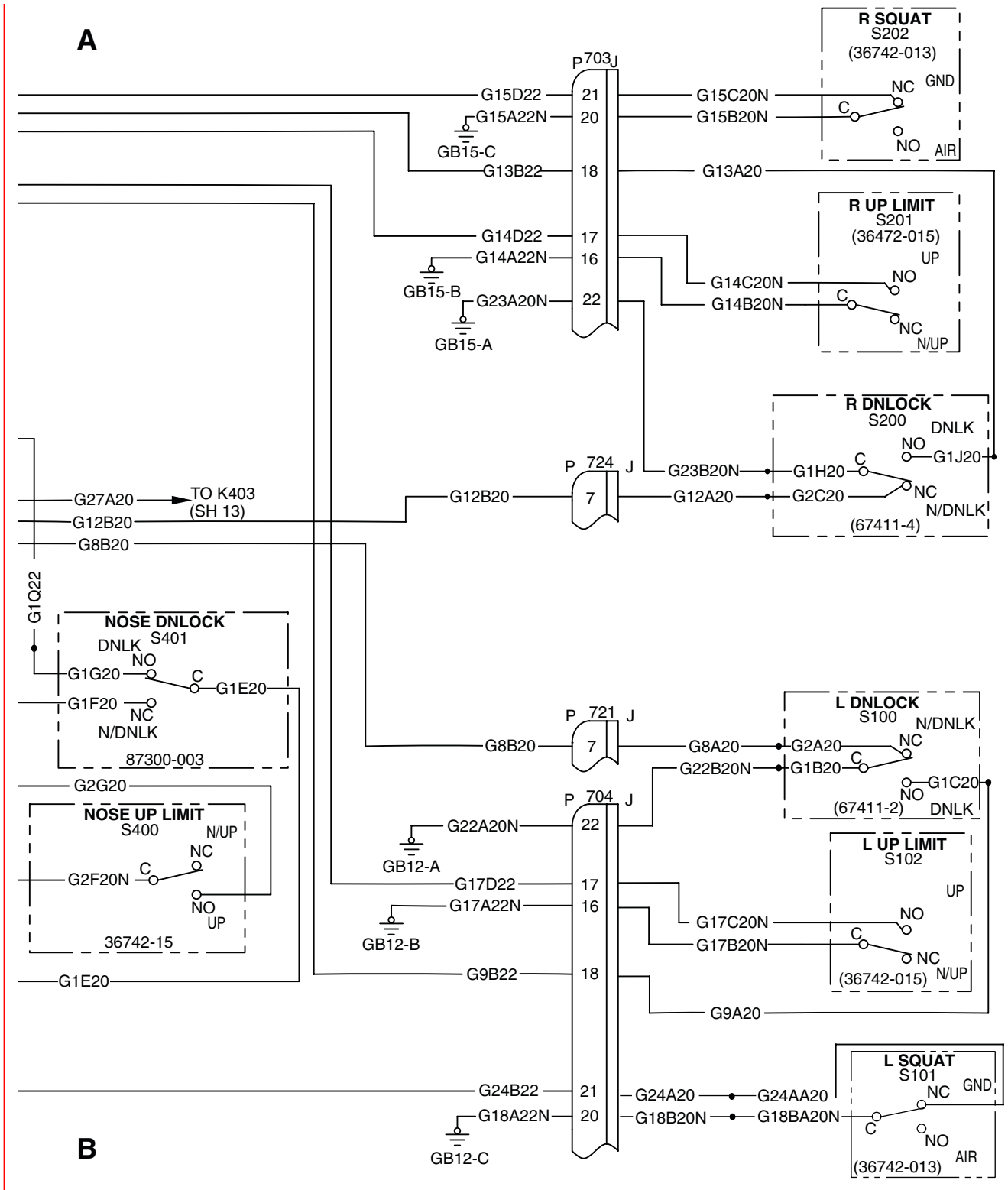
Landing Gear Indicators
Figure 3 (Sheet 2 of 2)

Effectivity
Seneca V
3449459, 3449492 - 3449508
with Garmin G1000 / GFC 700



Effectivity
Seneca V
3449459, 3449467 and Up
with Garmin G1000

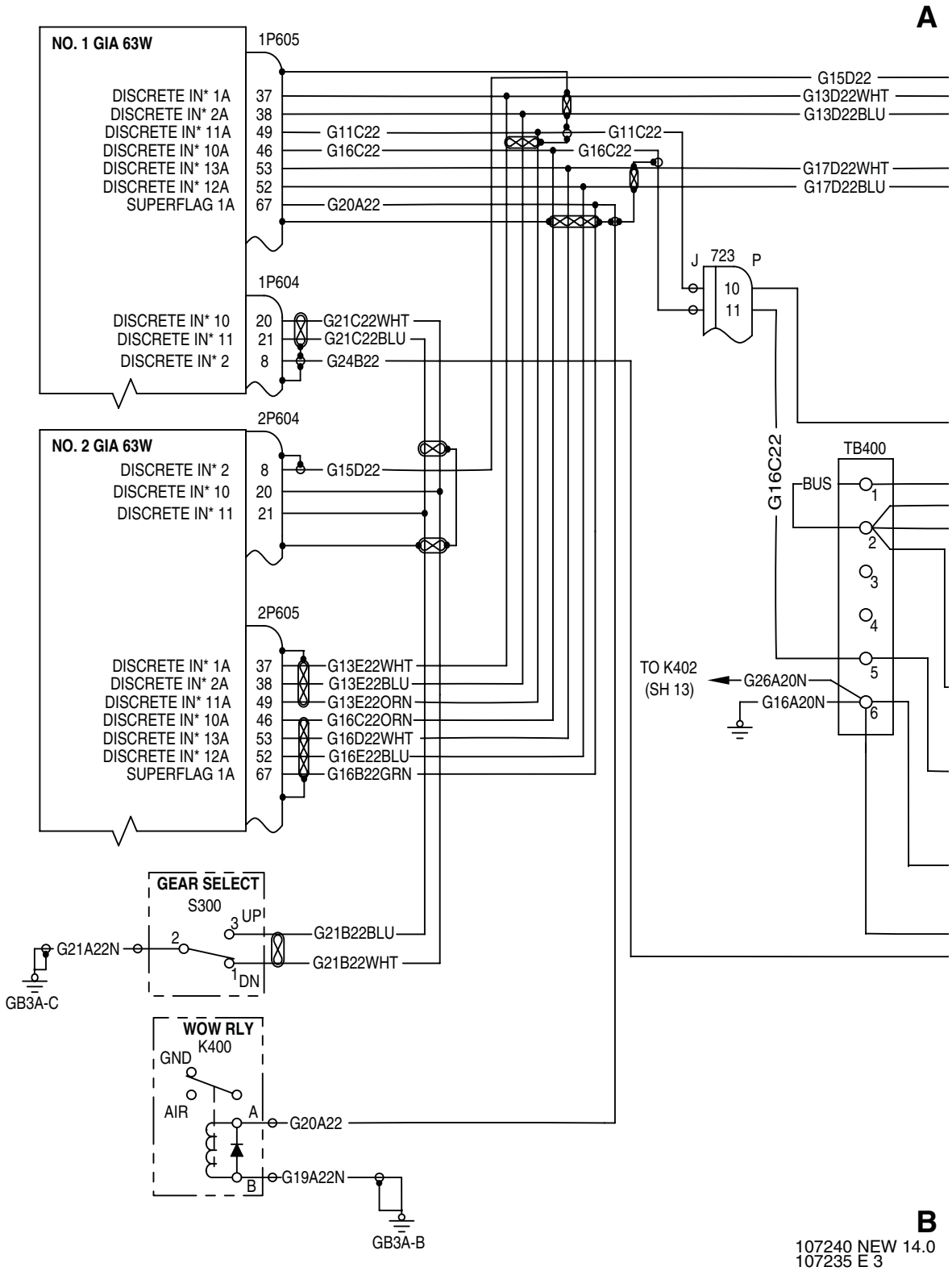
Landing Gear Control and Indicators
Figure 4 (Sheet 1 of 8)



107230 NEW 14.0

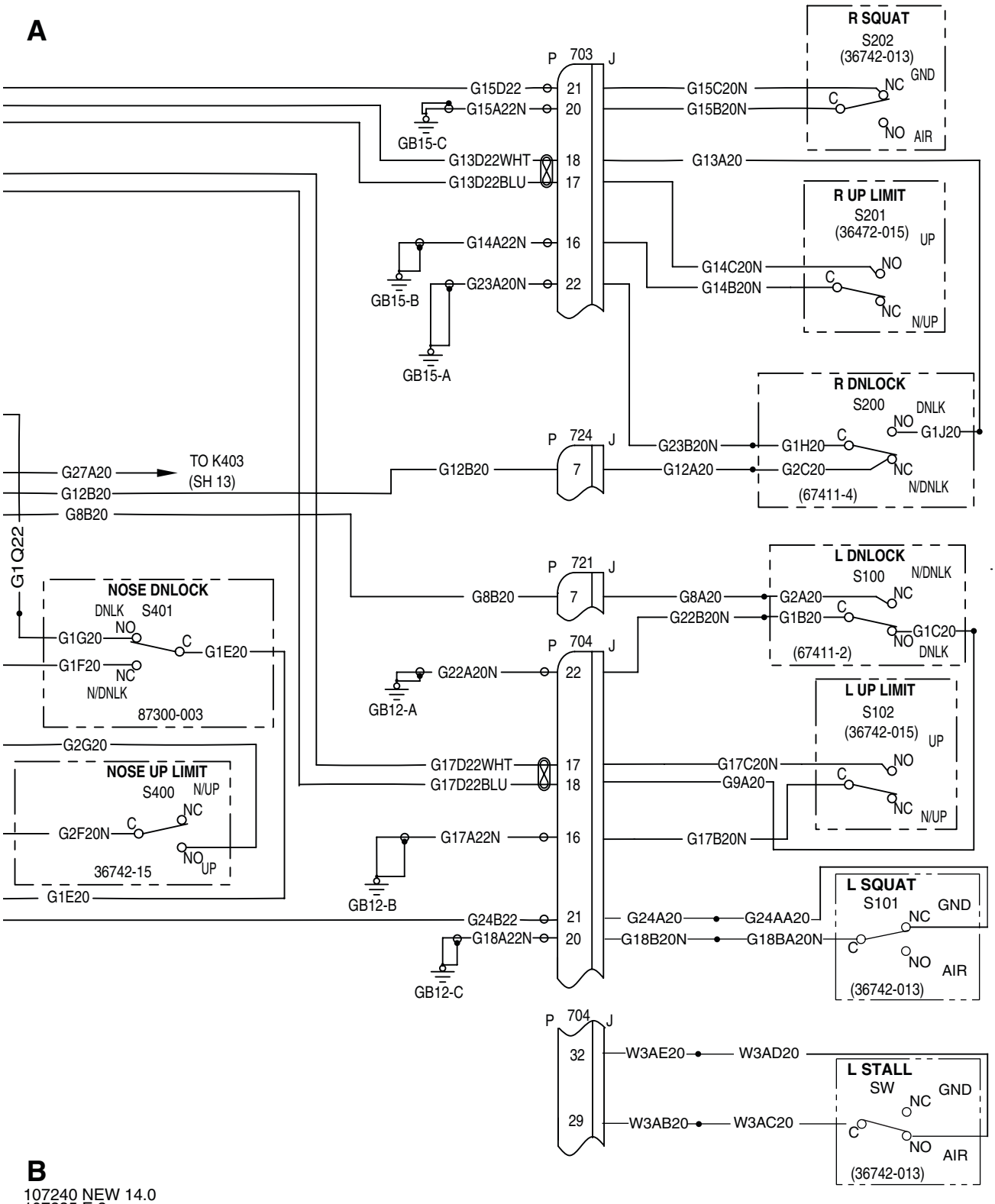
Landing Gear Control and Indicators
Figure 4 (Sheet 2 of 8)

Effectivity
Seneca V
3449459, 3449467 and Up
with Garmin G1000



Effectivity
Seneca V
3449459, 3449492 - 3449508
with Garmin G1000 / GFC 700

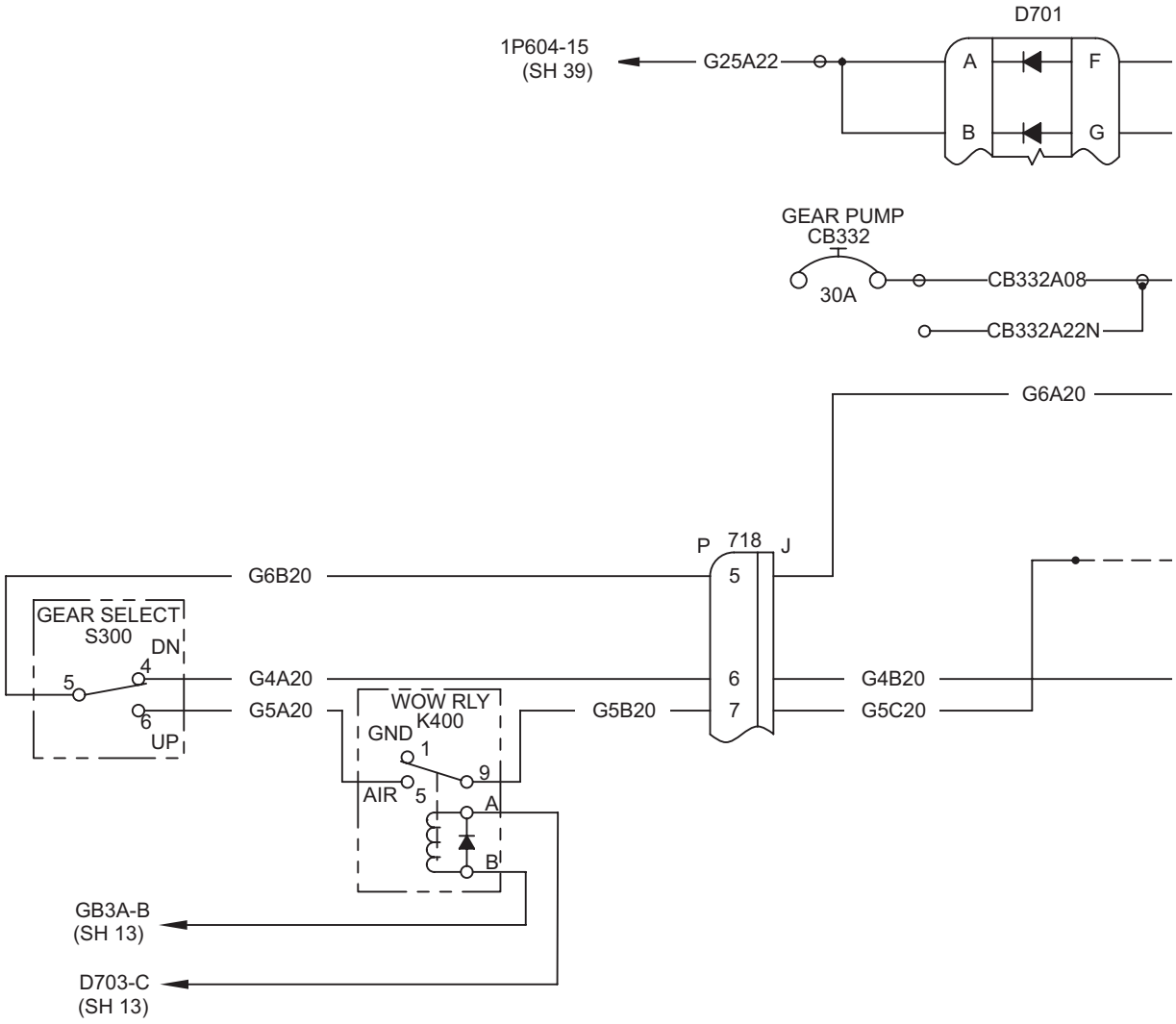
Landing Gear Control and Indicators
Figure 4 (Sheet 3 of 8)



Landing Gear Control and Indicators
Figure 4 (Sheet 4 of 8)

Effectivity
Seneca V
3449459, 3449492 - 3449508
with Garmin G1000 / GFC 700

A



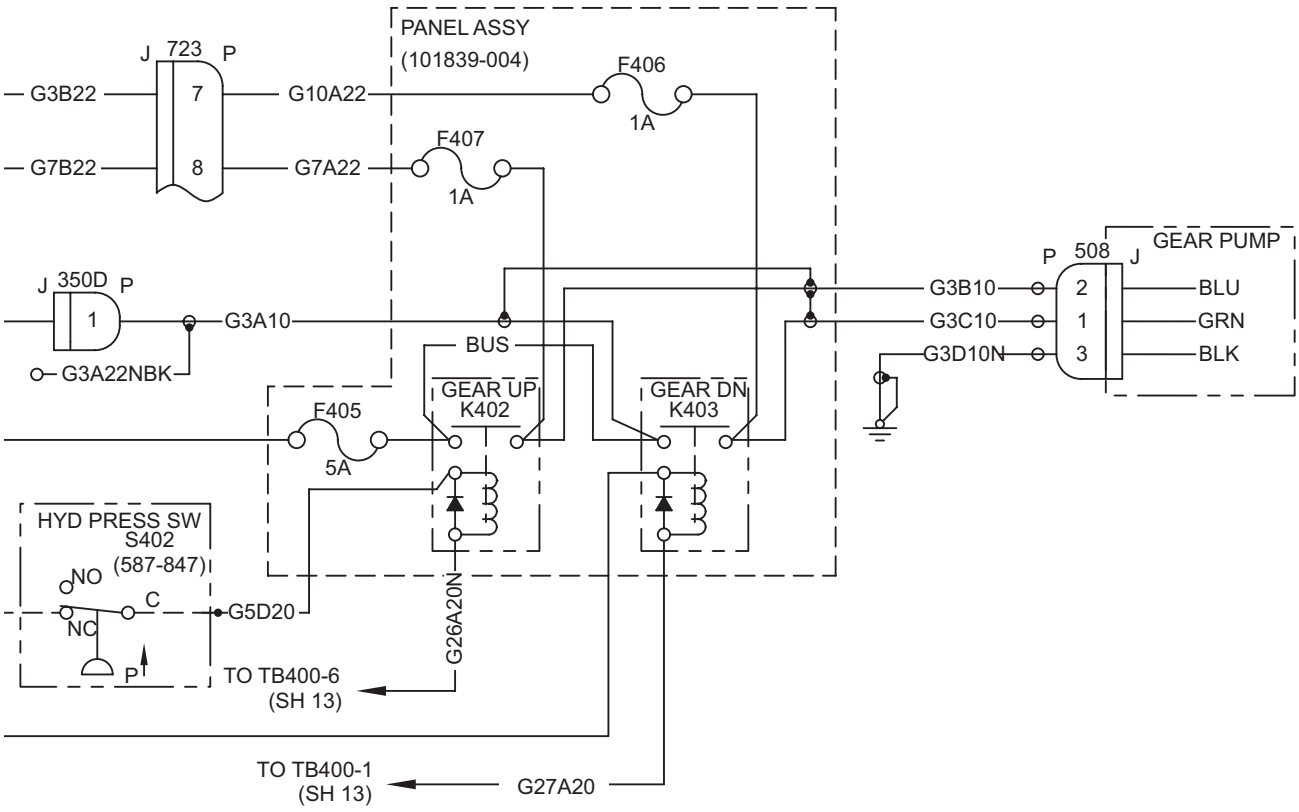
B

34F24A020 NEW 12.0
34F24A002 A 3.0

Effectivity
Seneca V
with Garmin G1000 NXi

Landing Gear Control and Indicators
Figure 4 (Sheet 5 of 8)

A

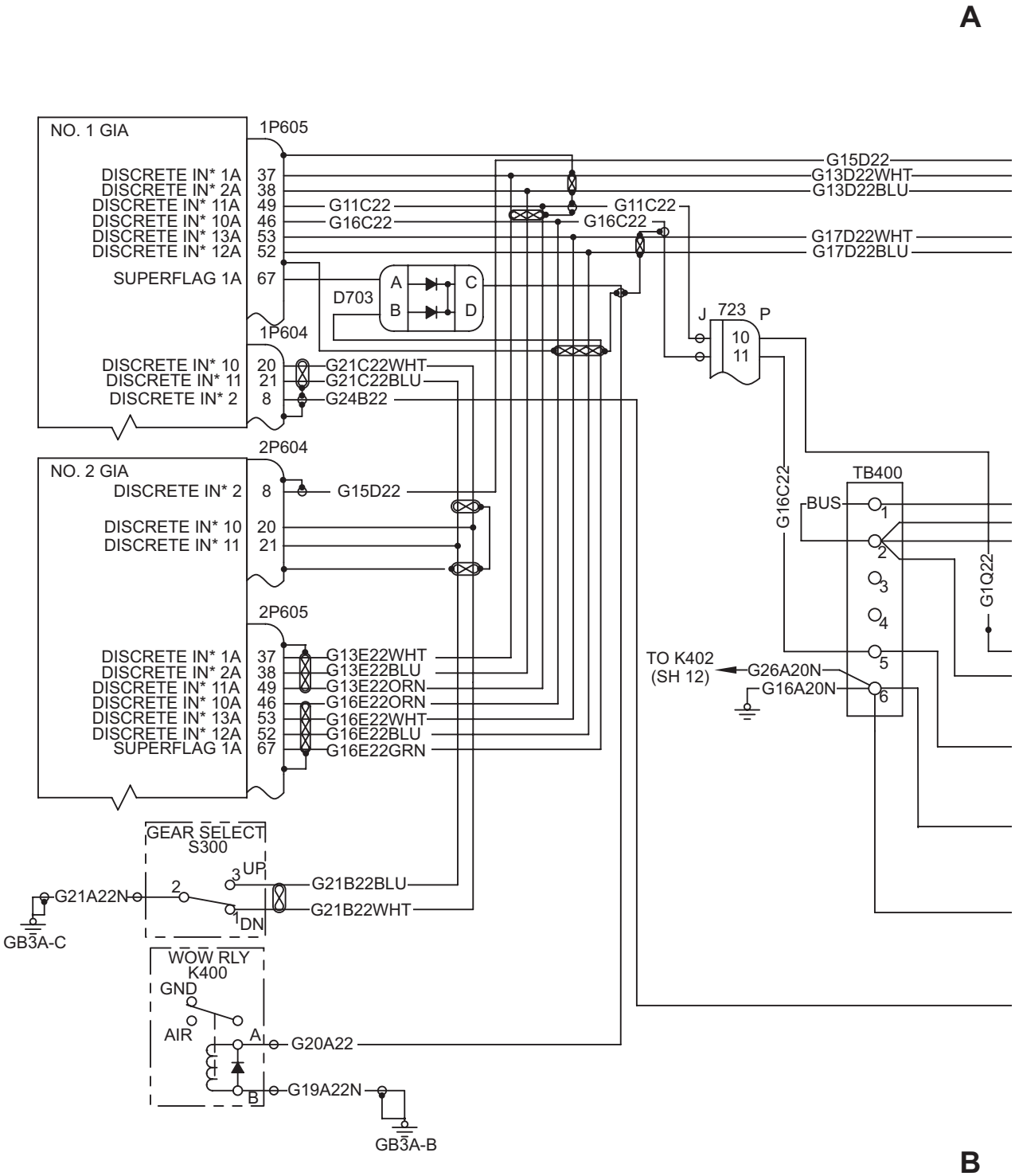


B

34F24A020 NEW 12.0
34F24A002 A 3.0

Landing Gear Control and Indicators
Figure 4 (Sheet 6 of 8)

Effectivity
Seneca V
with Garmin G1000 NXi

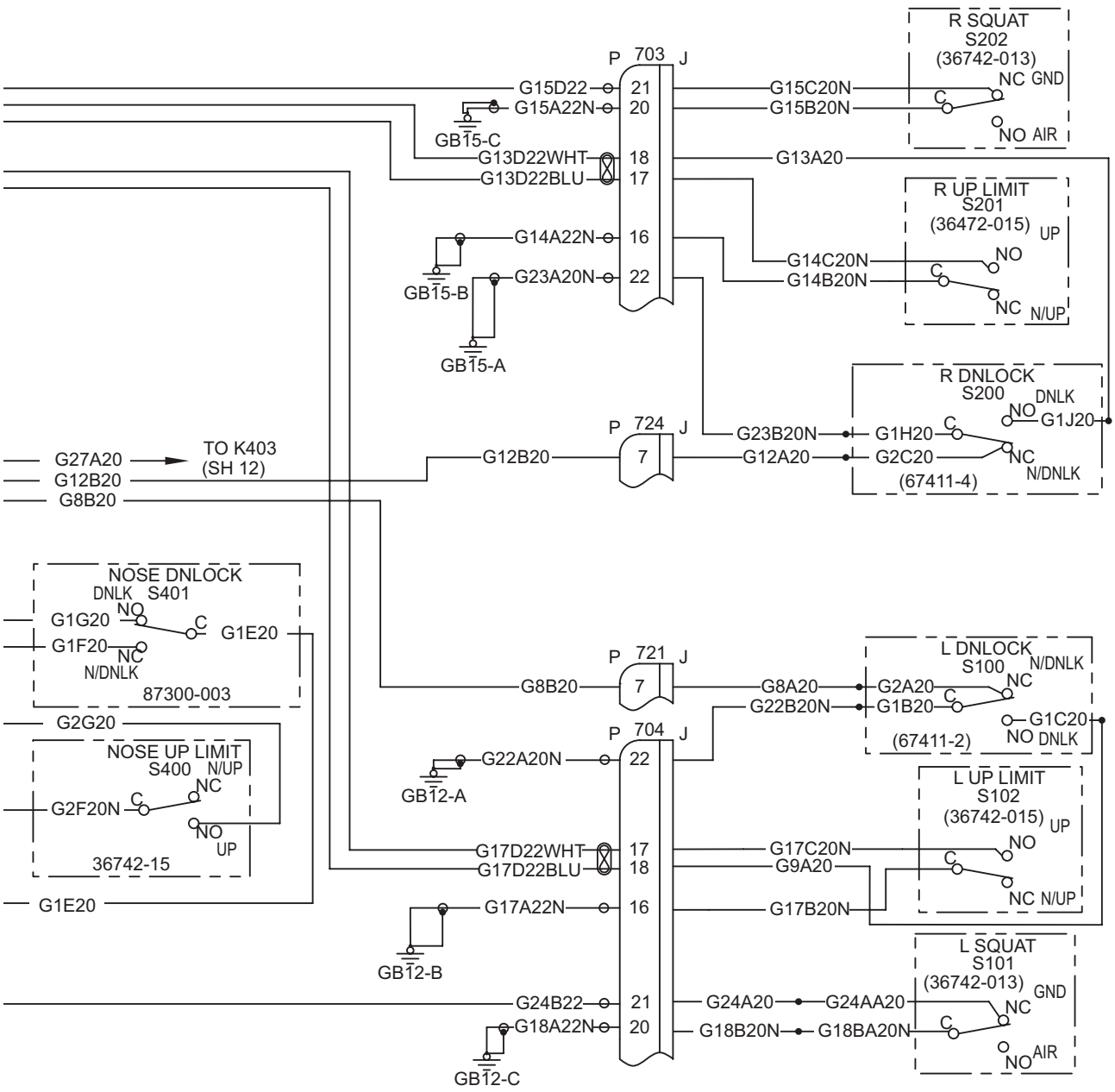


34F24A020 NEW 13
34W24A001 NEW 3, 7

Effectivity
Seneca V
with Garmin G1000 NXi

Landing Gear Control and Indicators
Figure 4 (Sheet 7 of 8)

A



B

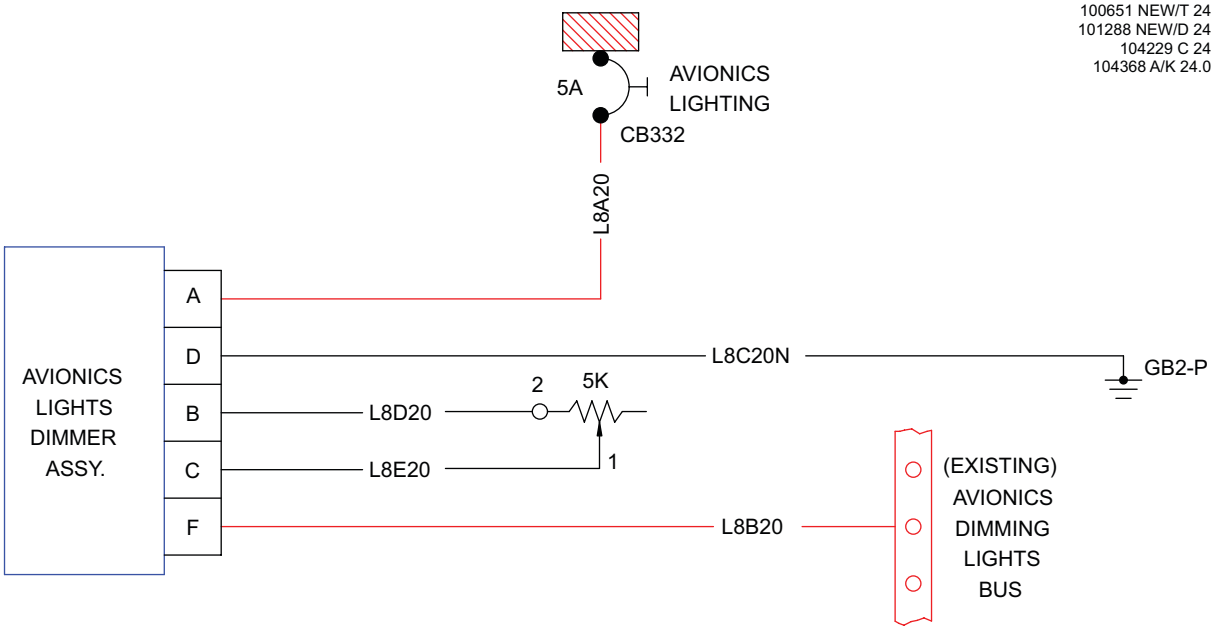
34F24A020 NEW 13
34W24A001 NEW 3, 7

Landing Gear Control and Indicators
Figure 4 (Sheet 8 of 8)

Effectivity
Seneca V
with Garmin G1000 NXi

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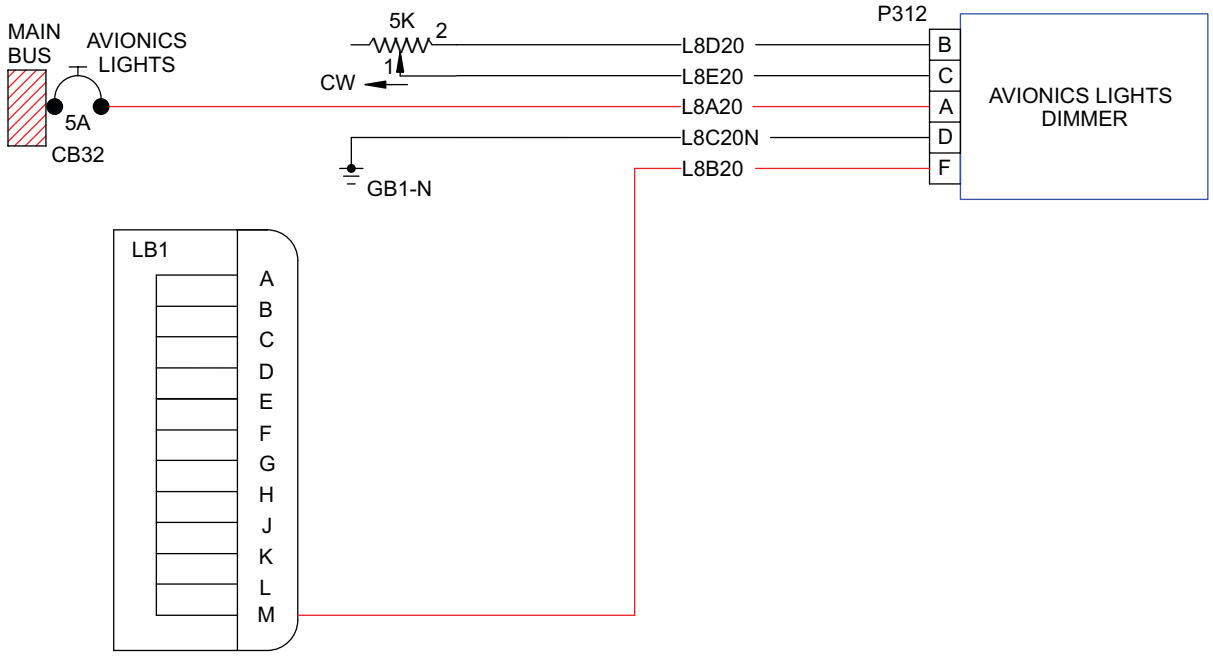
83353 NEW/G 24
100651 NEW/T 24
101288 NEW/D 24
104229 C 24
104368 A/K 24.0



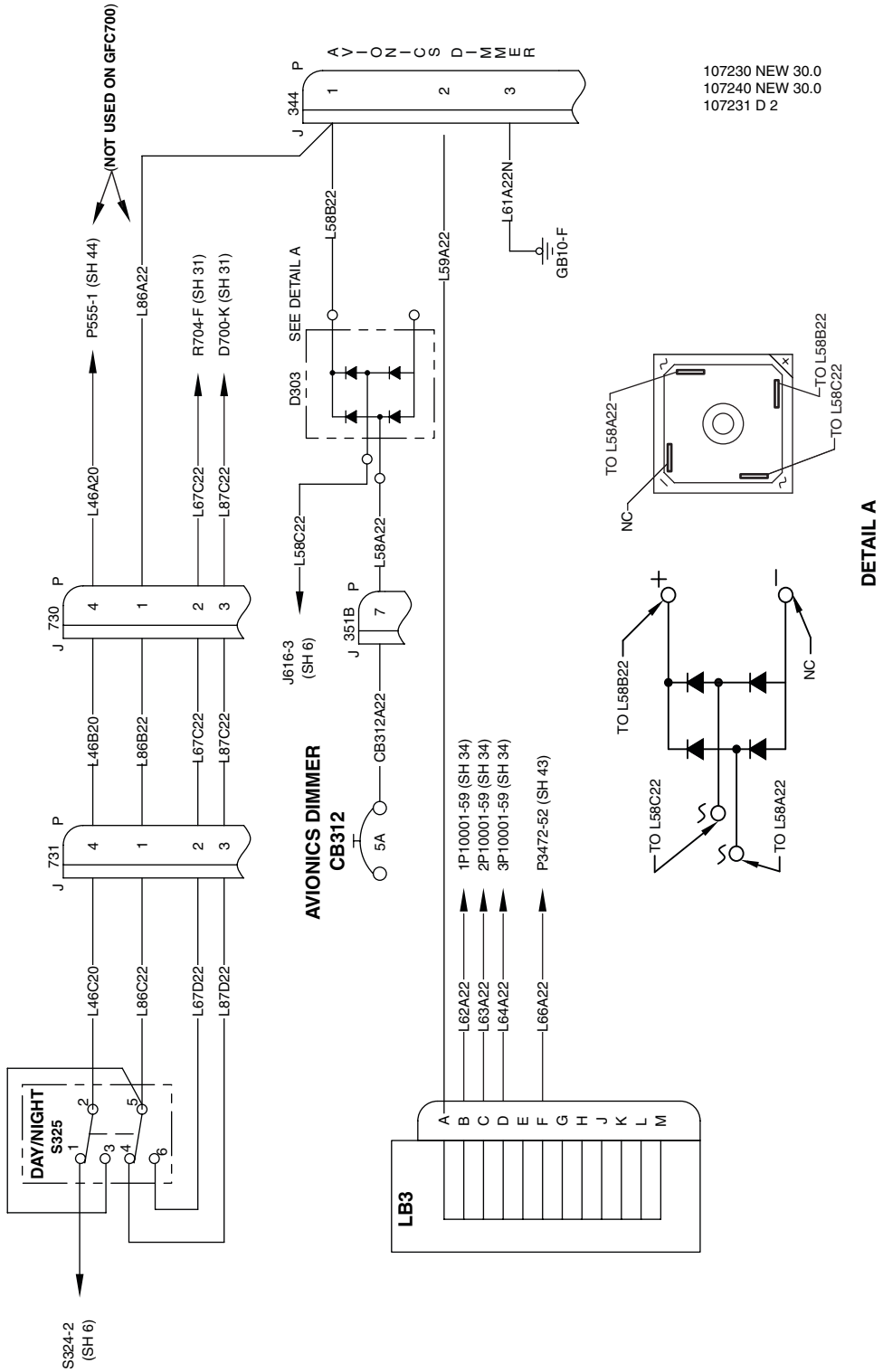
STANDARD

WITH AVIDYNE ENTEGRA

101840 NEW 17.0



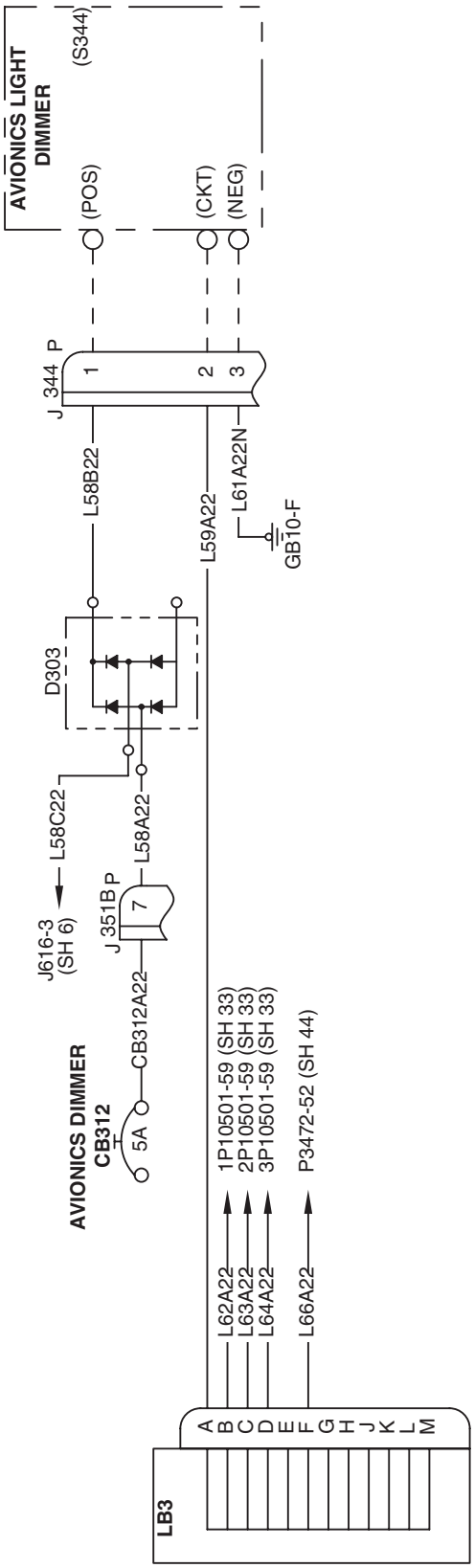
Avionics Lights Dimming
Figure 1 (Sheet 1 of 3)



Effectivity
Seneca V
with Garmin G1000

Avionics Lights Dimming
Figure 1 (Sheet 2 of 3)

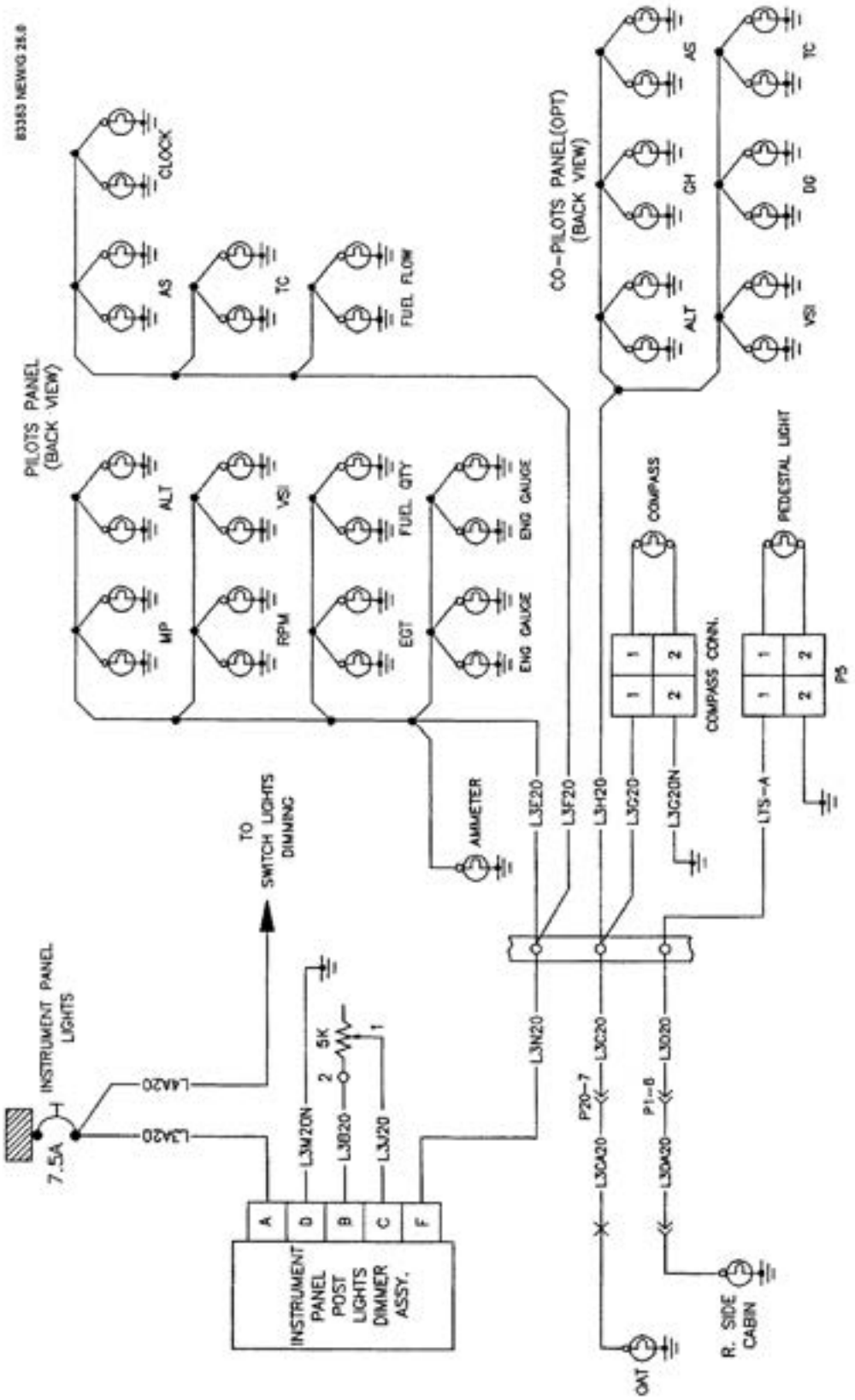
34F24A020 NEW 29



Avionics Lights Dimming
Figure 1 (Sheet 3 of 3)

Effectivity
Seneca V
with Garmin G1000 NXi

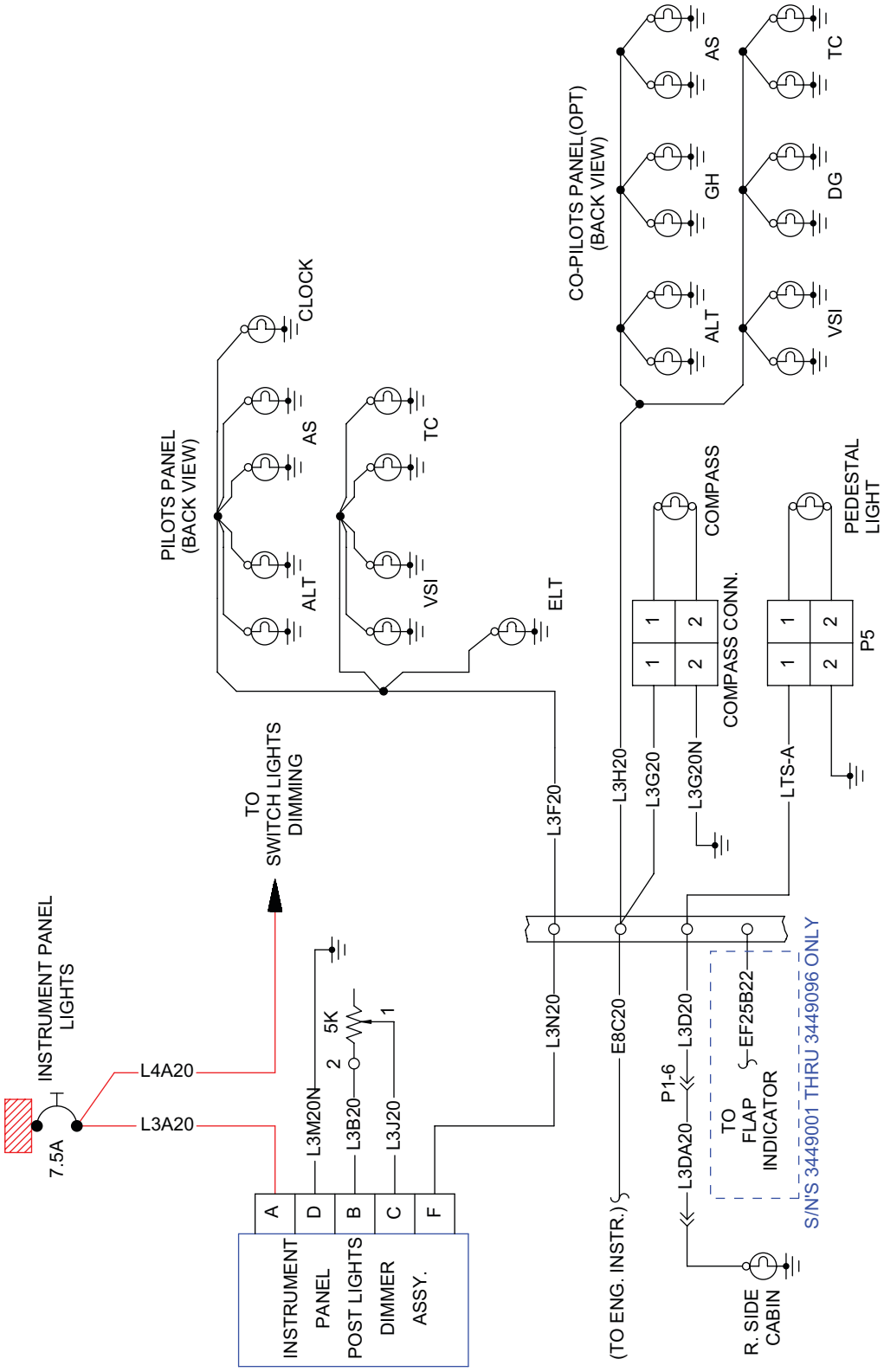
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Instrument Panel Post Lights
Figure 2 (Sheet 1 of 5)

Effectivity
Seneca IV

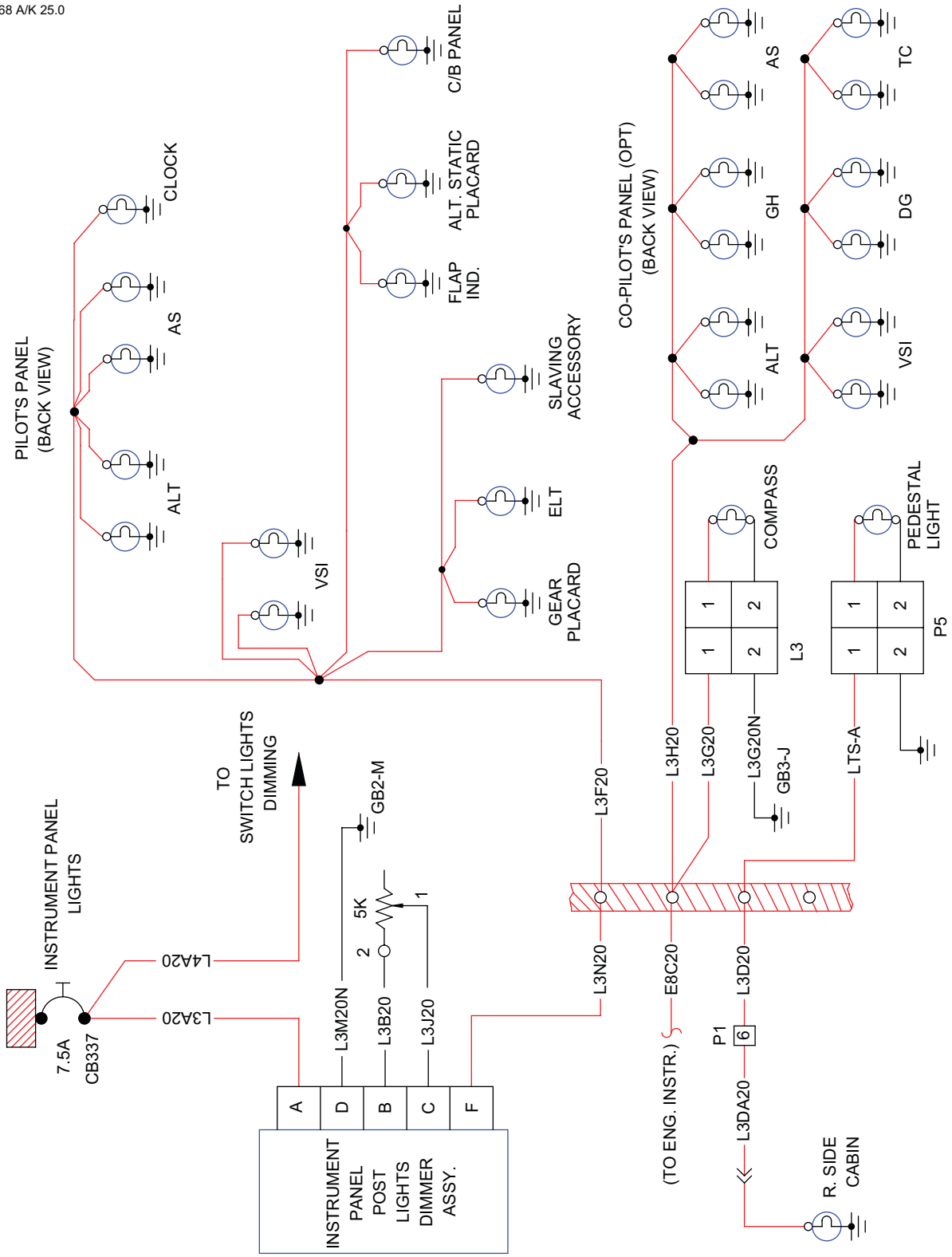
100651 K/T 25
101288 A/D 25.0



Effectivity
3449001 thru 3449151

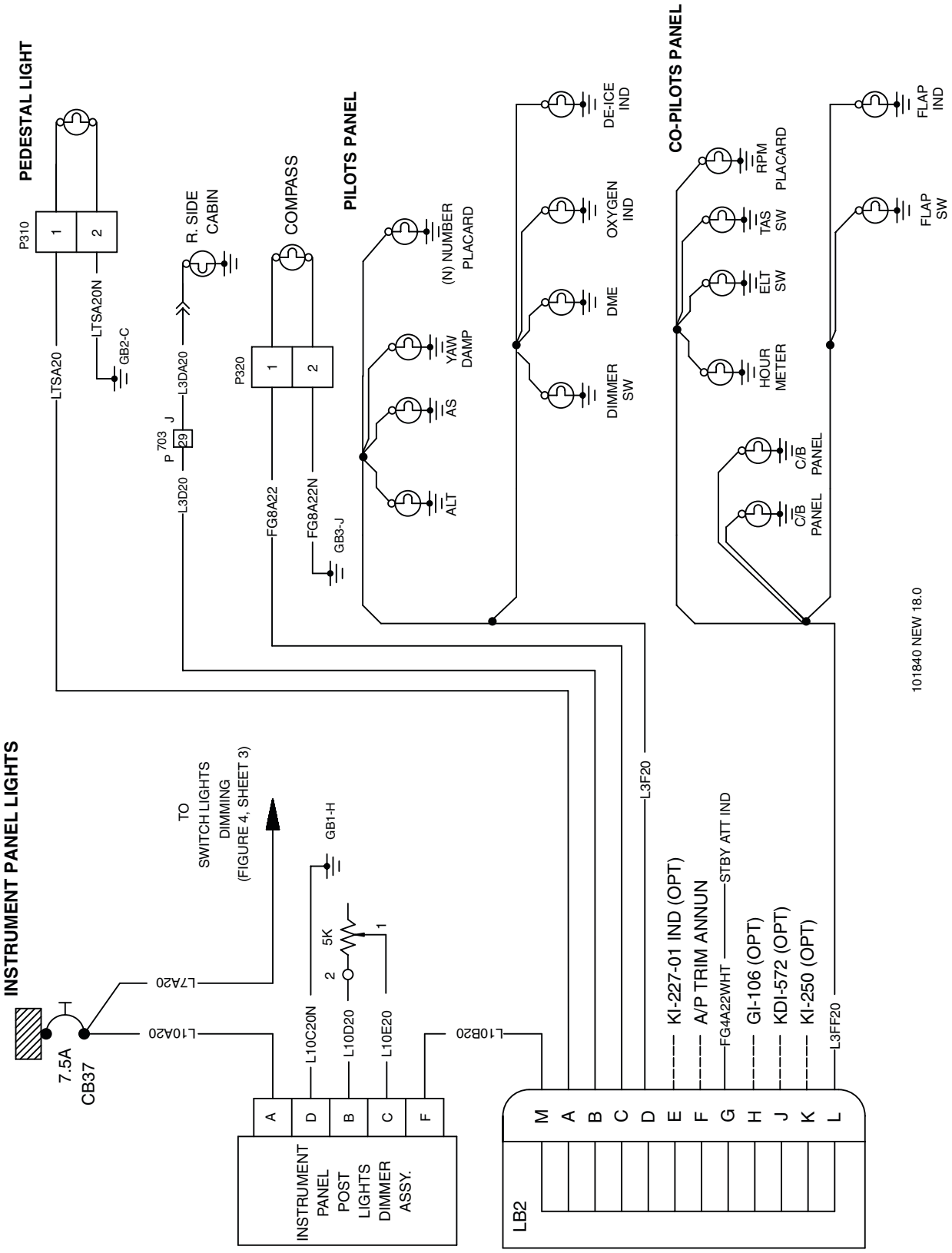
Instrument Panel Post Lights
Figure 2 (Sheet 2 of 5)

104229 C 25
104368 A/K 25.0



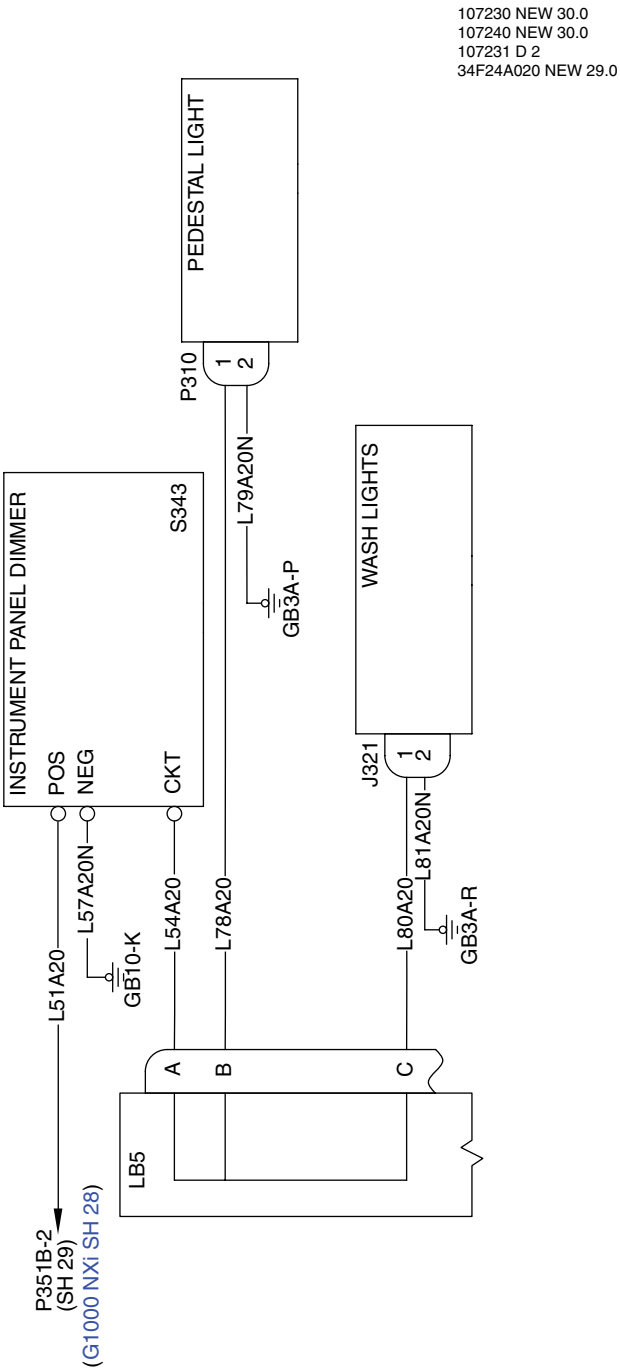
Instrument Panel Post Lights
Figure 2 (Sheet 3 of 5)

Effectivity
3449152 and up



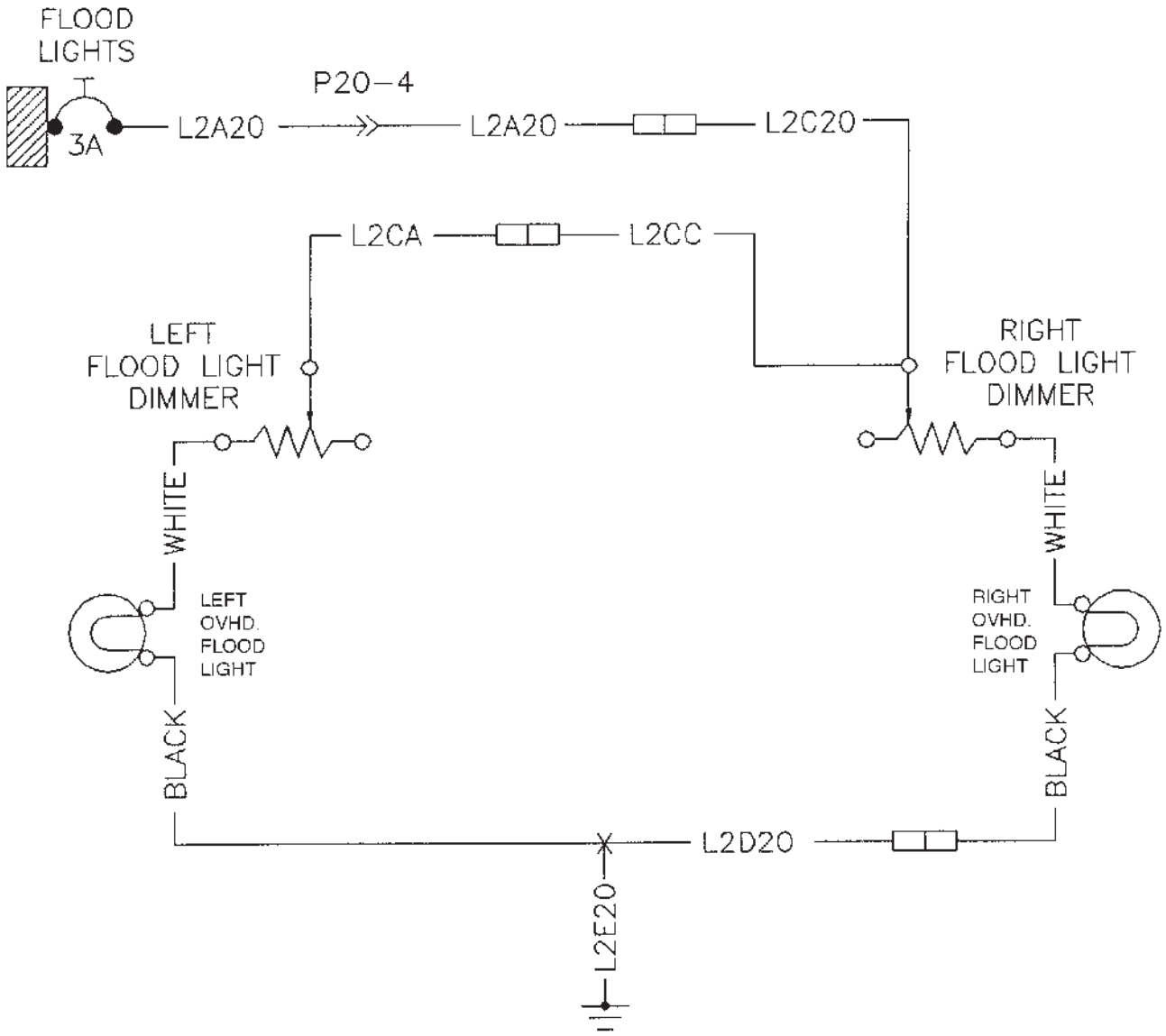
Instrument Panel Post Lights
Figure 2 (Sheet 4 of 5)

Effectivity
Seneca V
with Avidyne Entegra



Instrument Panel Post Lights
Figure 2 (Sheet 5 of 5)

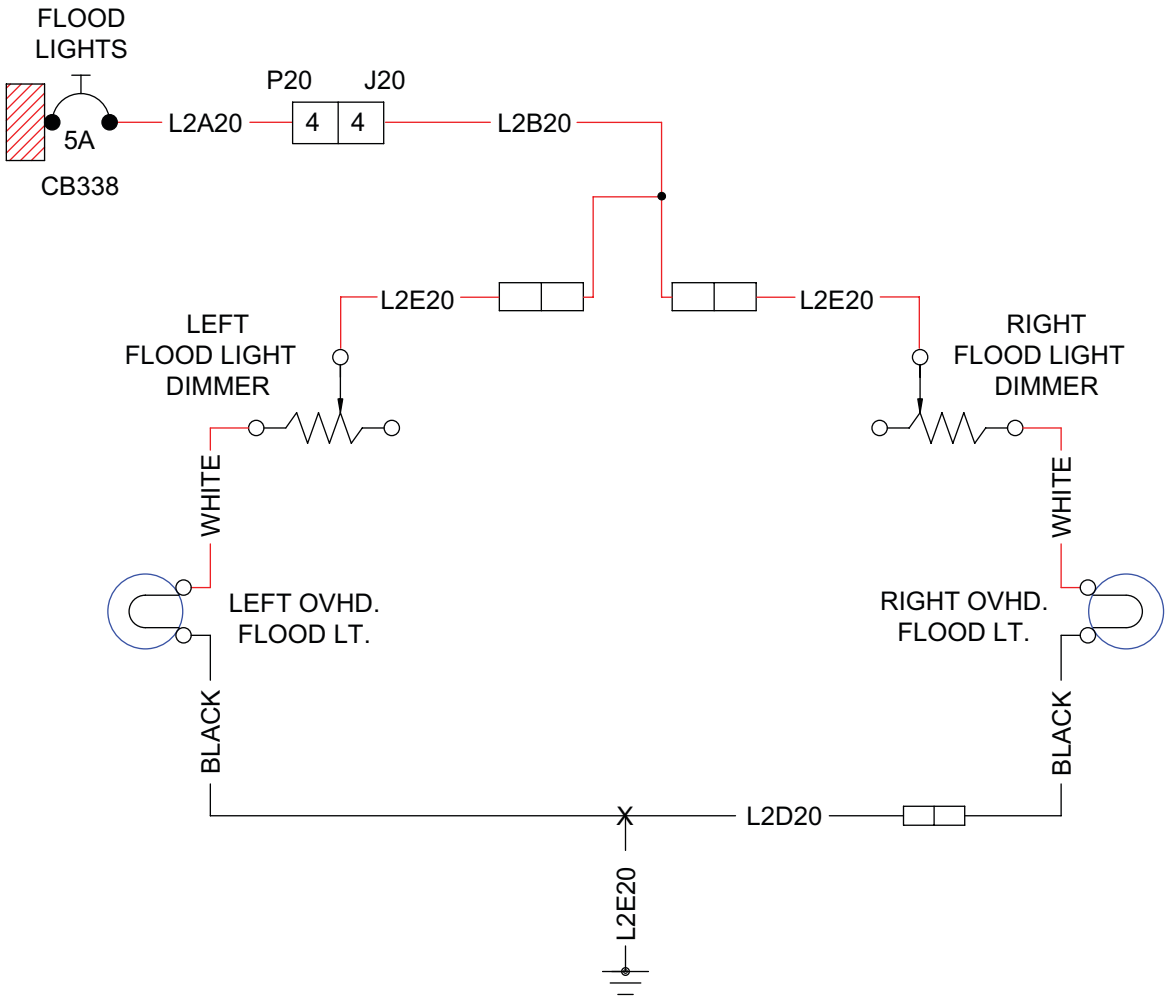
83353 NEW/G 21.0



Effectivity
Seneca IV

Overhead Flood Lights
Figure 3 (Sheet 1 of 3)

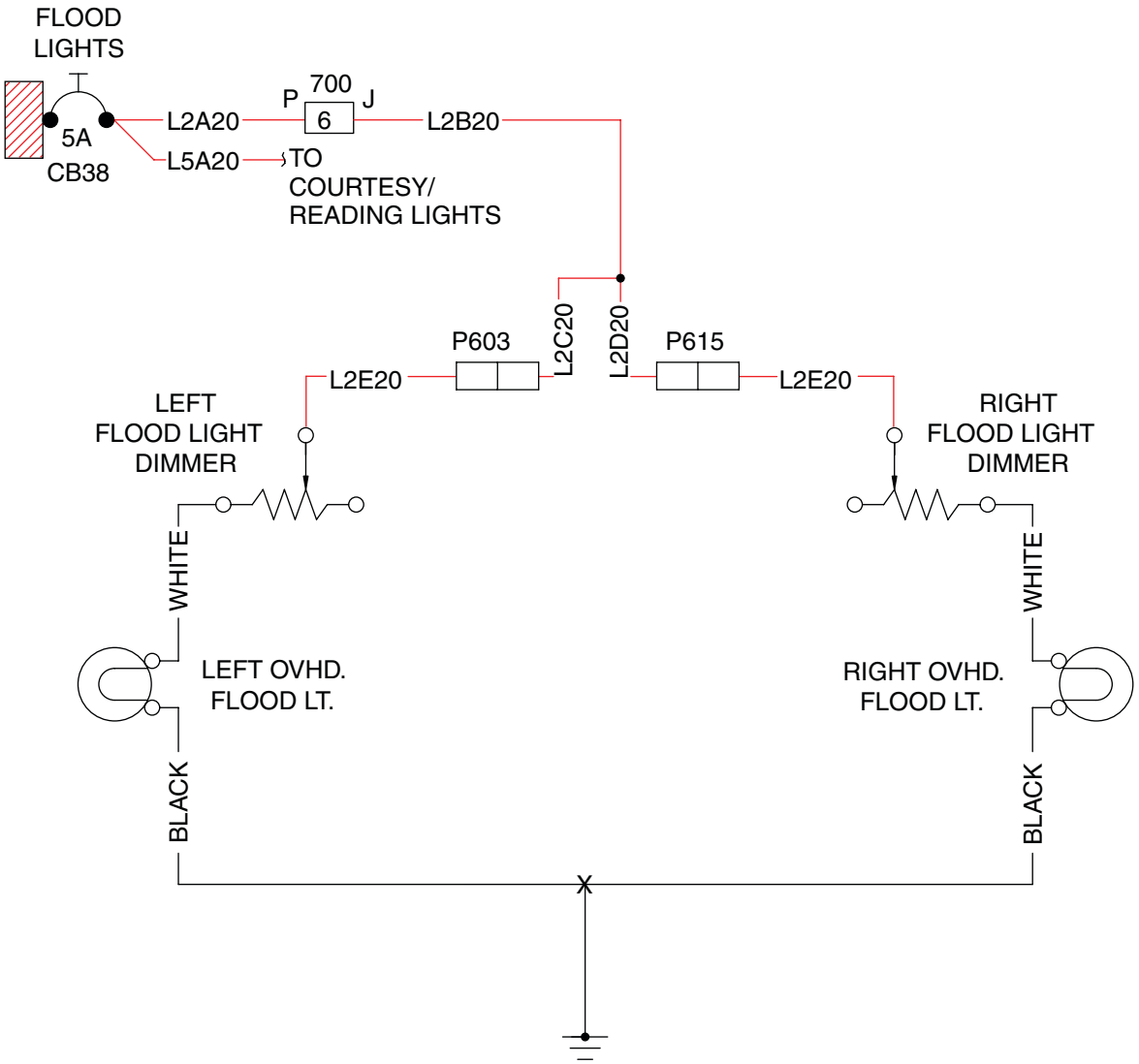
100651 K/T 21
101288 NEW/D 21
104229 C 21
104368 A/K 21.0



Overhead Flood Lights
Figure 3 (Sheet 2 of 3)

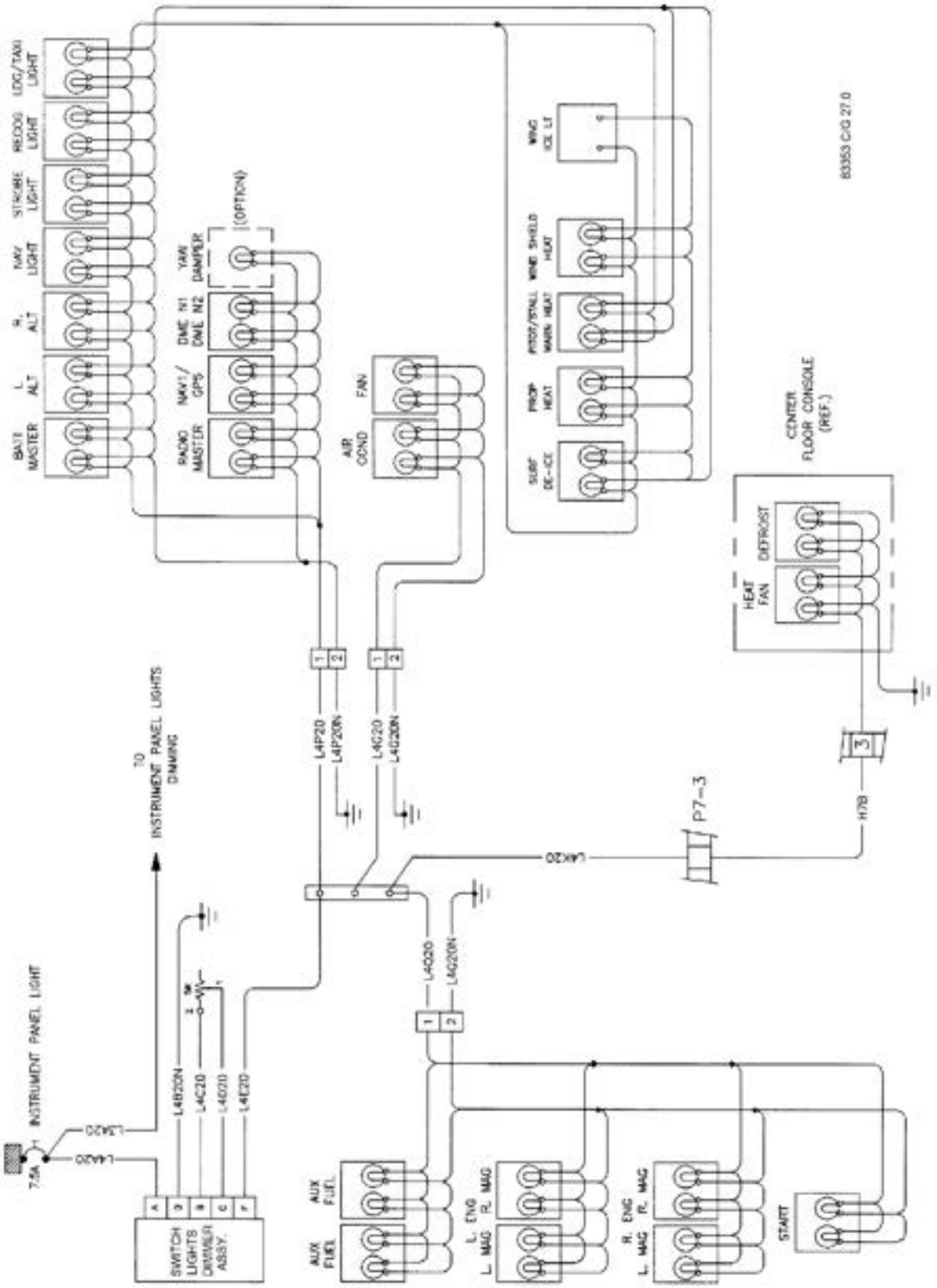
Effectivity
Seneca V

101840 NEW 14.0



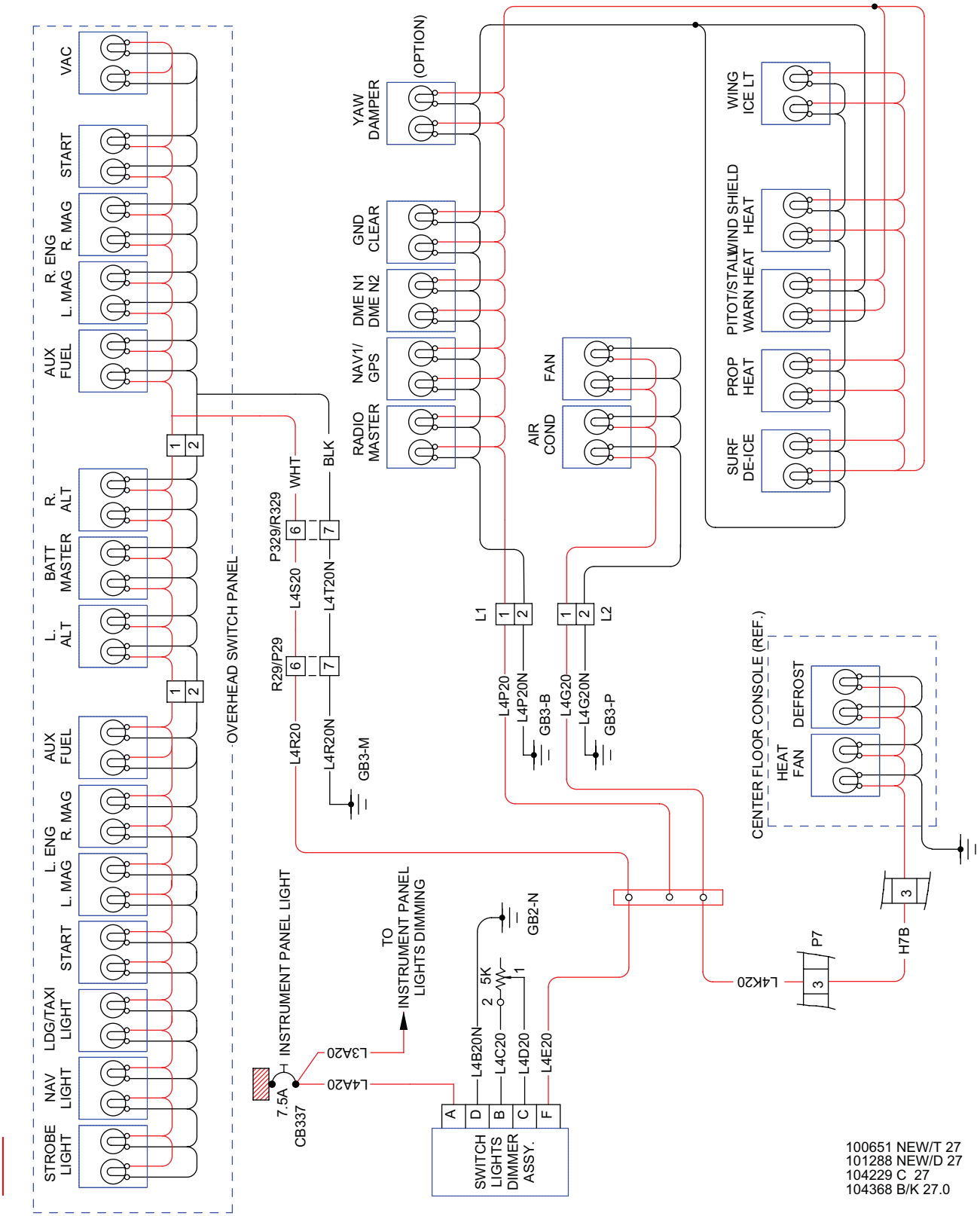
Effectivity
Seneca V
with Avidyne Entegra

Overhead Flood Lights
Figure 3 (Sheet 3 of 3)



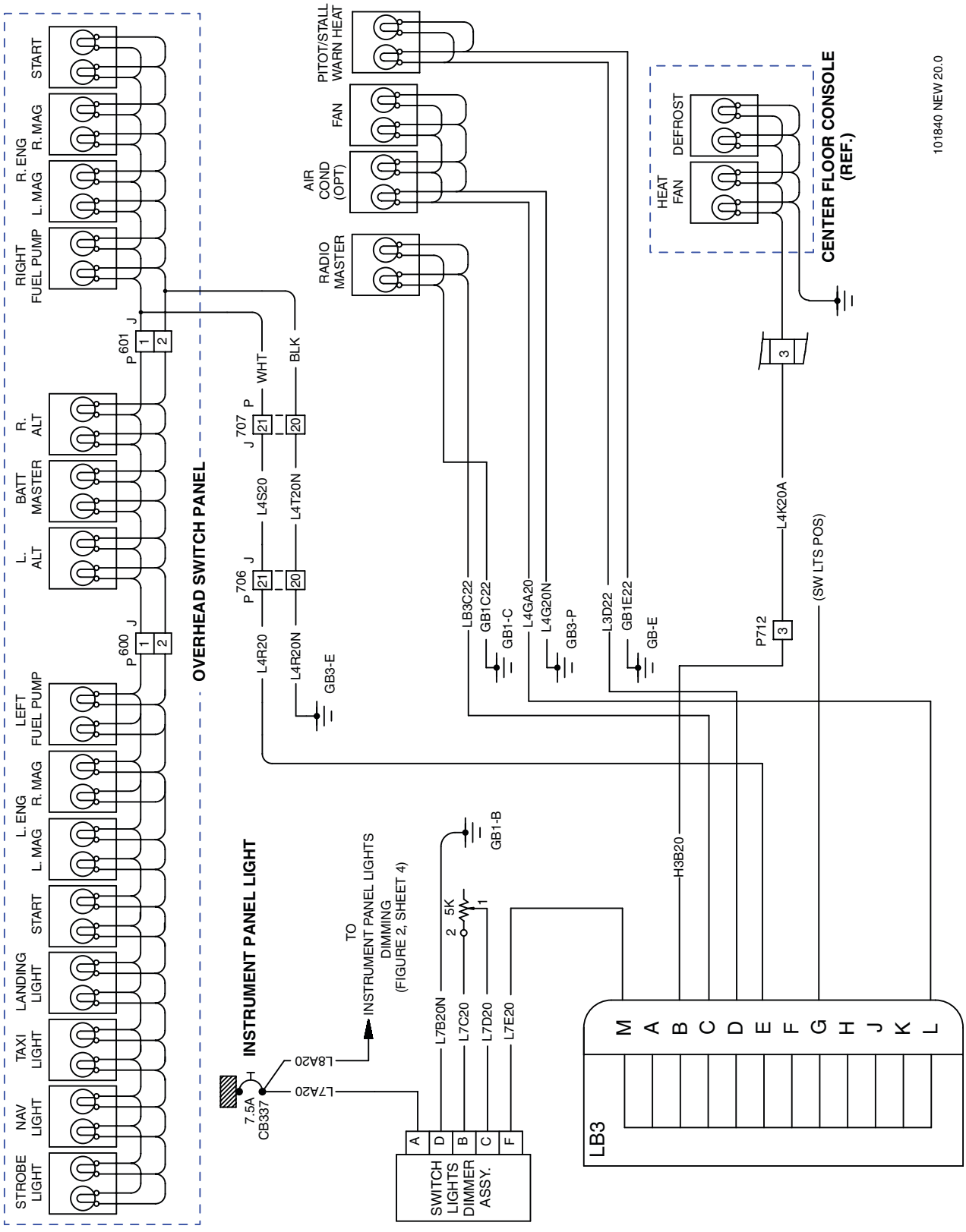
Switch Lights Dimming
Figure 4 (Sheet 1 of 5)

Effectivity
Seneca IV



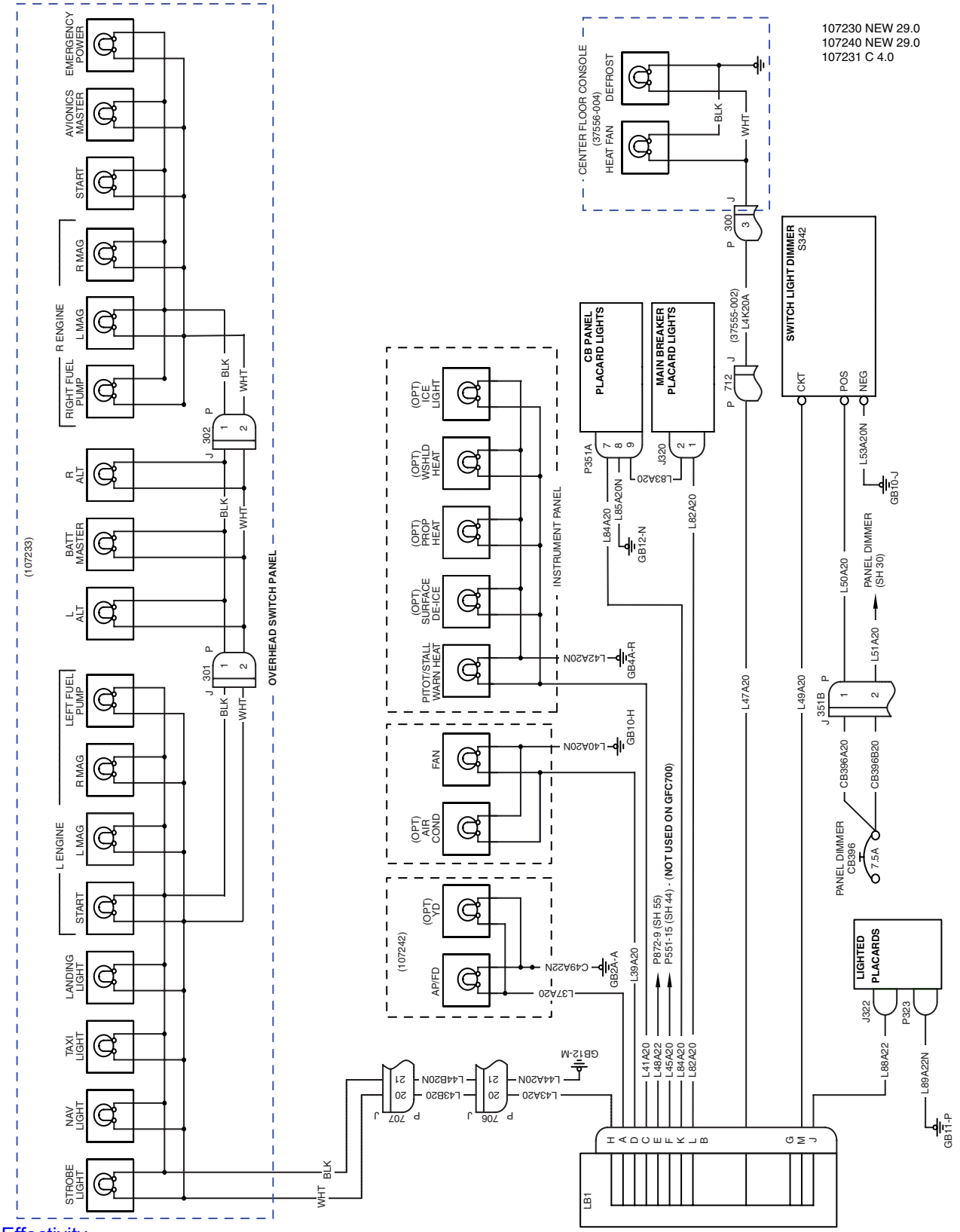
Effectivity
Seneca V

Switch Lights Dimming
Figure 4 (Sheet 2 of 5)



Switch Lights Dimming
Figure 4 (Sheet 3 of 5)

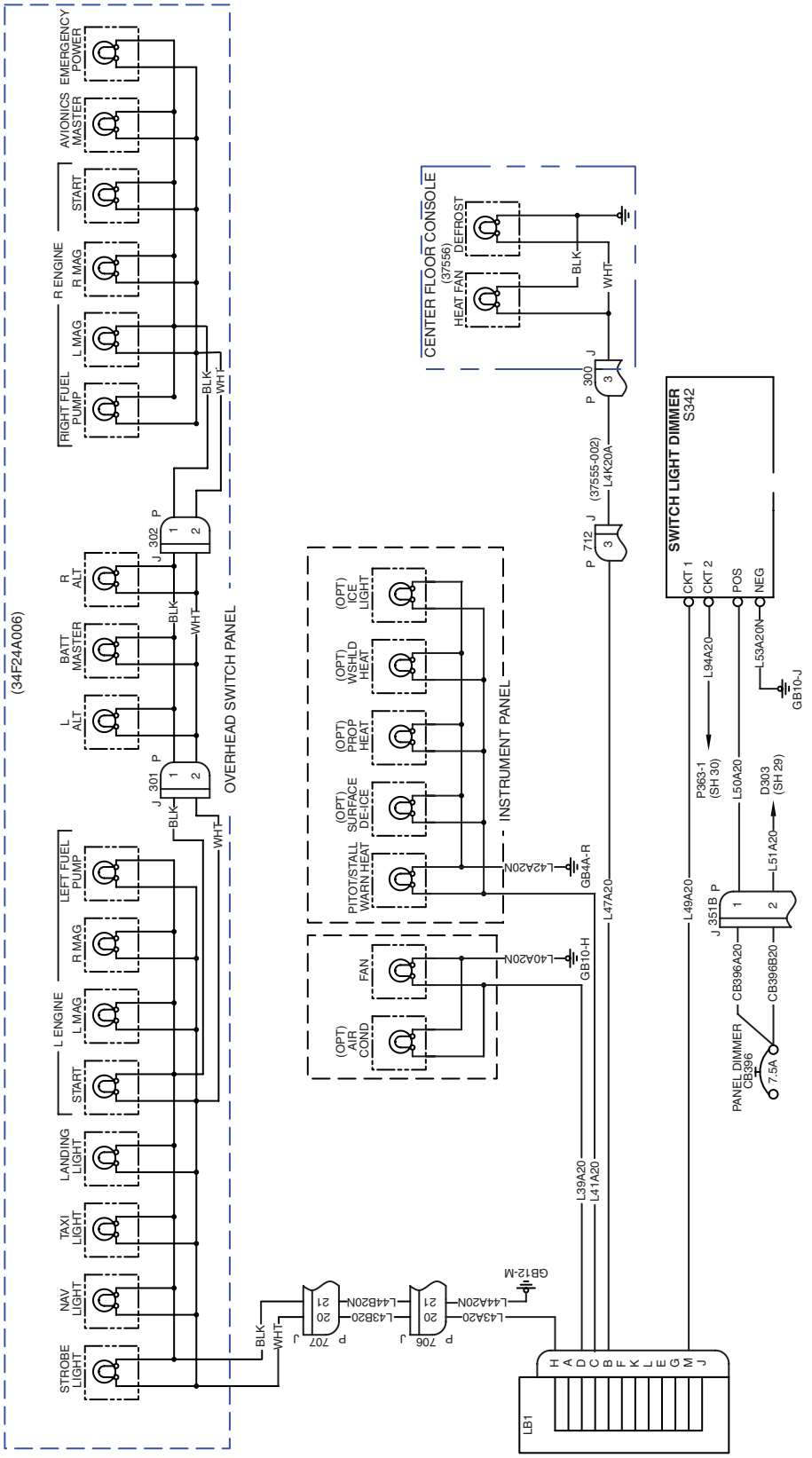
Effectivity
Seneca V
with Avidyne Entegra



Switch Lights Dimming
Figure 4 (Sheet 4 of 5)

Effectivity
Seneca V
with Garmin G1000

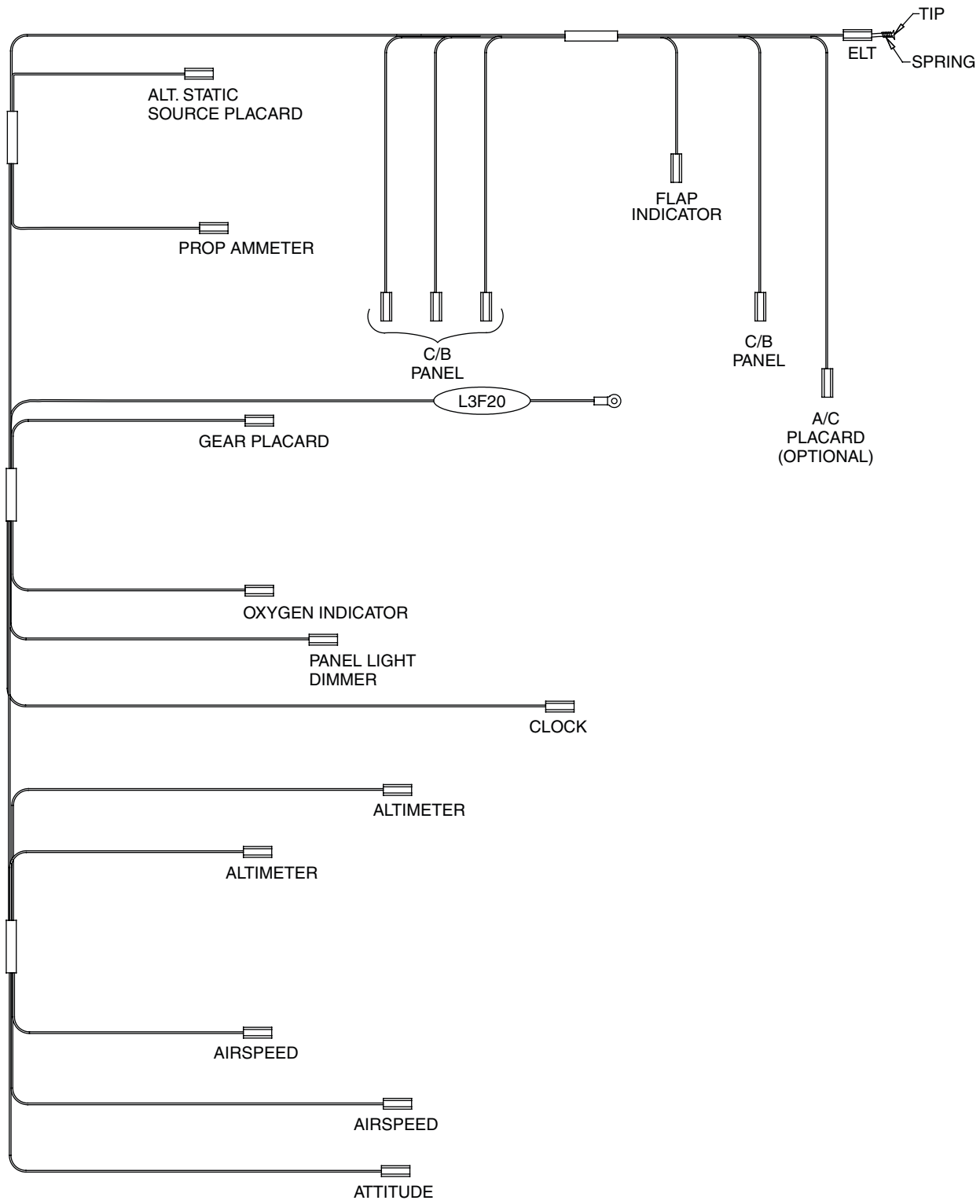
34F24A020 NEW 28.0



Switch Lights Dimming
Figure 4 (Sheet 5 of 5)

Effectivity
Seneca V
with Garmin G1000 NXi

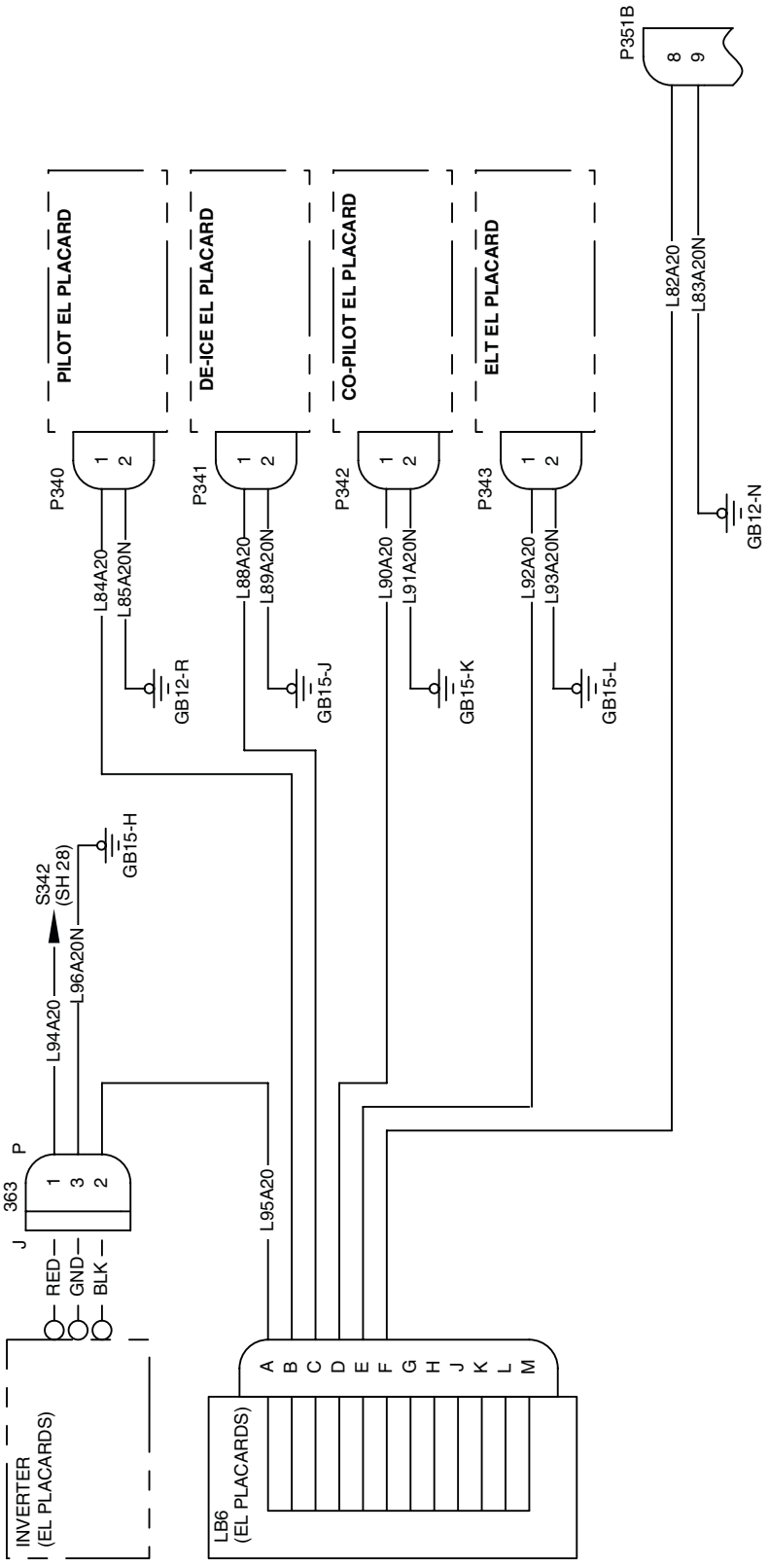
107314 B 1



Effectivity
Seneca V
with Garmin G600

Post Lights
Figure 5

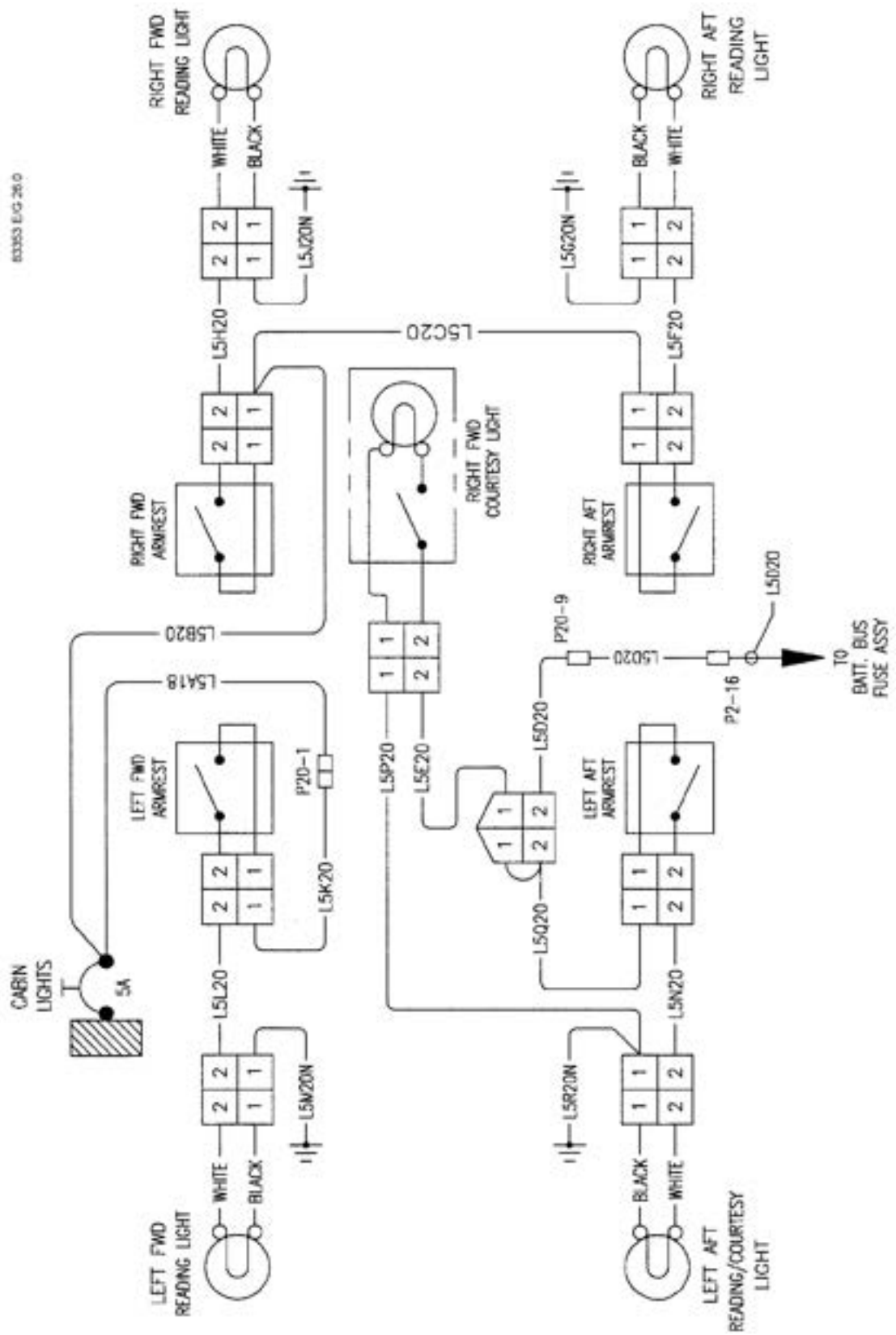
34F24A020 NEW 30.0



EL Placard Lighting
Figure 6

Effectivity
Seneca V
with Garmin G1000 NXi

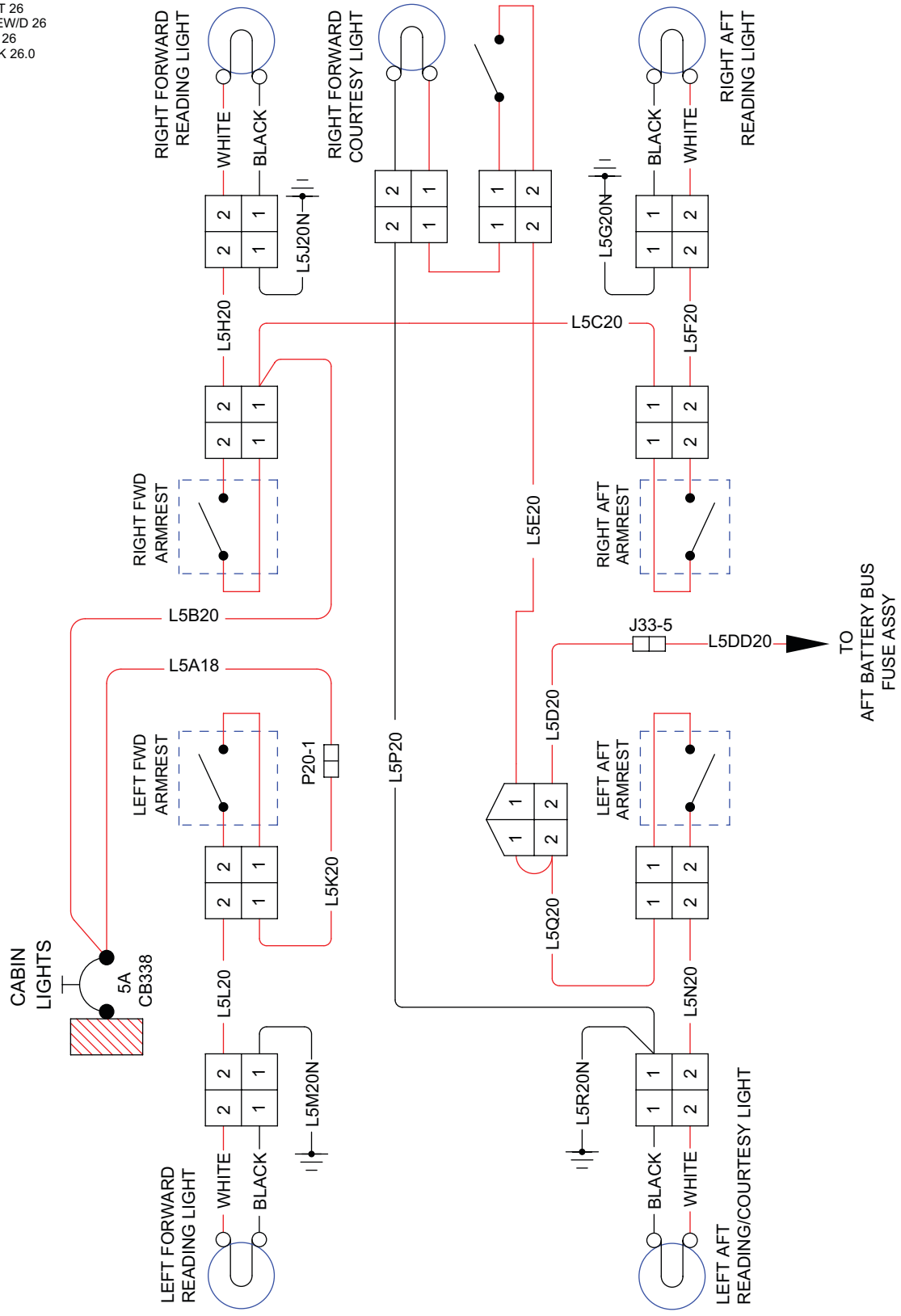
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Courtesy / Reading Lights
Figure 1 (Sheet 1 of 3)

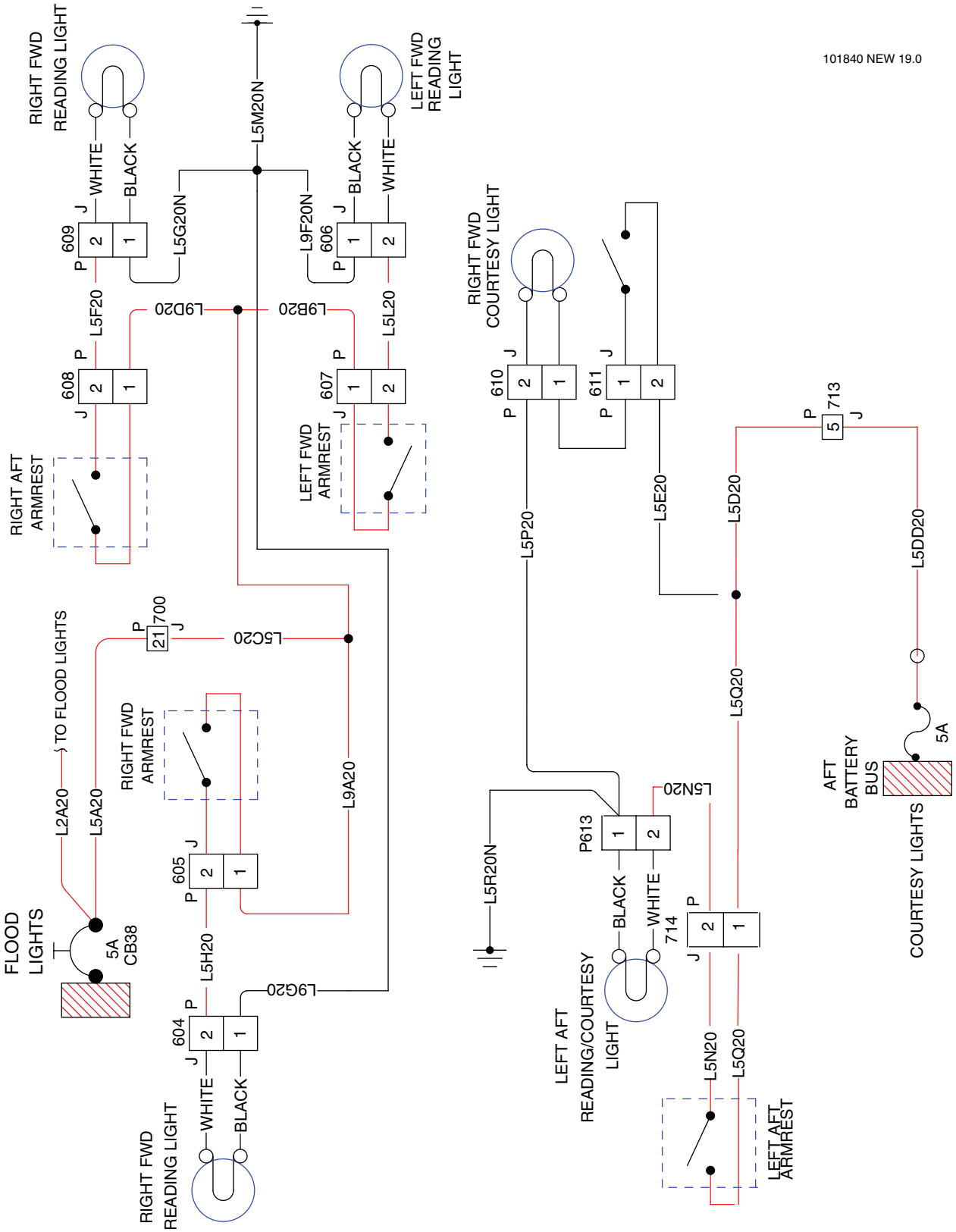
Effectivity
Seneca IV

100651 K/T 26
101288 NEW/D 26
104229 C 26
104368 A/K 26.0



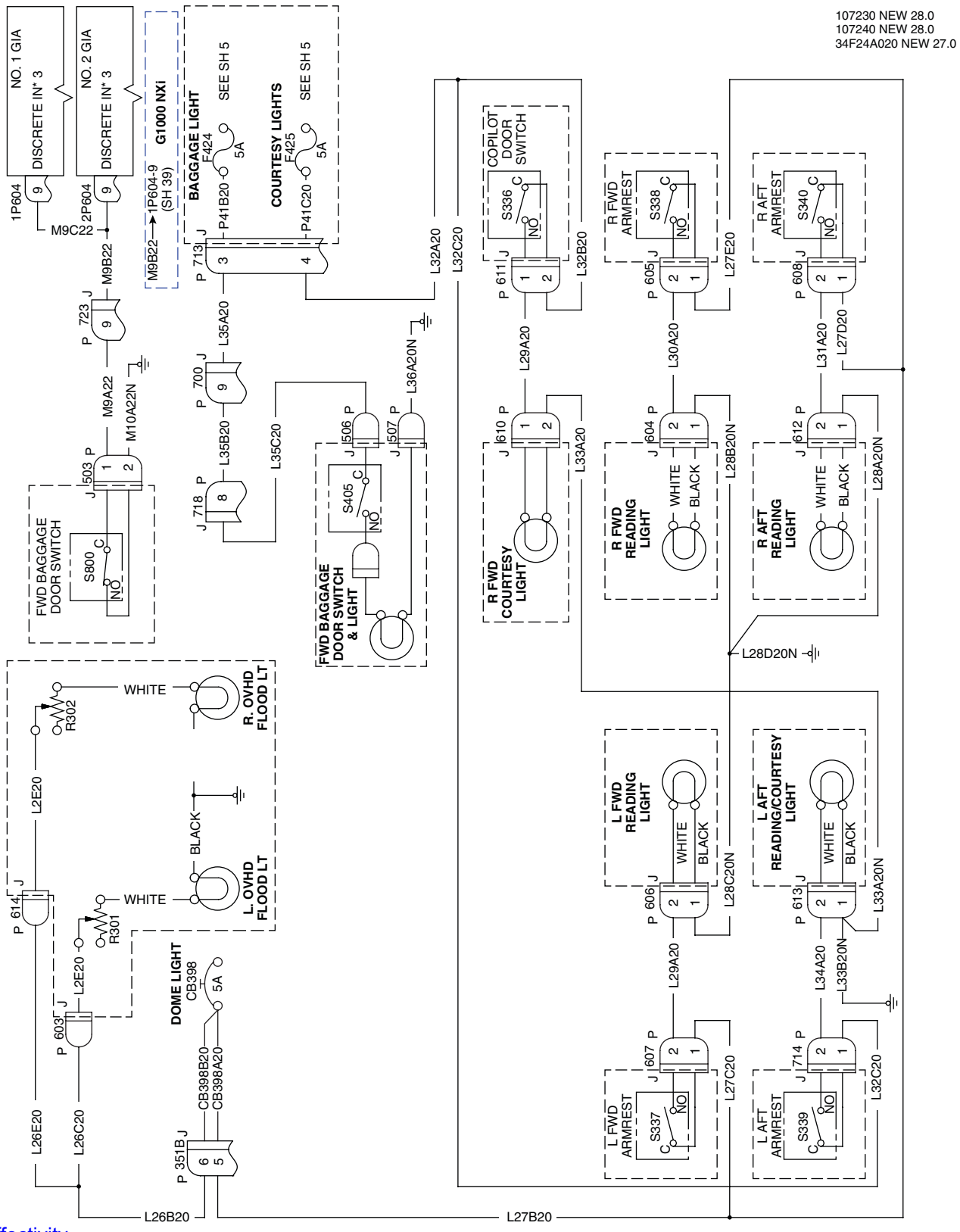
Effectivity
Seneca V

Courtesy / Reading Lights
Figure 1 (Sheet 2 of 3)



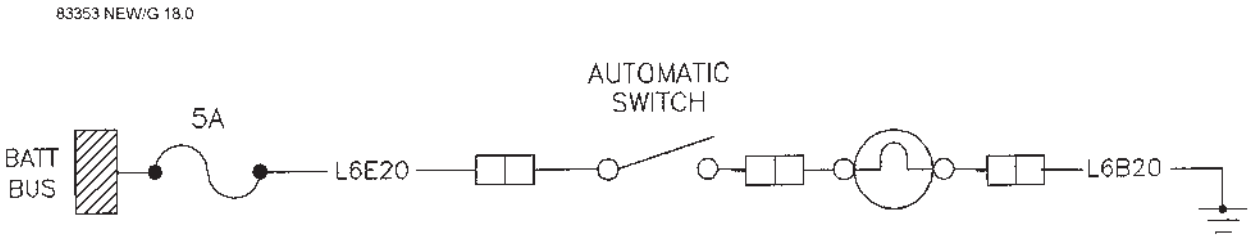
Courtesy / Reading Lights
Figure 1 (Sheet 3 of 3)

Effectivity
Seneca V
with Avidyne Entegra



Effectivity
Seneca V
with Garmin G1000 / G1000 NXi

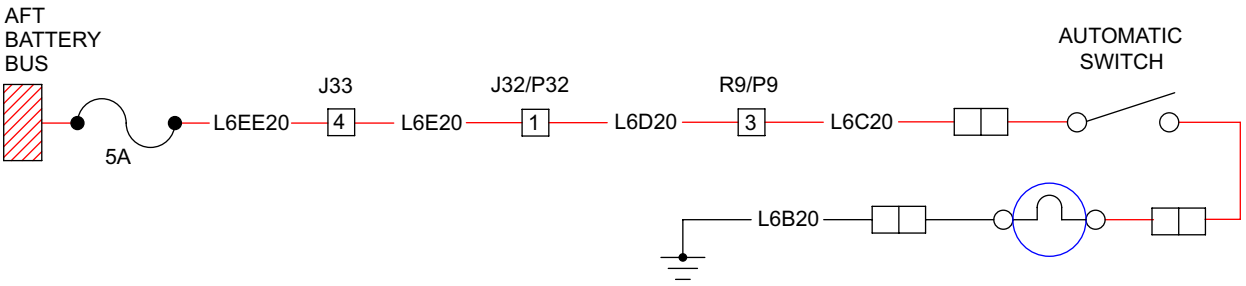
Interior / Door Lighting
Figure 2



Baggage Compartment Light
Figure 1 (Sheet 1 of 2)

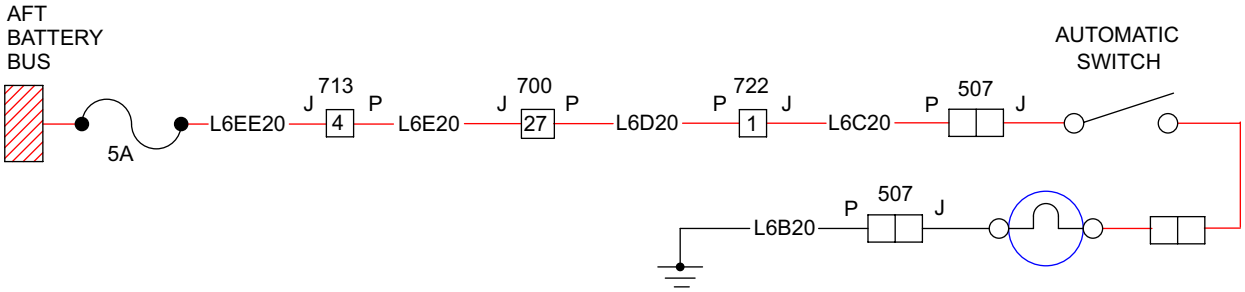
Effectivity
Seneca IV

100651 K/T 18
101288 NEW/D 18
104229 NEW/C 18
104368 NEW/K 18.0



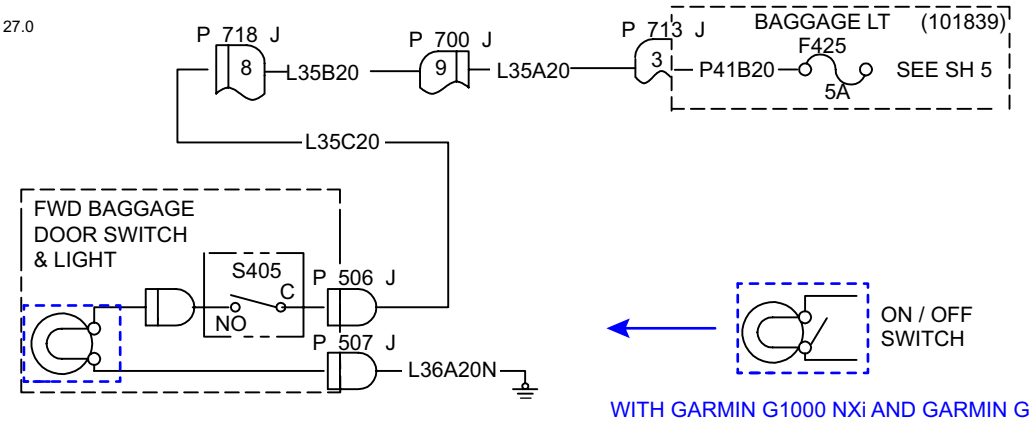
STANDARD

101840 NEW 11.0



WITH AVIDYNE ENTEGRA

107230 NEW 28.0
107240 A 28.0
34F24A020 NEW 27.0
34F24A001 1 8.0

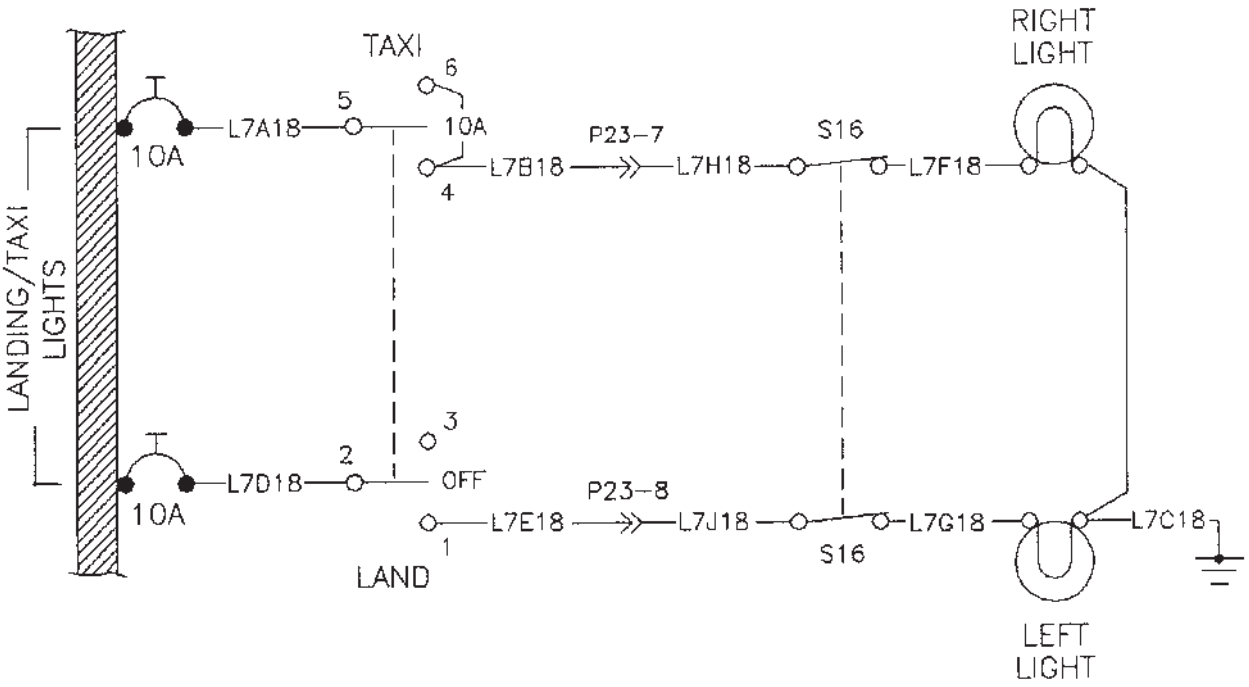


WITH GARMIN G1000

WITH GARMIN G1000 NXi AND GARMIN G1000, S/N's 3449487 – 34499507, WITH IMPROVED FORWARD BAGGAGE COMPARTMENT LIGHT

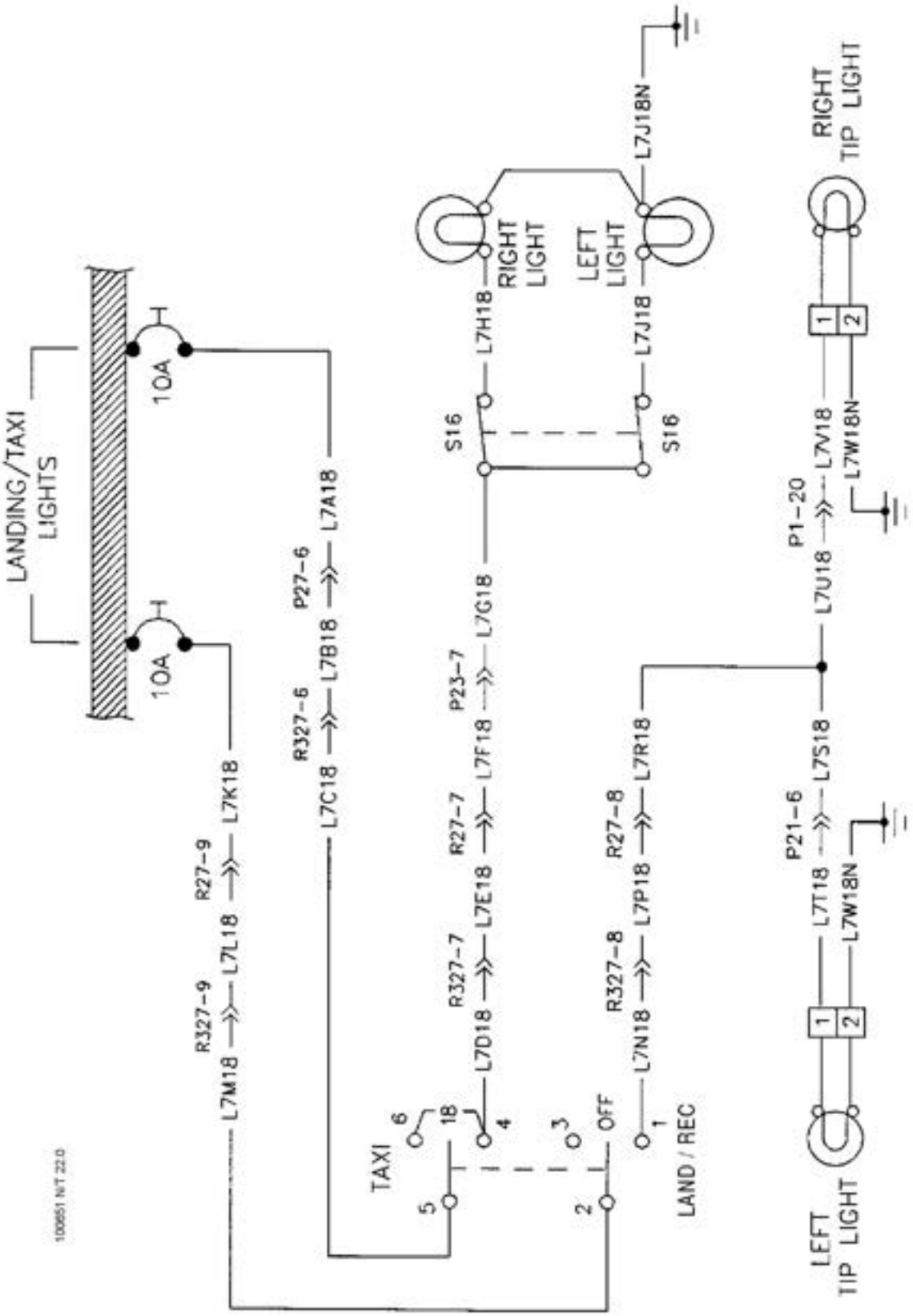
Effectivity
Seneca V

Baggage Compartment Light
Figure 1 (Sheet 2 of 2)



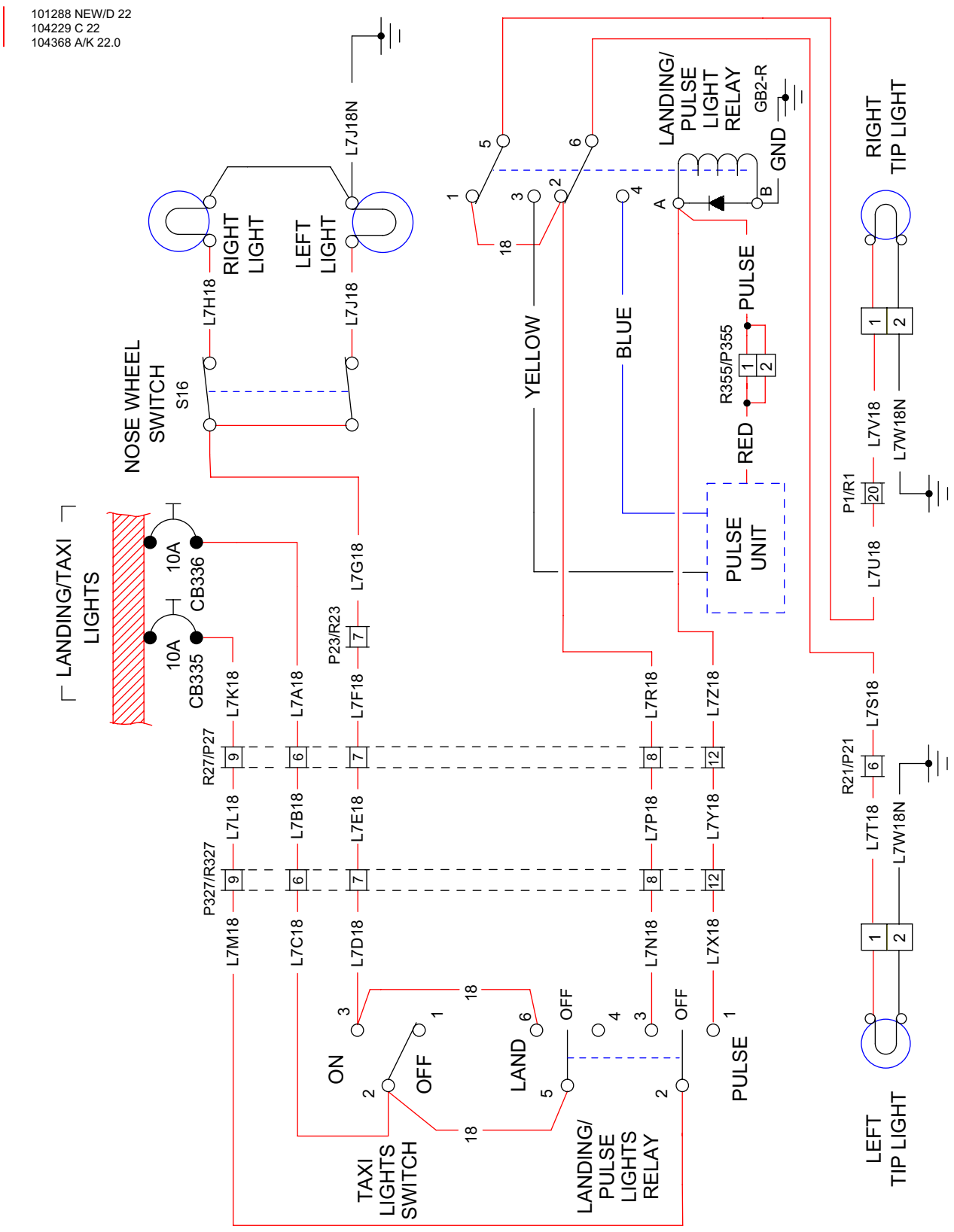
Landing / Taxi Lights
Figure 1 (Sheet 1 of 8)

Effectivity
Seneca IV



Effectivity
3449001 thru 3449096

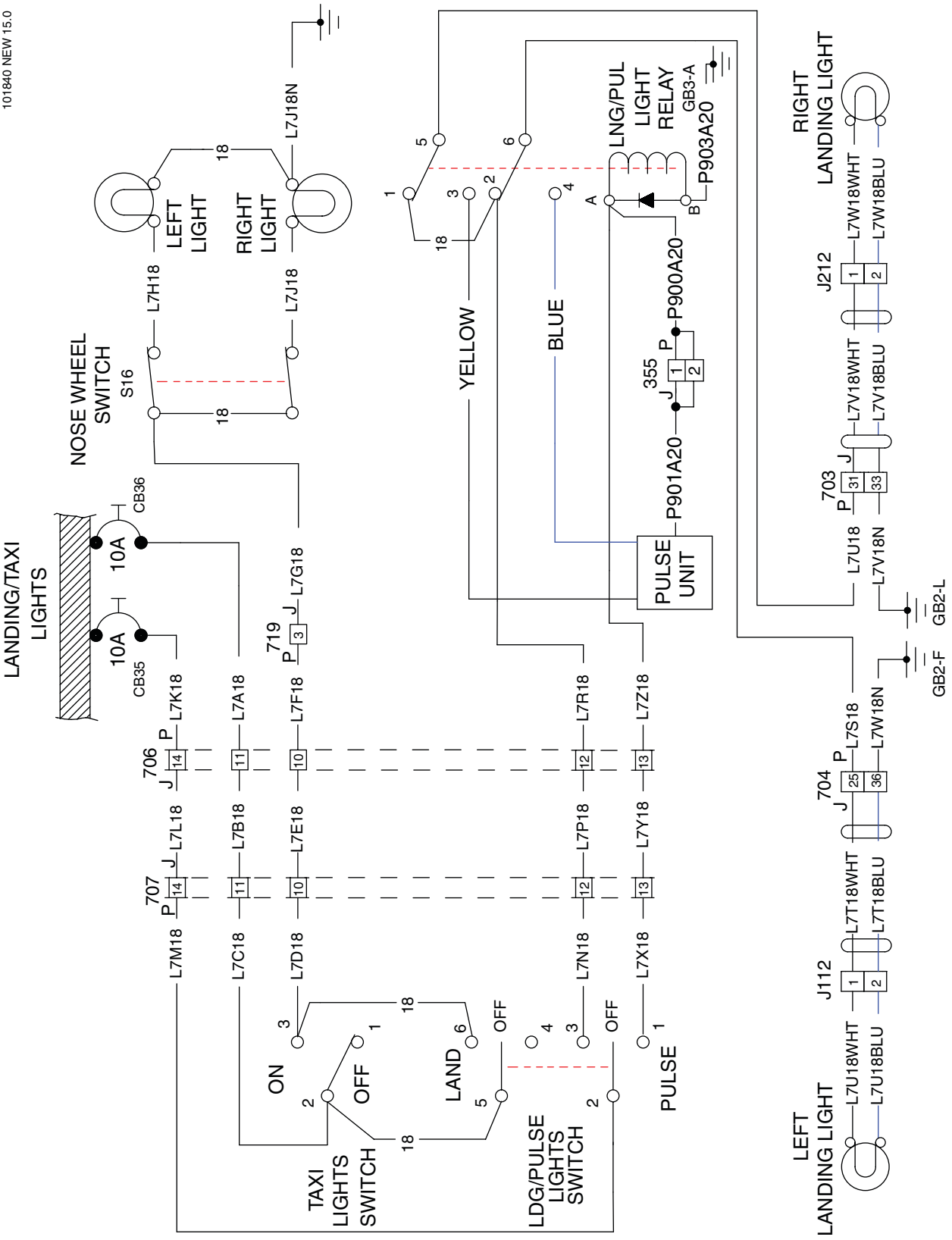
Landing / Taxi Lights
Figure 1 (Sheet 2 of 8)



Landing / Taxi Lights
Figure 1 (Sheet 3 of 8)

Effectivity
3449097 and up

101840 NEW 15.0

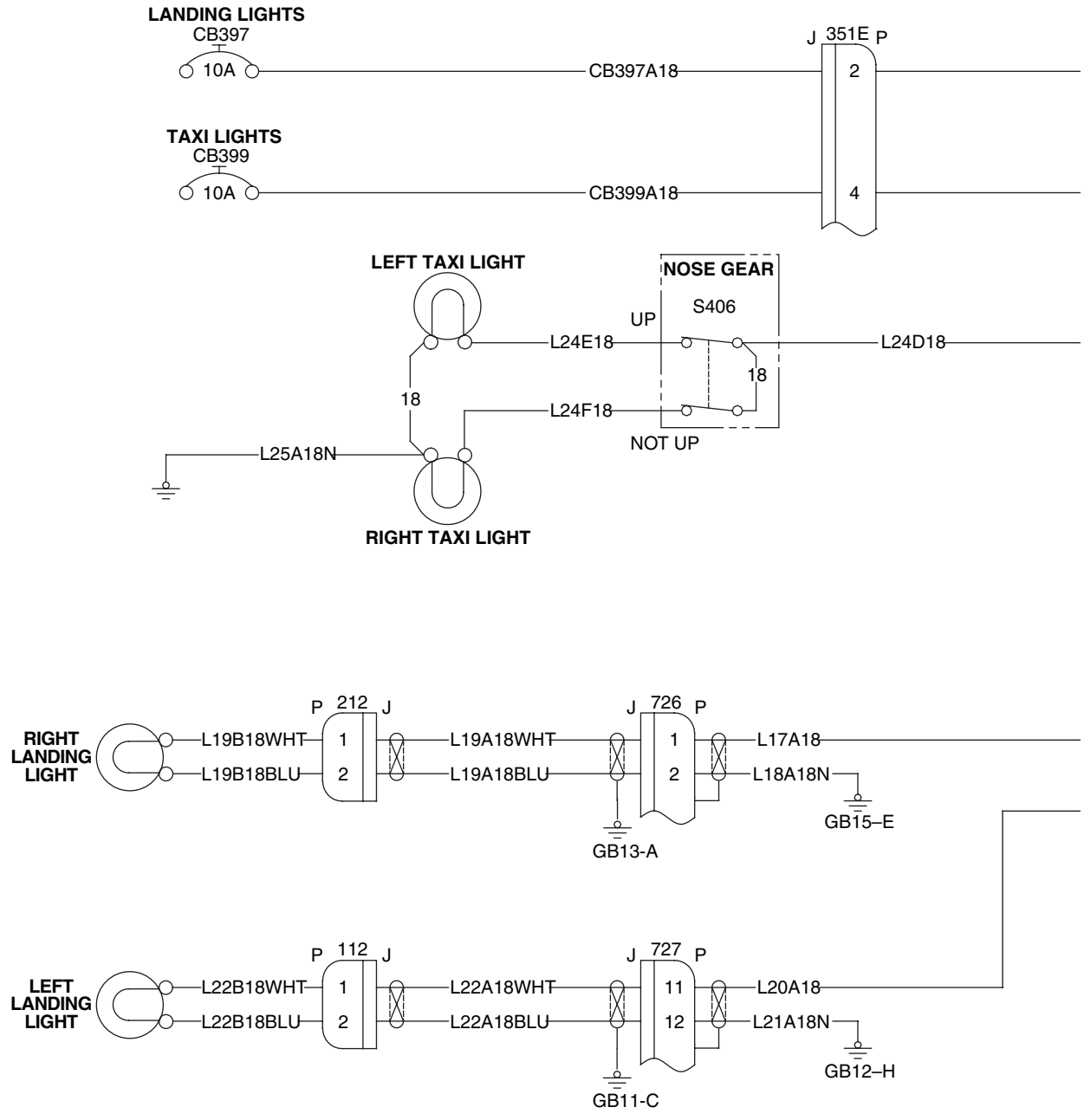


Effectivity
Seneca V
with Avidyne Entegra

Landing / Taxi Lights
Figure 1 (Sheet 4 of 8)

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A

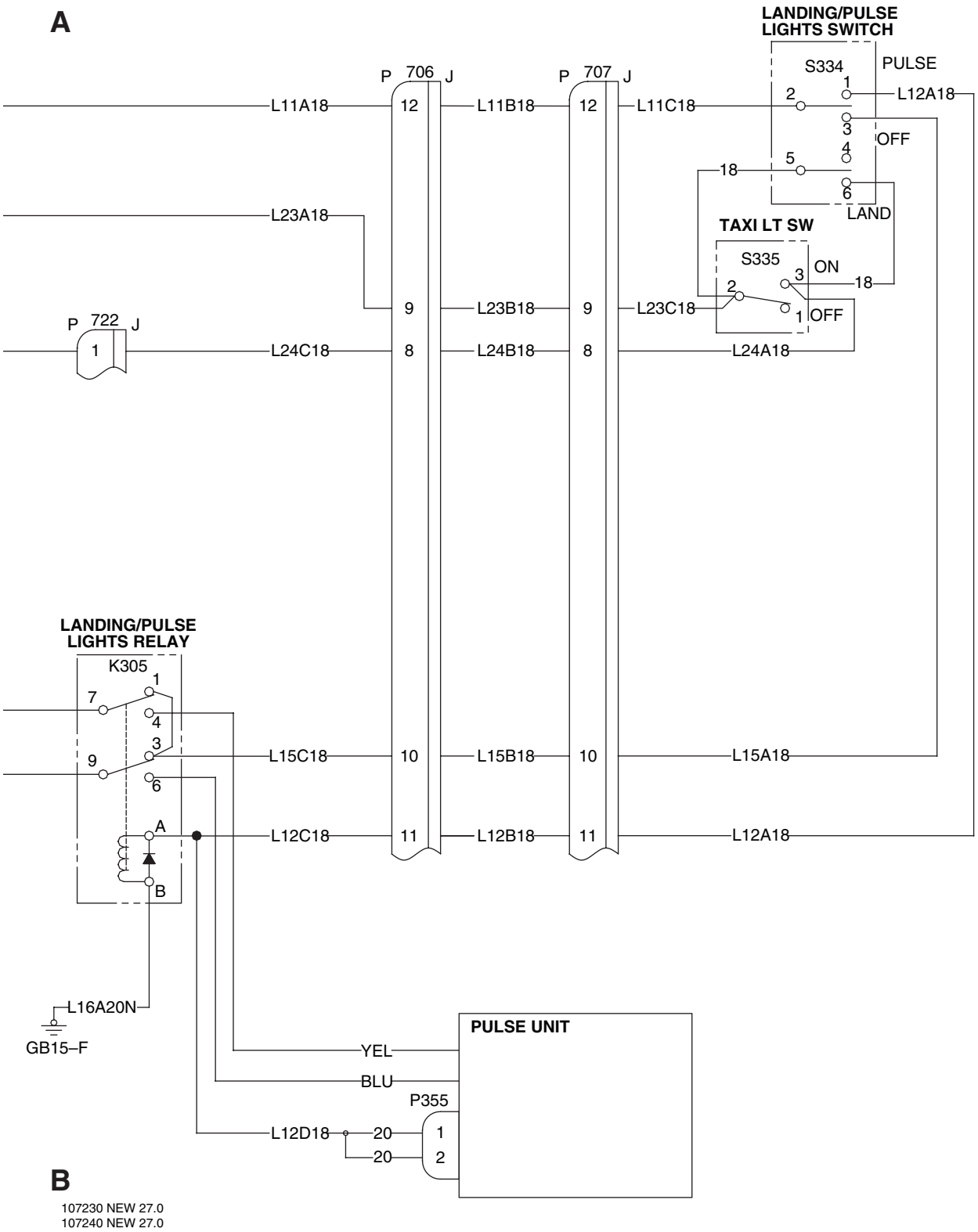


B

107230 NEW 27.0
107240 NEW 27.0

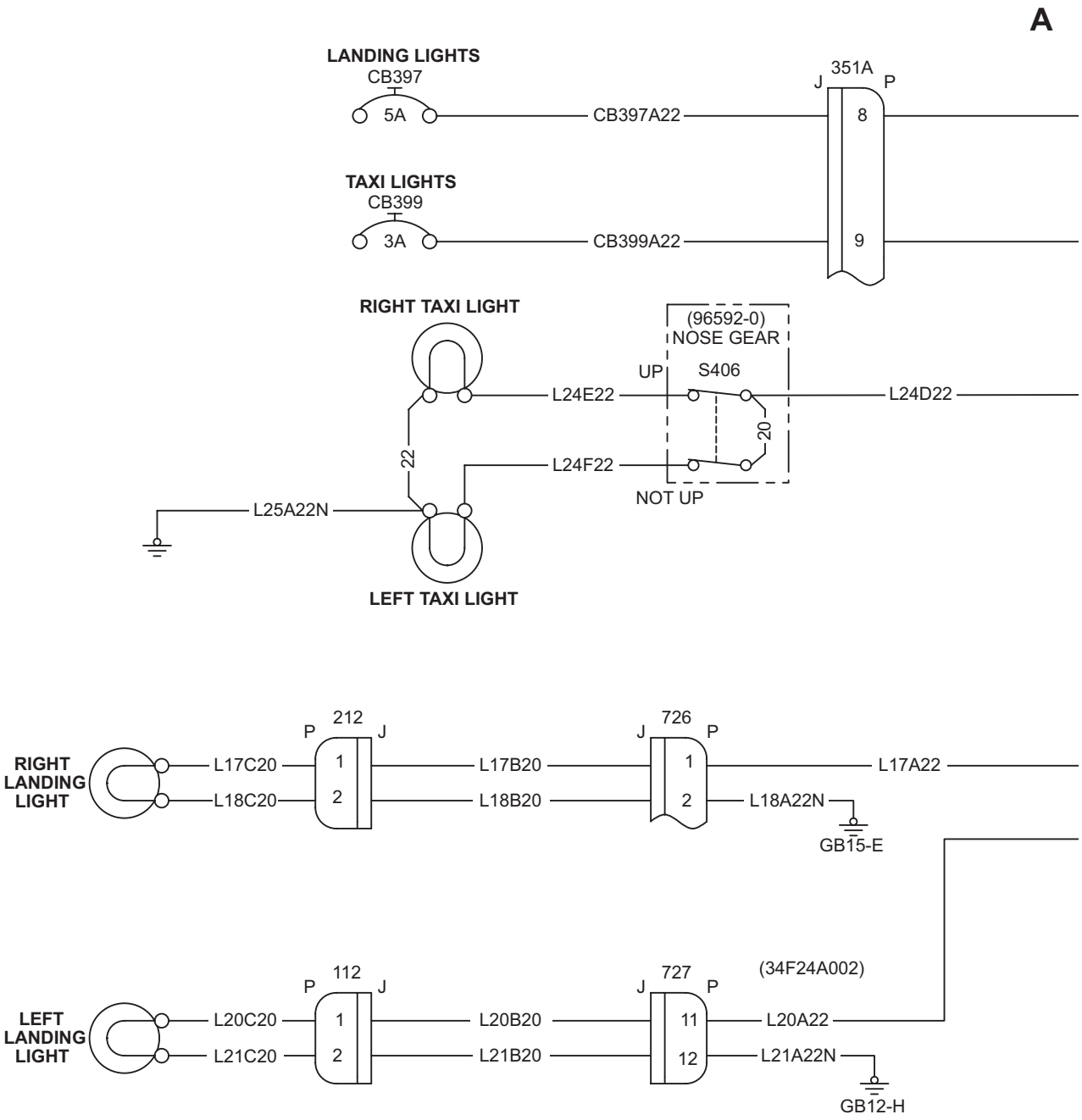
Effectivity
Seneca V
with Garmin G1000

Landing / Taxi Lights
Figure 1 (Sheet 5 of 8)



Landing / Taxi Lights
Figure 1 (Sheet 6 of 8)

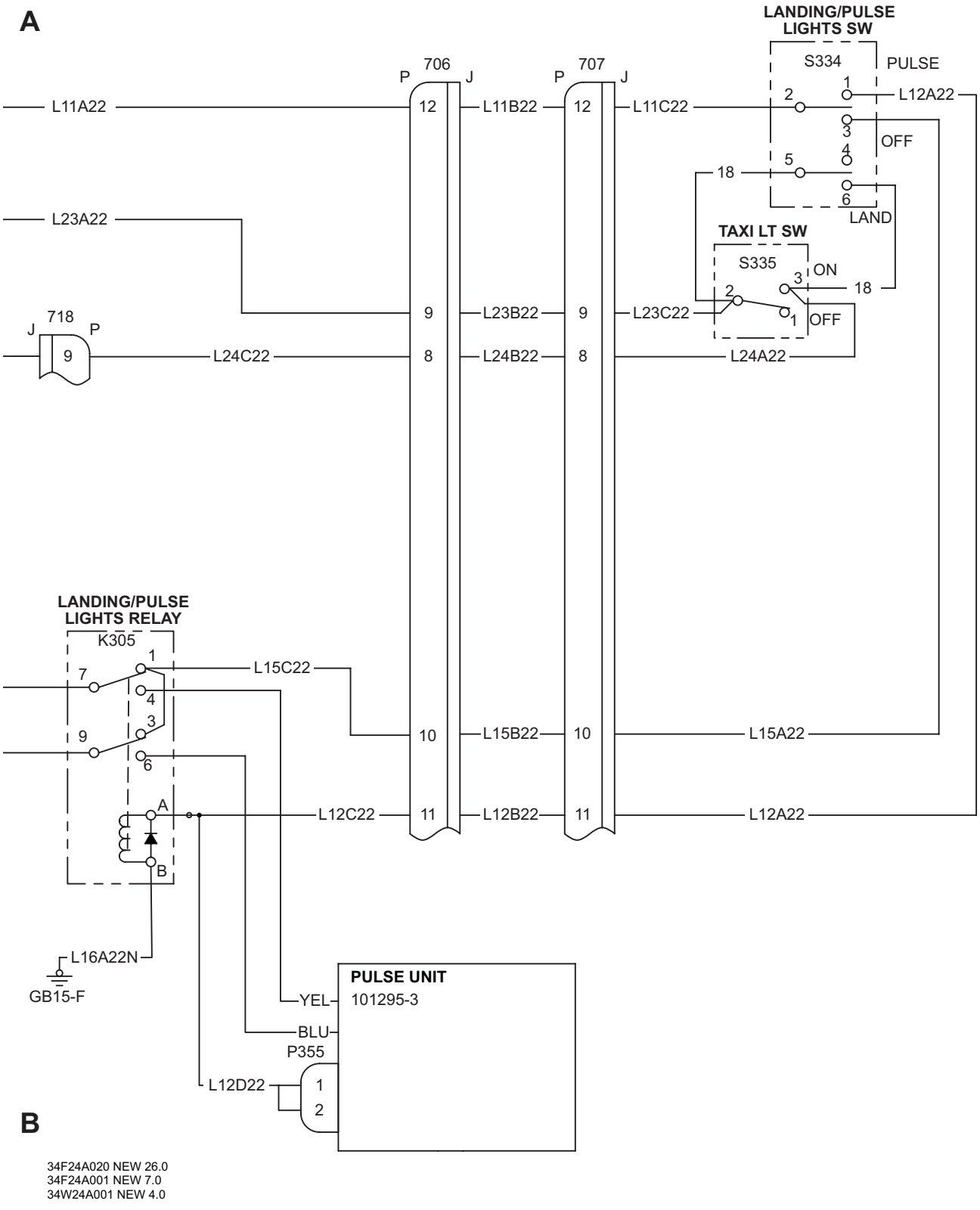
Effectivity
Seneca V
with Garmin G1000



34F24A020 NEW 26.0
34F24A001 NEW 7.0
34W24A001 NEW 4.0

Effectivity
Seneca V
with Garmin G1000 NXi

Landing / Taxi Lights
Figure 1 (Sheet 7 of 8)

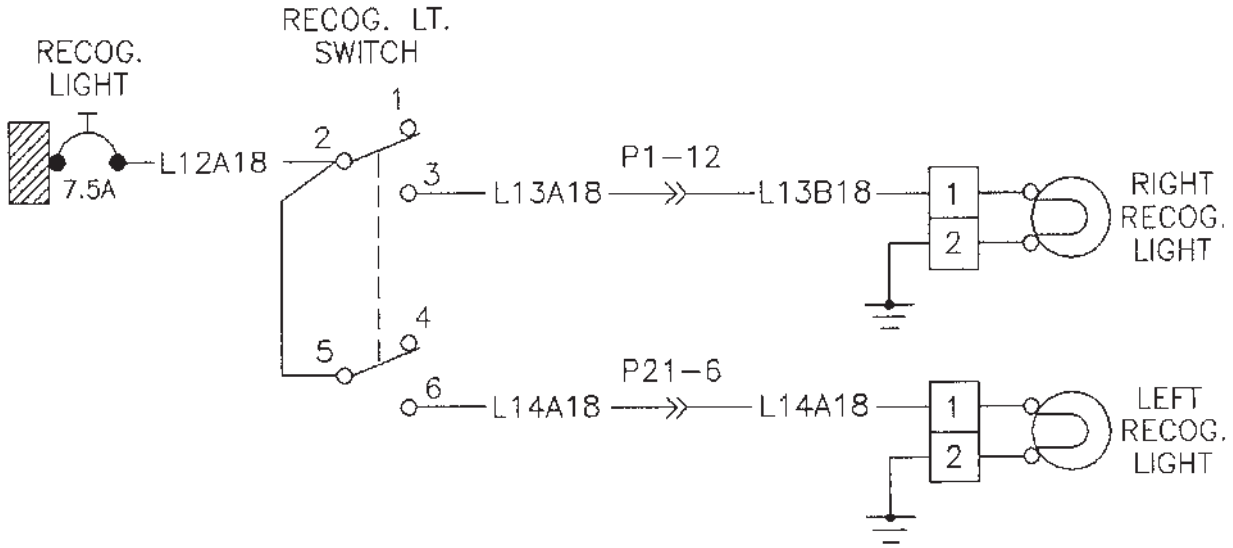


Landing / Taxi Lights
Figure 1 (Sheet 8 of 8)

Effectivity
Seneca V
with Garmin G1000 NXi

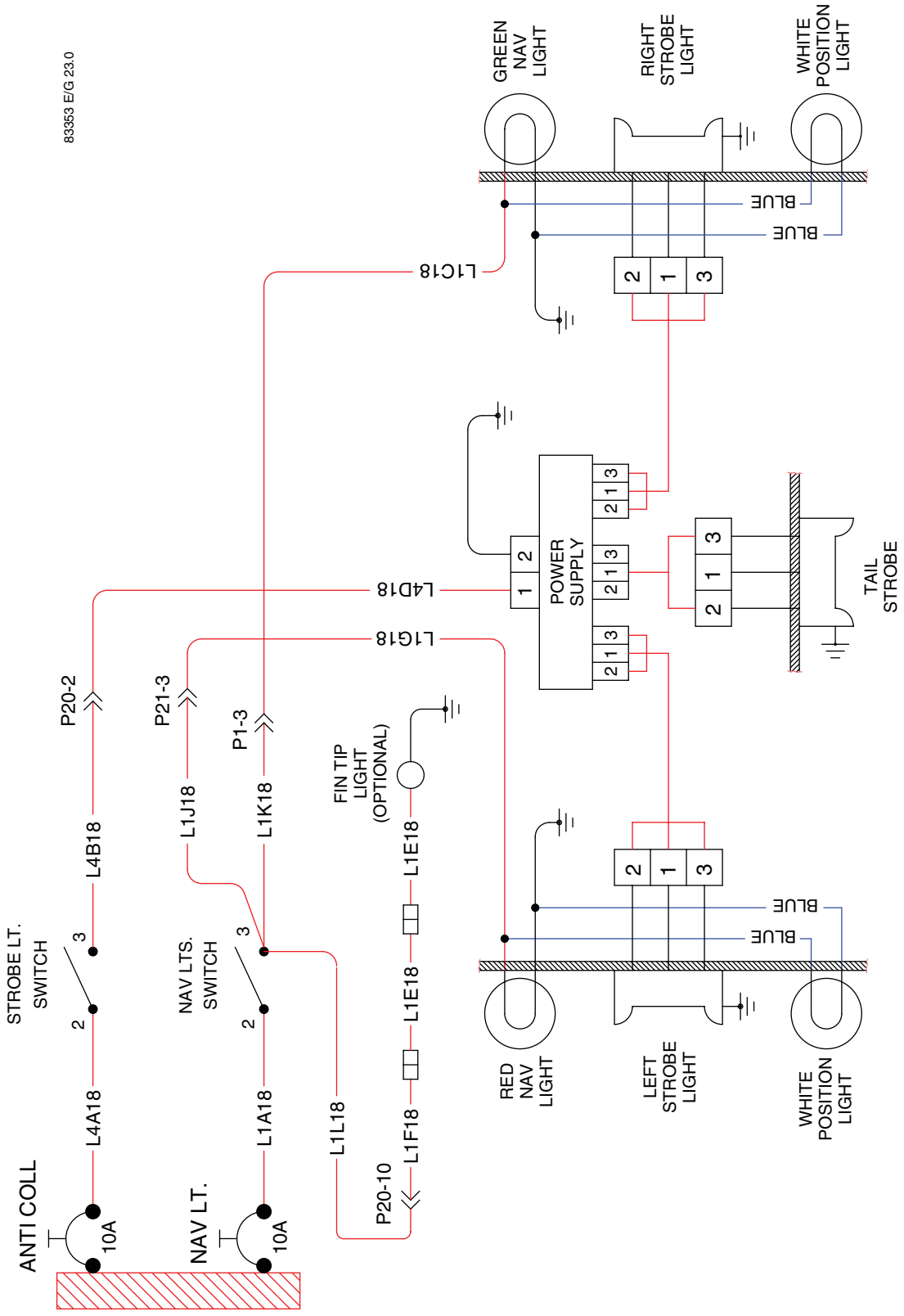
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83353 NEW/G 12.0

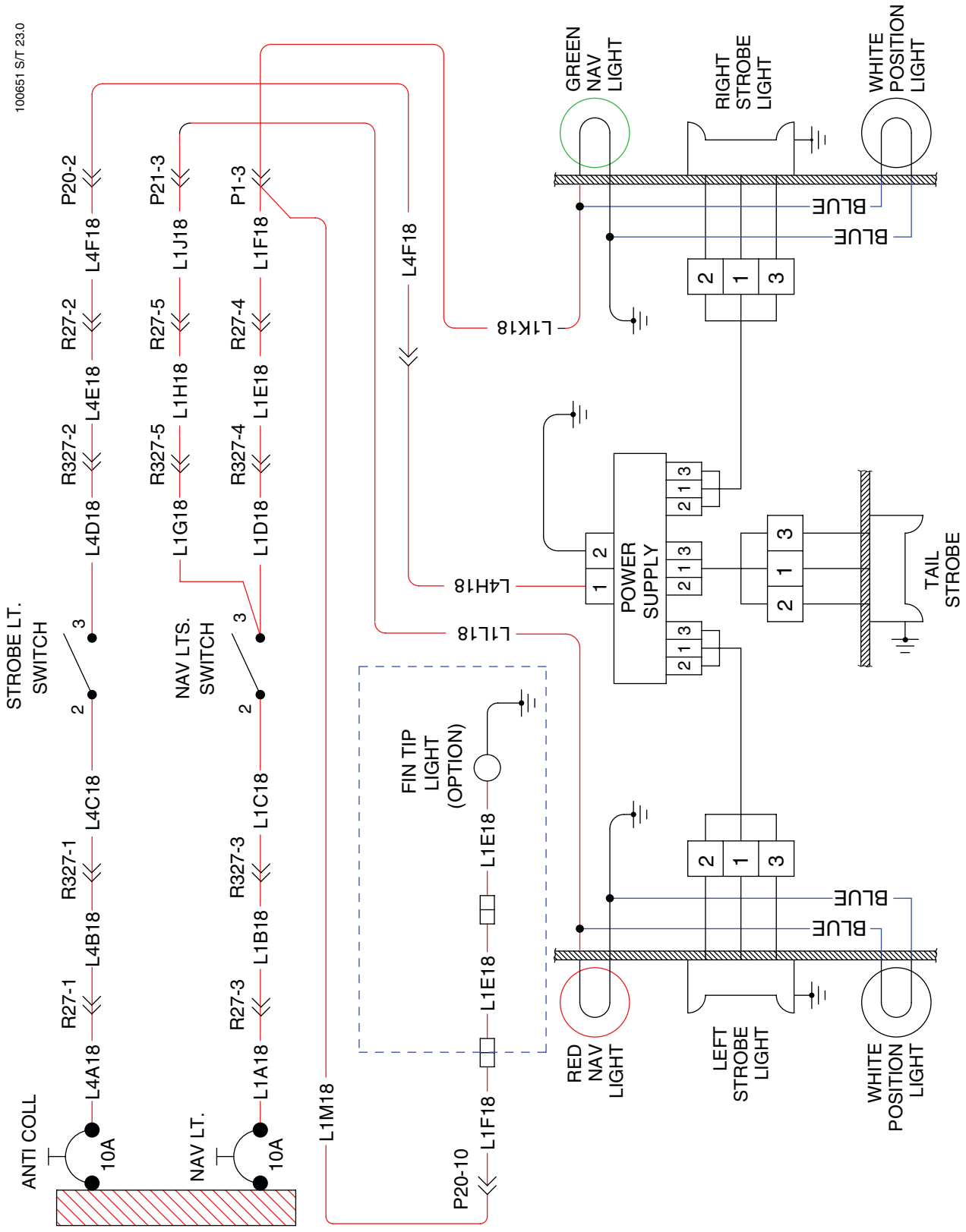


Recognition Lights
Figure 2

Effectivity
Seneca IV



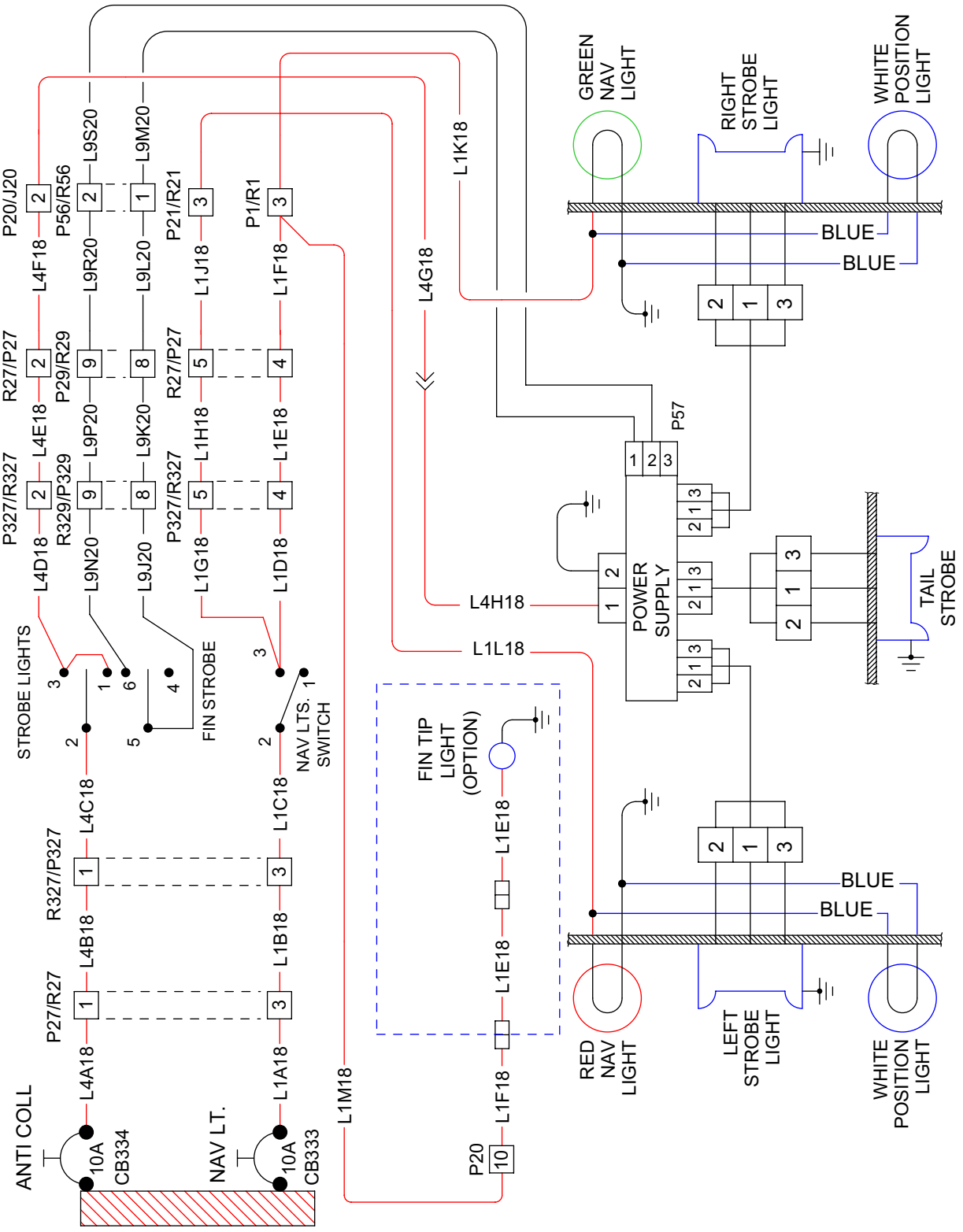
Navigation and Strobe Lights
Figure 3 (Sheet 1 of 8)



Navigation and Strobe Lights
Figure 3 (Sheet 2 of 8)

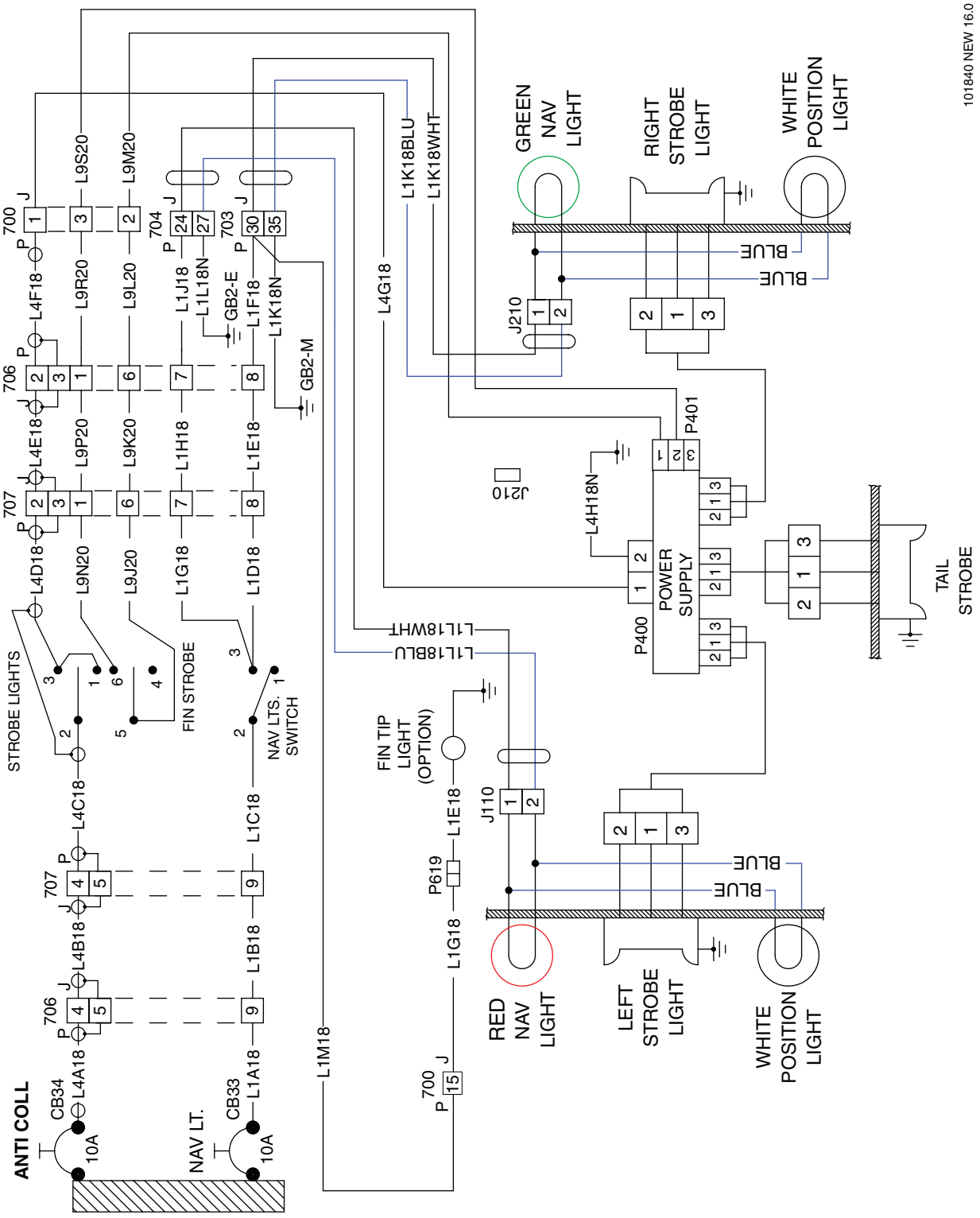
Effectivity
3449001 thru 3449096

101288 C/D
104229 C
104368 A/K 23.0



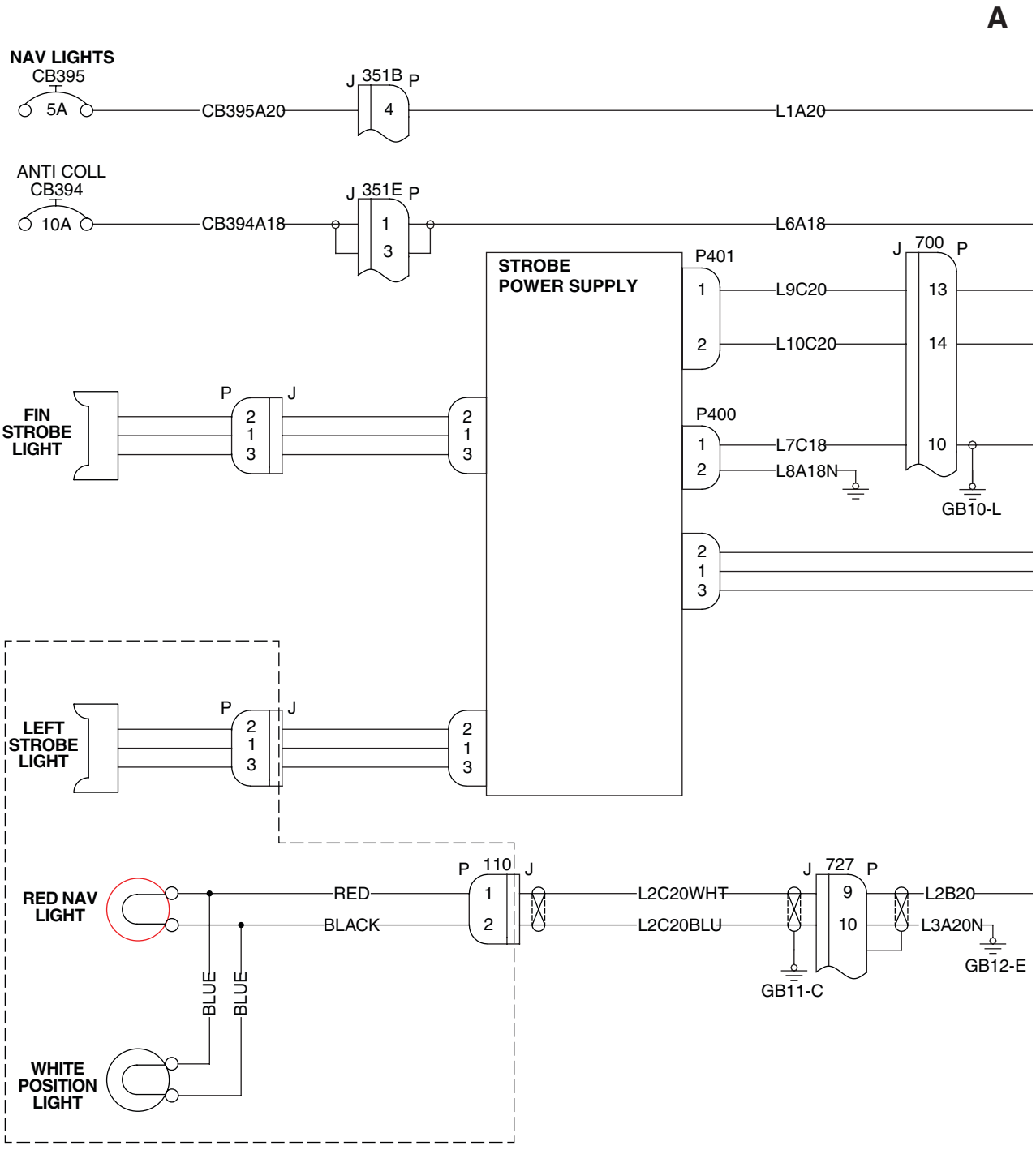
Effectivity
3449097 and up

Navigation and Strobe Lights
Figure 3 (Sheet 3 of 8)



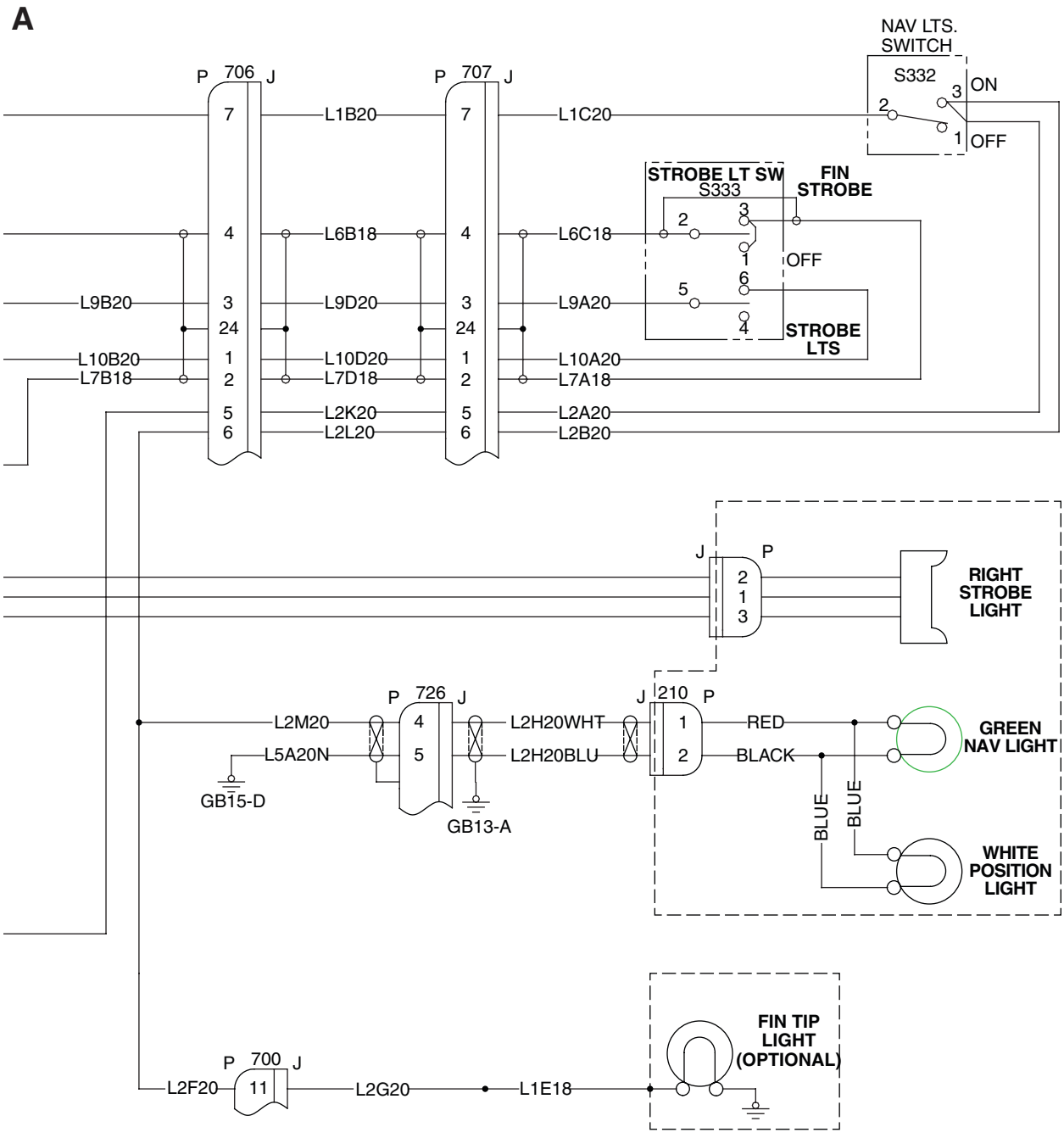
Navigation and Strobe Lights
Figure 3 (Sheet 4 of 8)

Effectivity
Seneca V
with Avidyne Entegra



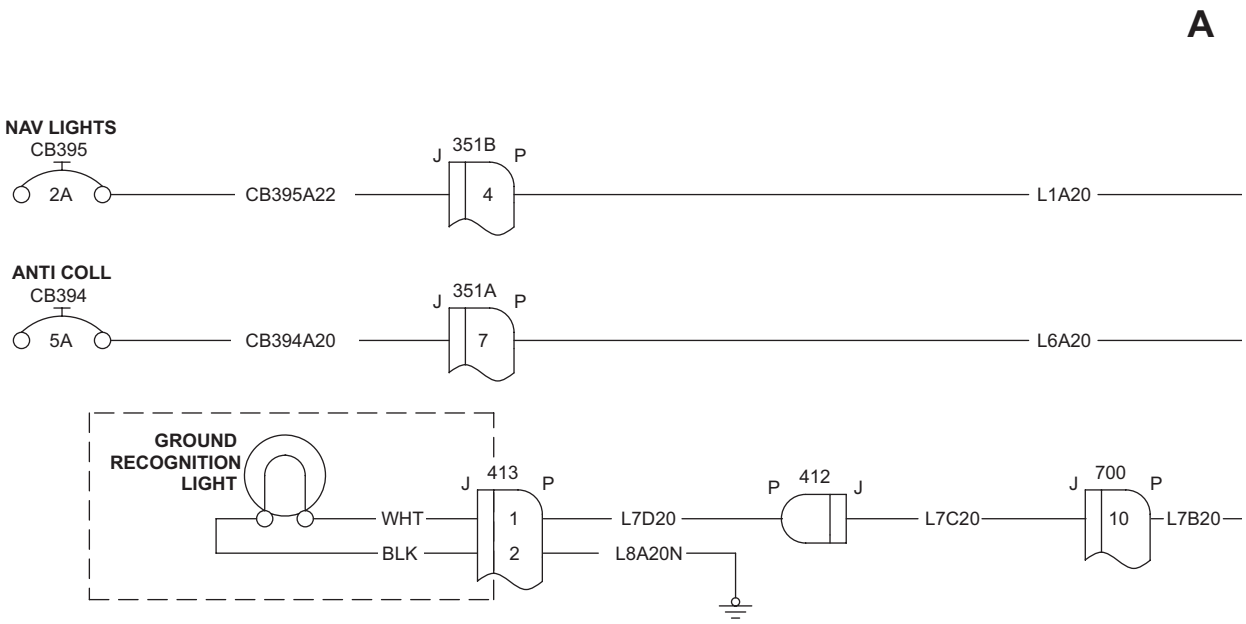
Effectivity
Seneca V
with Garmin G1000

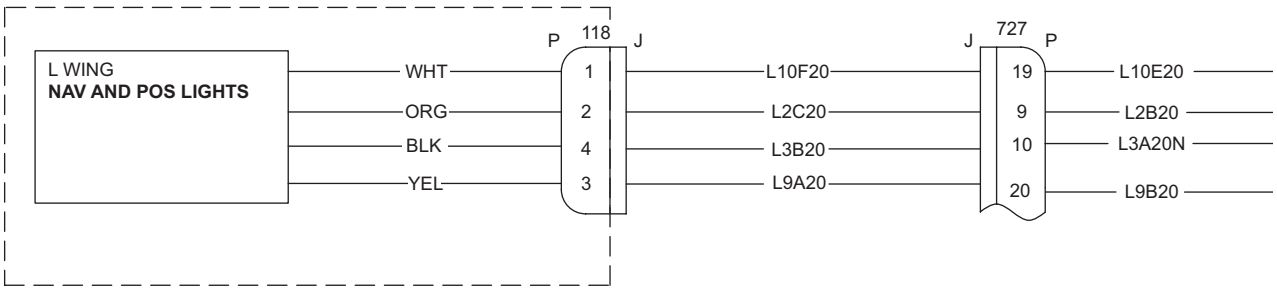
Navigation and Strobe Lights
Figure 3 (Sheet 5 of 8)



Navigation and Strobe Lights
Figure 3 (Sheet 6 of 8)

Effectivity
Seneca V
with Garmin G1000

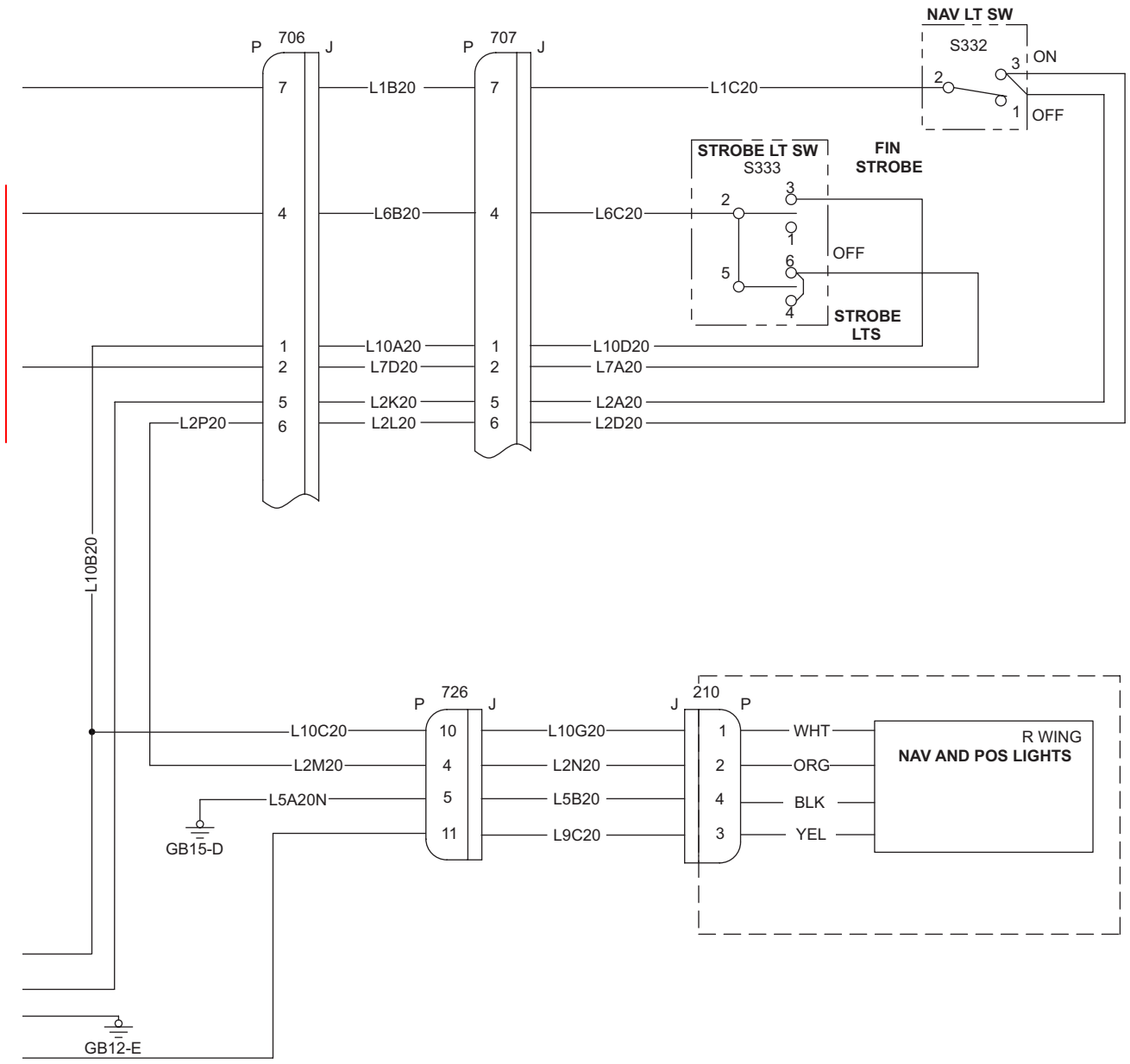




34F24A020 NEW 25
34F24A006 NEW 2
34F24A002 A 3
34W24001 NEW 4

Effectivity
Seneca V
with Garmin G1000 NXi

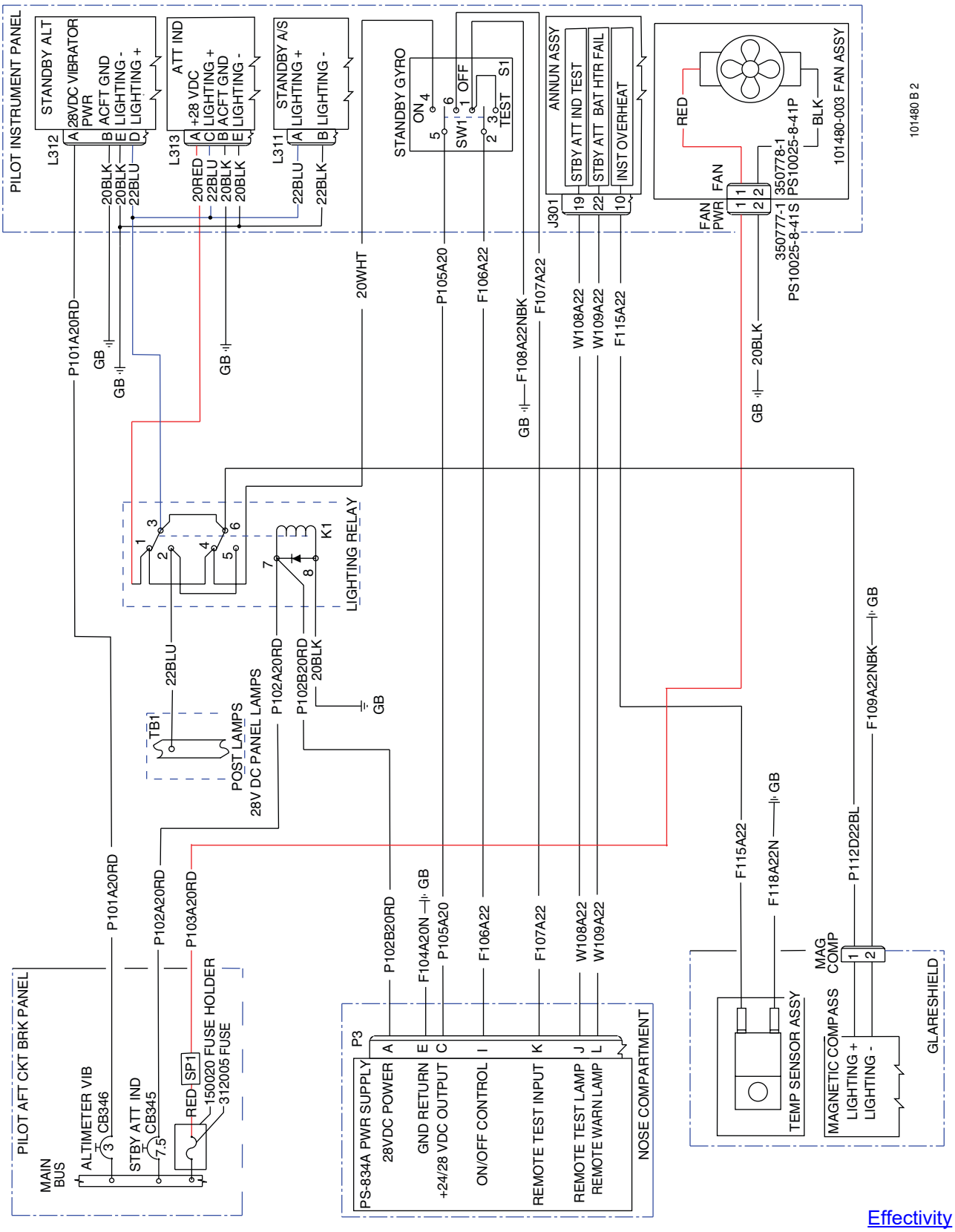
Navigation and Strobe Lights
Figure 3 (Sheet 7 of 8)



Navigation and Strobe Lights
Figure 3 (Sheet 8 of 8)

Effectivity
Seneca V
with Garmin G1000 NXi

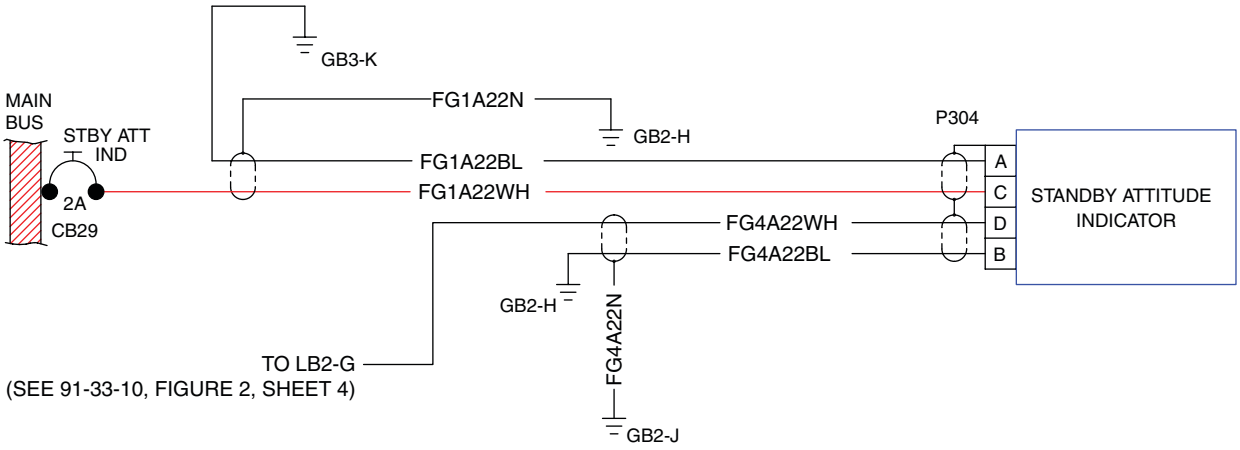
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Standby Instruments
Figure 1

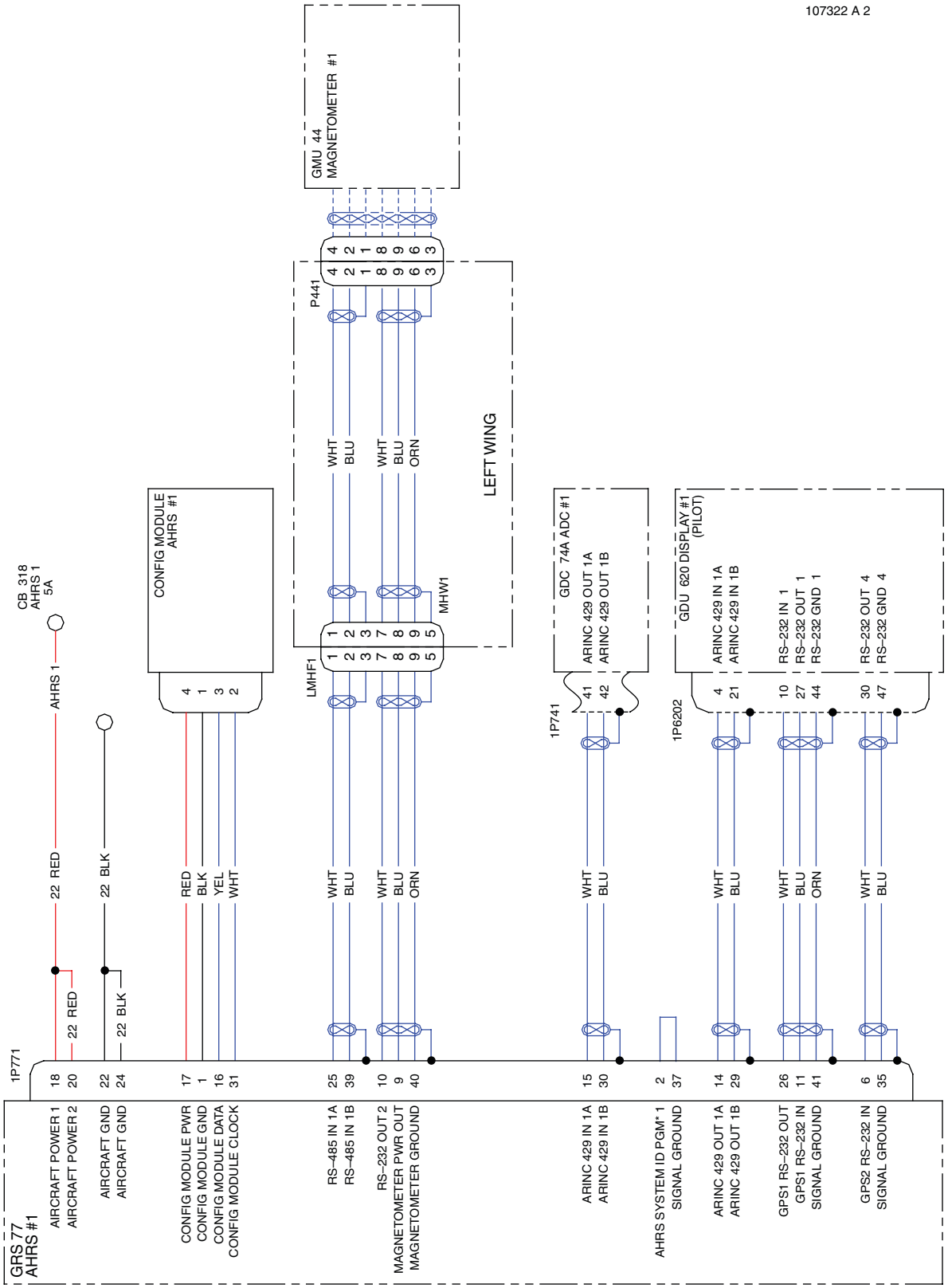
Effectivity
3449269 and up
with Single-side EFIS Option

101840 NEW 33.0



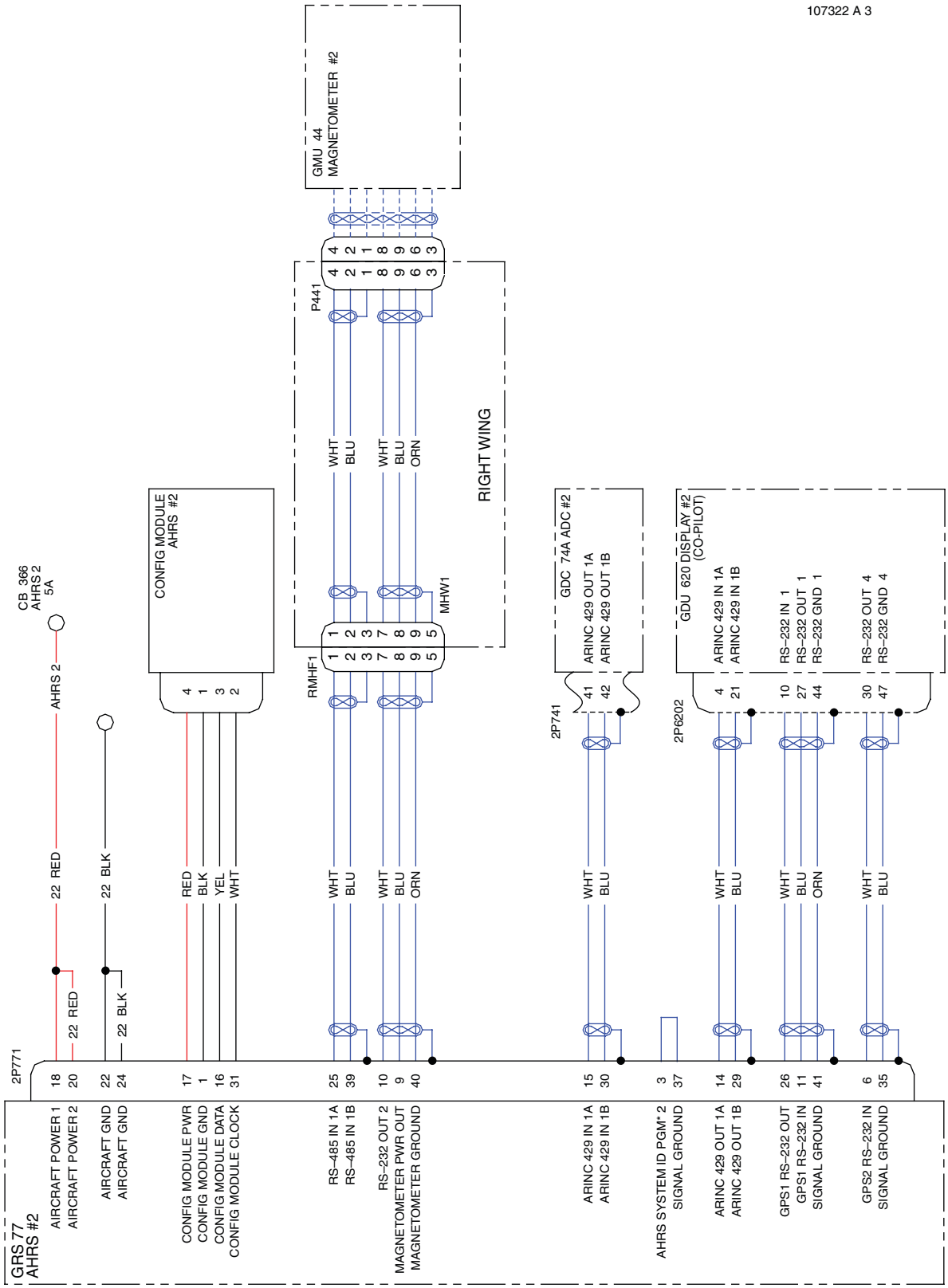
Effectivity
Seneca V
with Avidyne Entegra

Standby Attitude Indicator
Figure 2



AHRS
Figure 3 (Sheet 1 of 2)

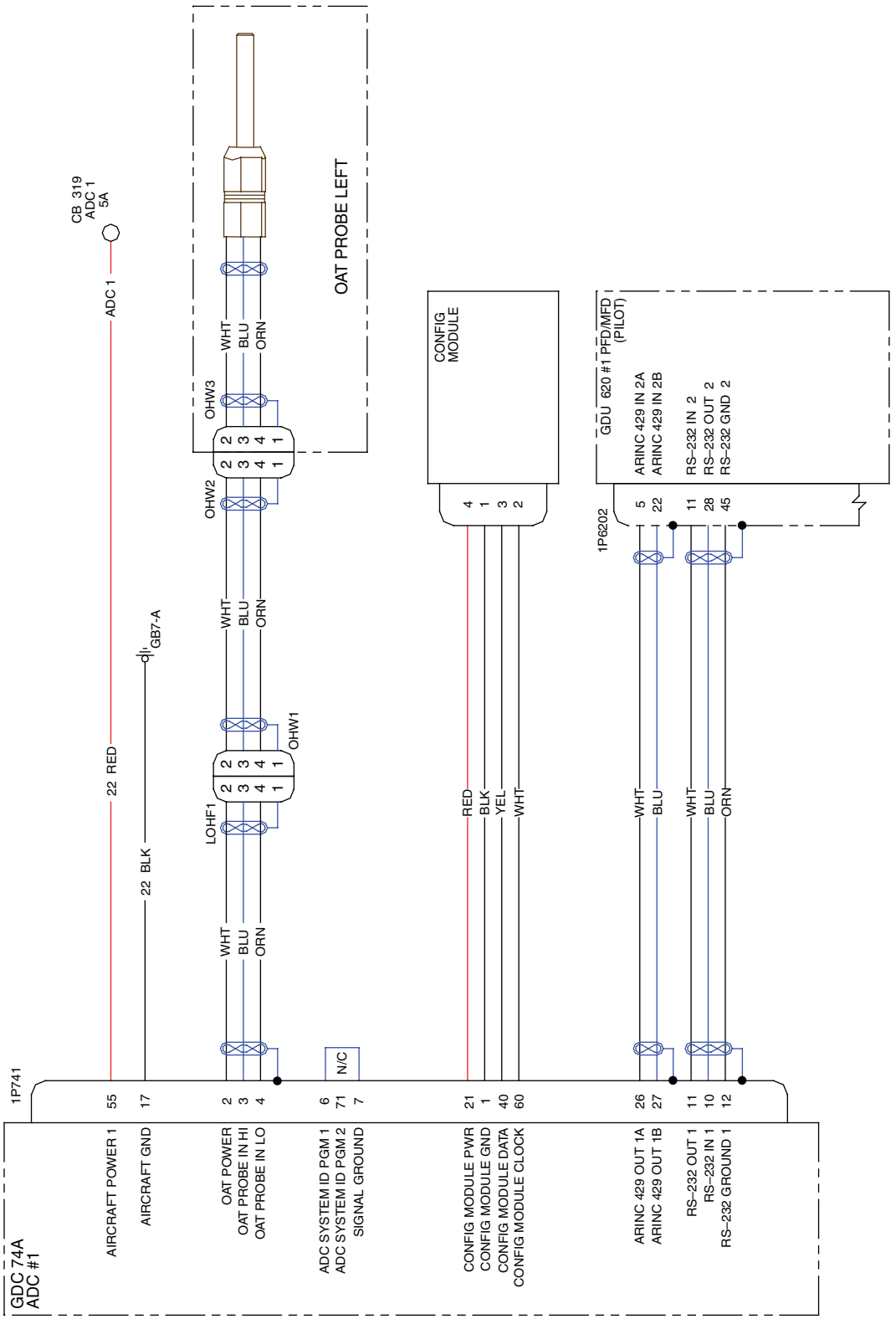
Effectivity
Seneca V
with Garmin G600



Effectivity
Seneca V
with Garmin G600

AHRS
Figure 3 (Sheet 2 of 2)

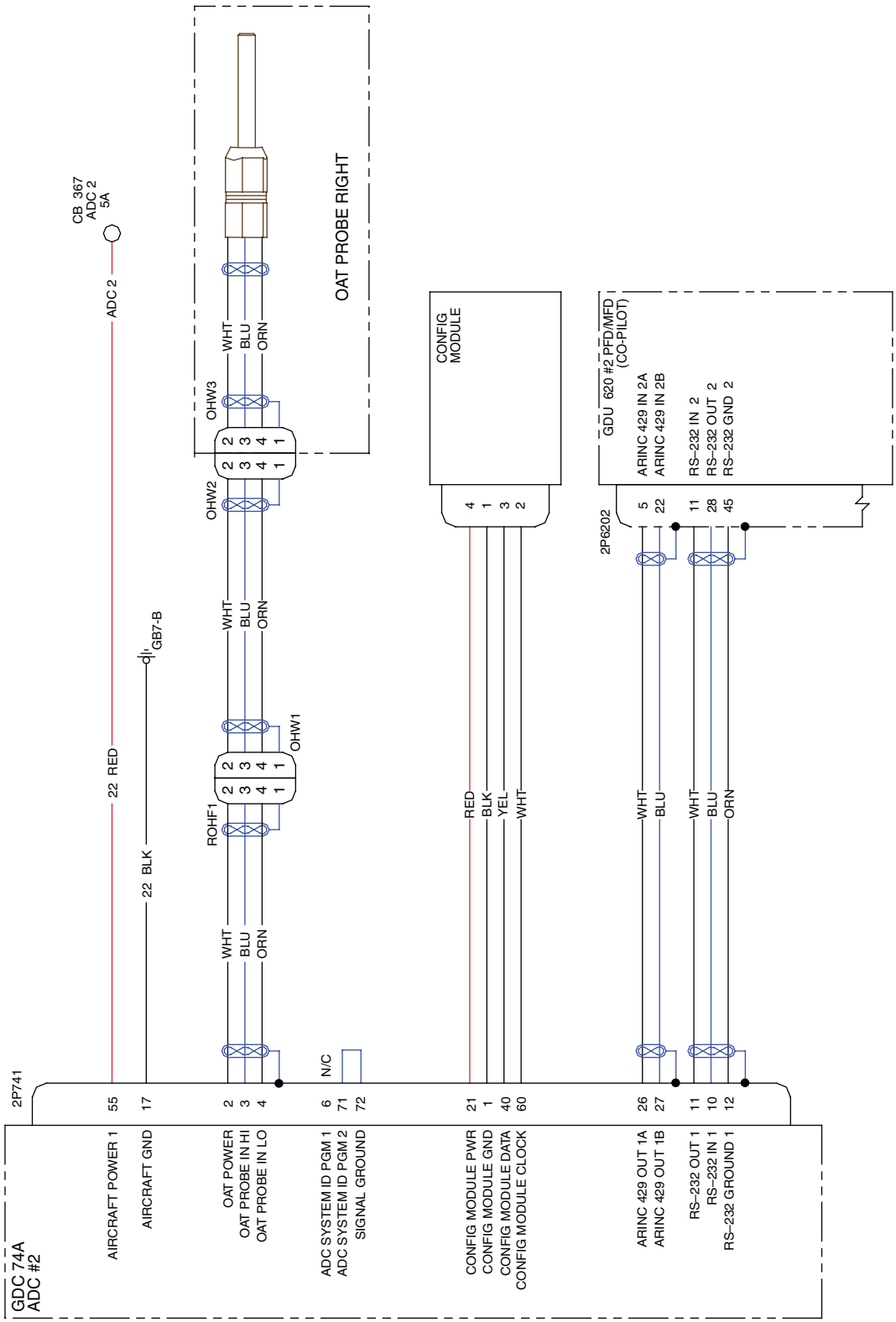
107332 C 2



Air Data Computer (ADC)/Outside Air Temp. (OAT)
Figure 4 (Sheet 1 of 2)

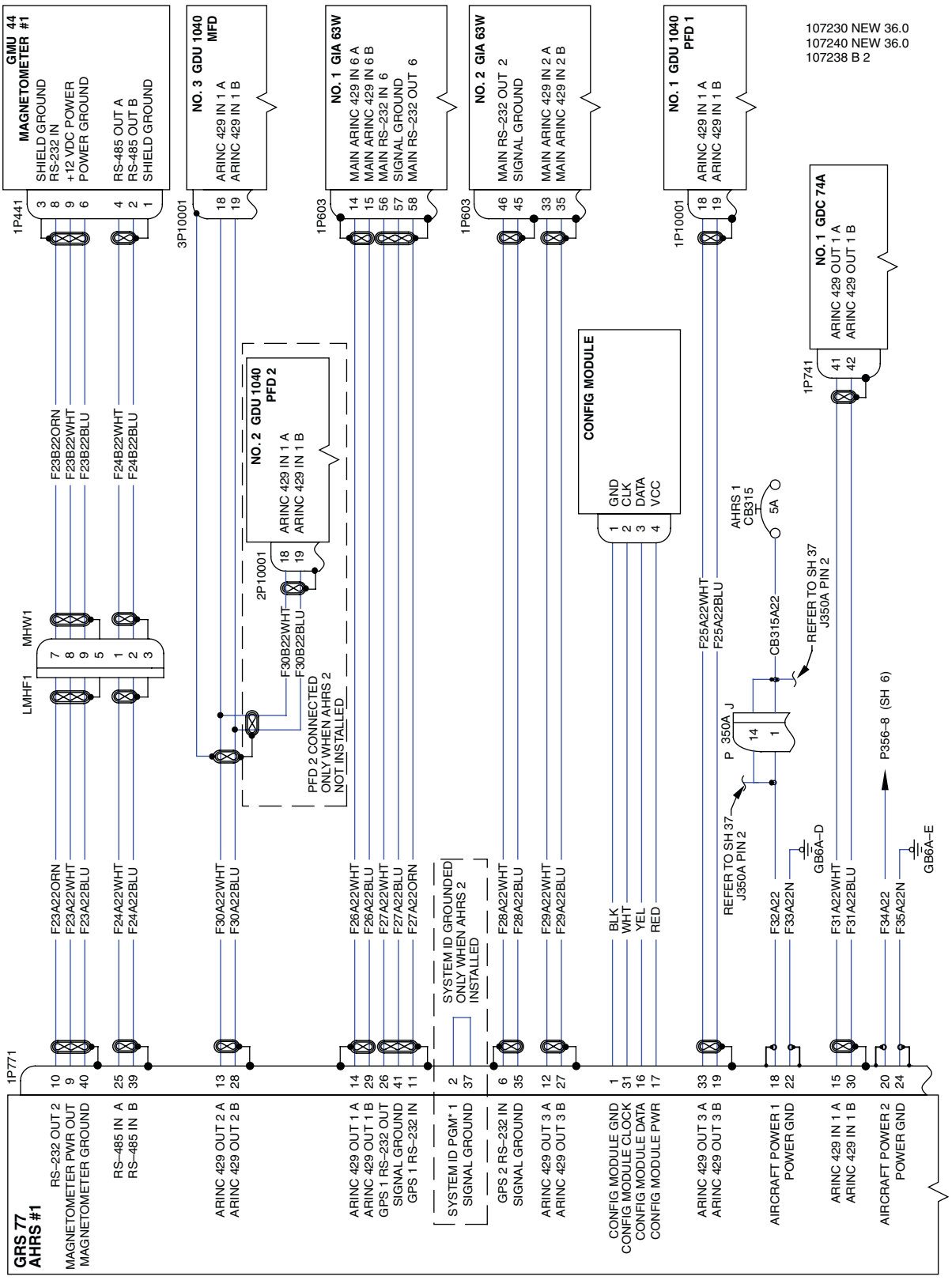
Effectivity
Seneca V
with Garmin G600

107332 C 3



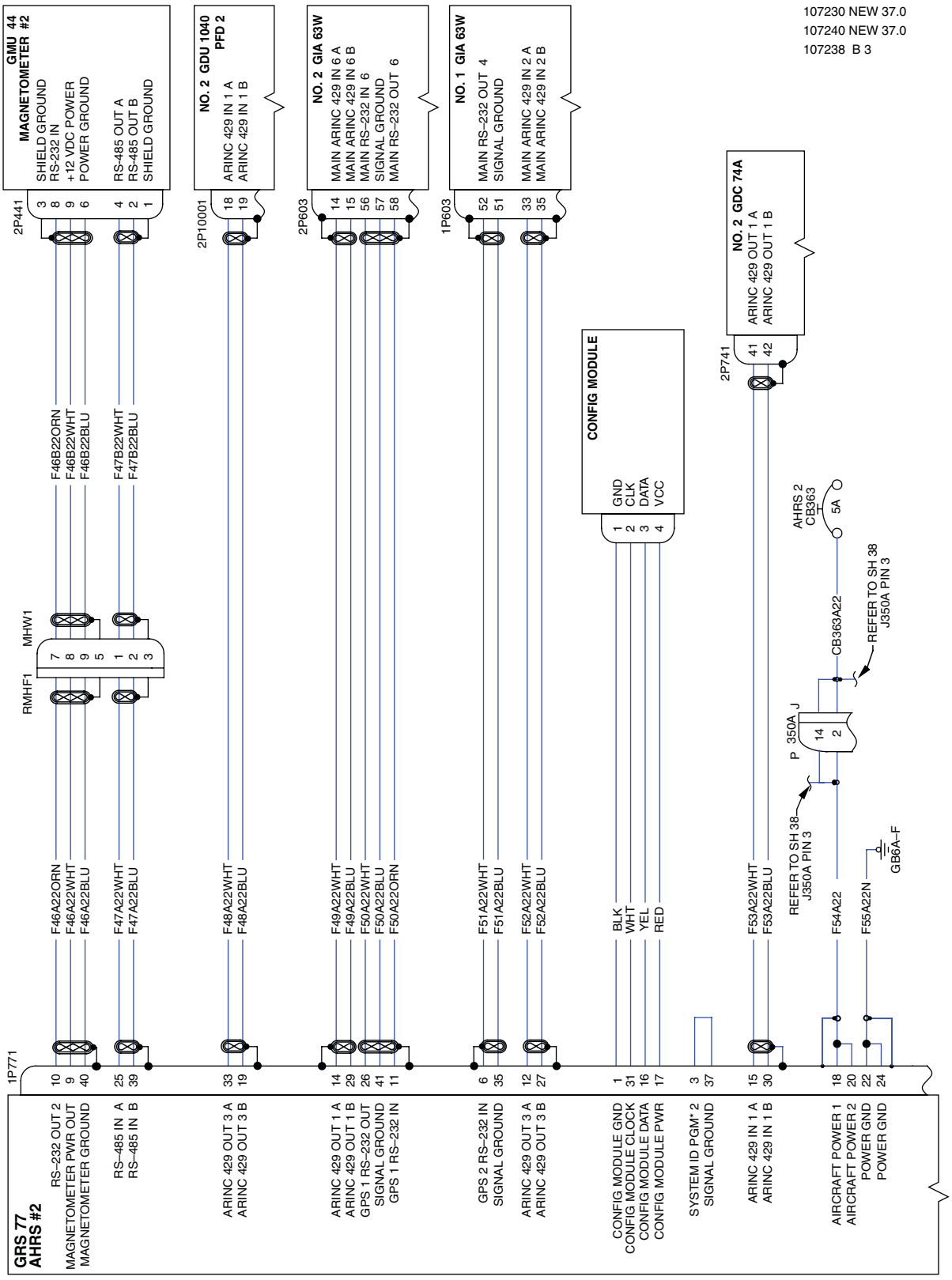
Effectivity
Seneca V
with Garmin G600

Air Data Computer (ADC)/Outside Air Temp. (OAT)
Figure 4 (Sheet 2 of 2)



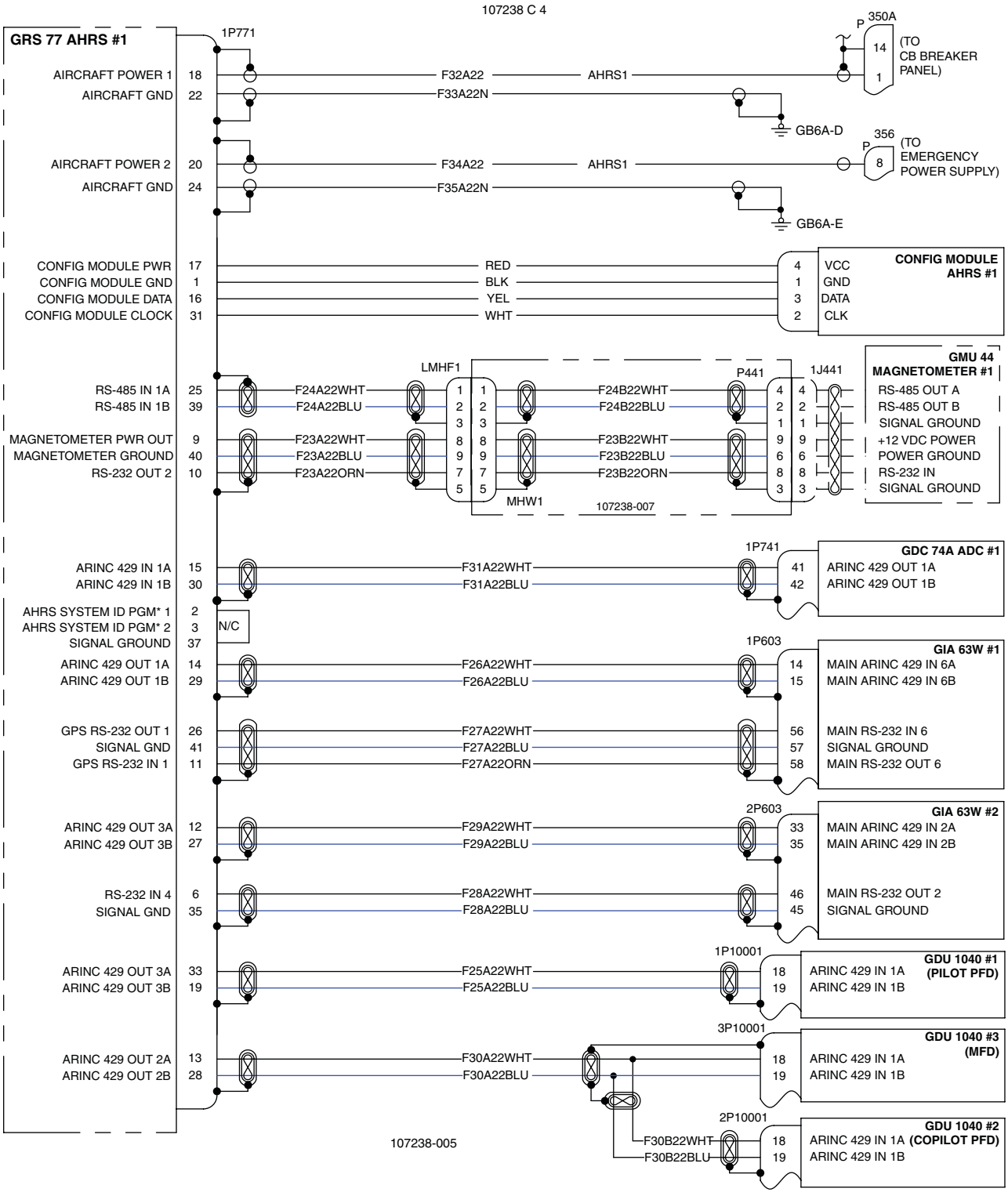
AHRS
Figure 5 (Sheet 1 of 6)

Effectivity
Seneca V
with Garmin G1000



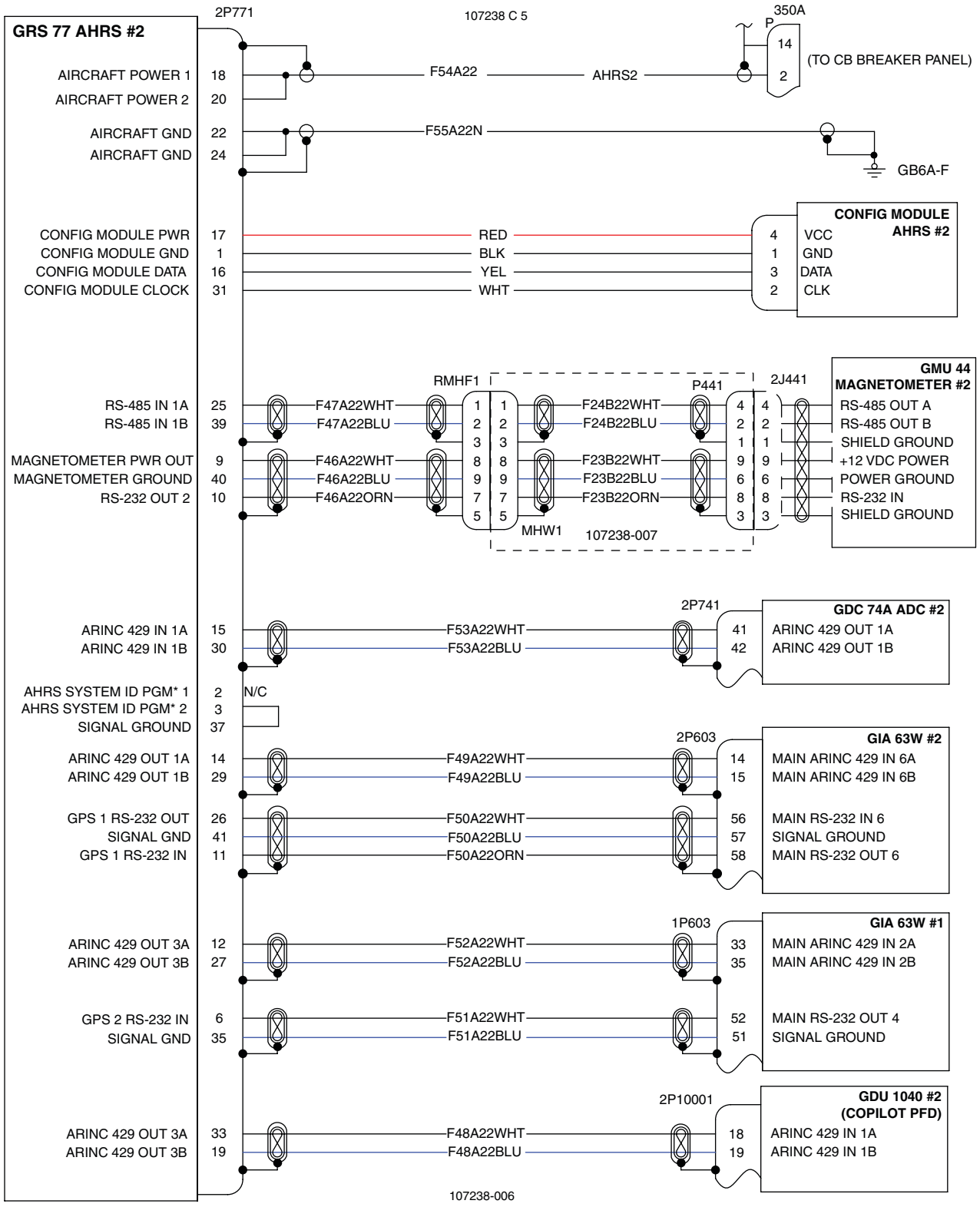
Effectivity
Seneca V
with Garmin G1000

AHRS
Figure 5 (Sheet 2 of 6)



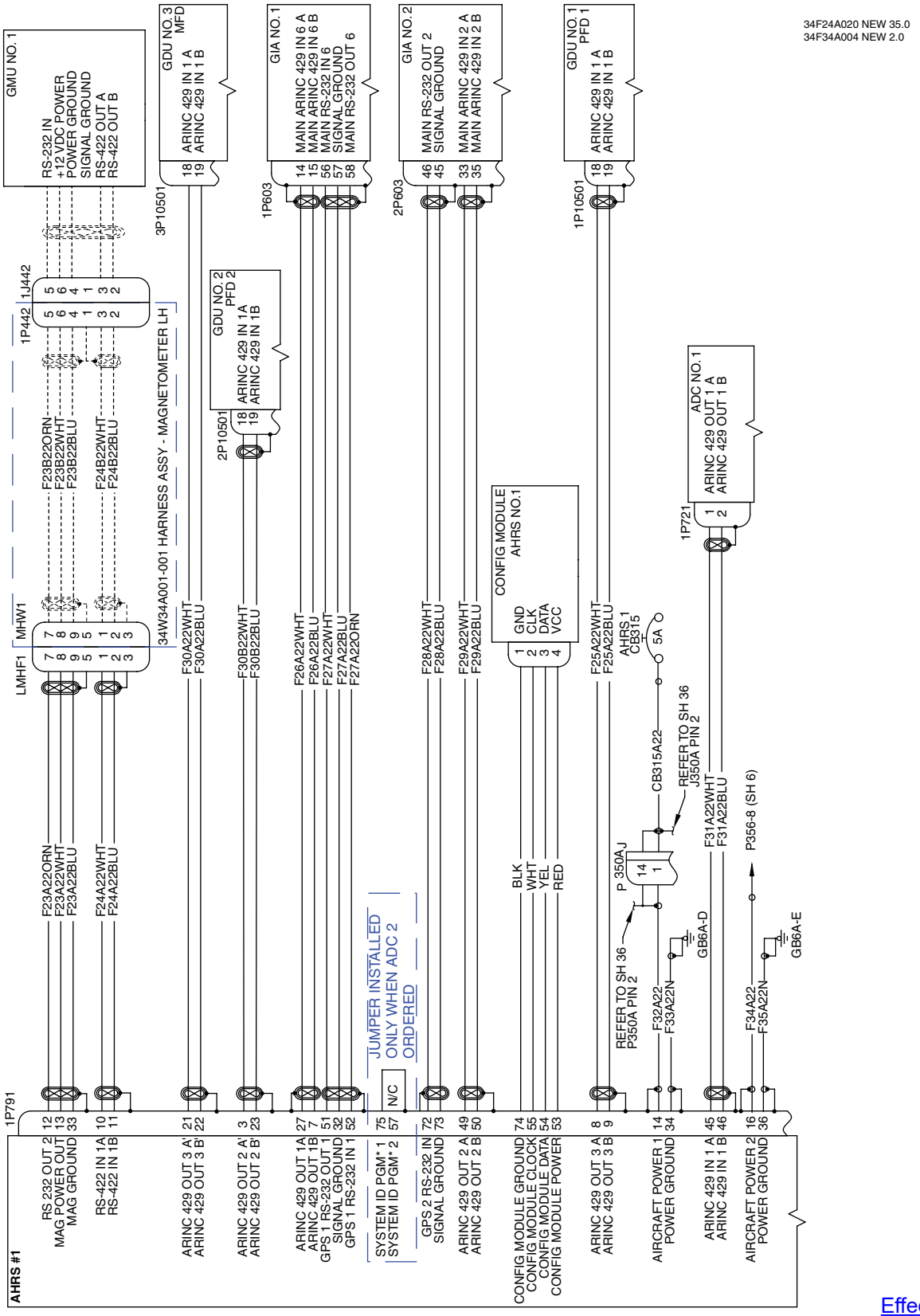
AHRS
Figure 5 (Sheet 3 of 6)

Effectivity
Seneca V
with Garmin GFC 700



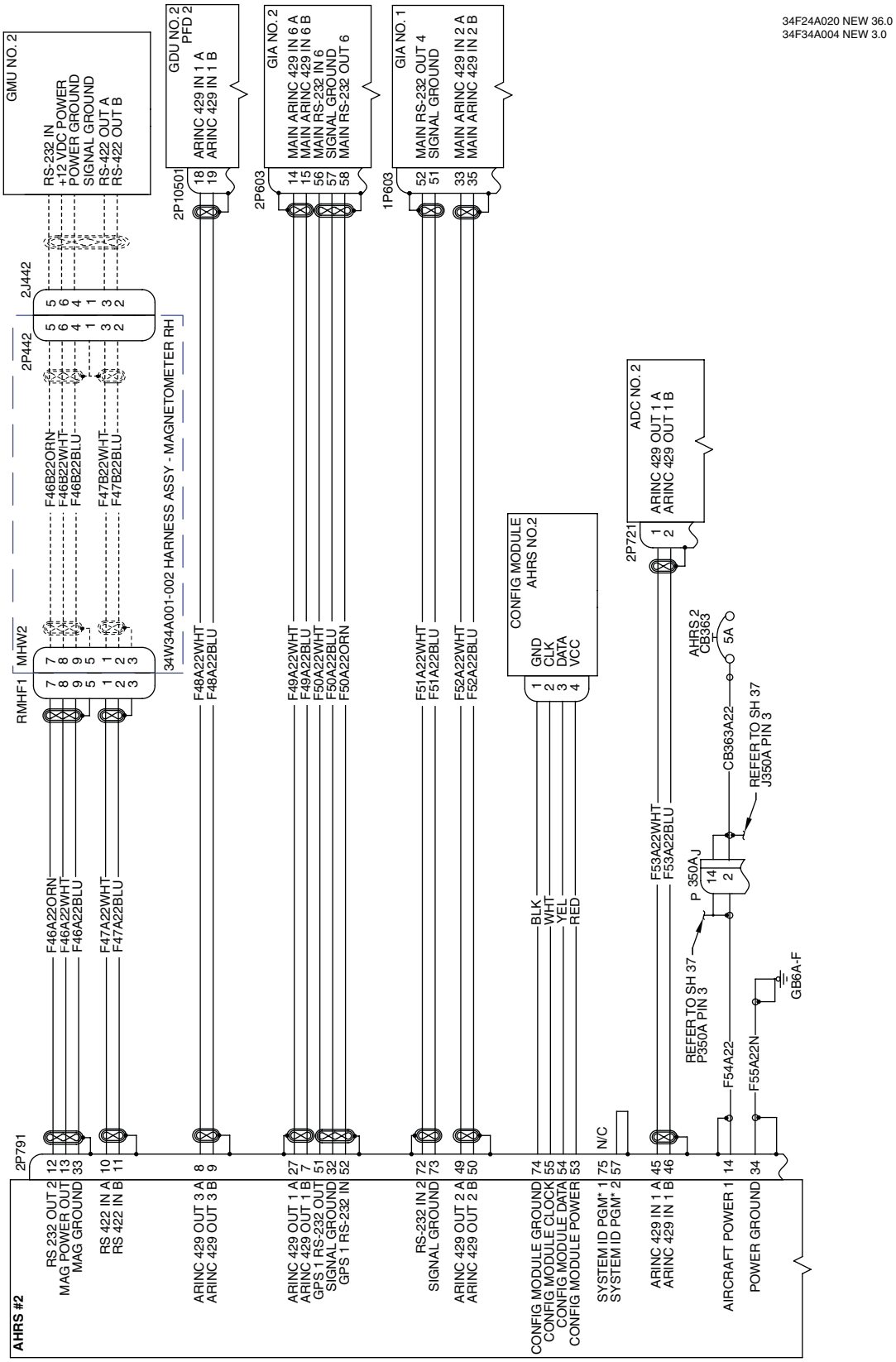
Effectivity
Seneca V
with Garmin GFC 700

AHRS
Figure 5 (Sheet 4 of 6)



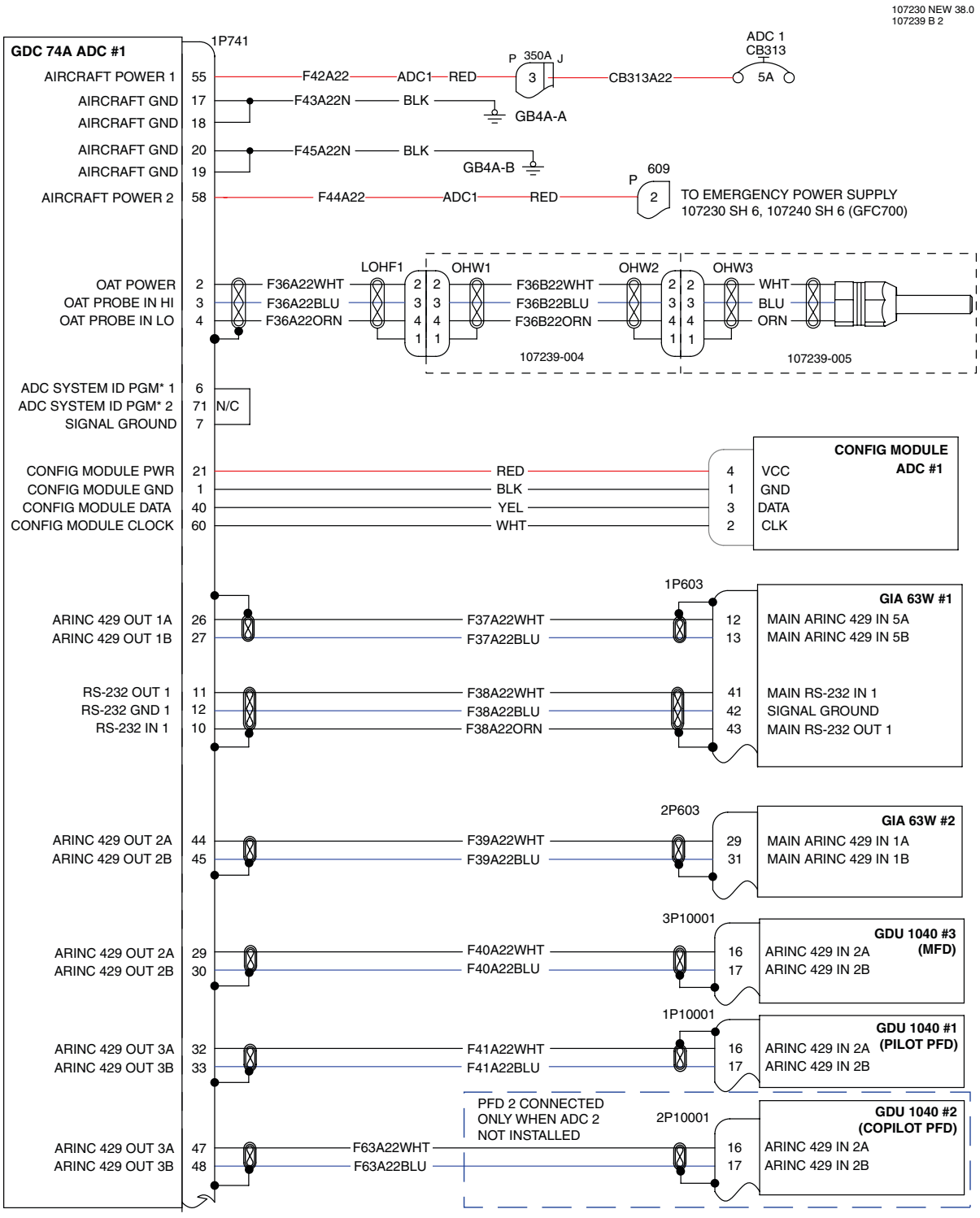
AHRS
Figure 5 (Sheet 5 of 6)

Effectivity
Seneca V
with Garmin G1000 NXi



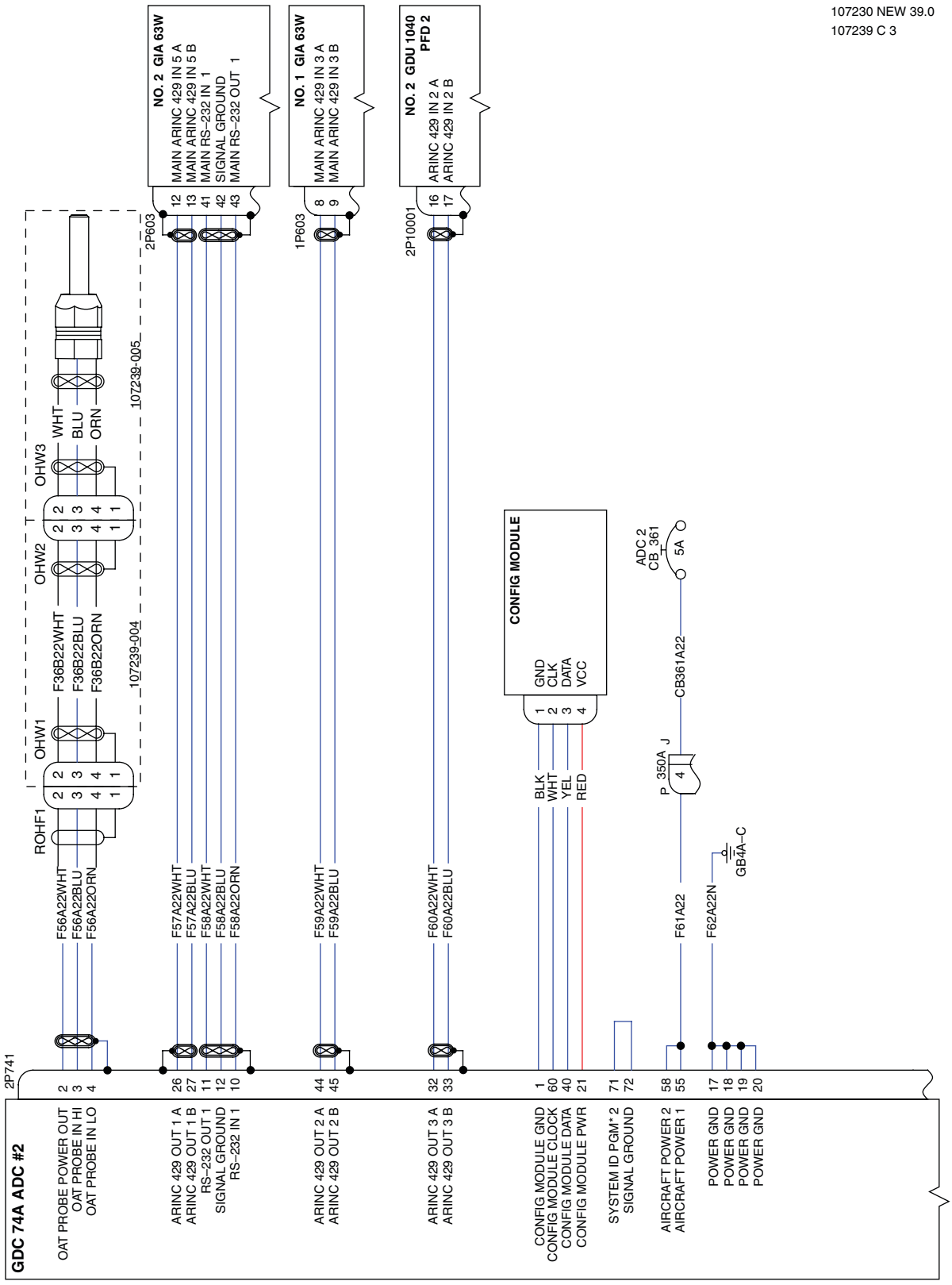
AHRS
Figure 5 (Sheet 6 of 6)

Effectivity
Seneca V
with Garmin G1000 NXi



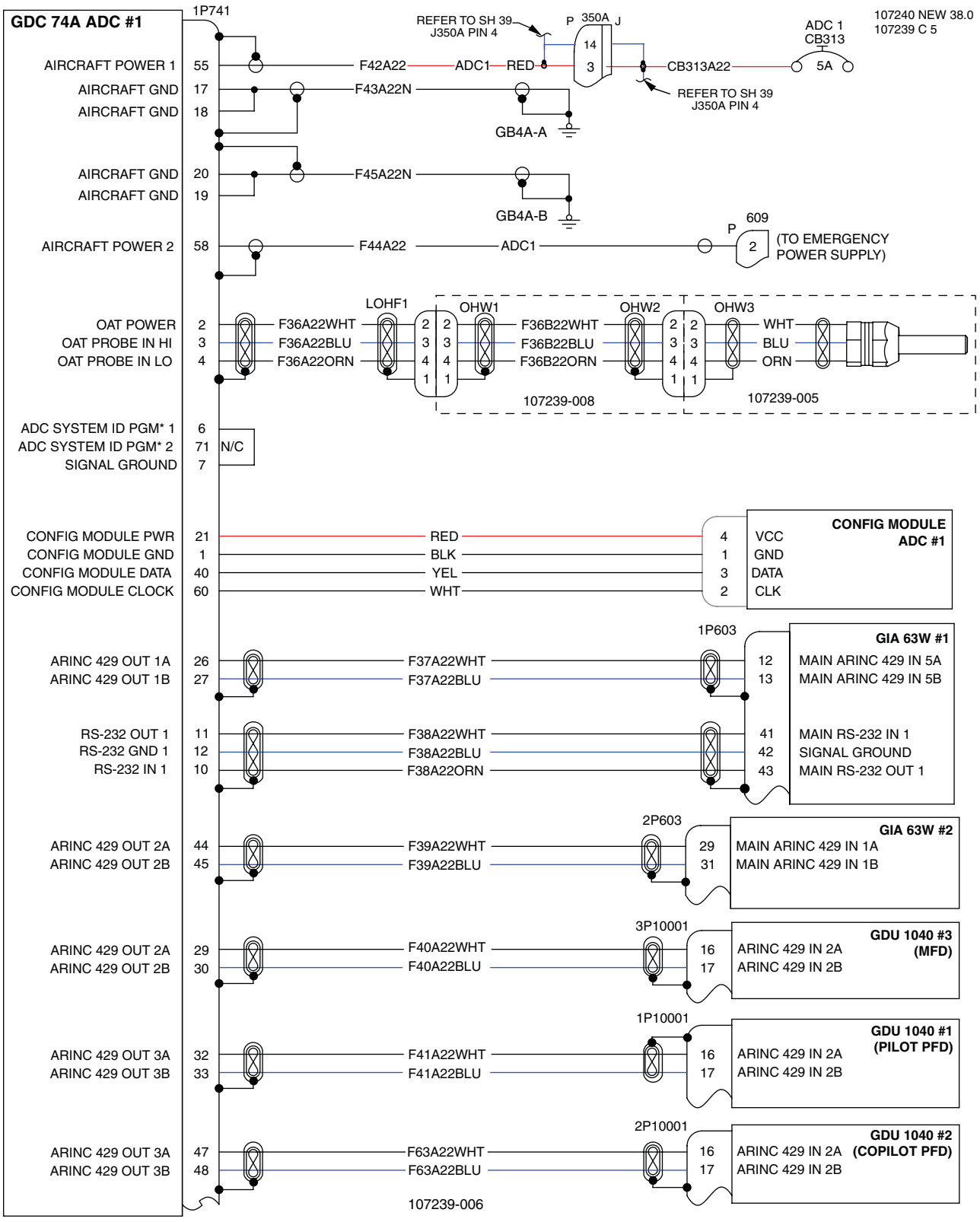
Air Data Computer (ADC) / Outside Air Temp. (OAT)
Figure 6 (Sheet 1 of 6)

Effectivity
Seneca V
with Garmin G1000



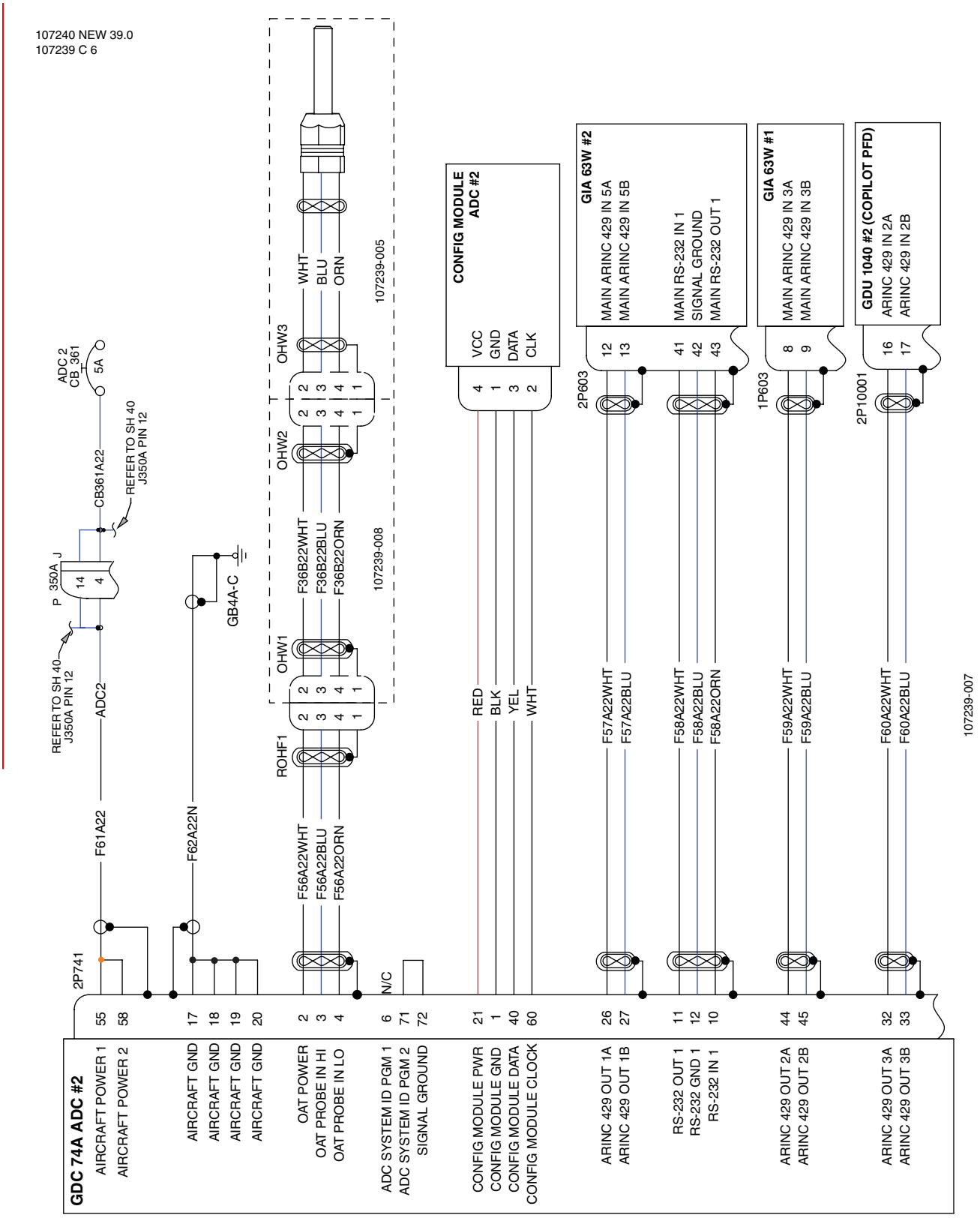
Effectivity
Seneca V
with Garmin G1000

Air Data Computer (ADC) / Outside Air Temp. (OAT)
Figure 6 (Sheet 2 of 6)



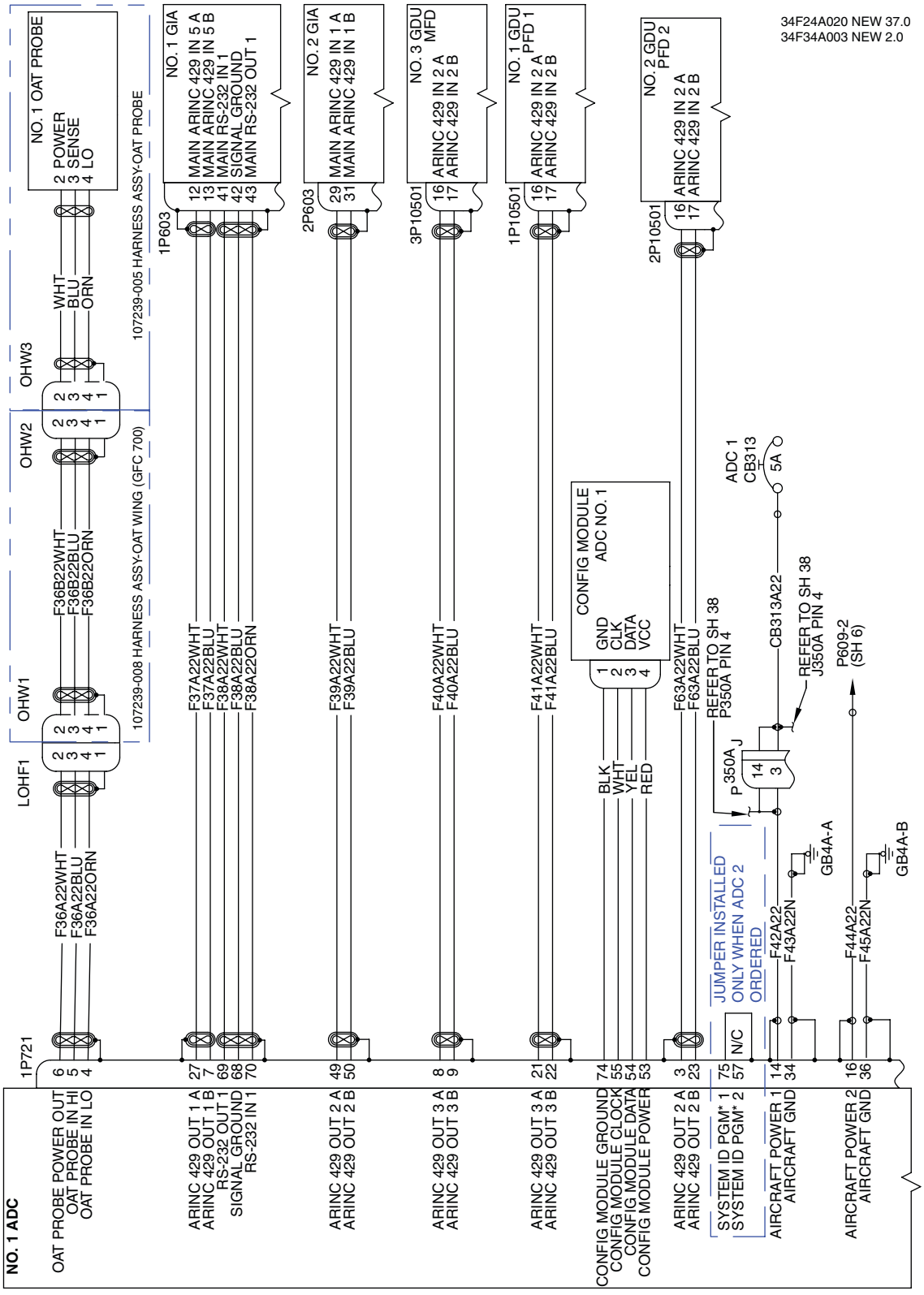
Air Data Computer (ADC) / Outside Air Temp. (OAT)
Figure 6 (Sheet 3 of 6)

Effectivity
Seneca V
with Garmin GFC 700



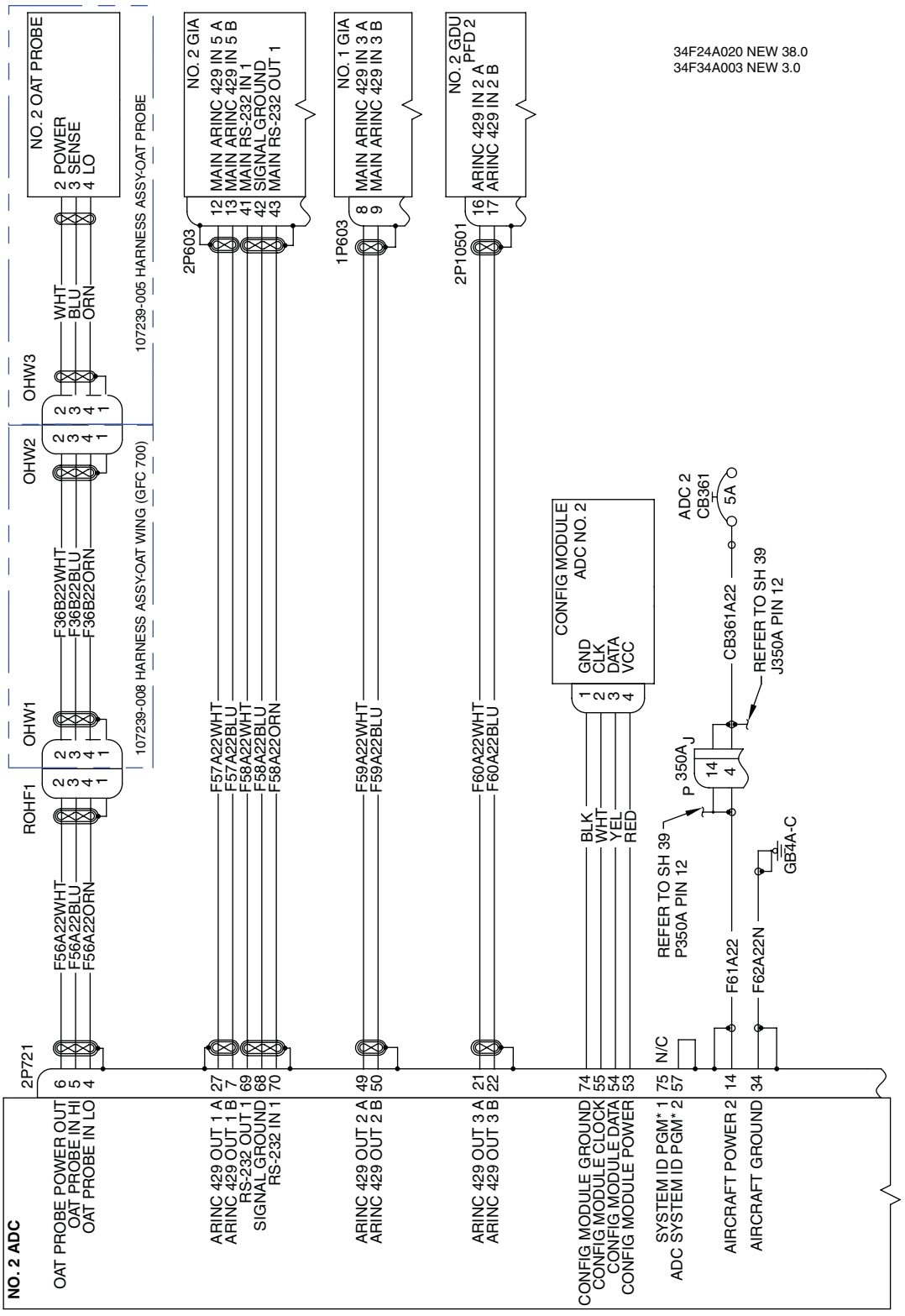
Effectivity
Seneca V
with Garmin GFC 700

Air Data Computer (ADC) / Outside Air Temp. (OAT)
Figure 6 (Sheet 4 of 6)



Air Data Computer (ADC) / Outside Air Temp. (OAT)
Figure 6 (Sheet 5 of 6)

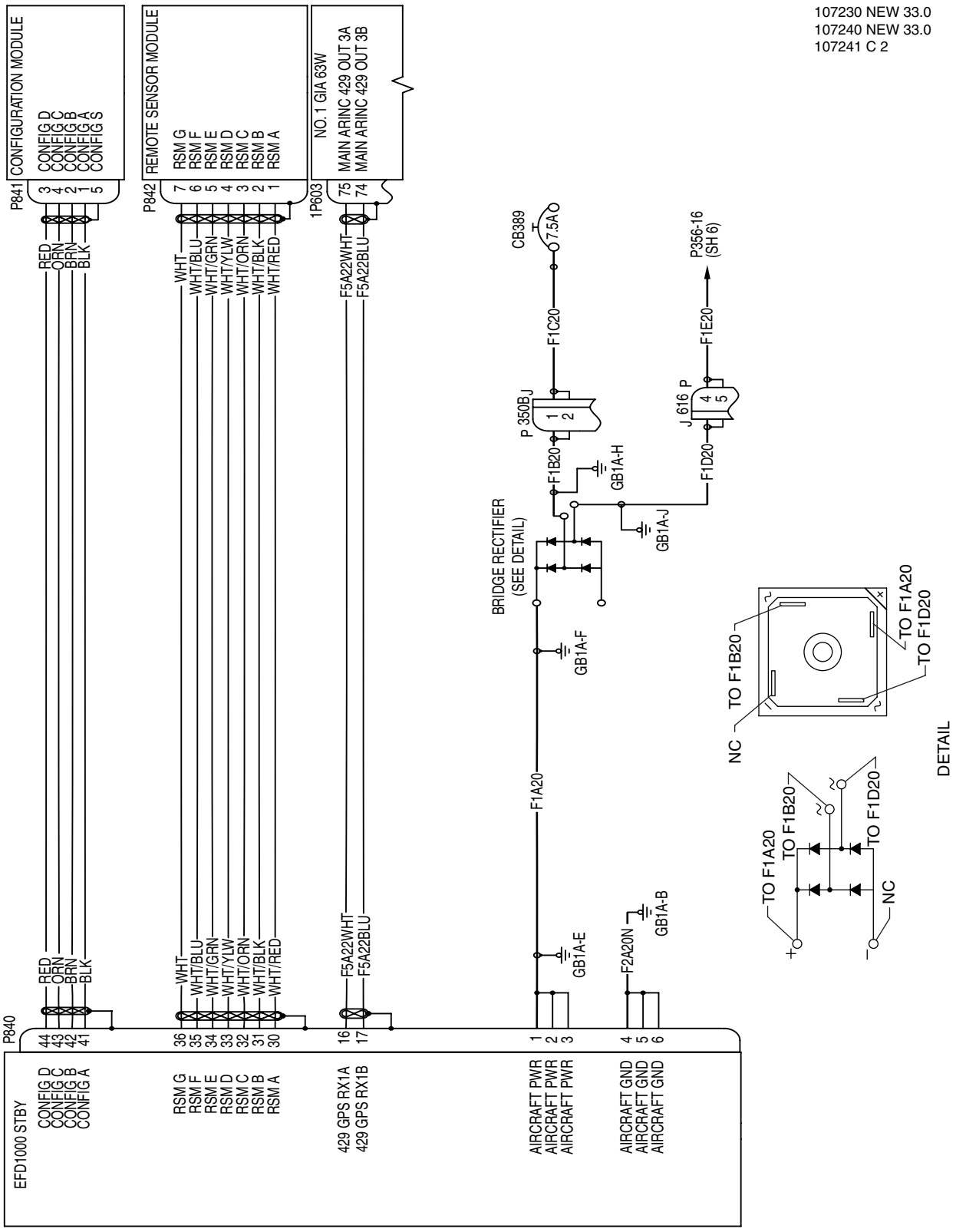
Effectivity
Seneca V
with Garmin G1000 NXi



Effectivity
Seneca V
with Garmin G1000 NXi

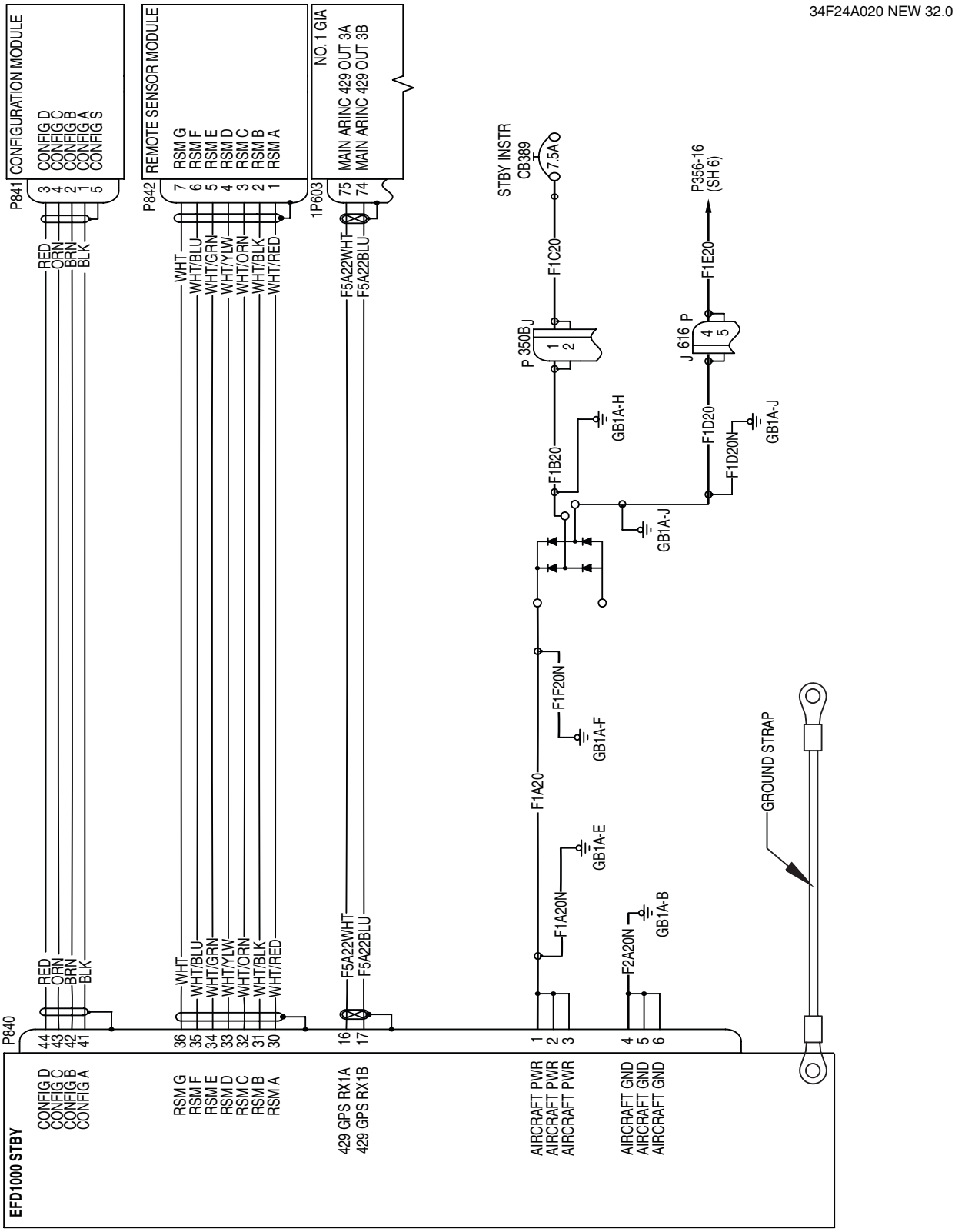
Air Data Computer (ADC) / Outside Air Temp. (OAT)
Figure 6 (Sheet 6 of 6)

107230 NEW 33.0
107240 NEW 33.0
107241 C 2



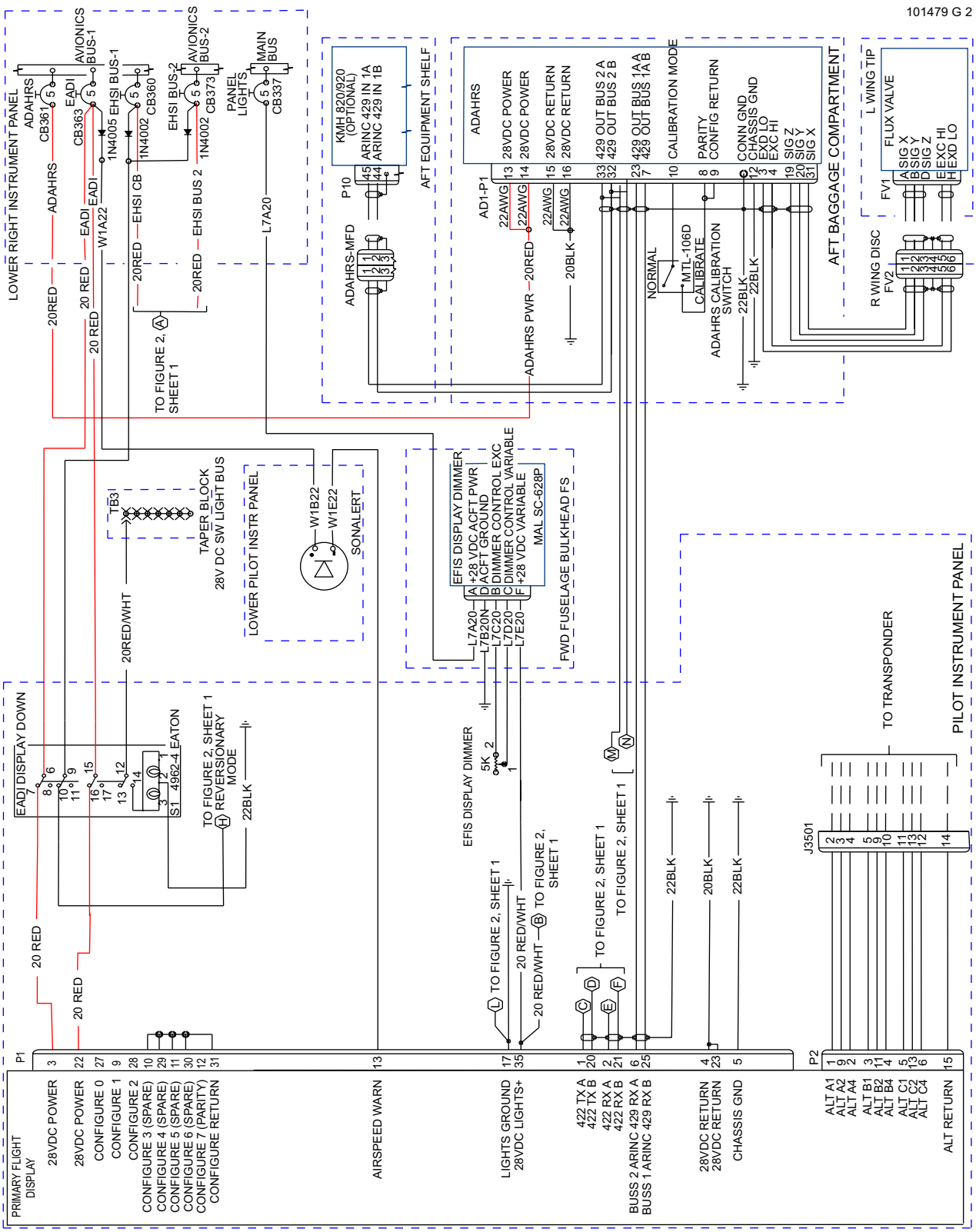
EBD Standby Instruments
Figure 7 (Sheet 1 of 2)

Effectivity
Seneca V
with Garmin G1000



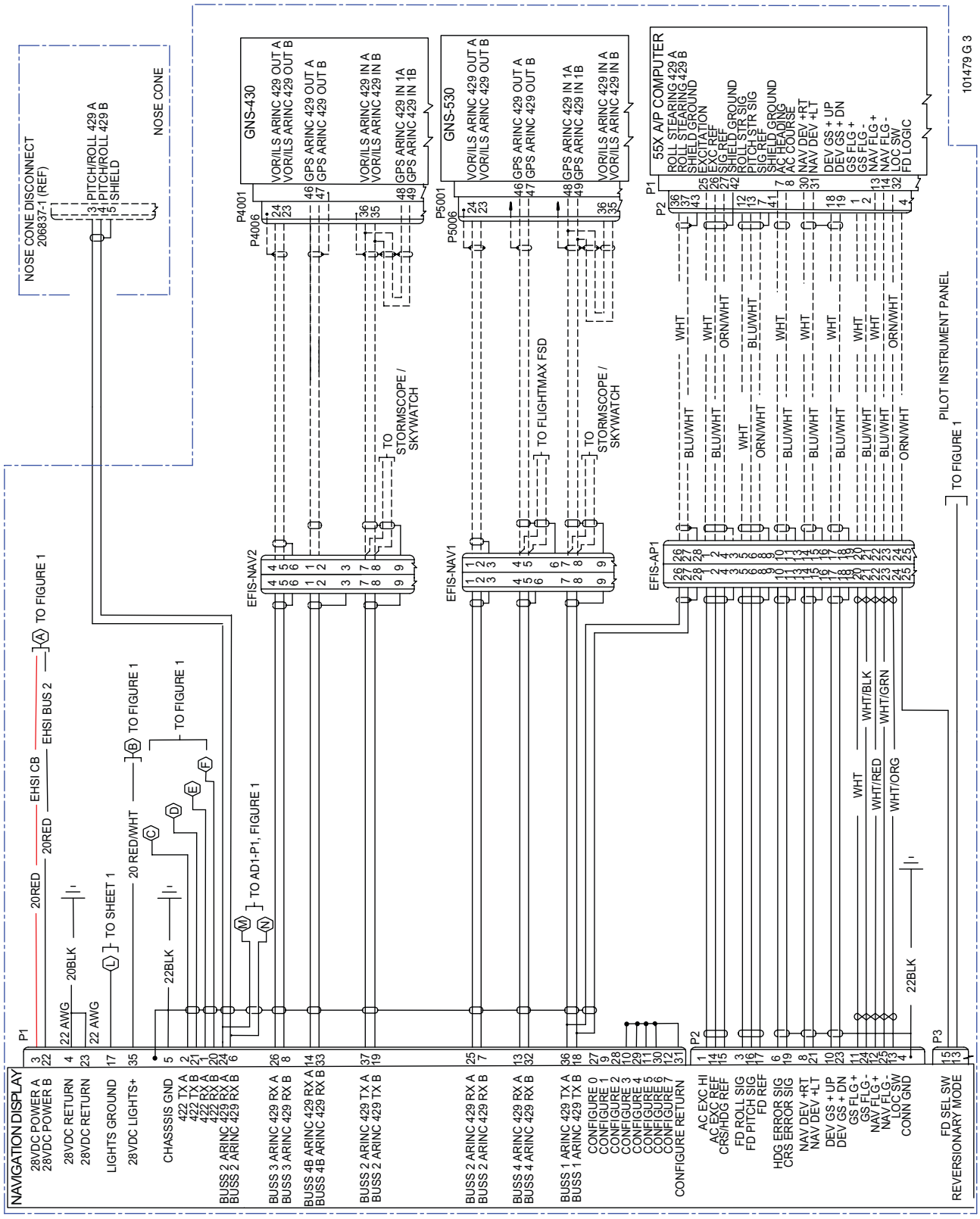
Effectivity
Seneca V
with Garmin G1000 NXi

EBD Standby Instruments
Figure 7 (Sheet 2 of 2)



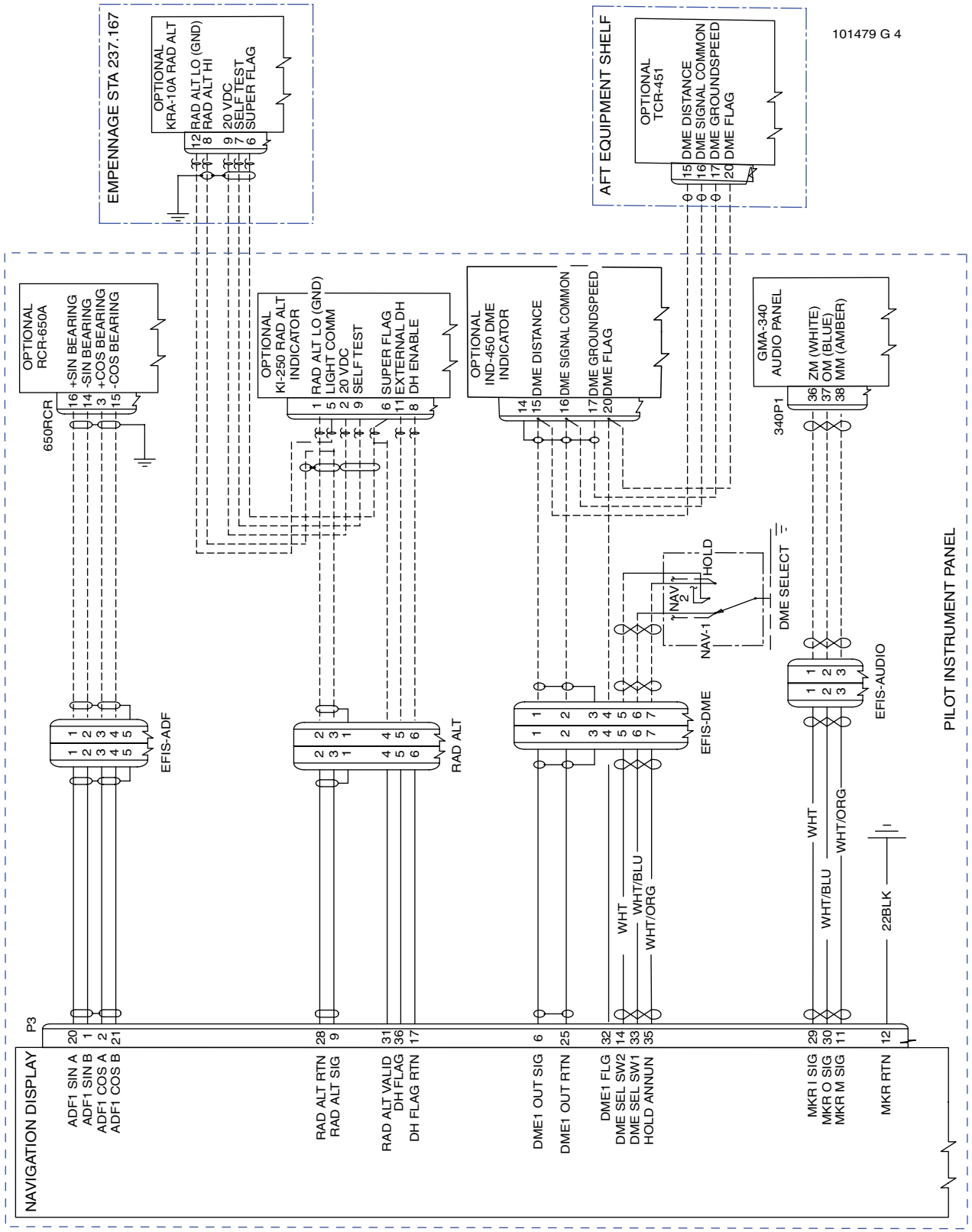
Primary Flight Display (PFD) (Meggitt)
Figure 1

Effectivity
3449269 and up
with Single-Side EFIS



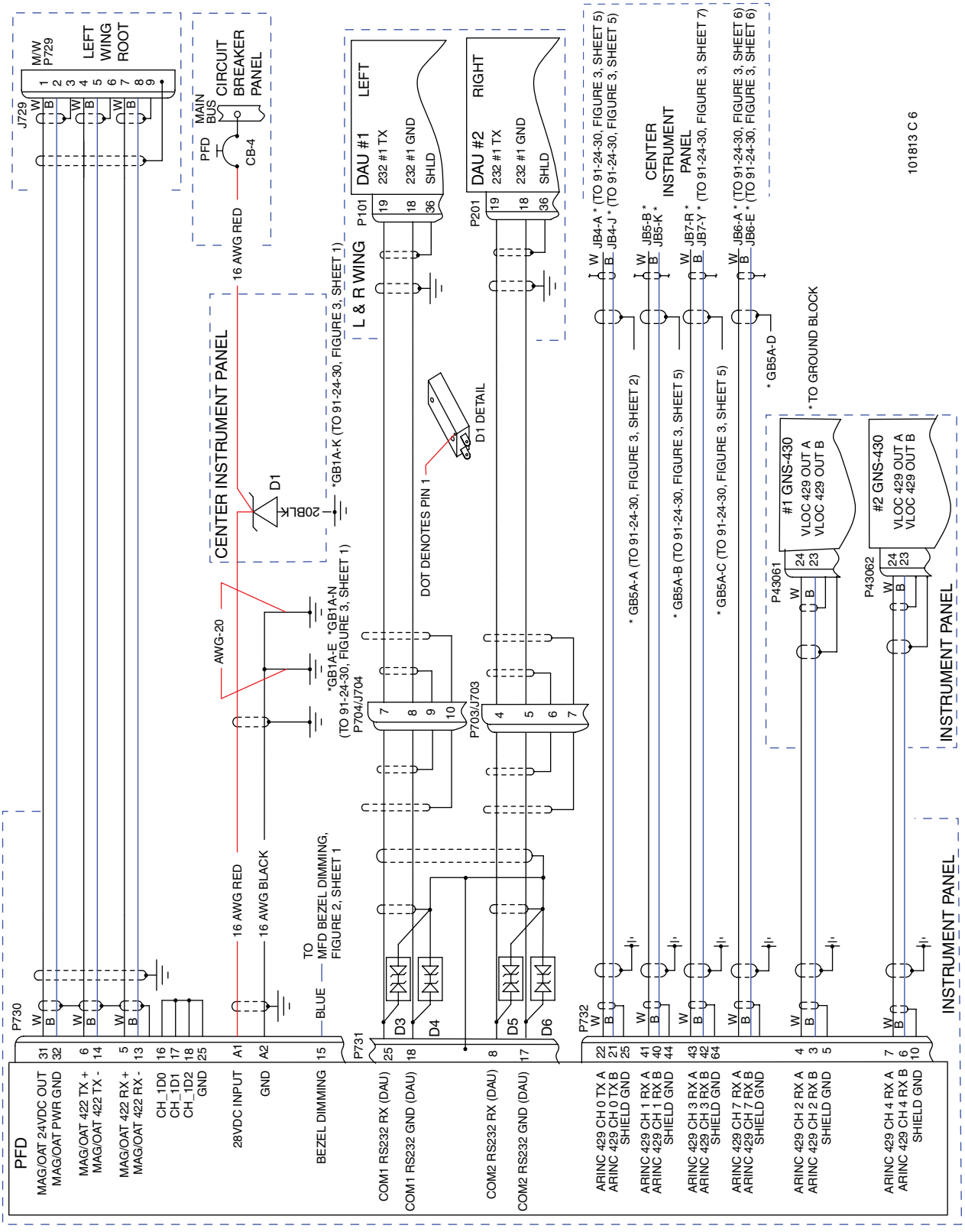
Effectivity
3449269 and up
with Single-side EFIS

Navigation Display (ND) (Meggitt)
Figure 2 (Sheet 1 of 2)



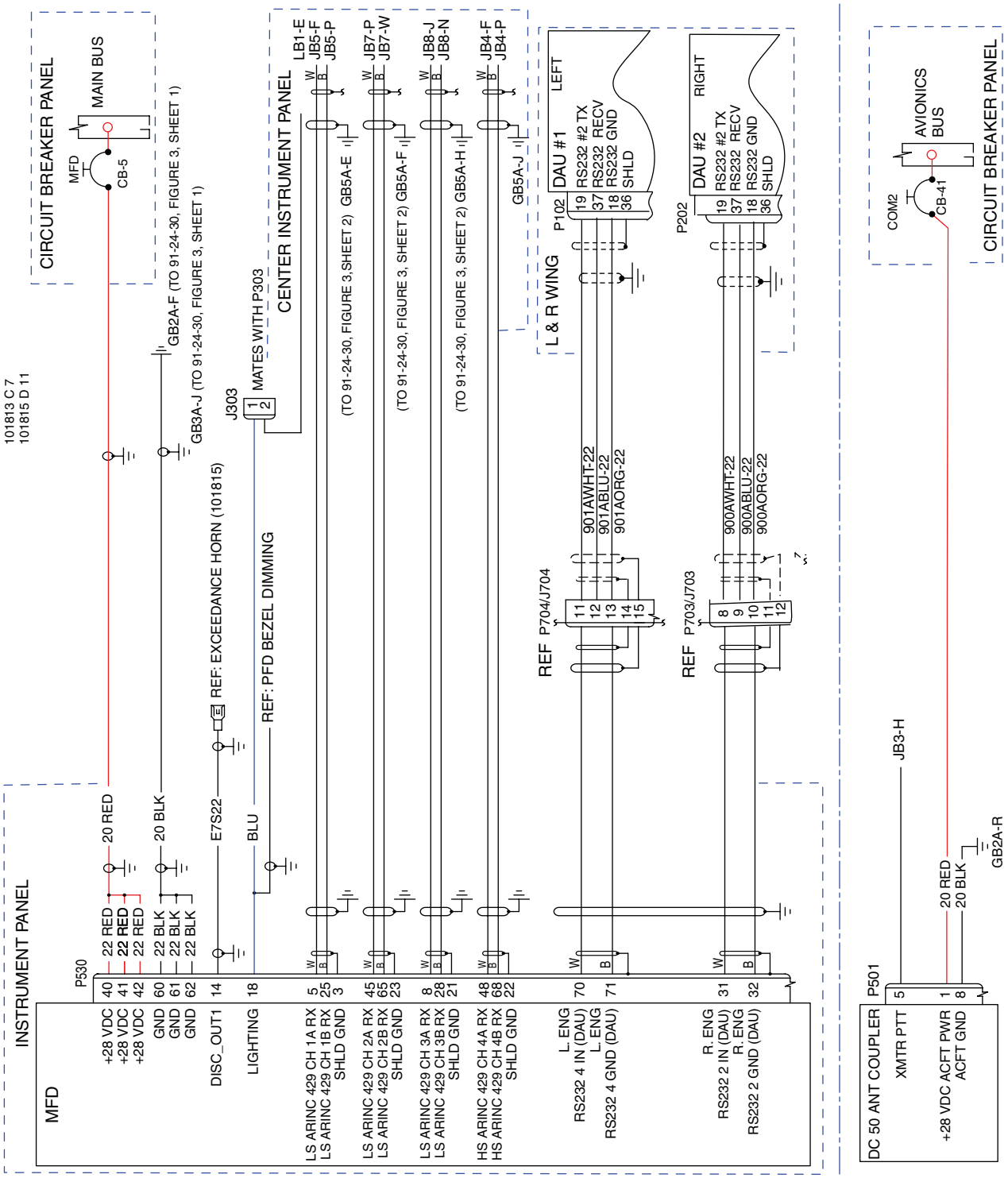
Navigation Display (ND) (Meggitt))
Figure 2 (Sheet 2 of 2)

Effectivity
3449269 and up
with Single-side EFIS



Effectivity
Seneca V
with Avidyne Entegra

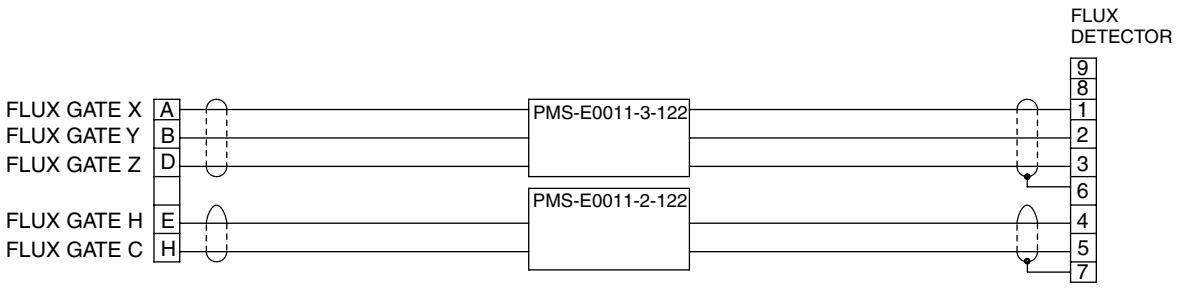
Primary Flight Display (PFD) (Avidyne IFDS)
Figure 3



Multi-Function Display (MFD) (Avidyne IFDS)
Figure 4

Effectivity
Seneca V
with Avidyne Entegra

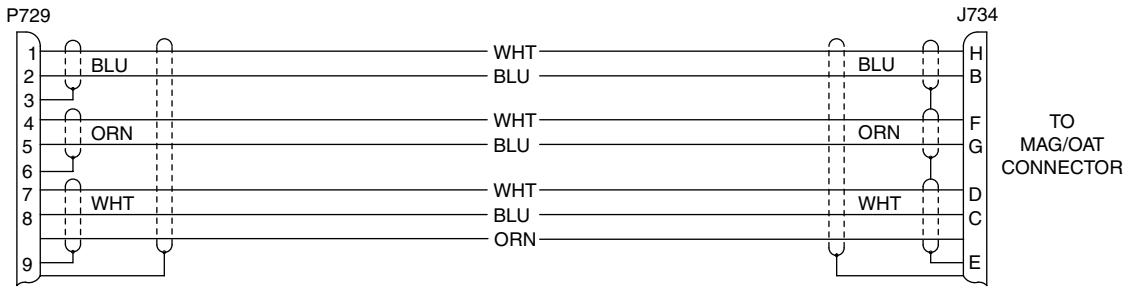
101261 J



STANDARD

WITH AVIDYNE ENTEGRA

101753 C

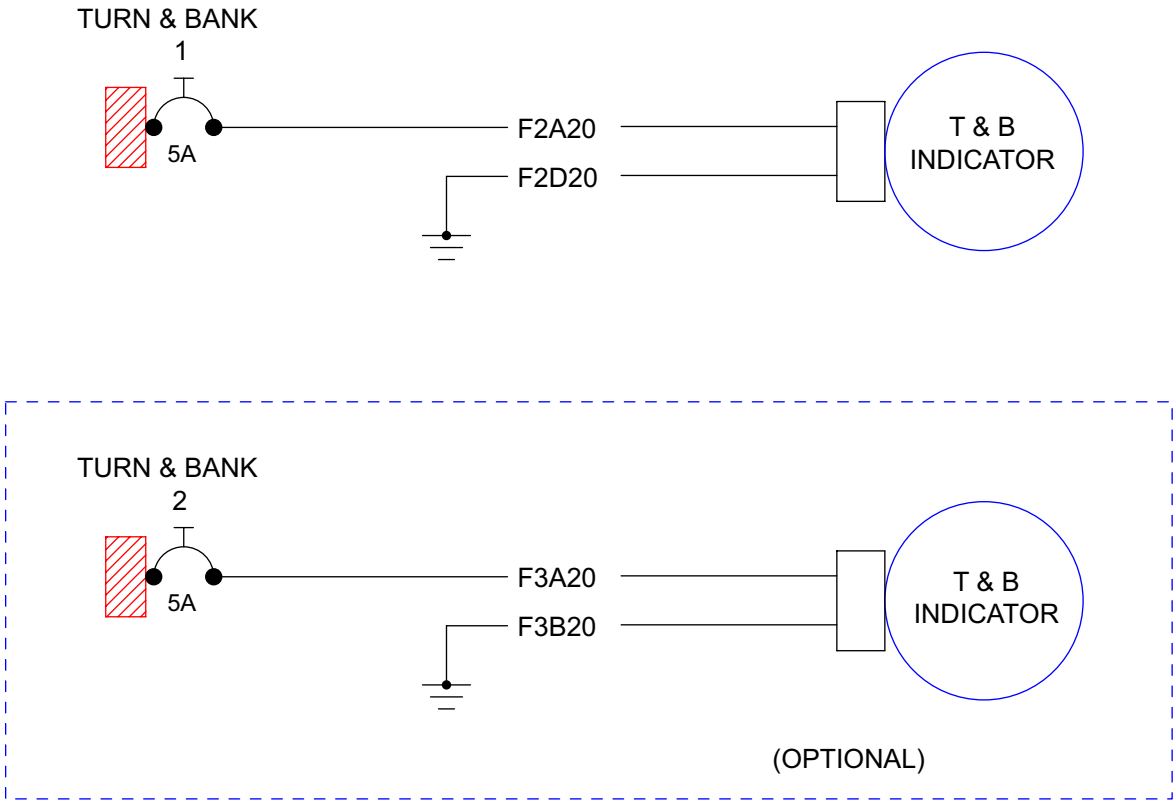


ALL WIRE IS MINIMUM 22 AWG UNLESS OTHERWISE SPECIFIED.

Effectivity
3449152 and up

Flux Detector
Figure 5

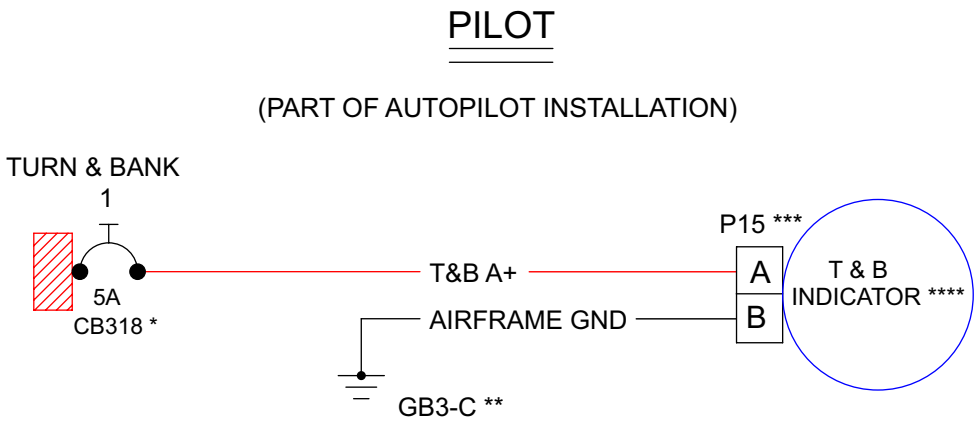
83353 NEW/G 9
100651 NEW/T 9
101288 NEW/D 9.0

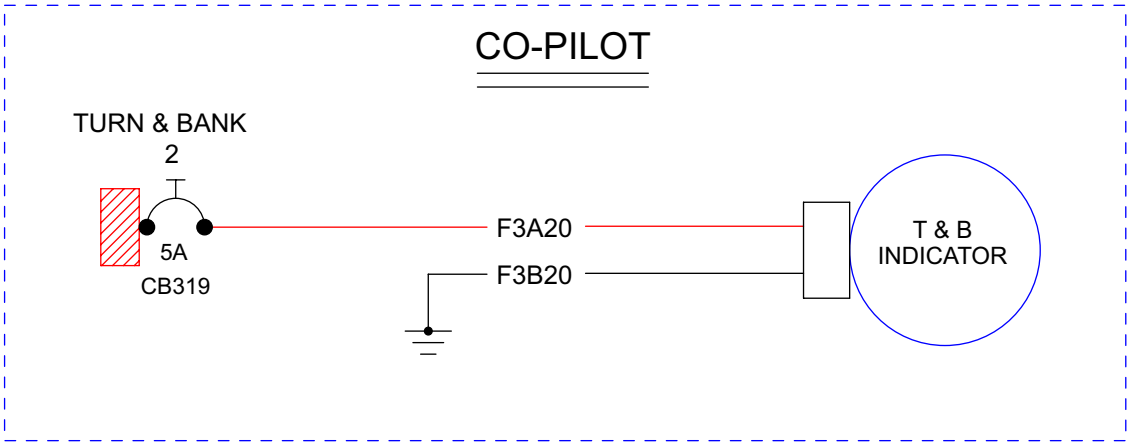


Turn and Bank
Figure 6 (Sheet 1 of 2)

Effectivity
Seneca IV and,
3449001 thru 3449151

104229 C 9.0
104368 A/K 9.0
101810 6.0
101840 NEW 6.0

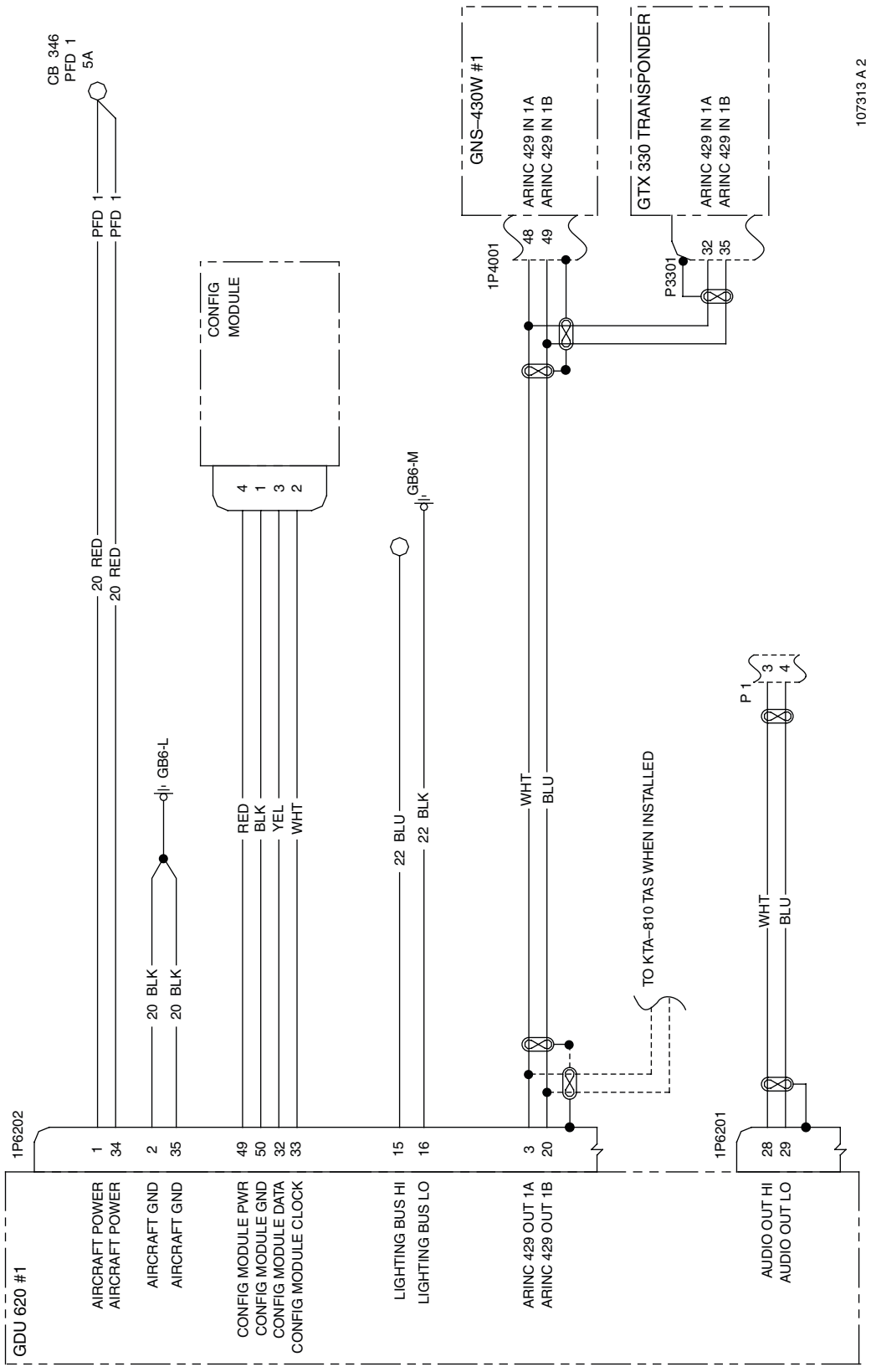




(OPTIONAL)
(NOT AVAILABLE WITH AVIDYNE ENTEGRA)

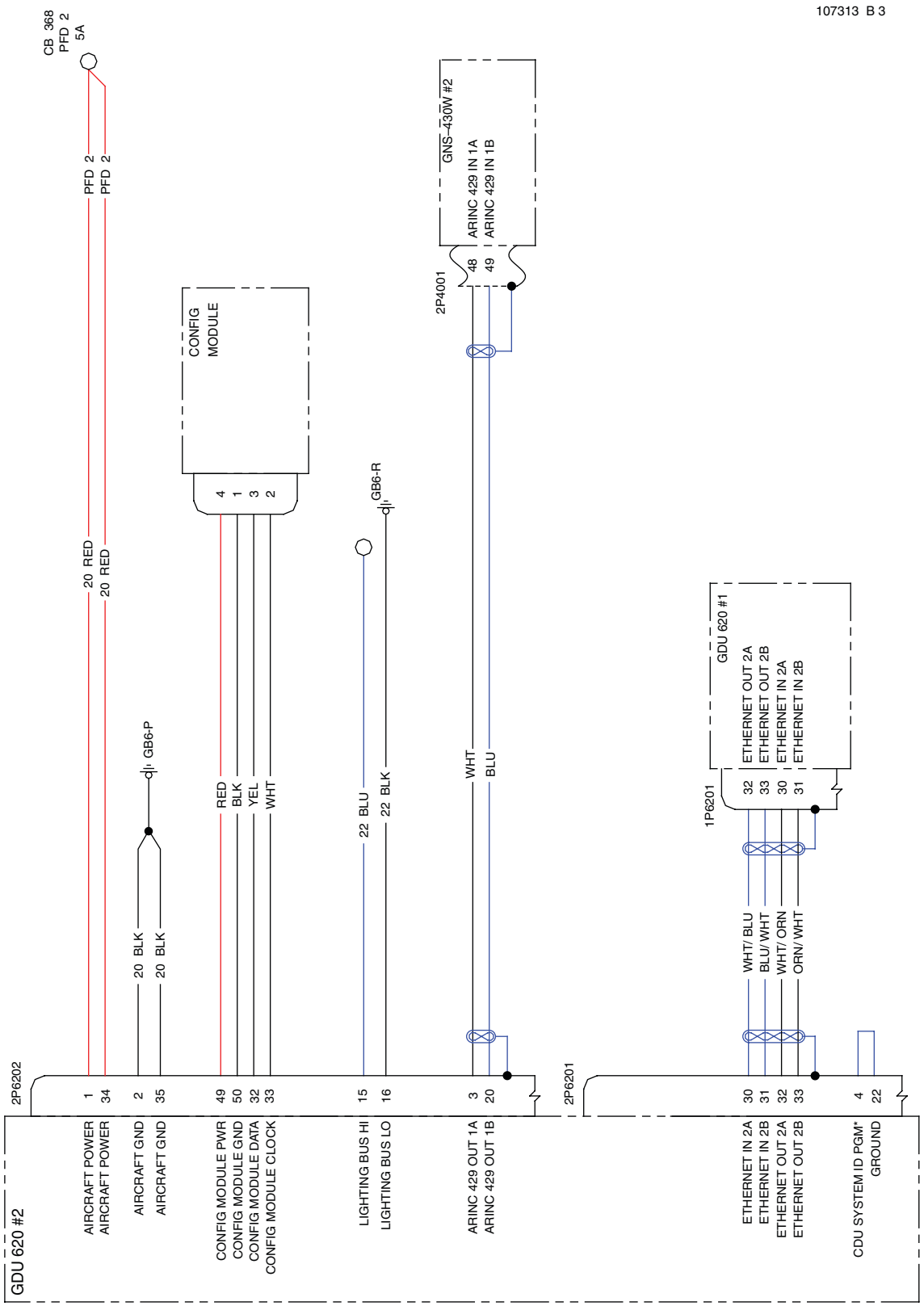
Effectivity
3449152 and up

Turn and Bank
Figure 6 (Sheet 2 of 2)



Primary Flight Display (PFD)
Figure 7 (Sheet 1 of 2)

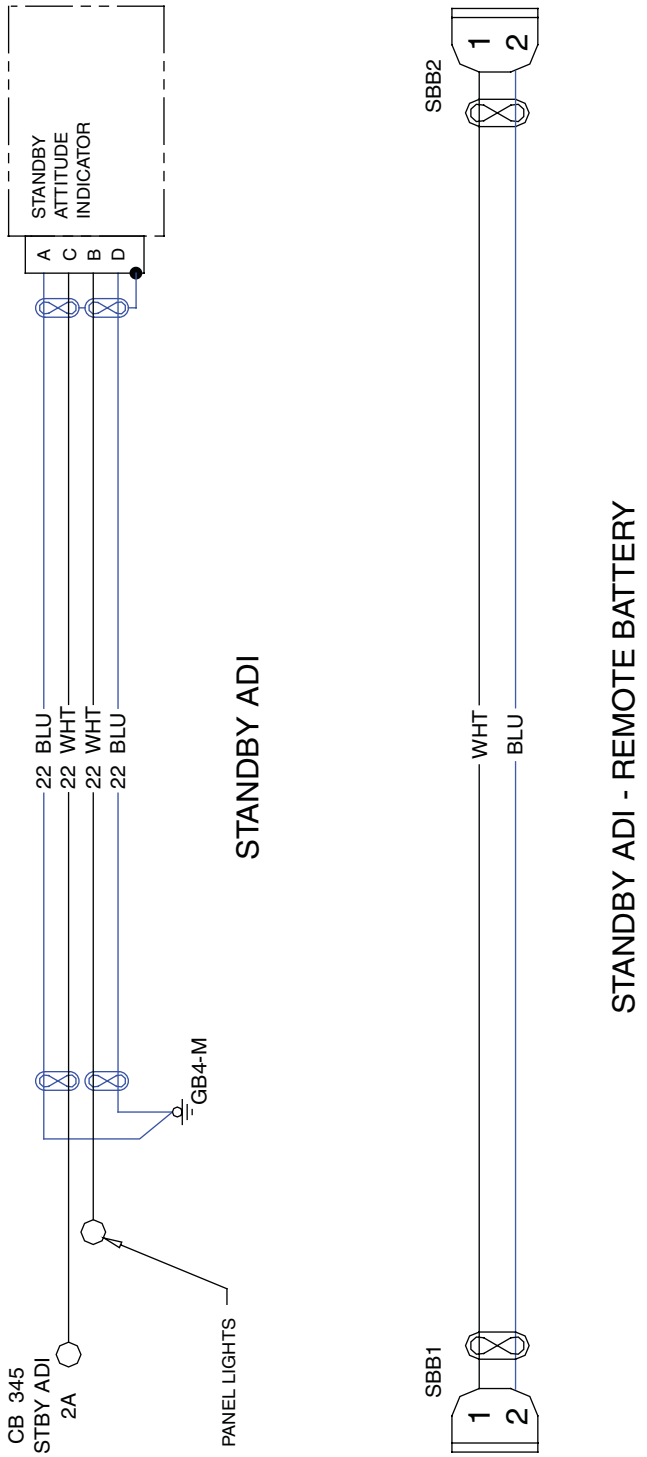
Effectivity
Seneca V
with Garmin G600



Effectivity
Seneca V
with Garmin G600

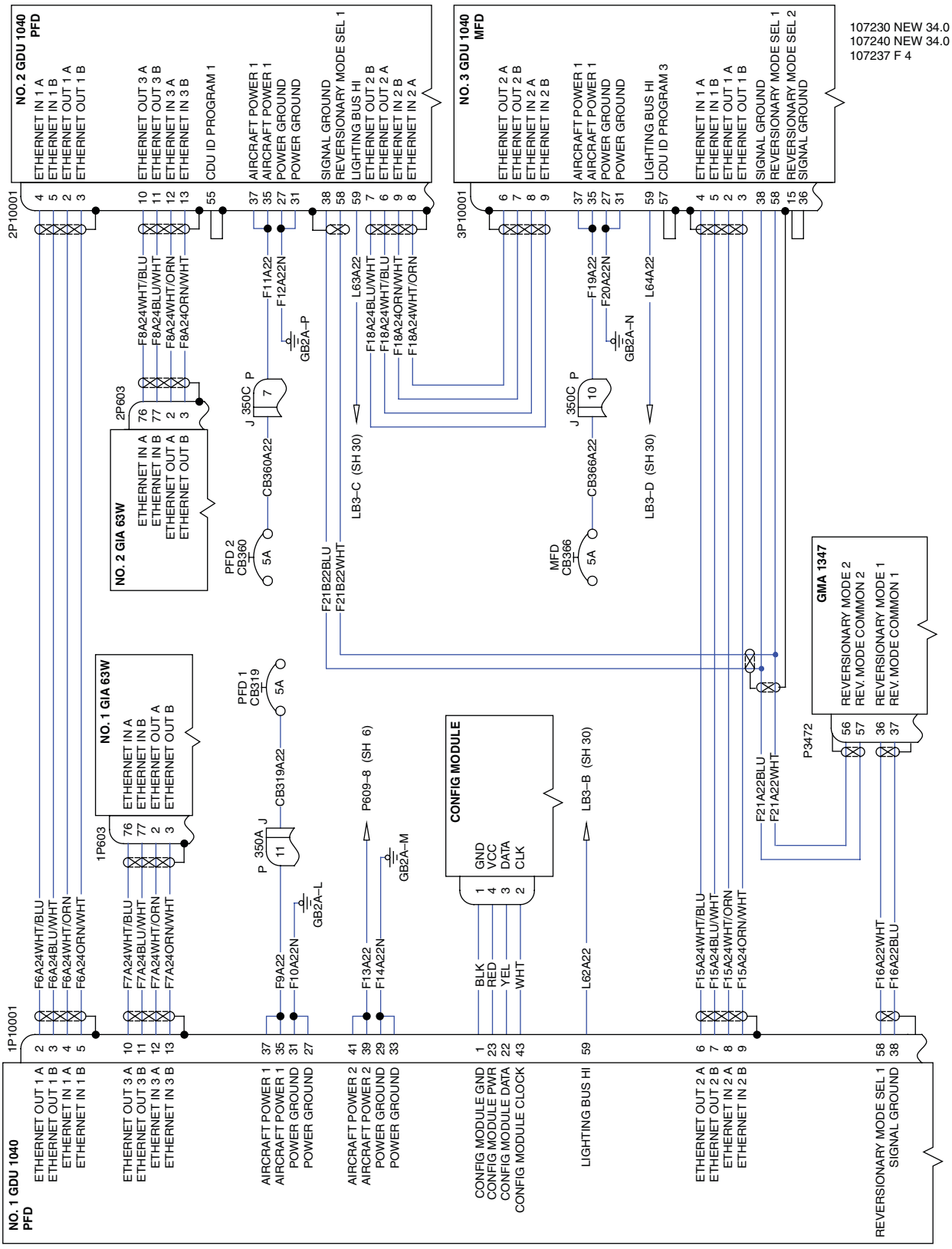
Primary Flight Display (PFD)
Figure 7 (Sheet 2 of 2)

107313 B 2



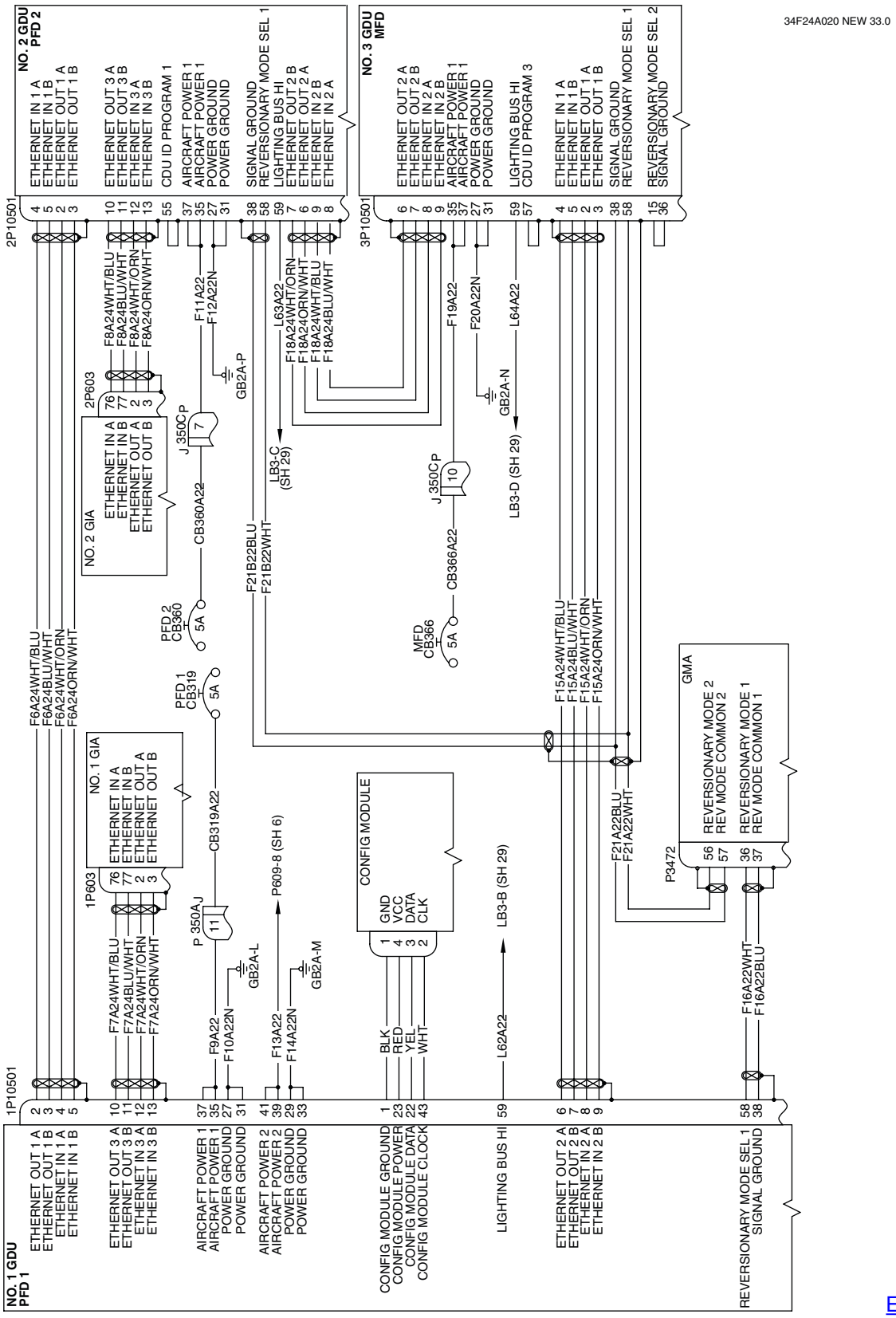
Standby Attitude Indicator
Figure 8

Effectivity
Seneca V
with Garmin G600



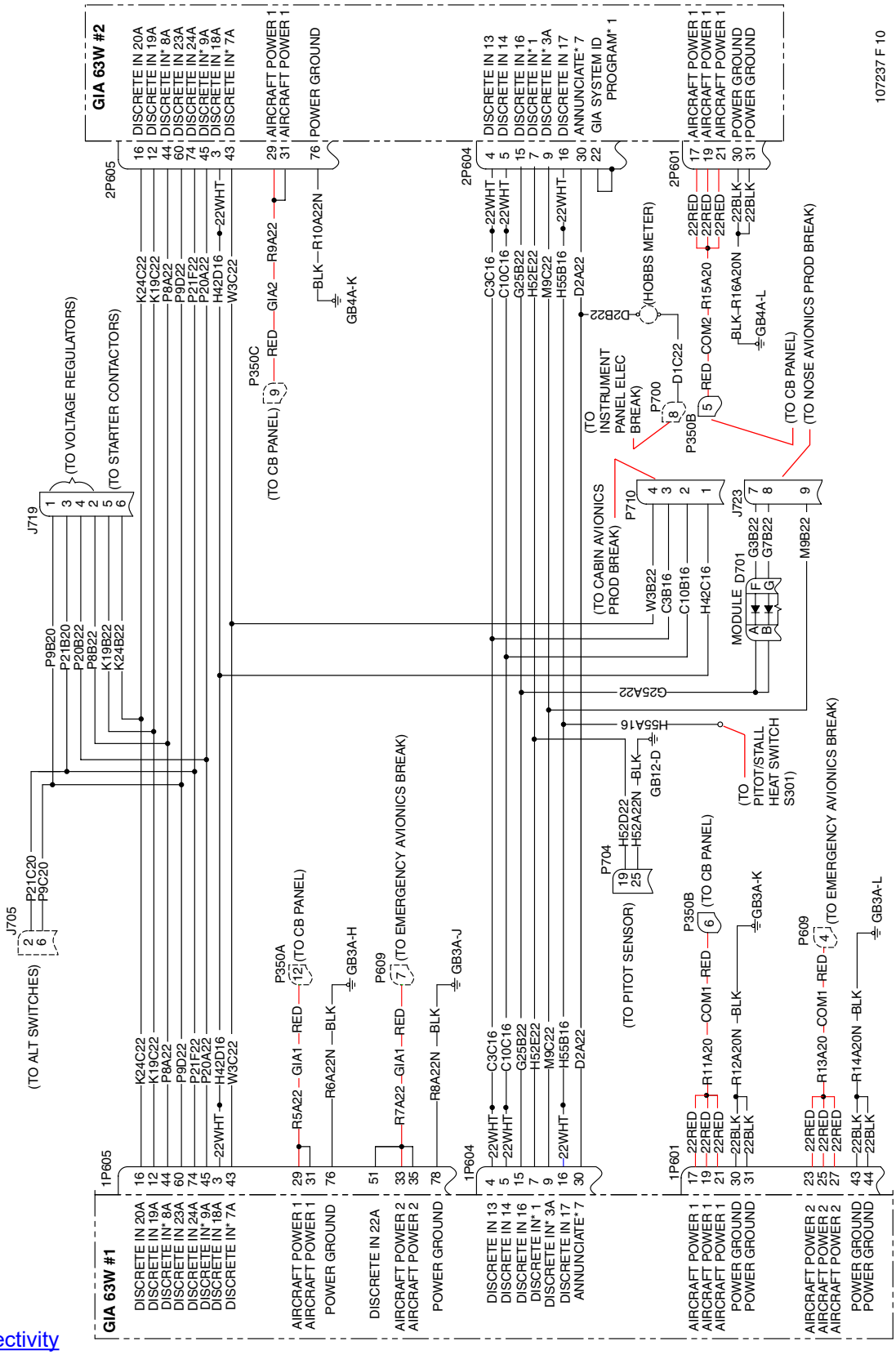
Effectivity
Seneca V
with Garmin G1000

Primary Flight Display /Multi-Function Display (PFD/MFD)
Figure 9 (Sheet 1 of 2)



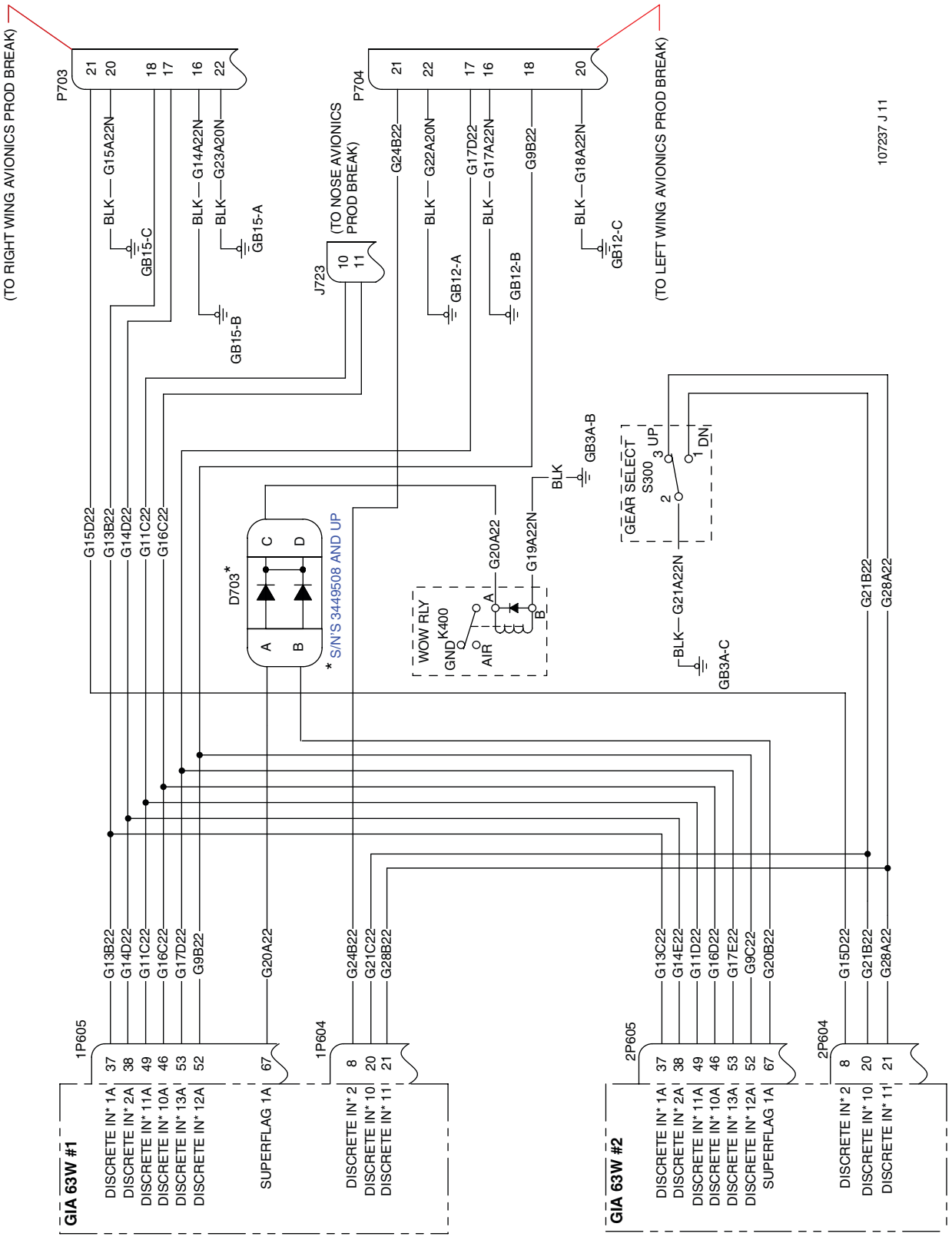
Primary Flight Display/Multi-Function Display (PFD/MFD)
Figure 9 (Sheet 2 of 2)

Effectivity
Seneca V
with Garmin G1000 NXi



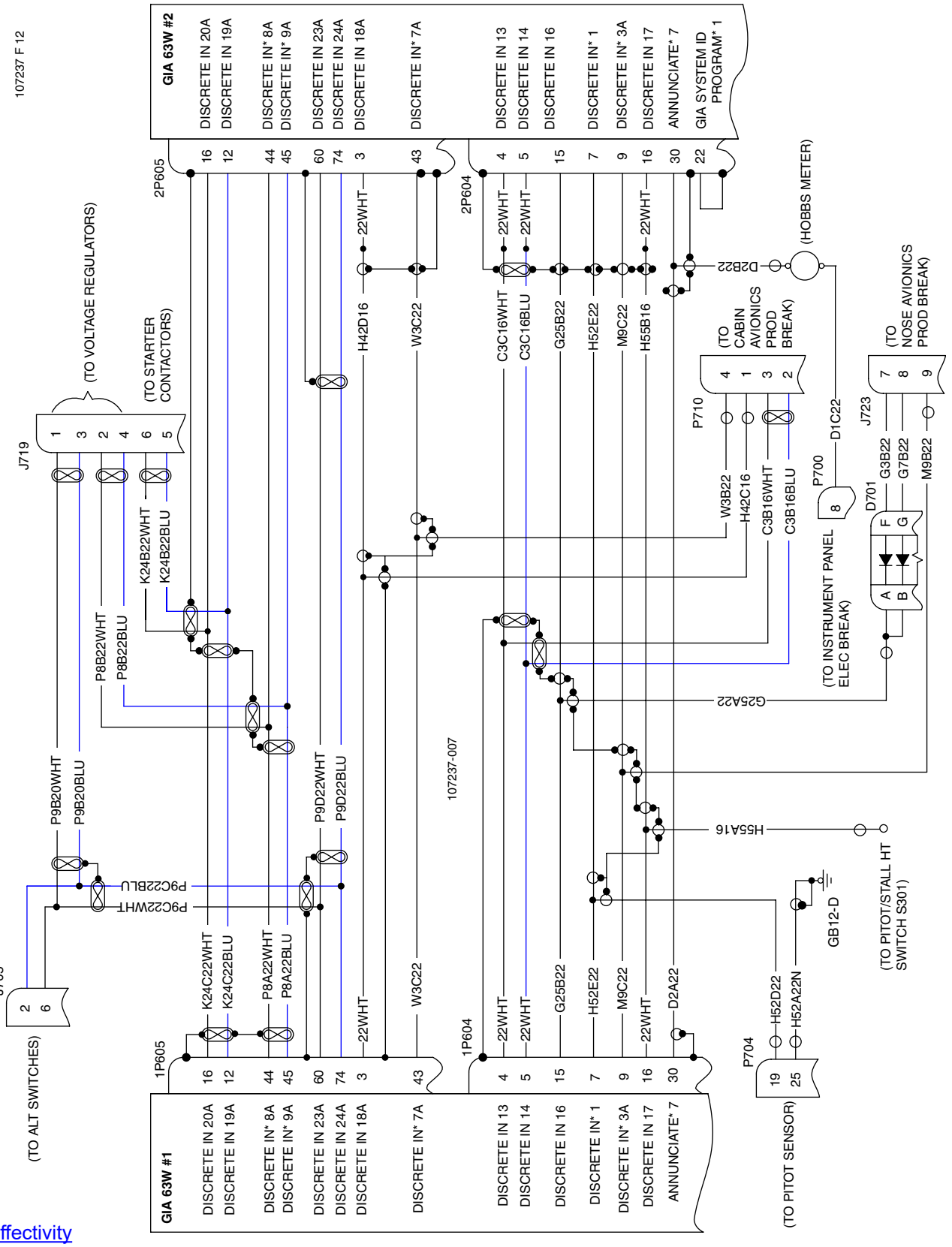
GIA Integrated Avionics Unit
Figure 10 (Sheet 1 of 7)

Effectivity
Seneca V
with Garmin G1000



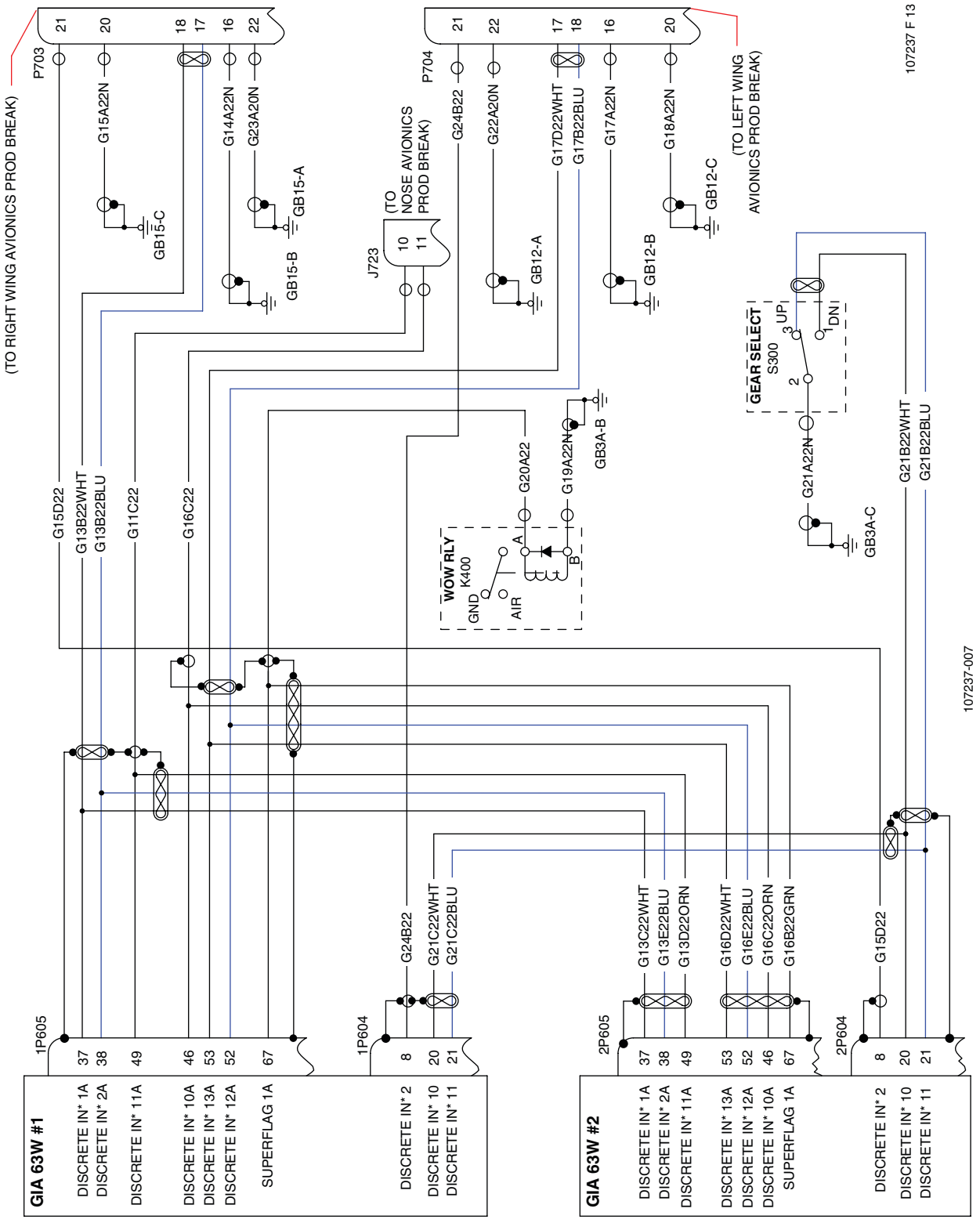
GIA Integrated Avionics Unit
Figure 10 (Sheet 2 of 7)

Effectivity
Seneca V
with Garmin G1000



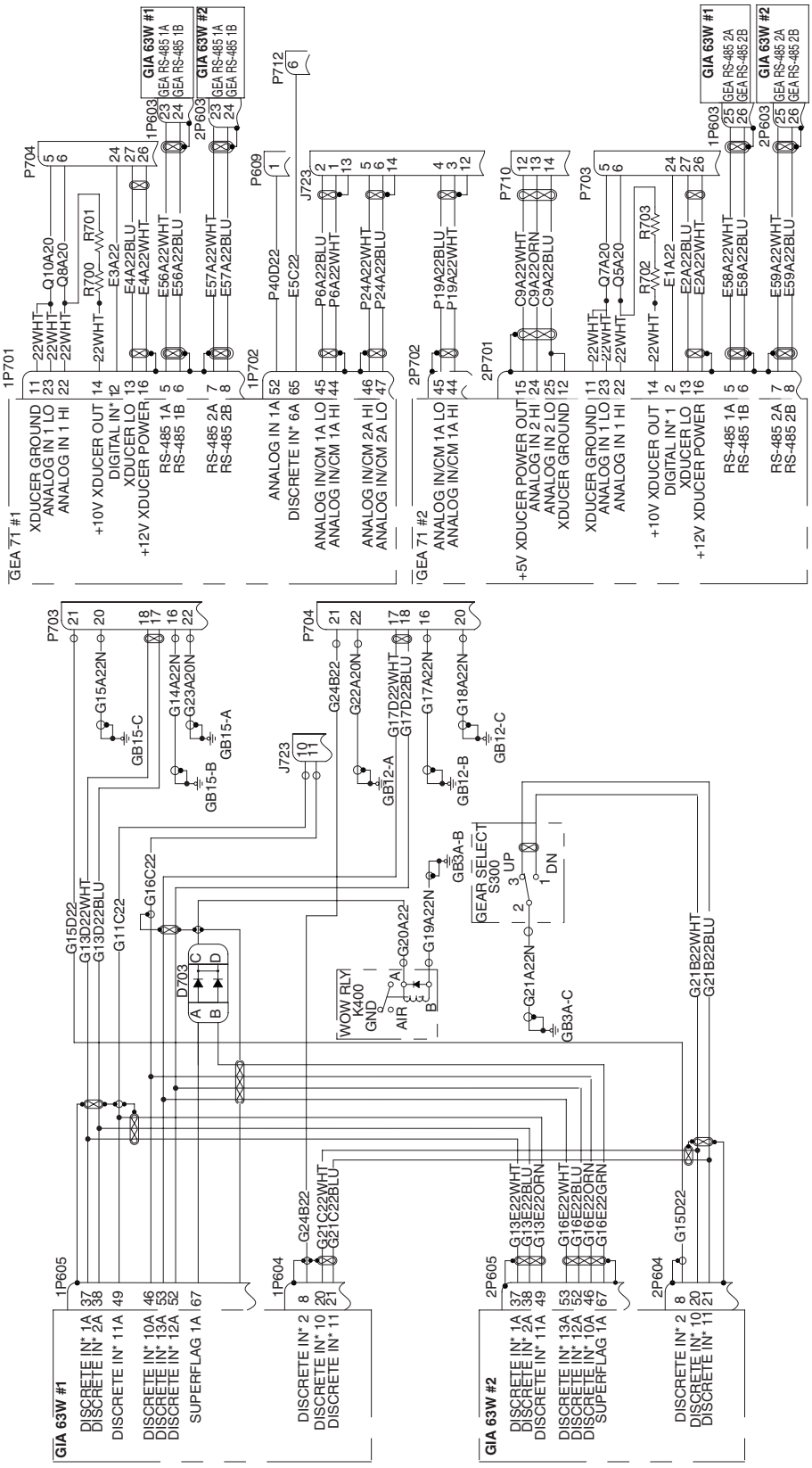
Effectivity
Seneca V
with Garmin GFC 700

GIA Integrated Avionics Unit
Figure 10 (Sheet 3 of 7)



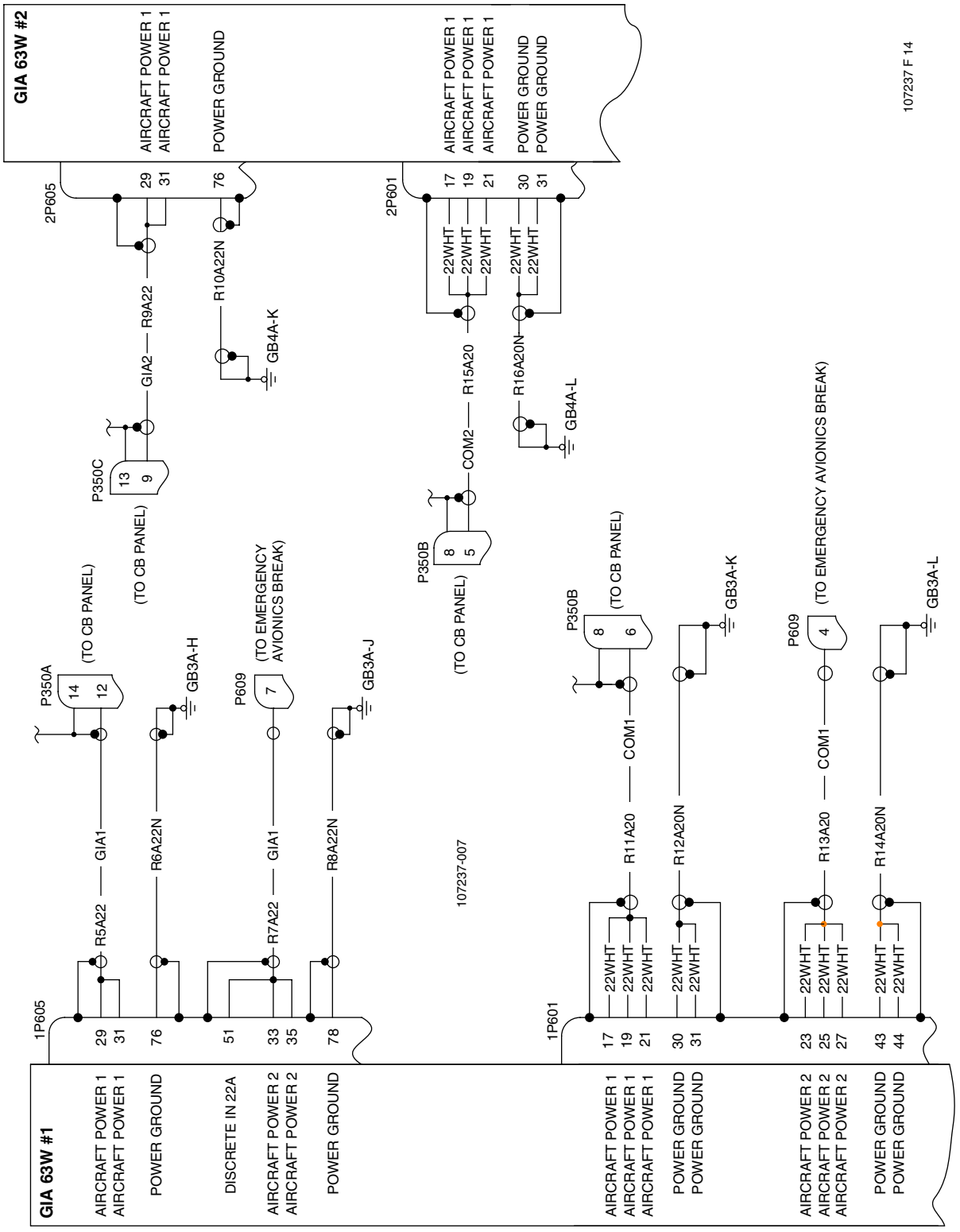
GIA Integrated Avionics Unit
Figure 10 (Sheet 4 of 7)

Effectivity
Seneca V
with Garmin GFC 700,
S/N'S 3449459, 3449492-3449508



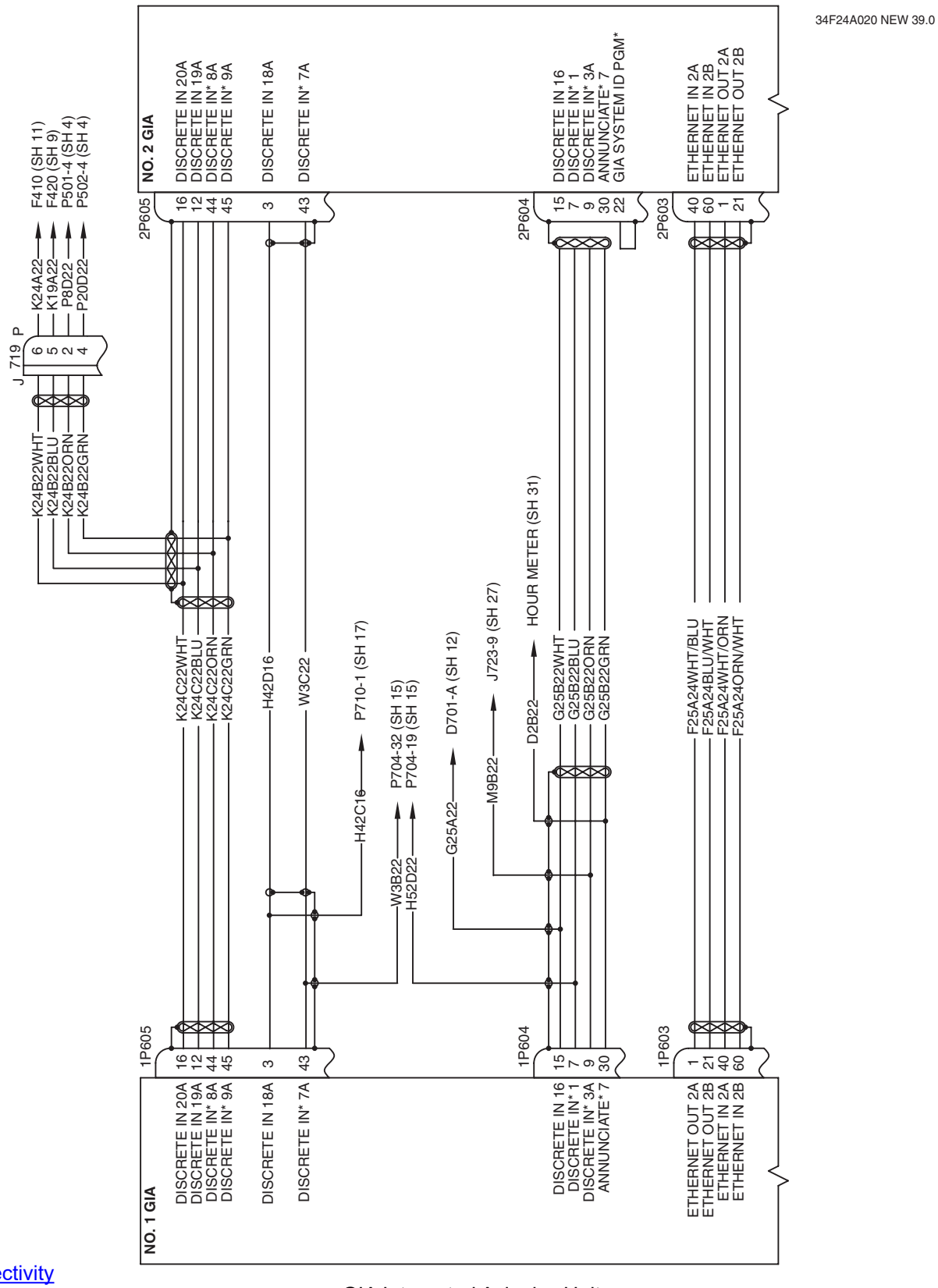
GIA Integrated Avionics Unit
Figure 10 (Sheet 5 of 7)

Effectivity
Seneca V
with Garmin GFC 700,
S/N'S 3449459, 3449492-3449508



GIA Integrated Avionics Unit
Figure 10 (Sheet 6 of 7)

Effectivity
Seneca V
with Garmin GFC 700,
S/N'S 3449459, 3449492-3449508

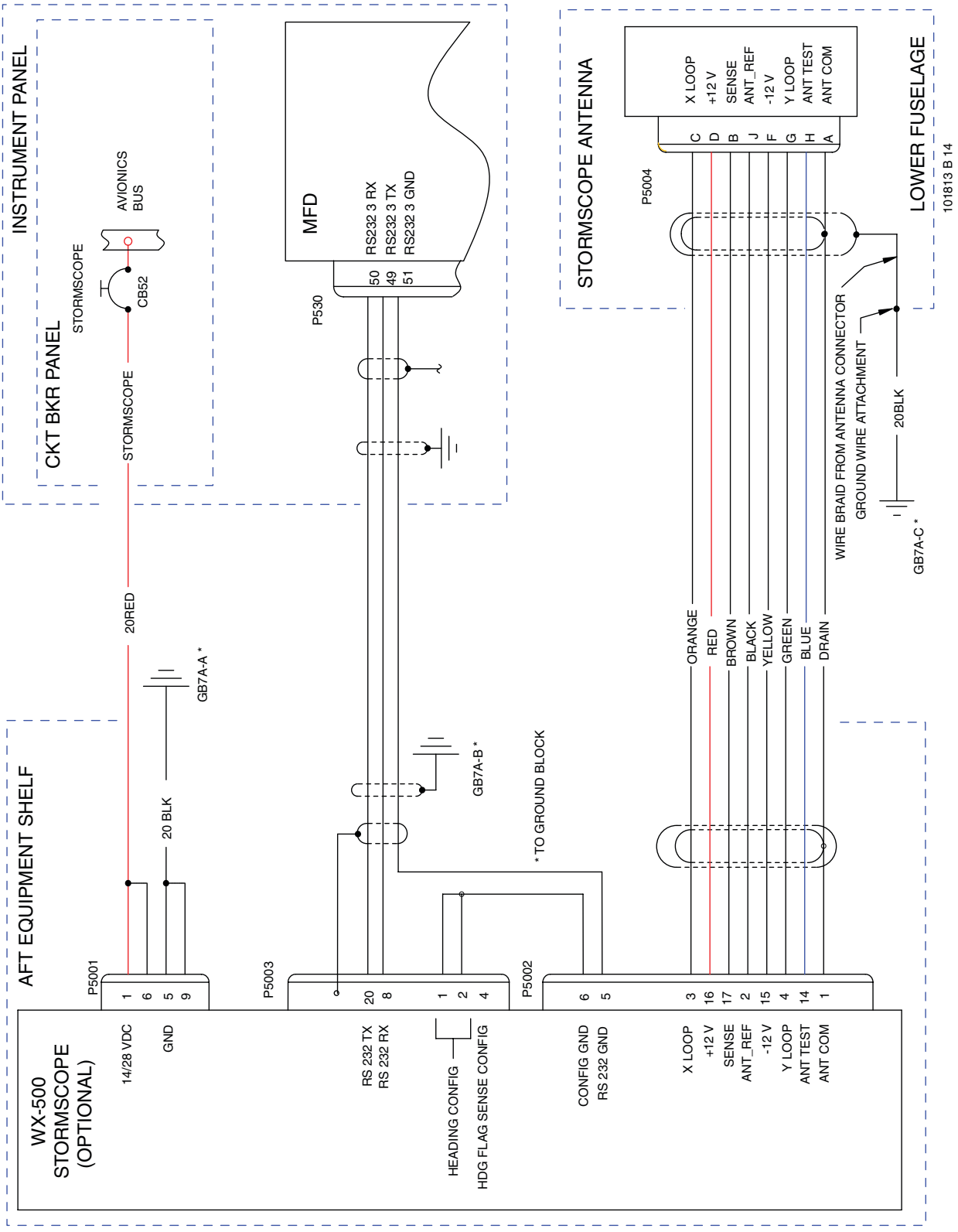


Effectivity
Seneca V
with Garmin G1000 NXi

GIA Integrated Avionics Unit
Figure 10 (Sheet 7 of 7)

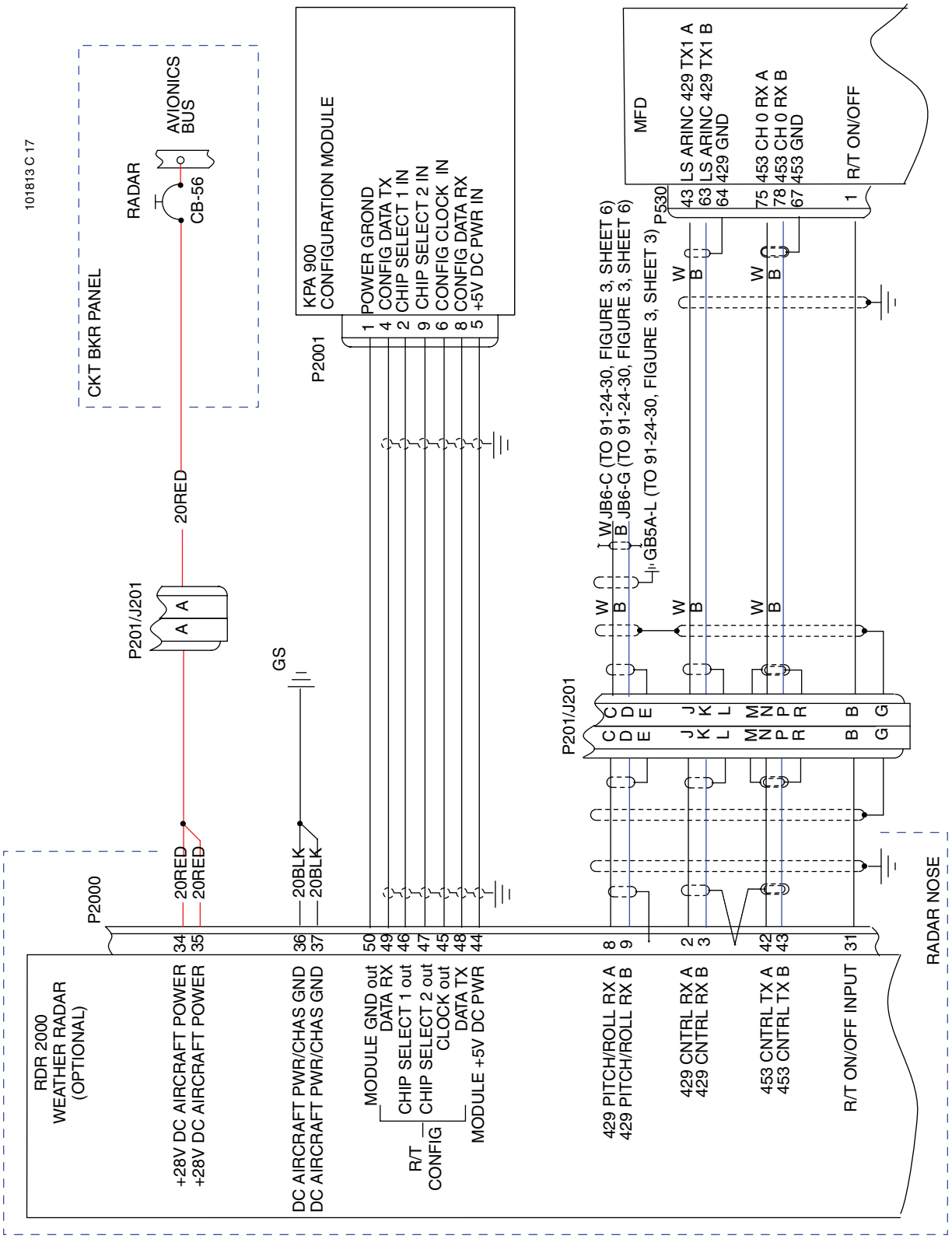
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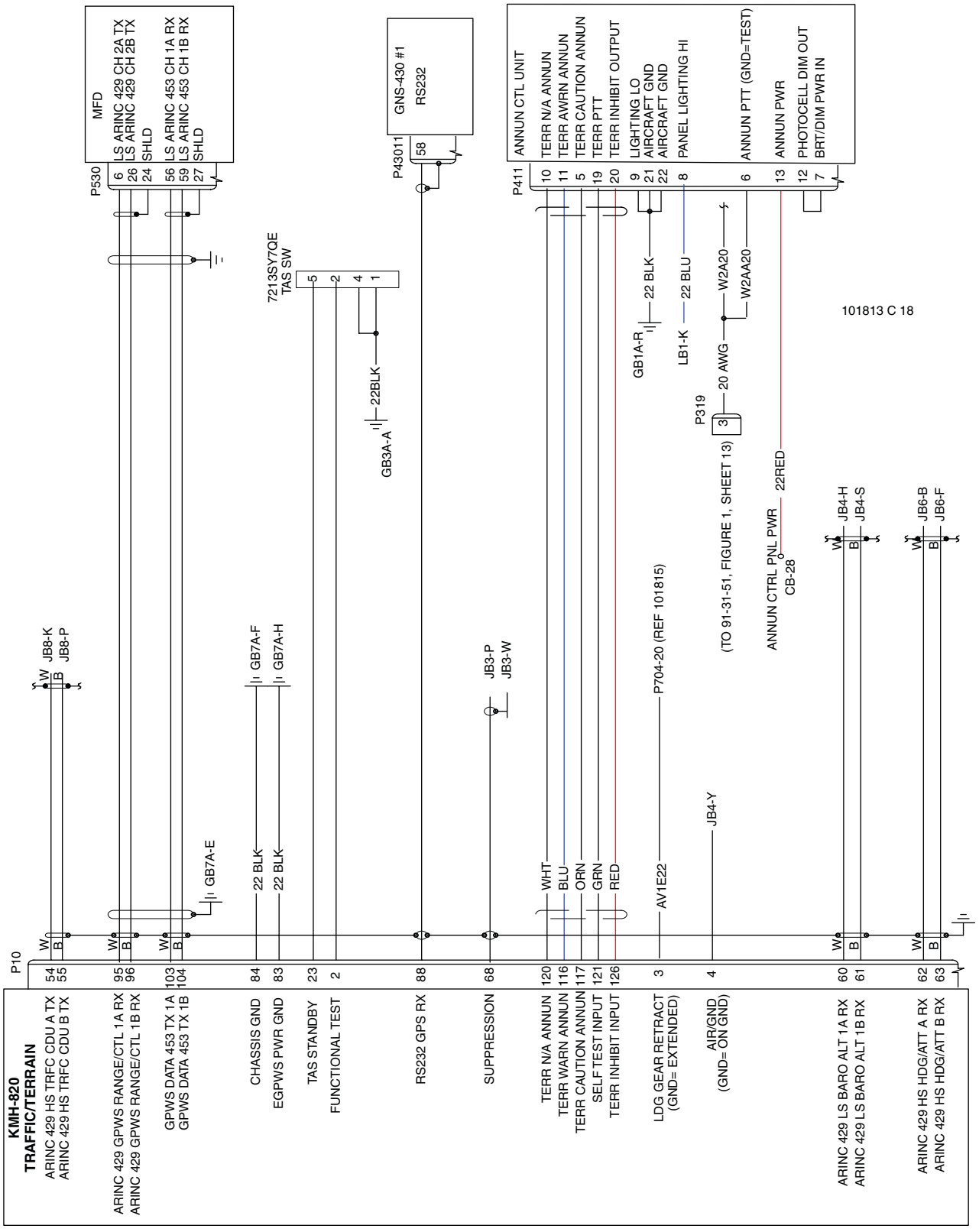
Stormscope
Figure 1

Effectivity
Seneca V
with Avidyne Entegra



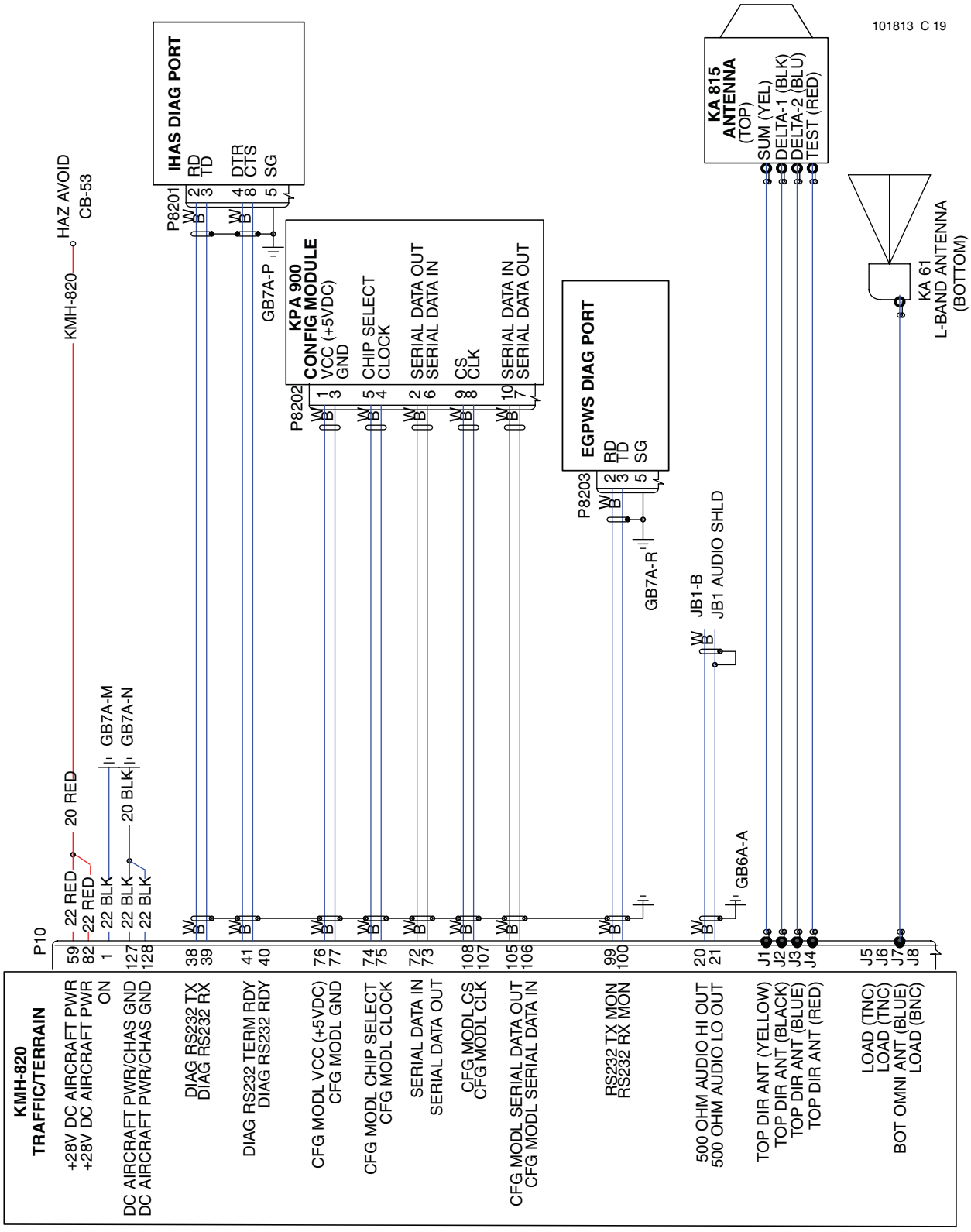
Effectivity
Seneca V
with Avidyne Entegra

Radar
Figure 2



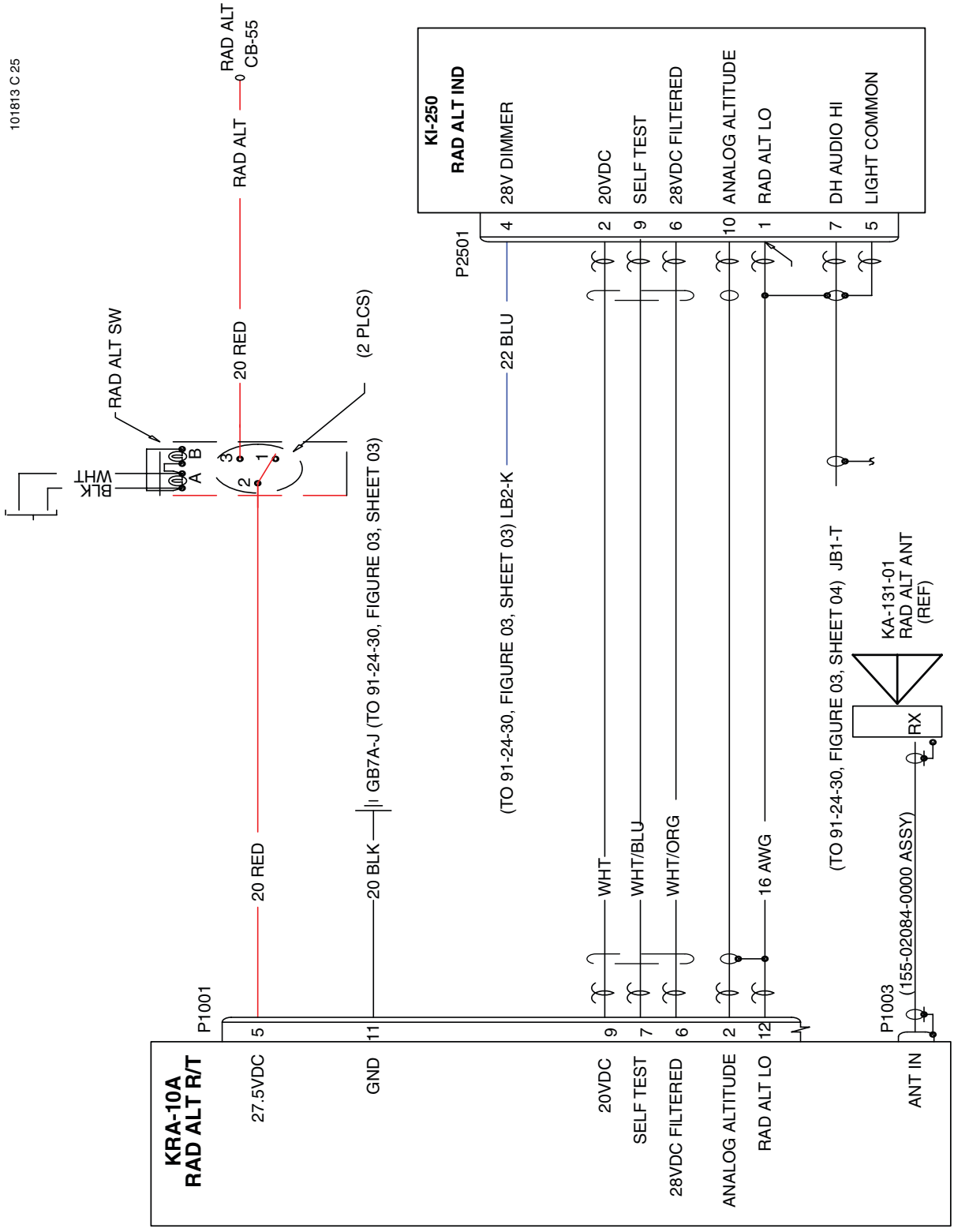
Traffic Alert / Terrain Avioidance
Figure 3 (Sheet 1 of 2)

Effectivity
Seneca V
with Avidyne Entegra



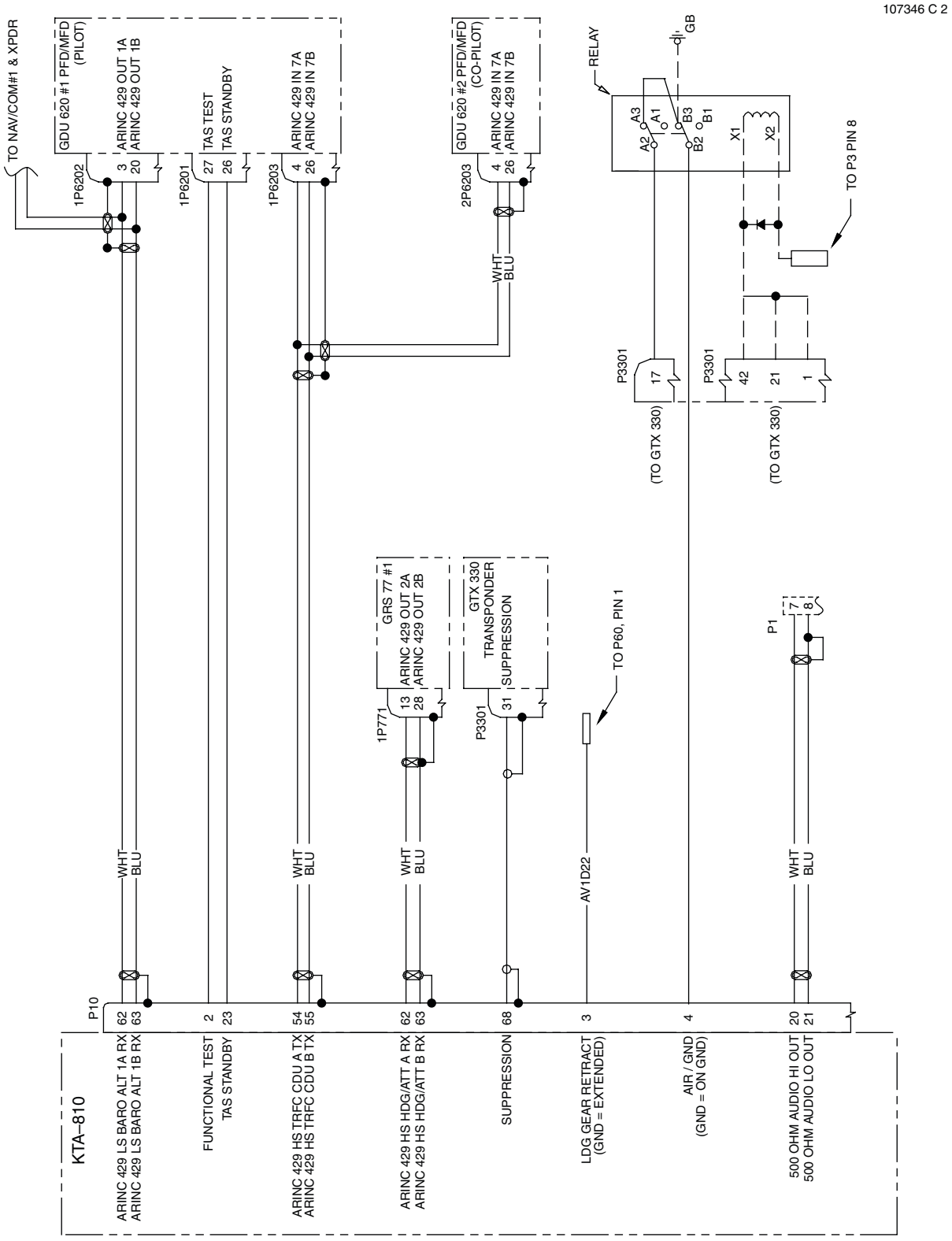
Effectivity
Seneca V
with Avidyne Entegra

Traffic Alert / Terrain Avoidance
Figure 3 (Sheet 2 of 2)



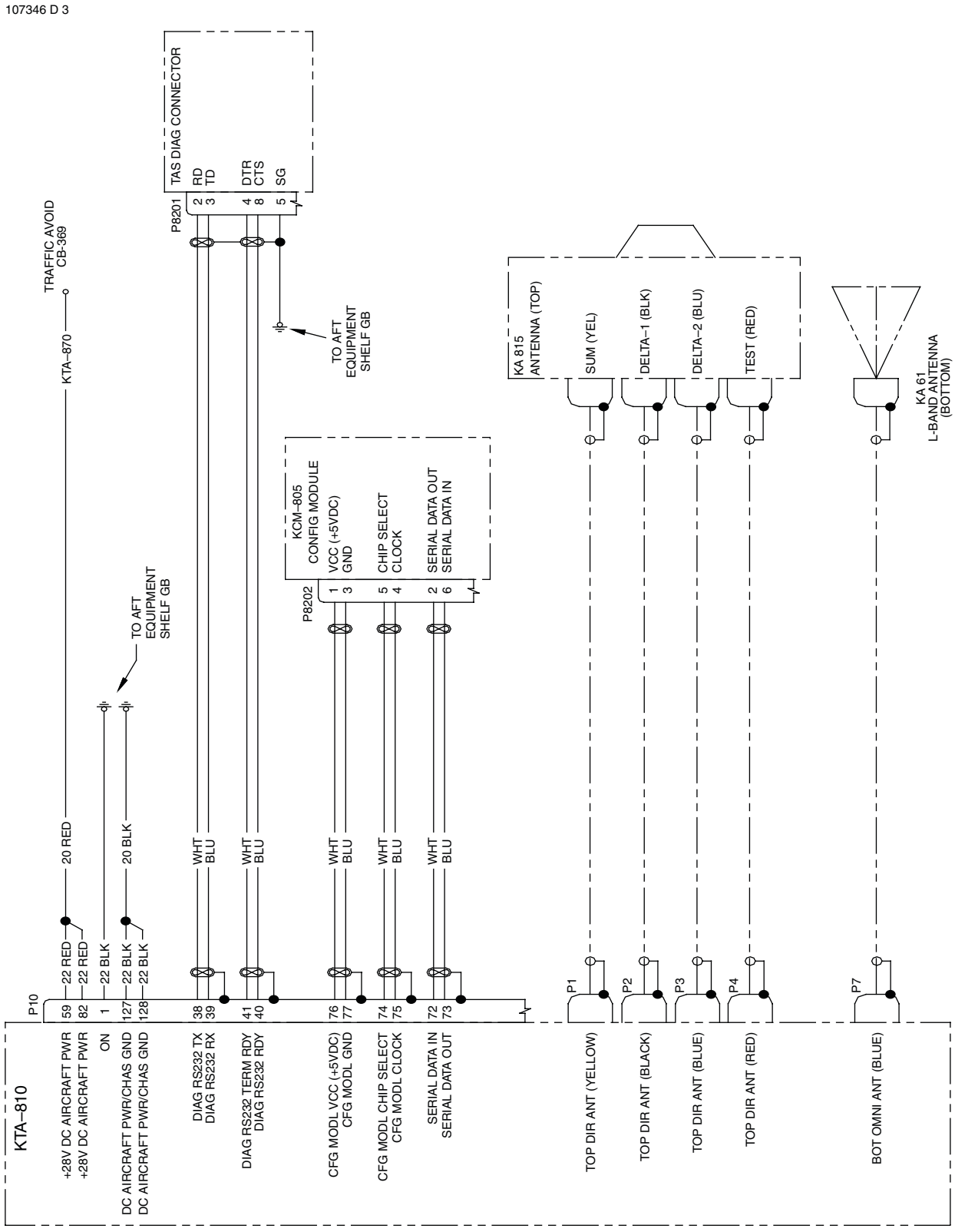
Radar Altimeter
Figure 4

Effectivity
Seneca V
with Avidyne Entegra



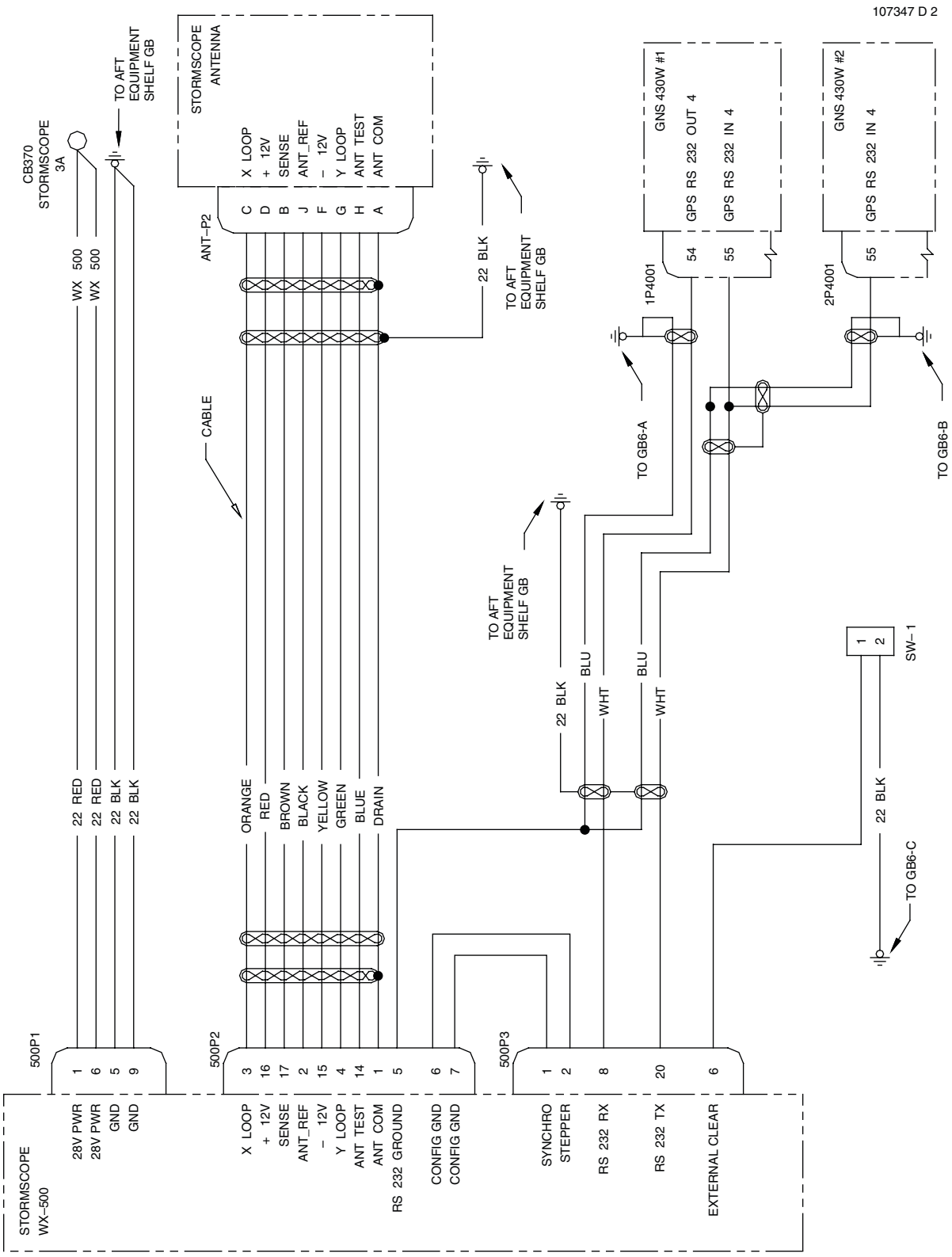
Effectivity
Seneca V
with Garmin G600

Traffic Advisory System
Figure 5 (Sheet 1 of 2)



Traffic Advisory System
Figure 5 (Sheet 2 of 2)

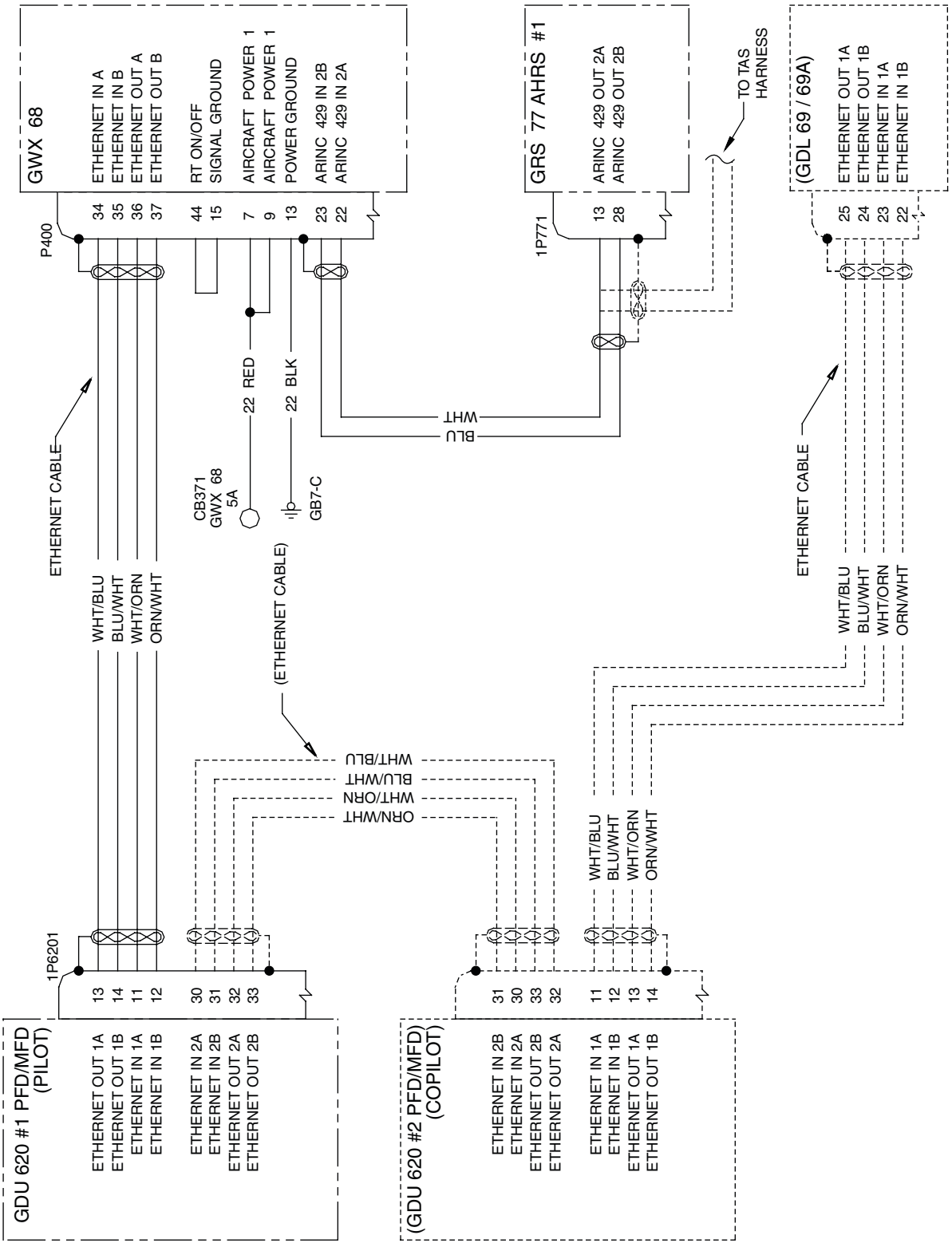
Effectivity
Seneca V
with Garmin G600



Effectivity
Seneca V
with Garmin G600

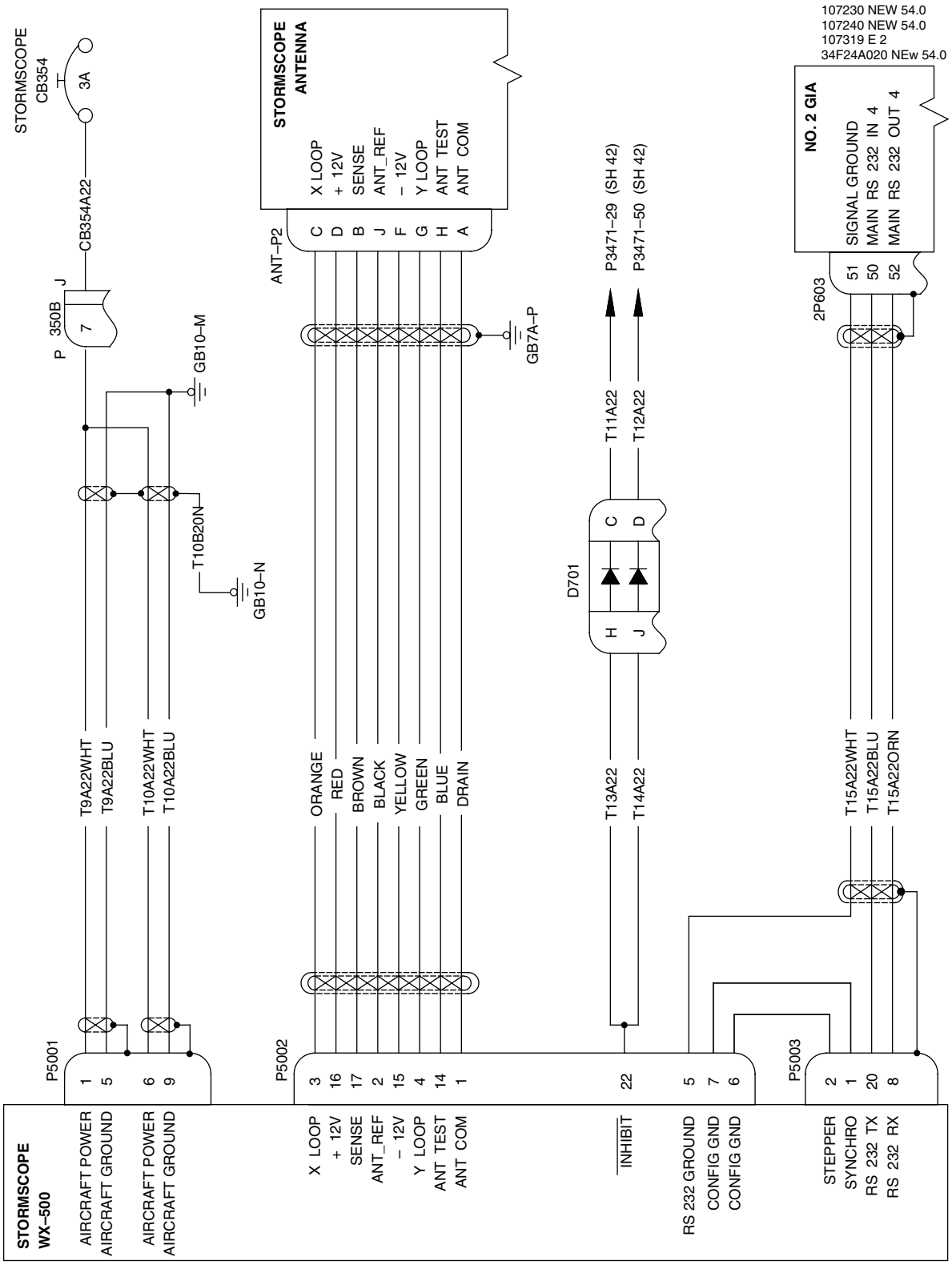
Stormscope
Figure 6

107349 B 2



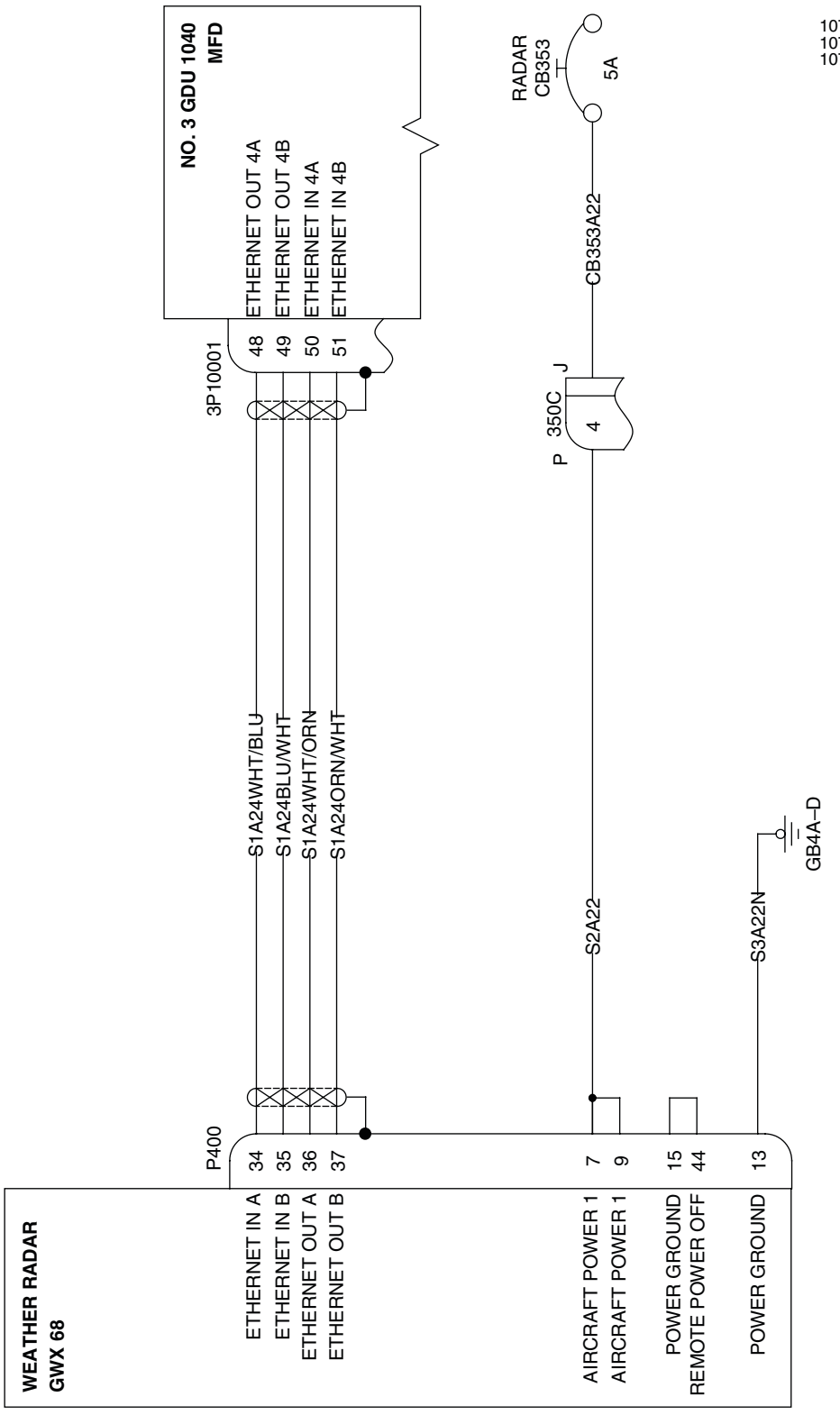
Weather Radar
Figure 7

Effectivity
Seneca V
with Garmin G600



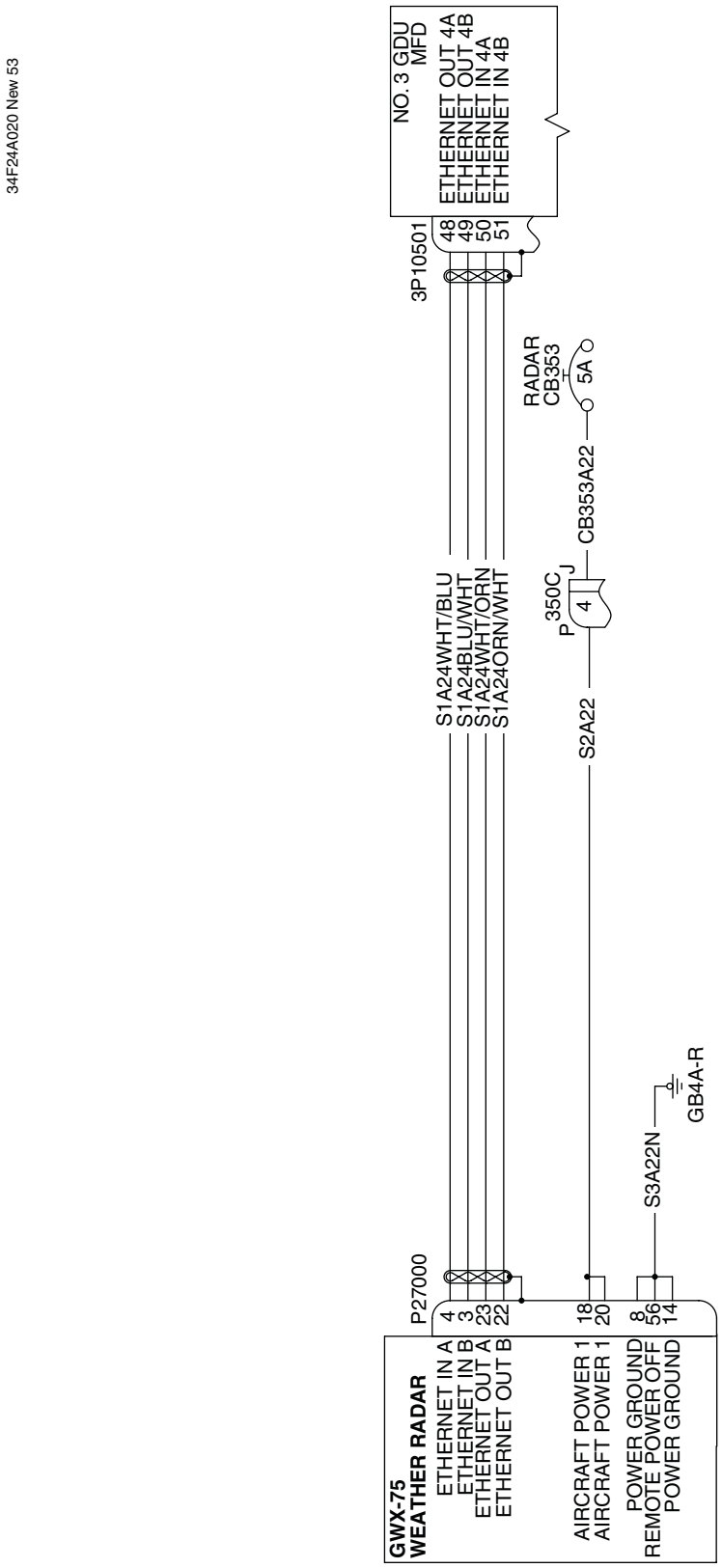
Effectivity
Seneca V
with Garmin G1000 / G1000 NXi

Stormscope
Figure 8



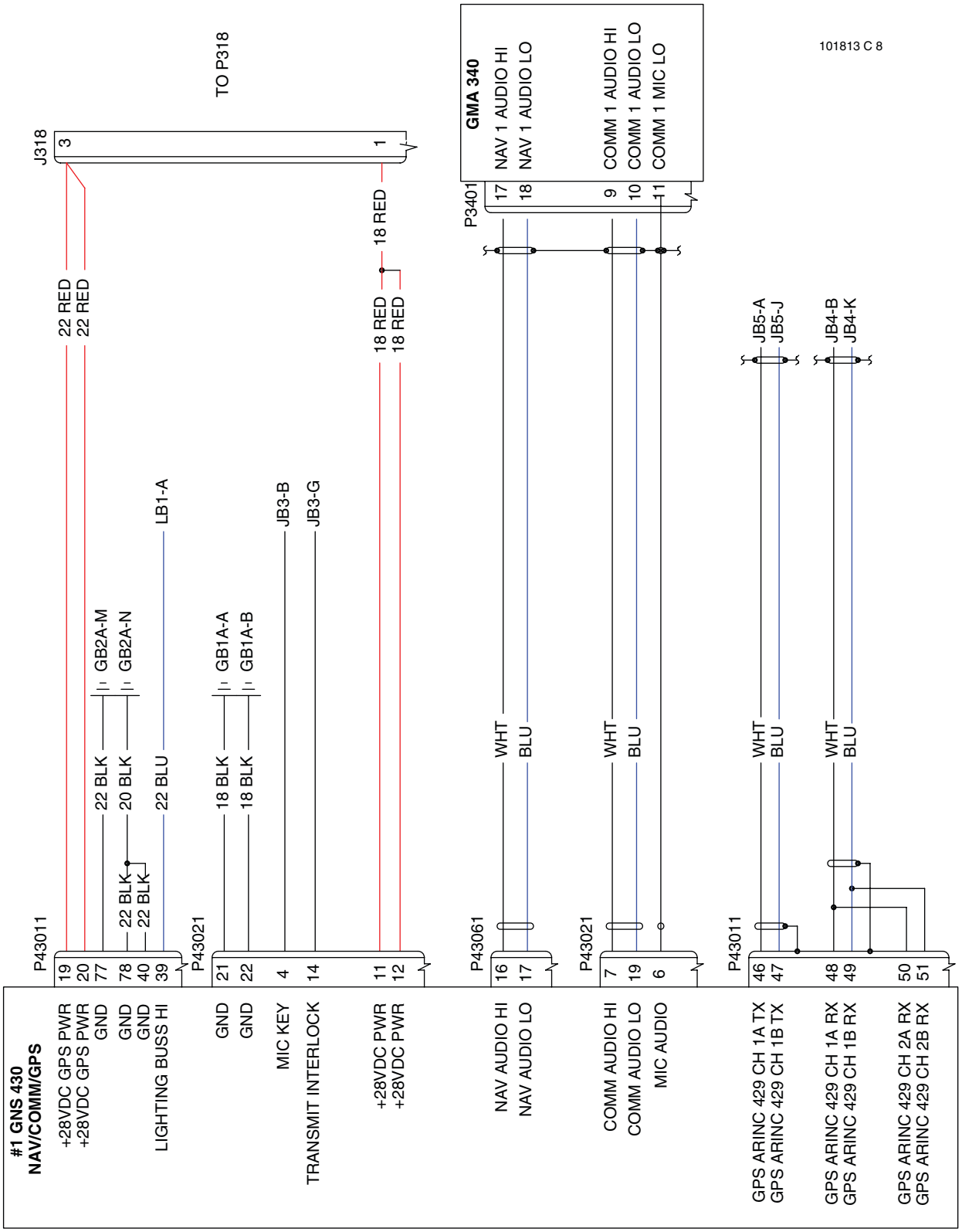
Weather Radar
Figure 9 (Sheet 1 of 2)

Effectivity
Seneca V
with Garmin G1000



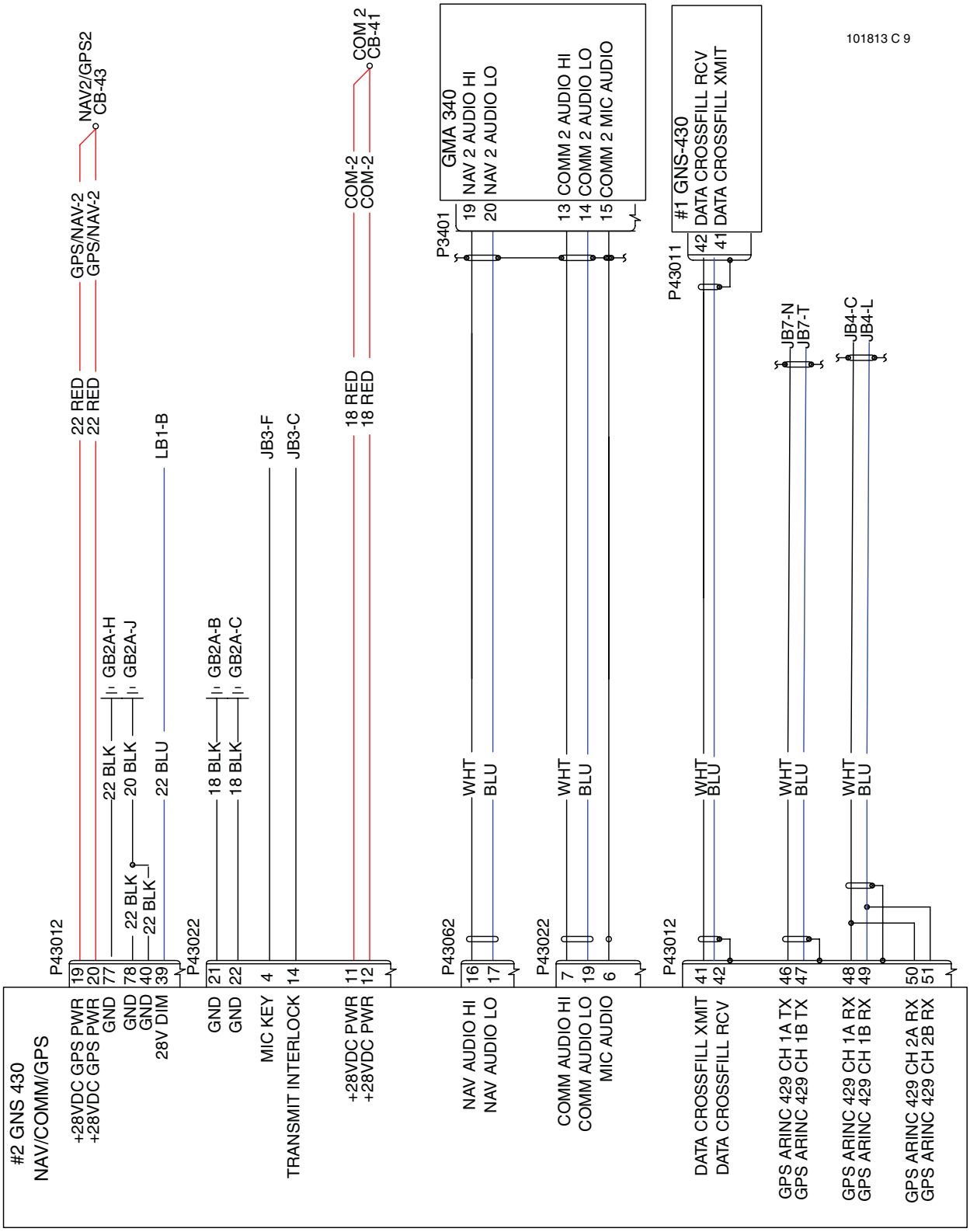
Weather Radar
Figure 9 (Sheet 2 of 2)

Effectivity
Seneca V
with Garmin G1000 NXi



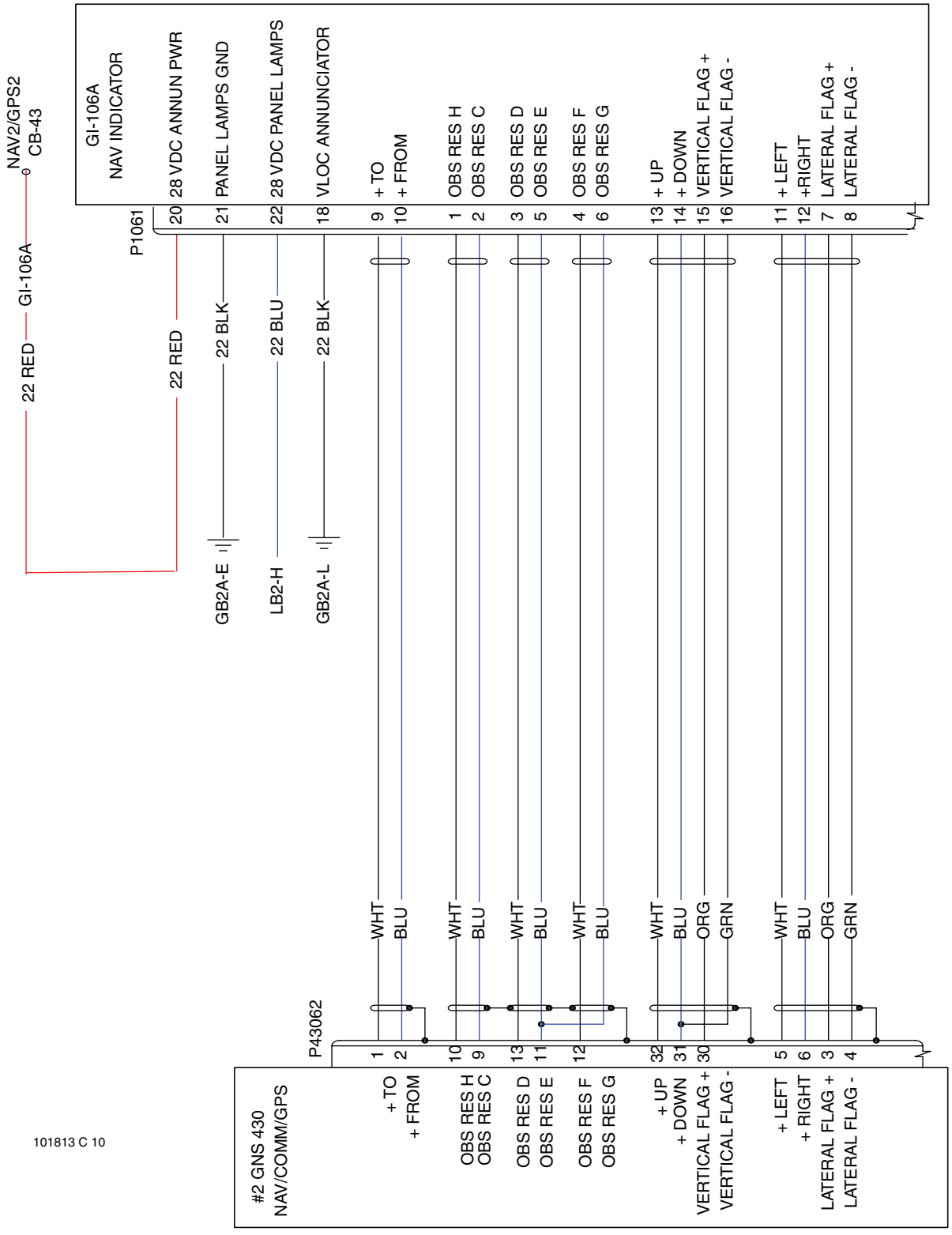
NAV / COM / GPS NO. 1
Figure 1

Effectivity
Seneca V
with Avidyne Entegra



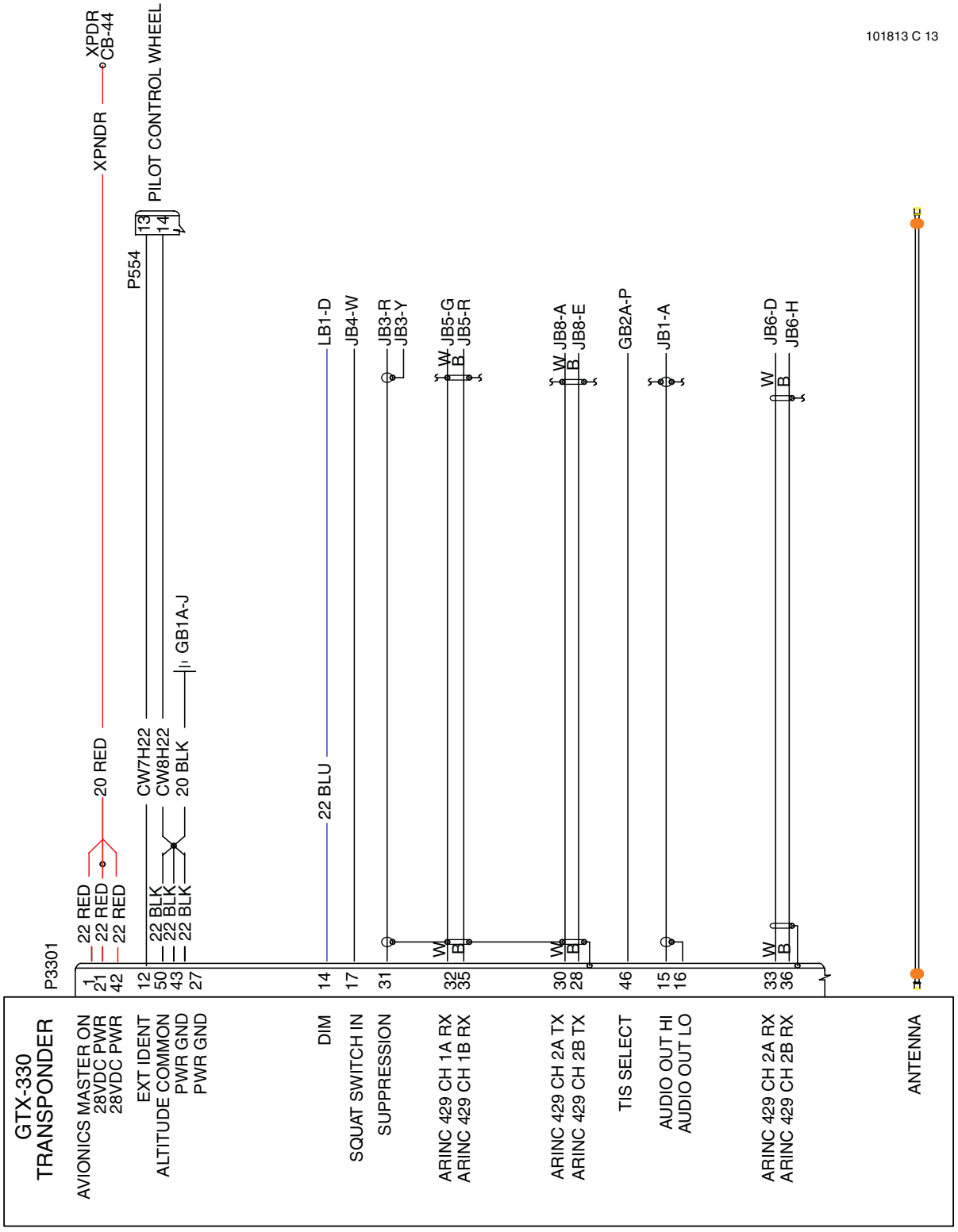
Effectivity
Seneca V
with Avidyne Entegra

NAV / COM / GPS NO. 2
Figure 2



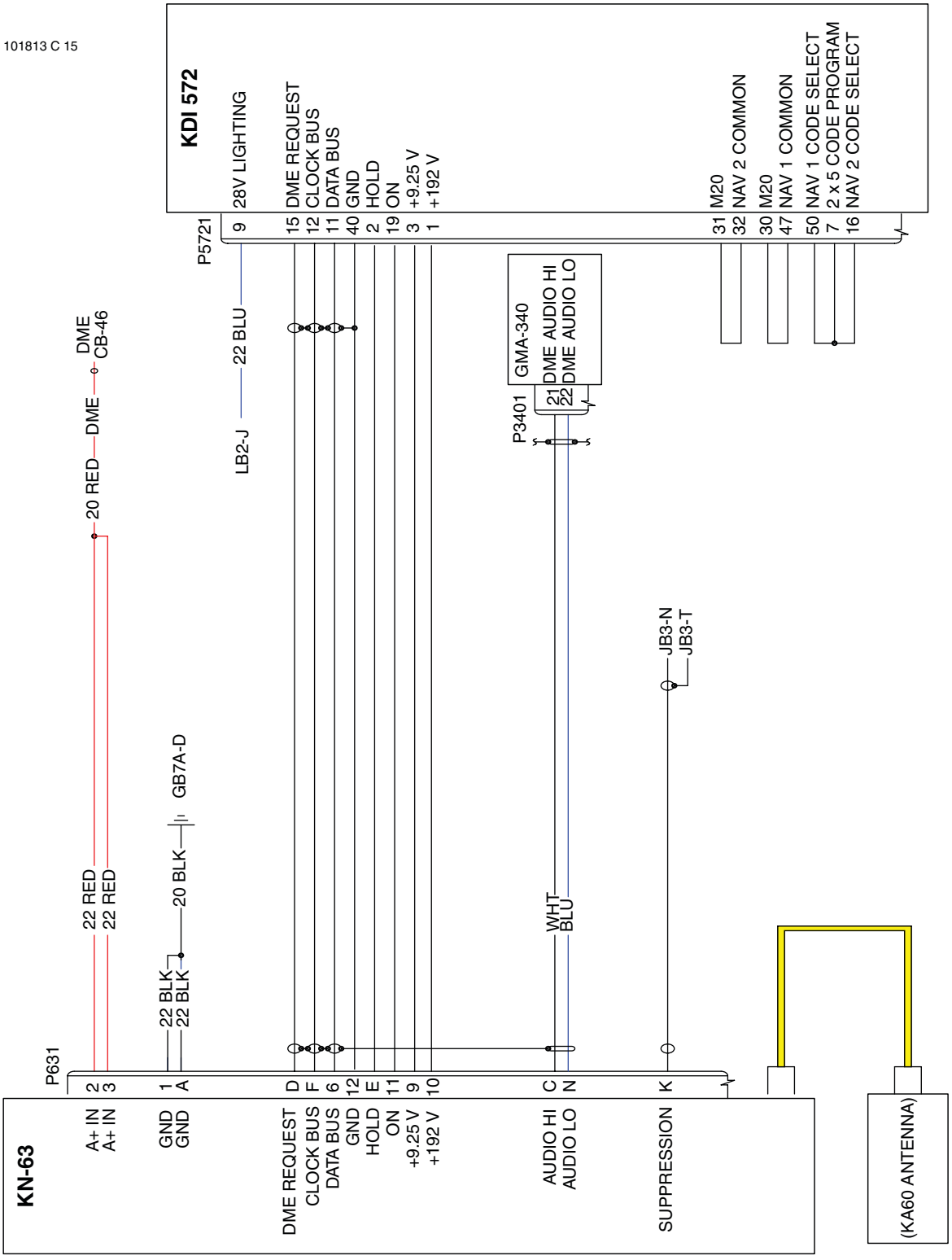
Course Deviation Indicator (CDI)
Figure 3

Effectivity
Seneca V
with Avidyne Entegra



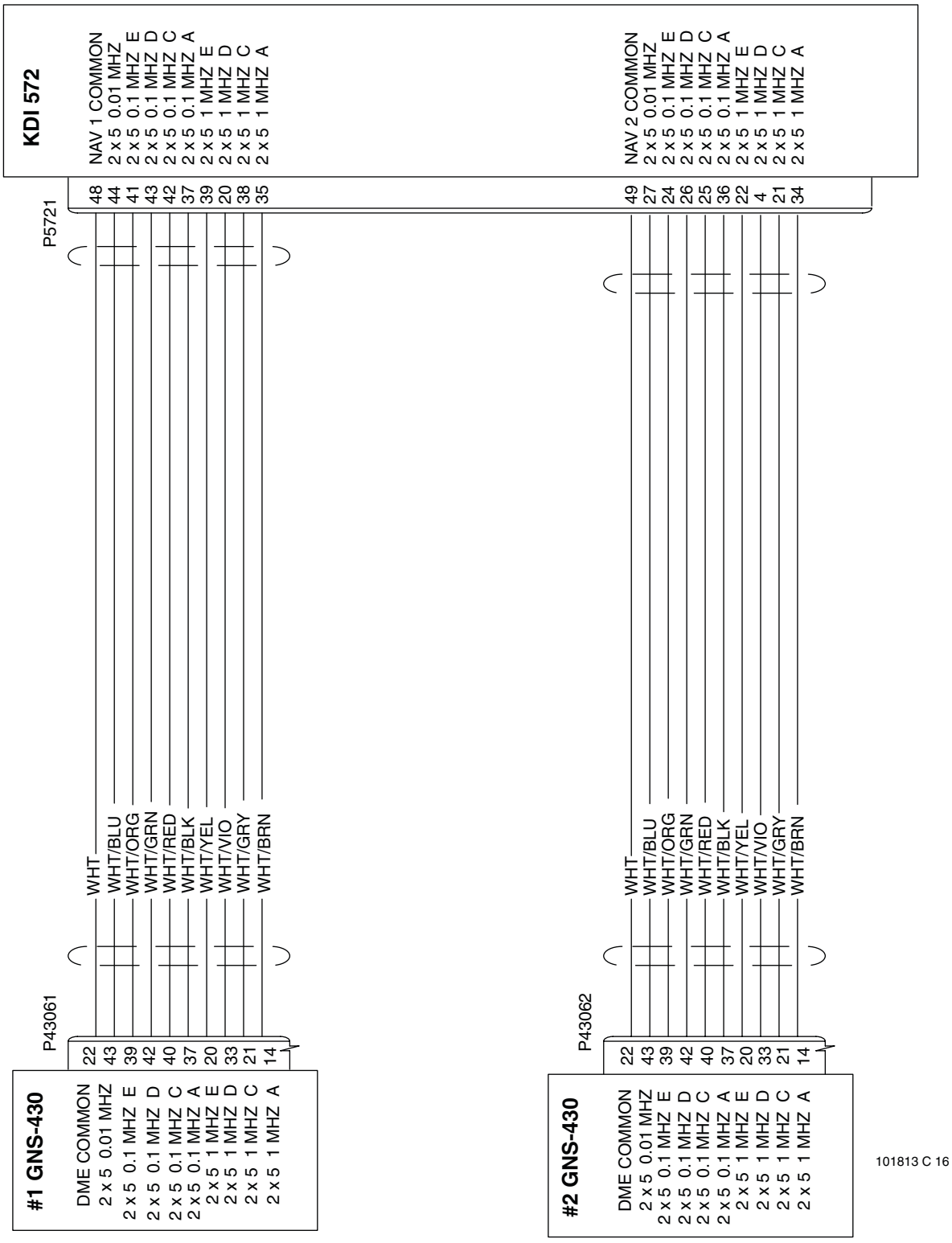
Effectivity
Seneca V
with Avidyne Entegra

Transponder
Figure 4



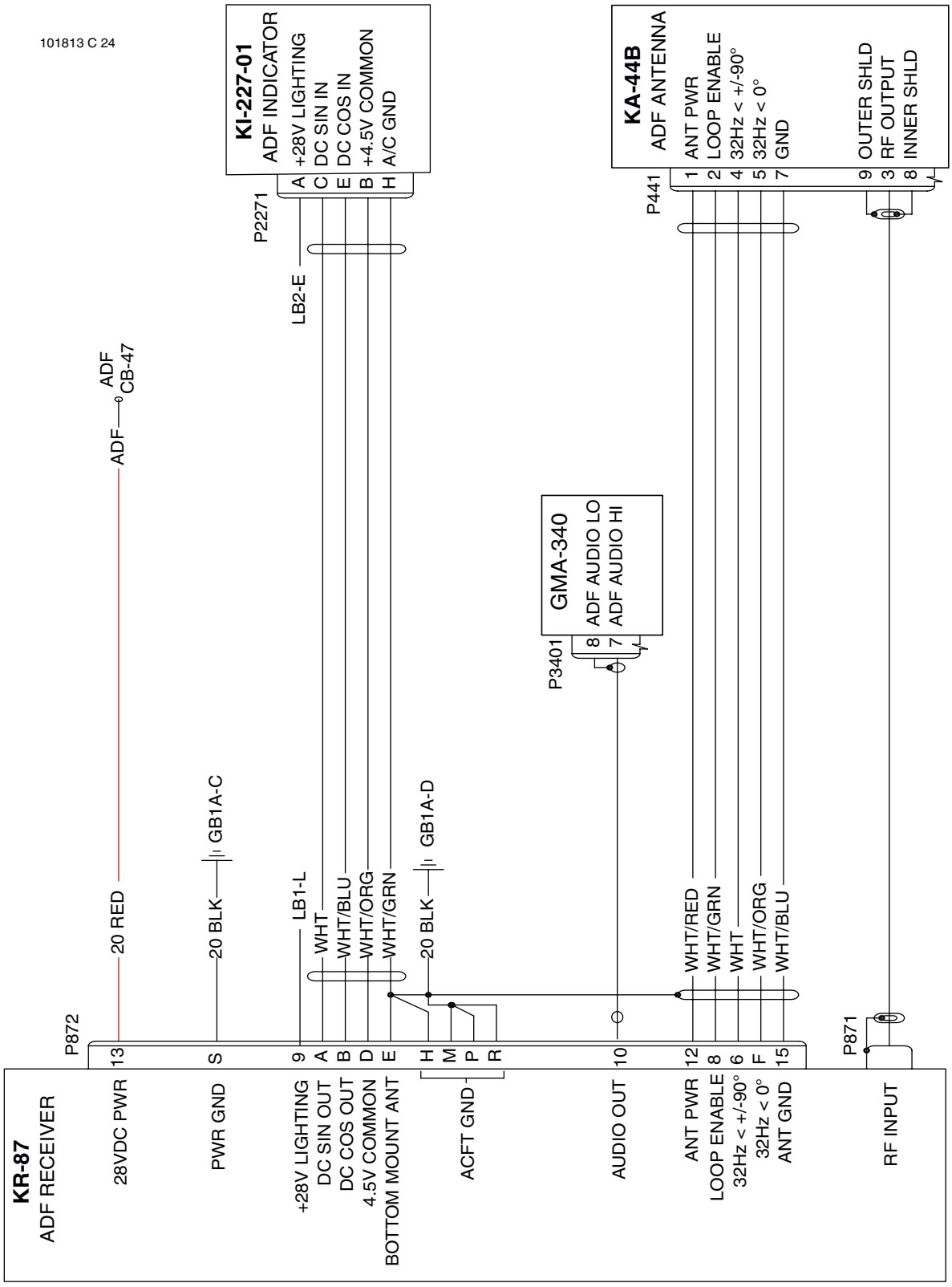
DME
Figure 5 (Sheet 1 of 2)

Effectivity
Seneca V
with Avidyne Entegra



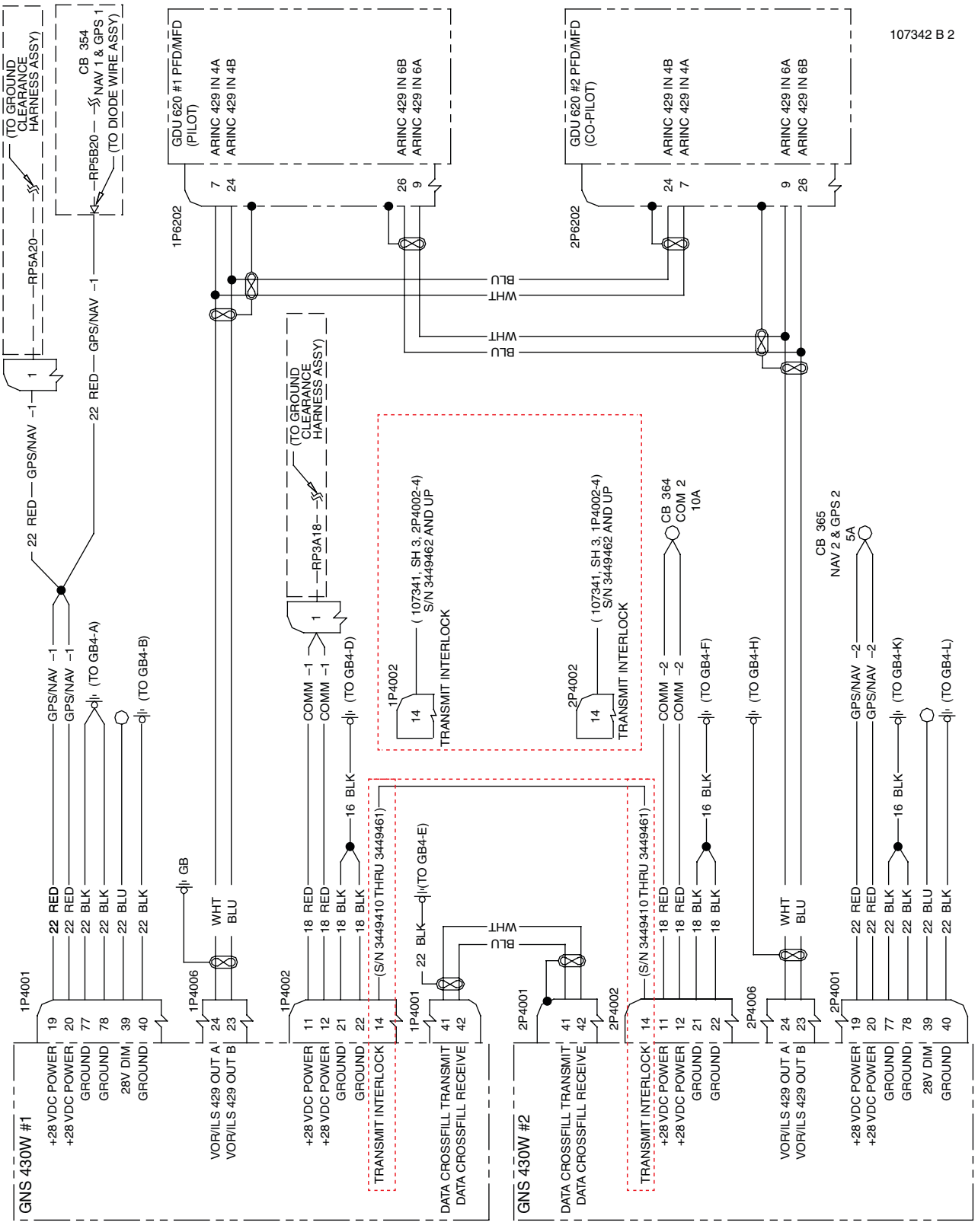
Effectivity
Seneca V
with Avidyne Entegra

DME
Figure 5 (Sheet 2 of 2)



ADF
Figure 6

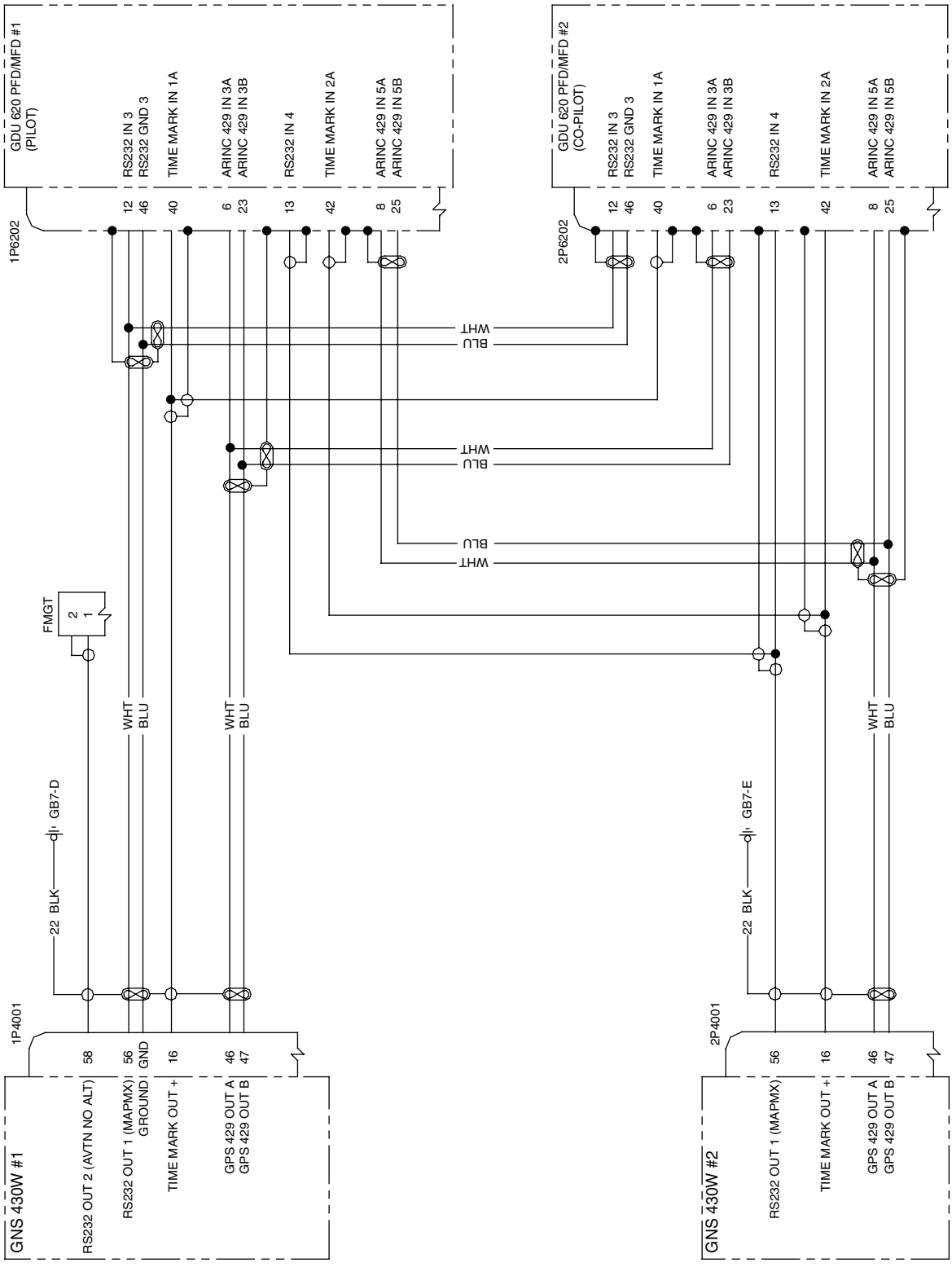
Effectivity
Seneca V
with Avidyne Entegra



Effectivity
Seneca V
with Garmin G600

NAV / COM / GPS
Figure 7 (Sheet 1 of 4)

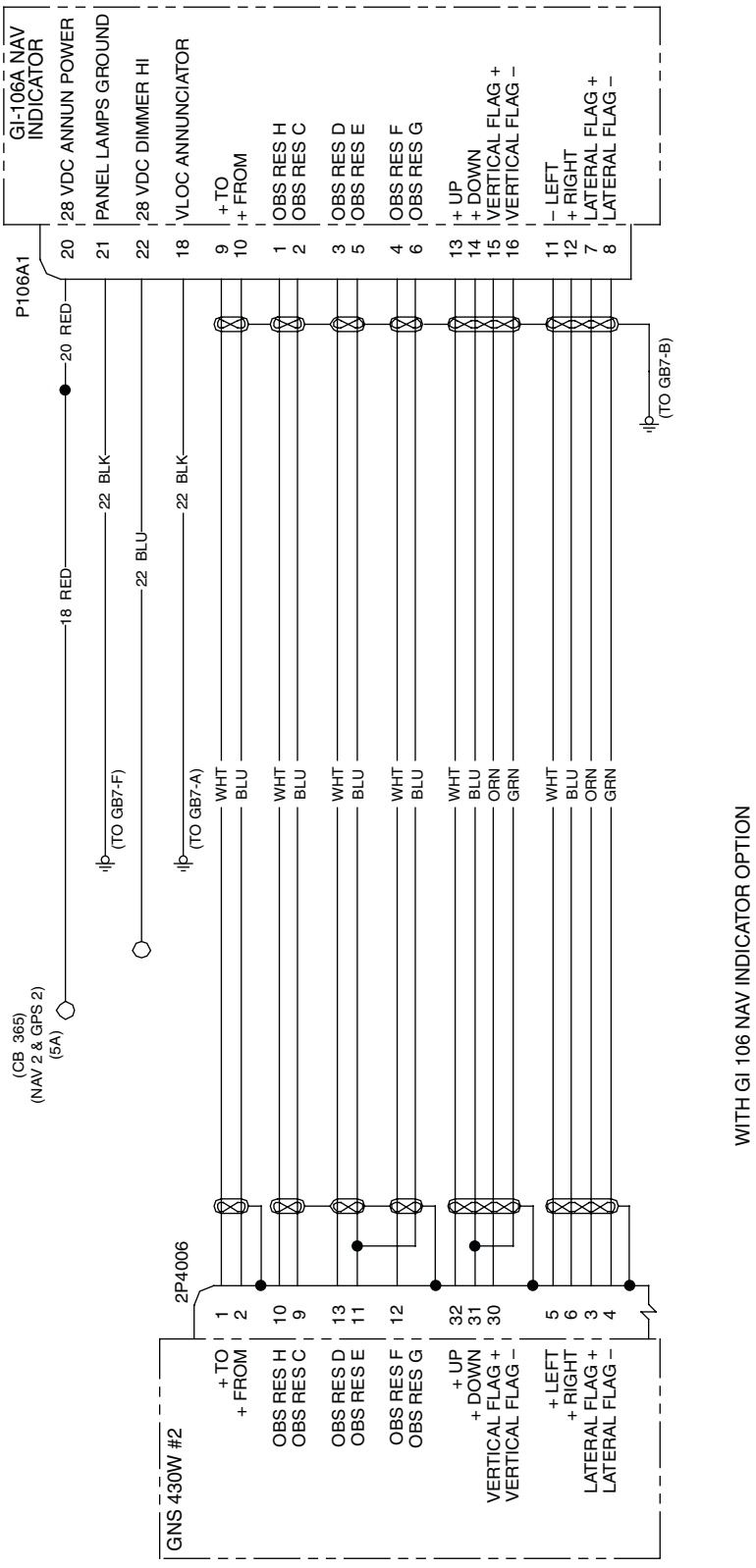
107342 A 3



NAV / COM / GPS
Figure 7 (Sheet 2 of 4)

Effectivity
Seneca V
with Garmin G600

107342 A 4

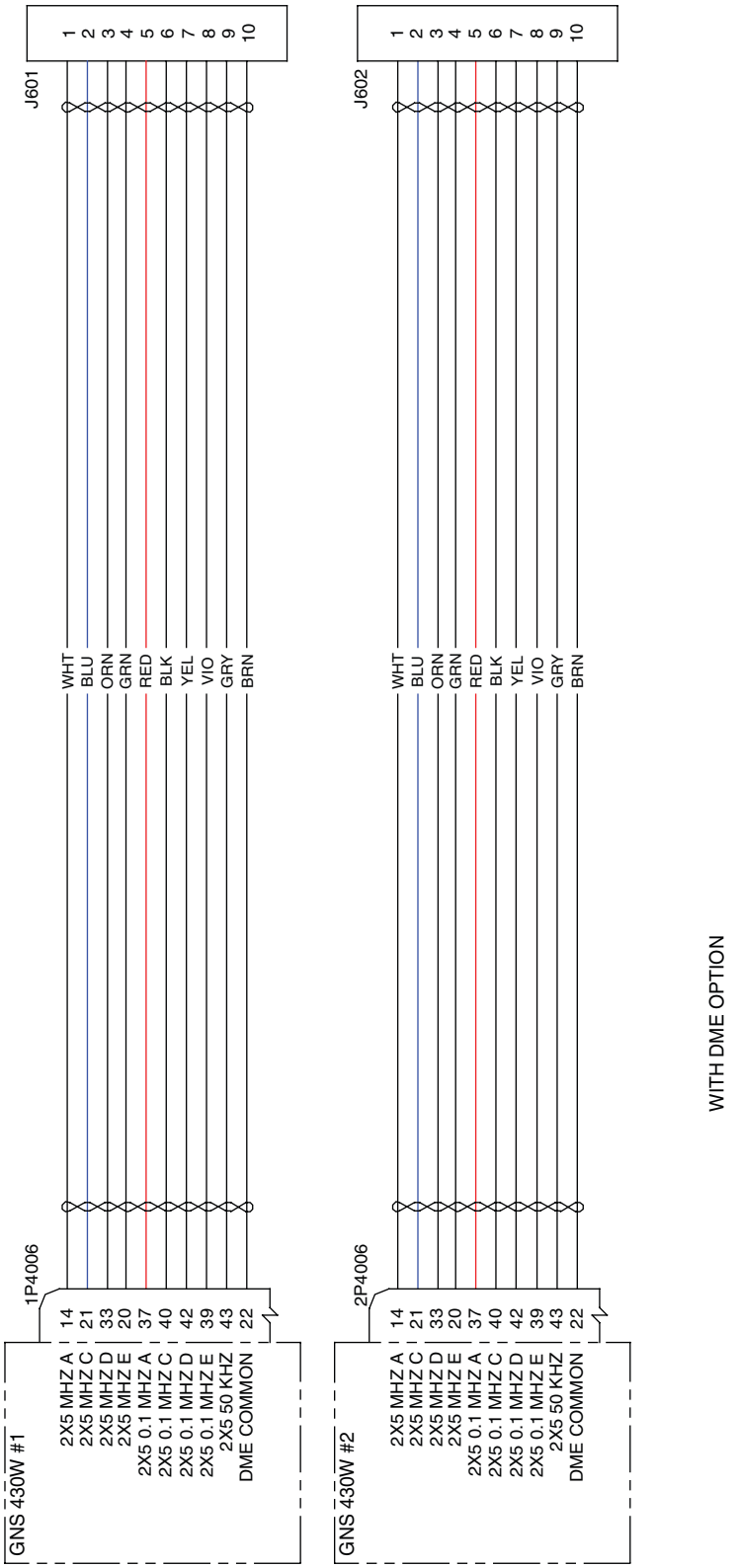


WITH GI 106 NAV INDICATOR OPTION

Effectivity
Seneca V
with Garmin G600

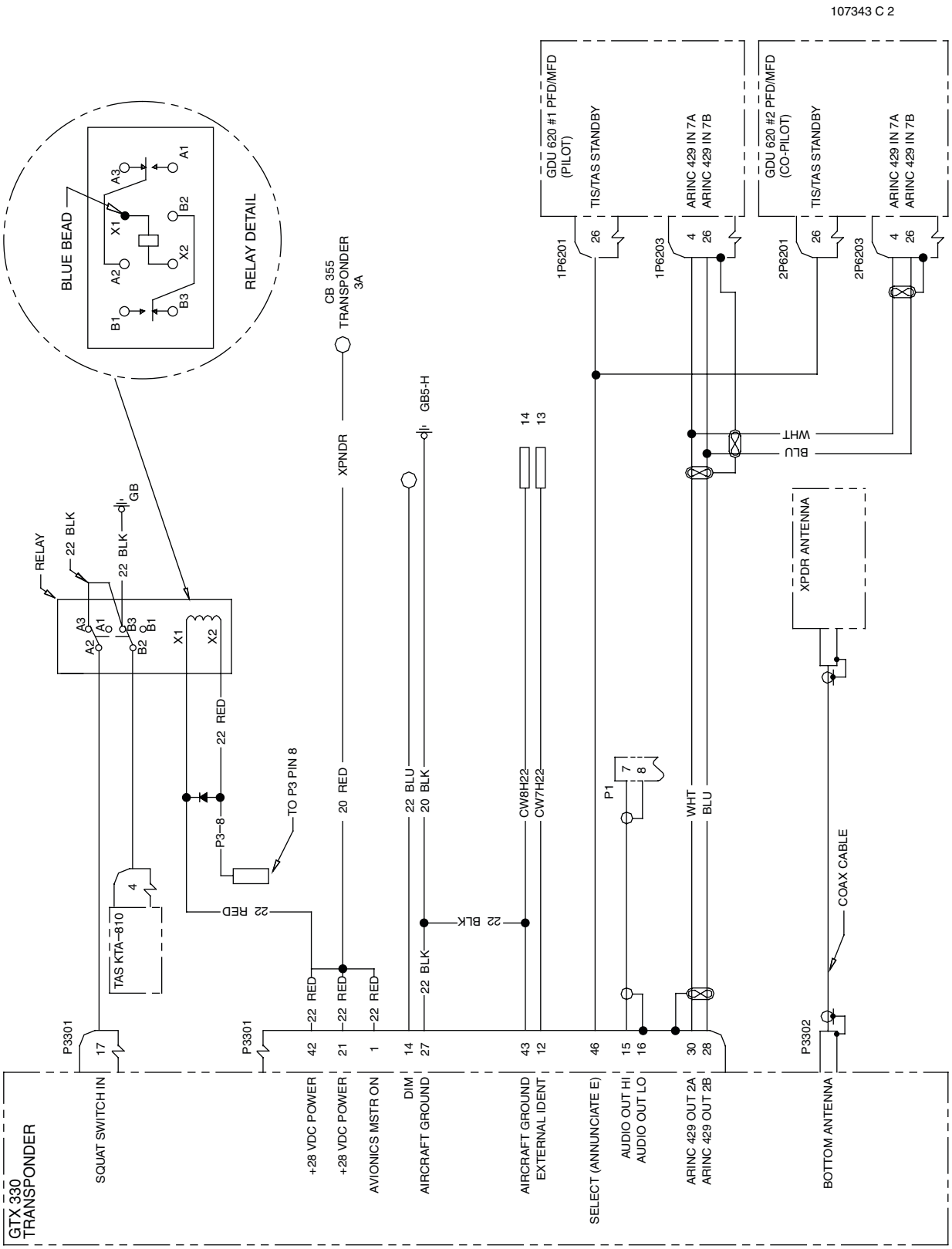
NAV / COM / GPS
Figure 7 (Sheet 3 of 4)

107342 A 4



NAV / COM / GPS
Figure 7 (Sheet 4 of 4)

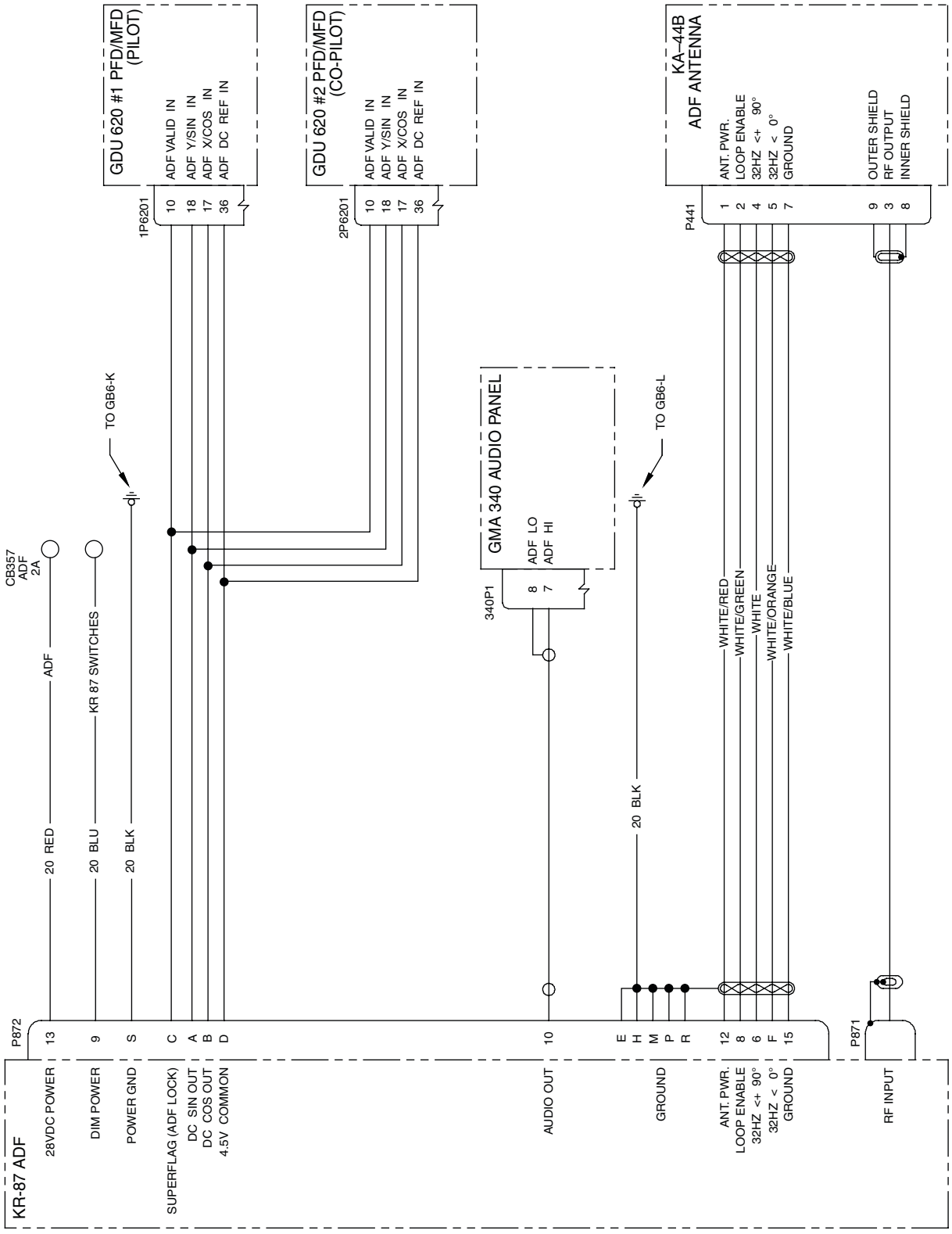
Effectivity
Seneca V
with Garmin G600



Effectivity
Seneca V
with Garmin G600

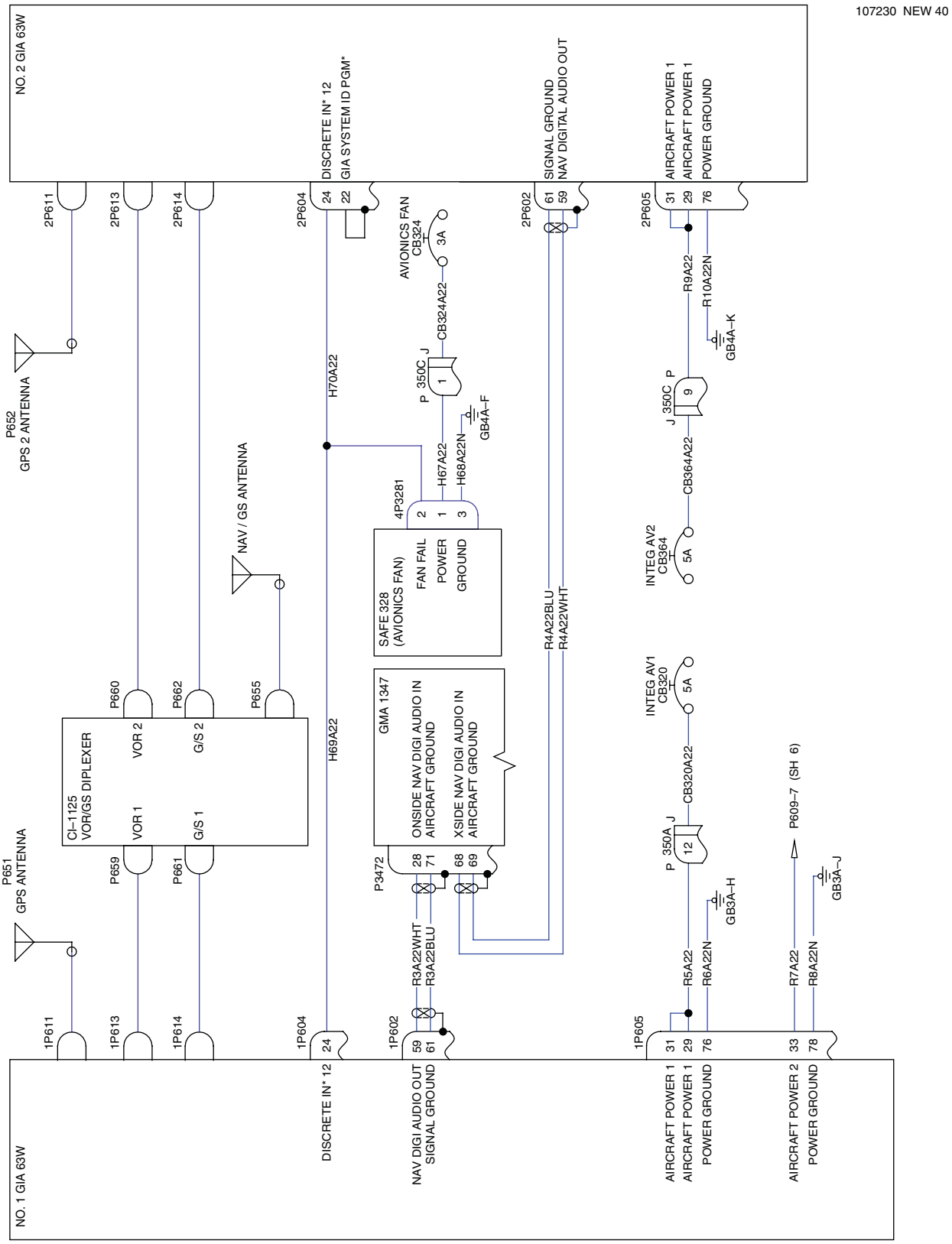
Transponder
Figure 8

107345 C 2



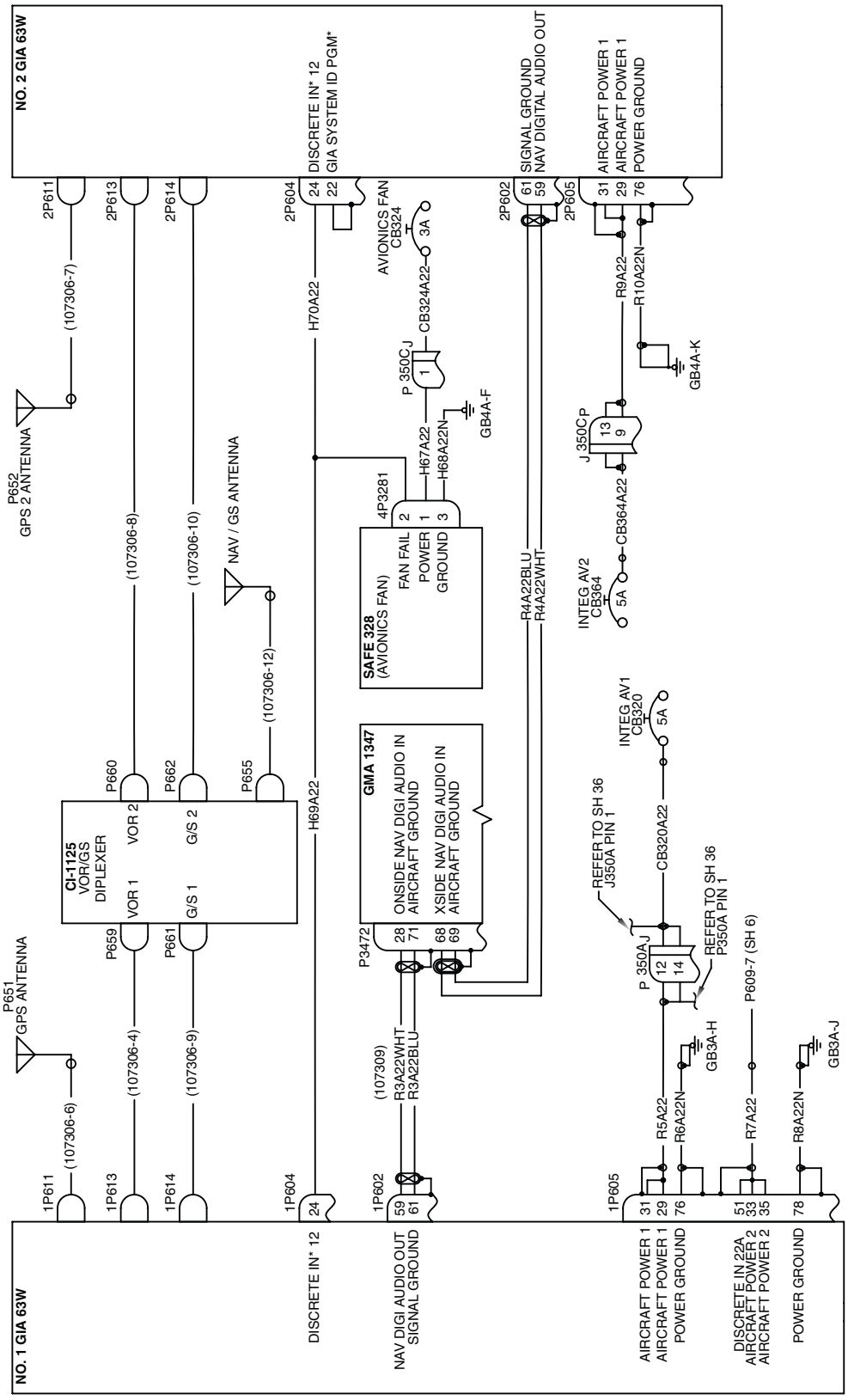
ADF
Figure 9

Effectivity
Seneca V
with Garmin G600



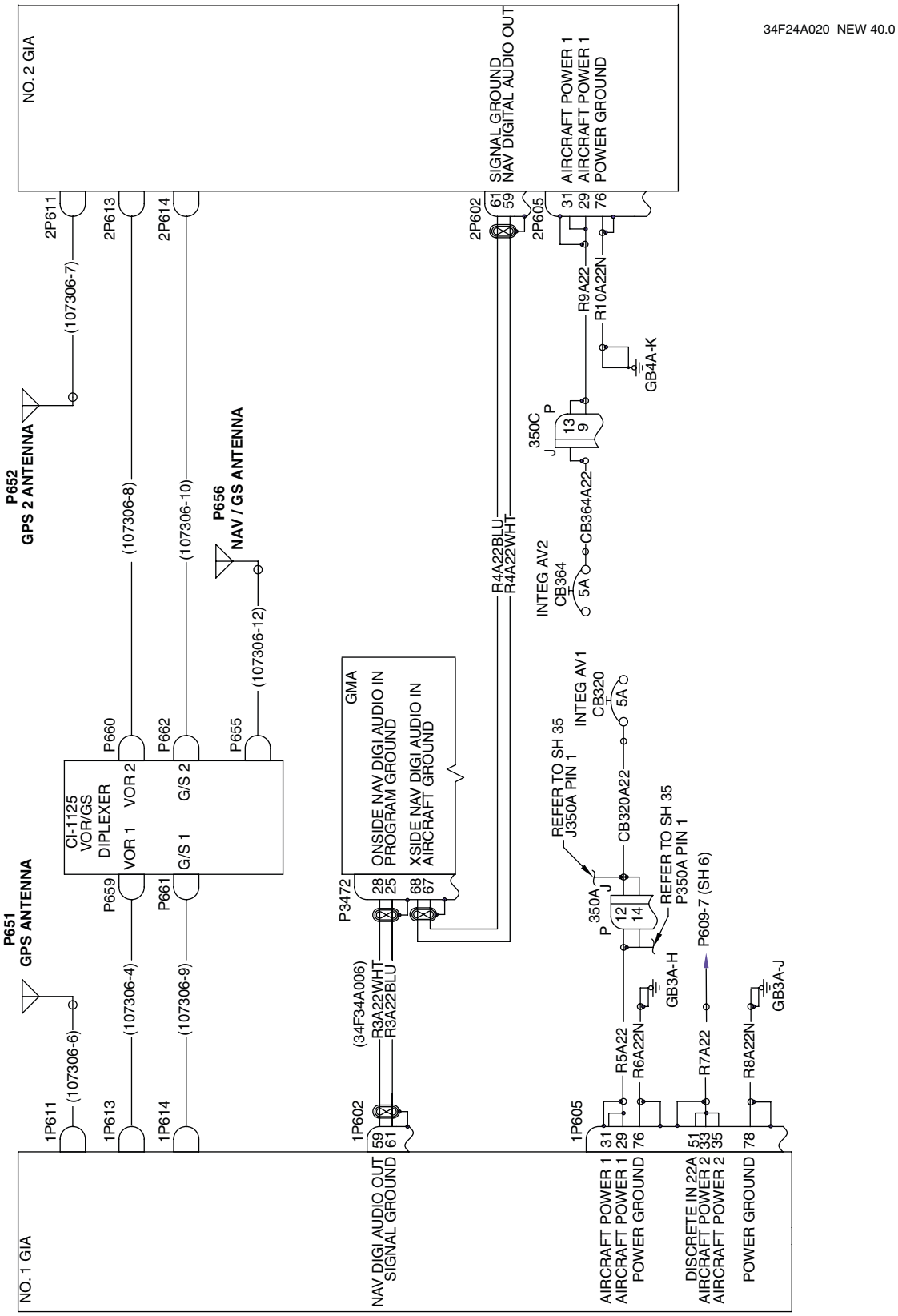
Effectivity
Seneca V
with Garmin G1000

NAV / GPS
Figure 10 (Sheet 1 of 3)



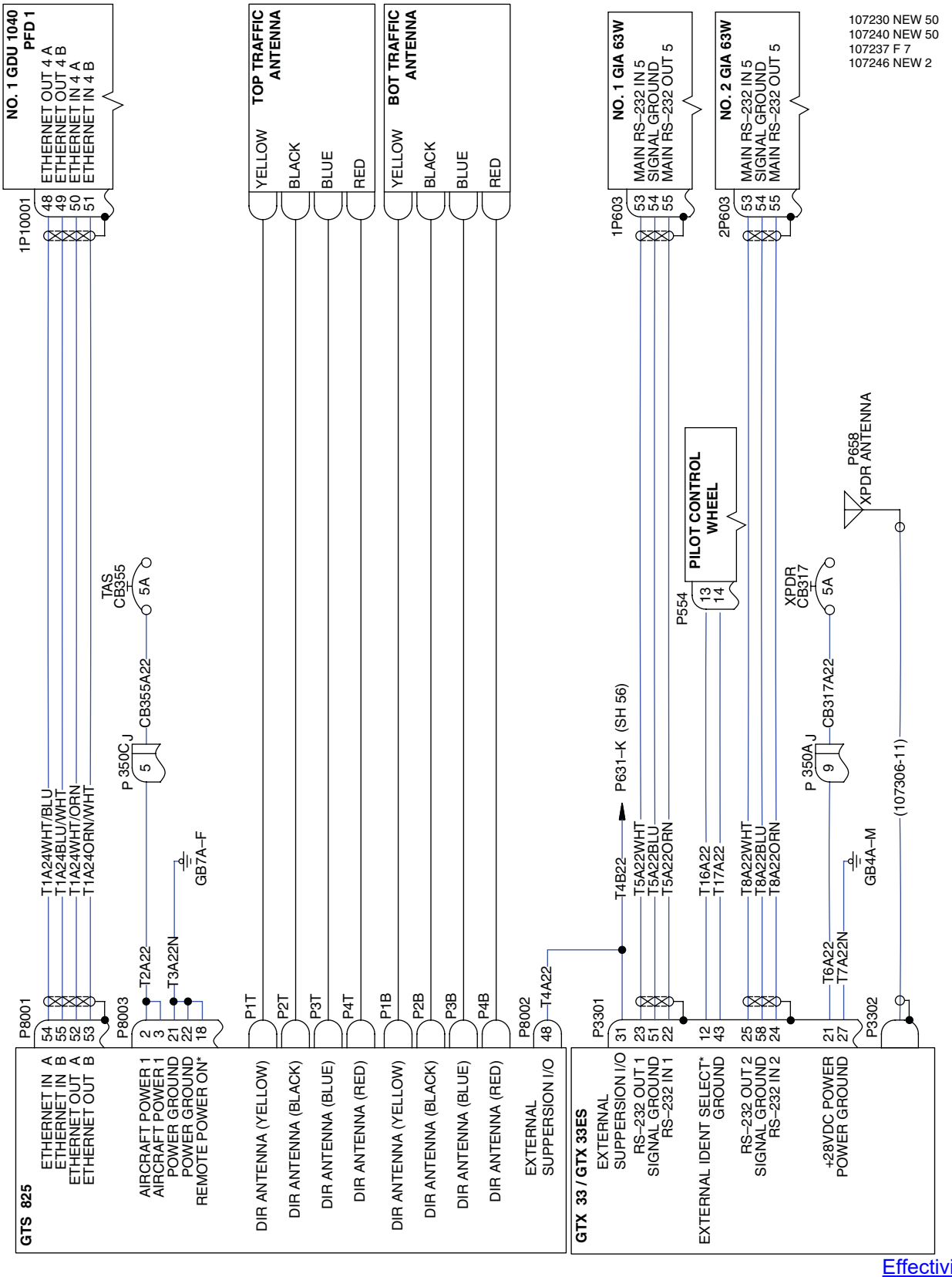
NAV / GPS
Figure 10 (Sheet 2 of 3)

Effectivity
Seneca V
with Garmin GFC 700



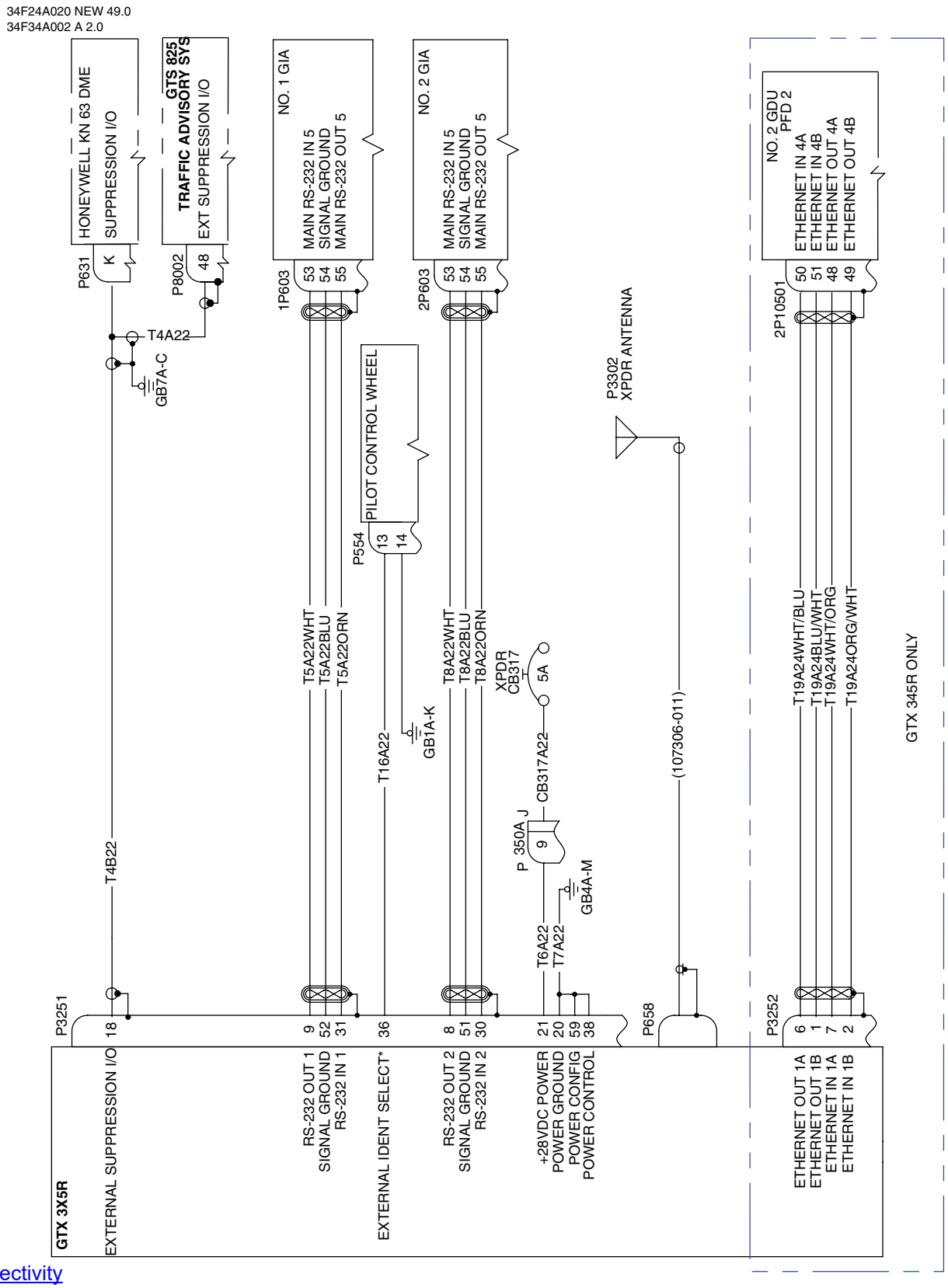
NAV / GPS
Figure 10 (Sheet 3 of 3)

Effectivity
Seneca V
with Garmin G1000 NXi



Traffic System / Transponder
Figure 11 (Sheet 1 of 3)

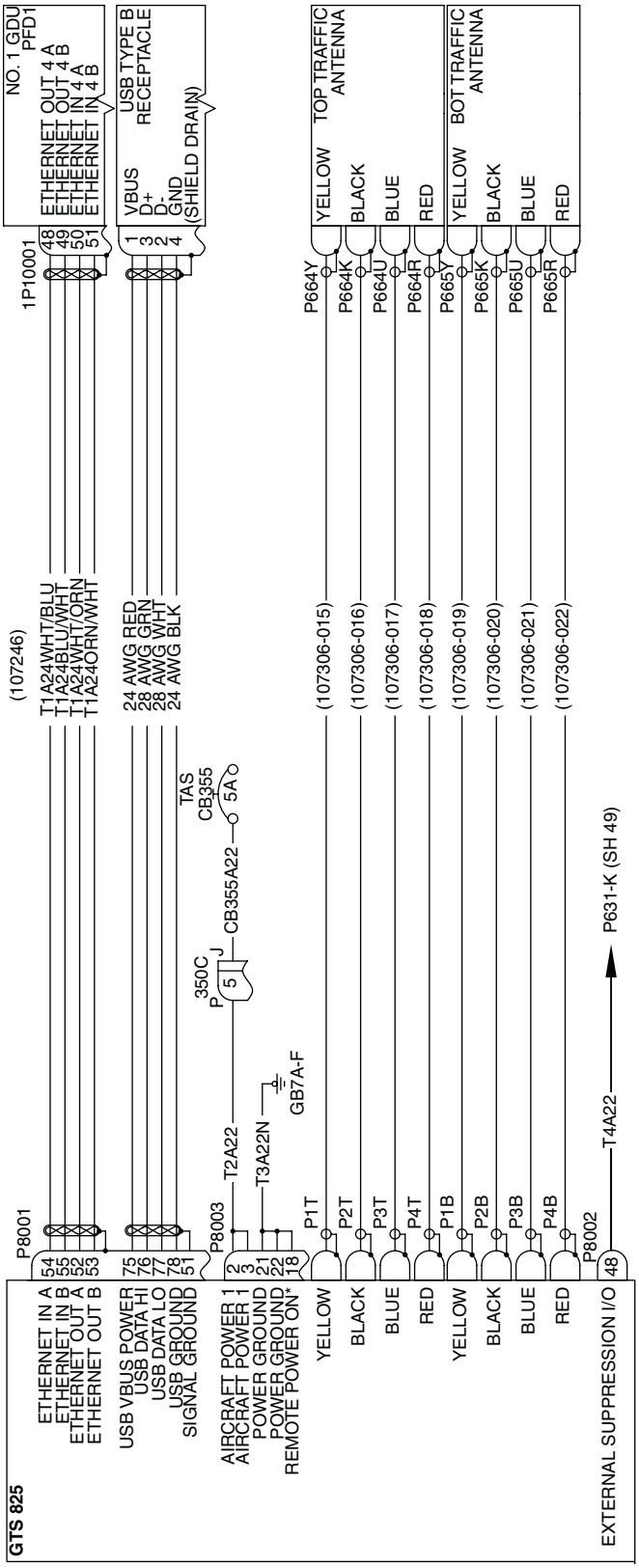
Effectivity
Seneca V
with Garmin G1000



Effectivity
Seneca V
with Garmin G1000 NXi

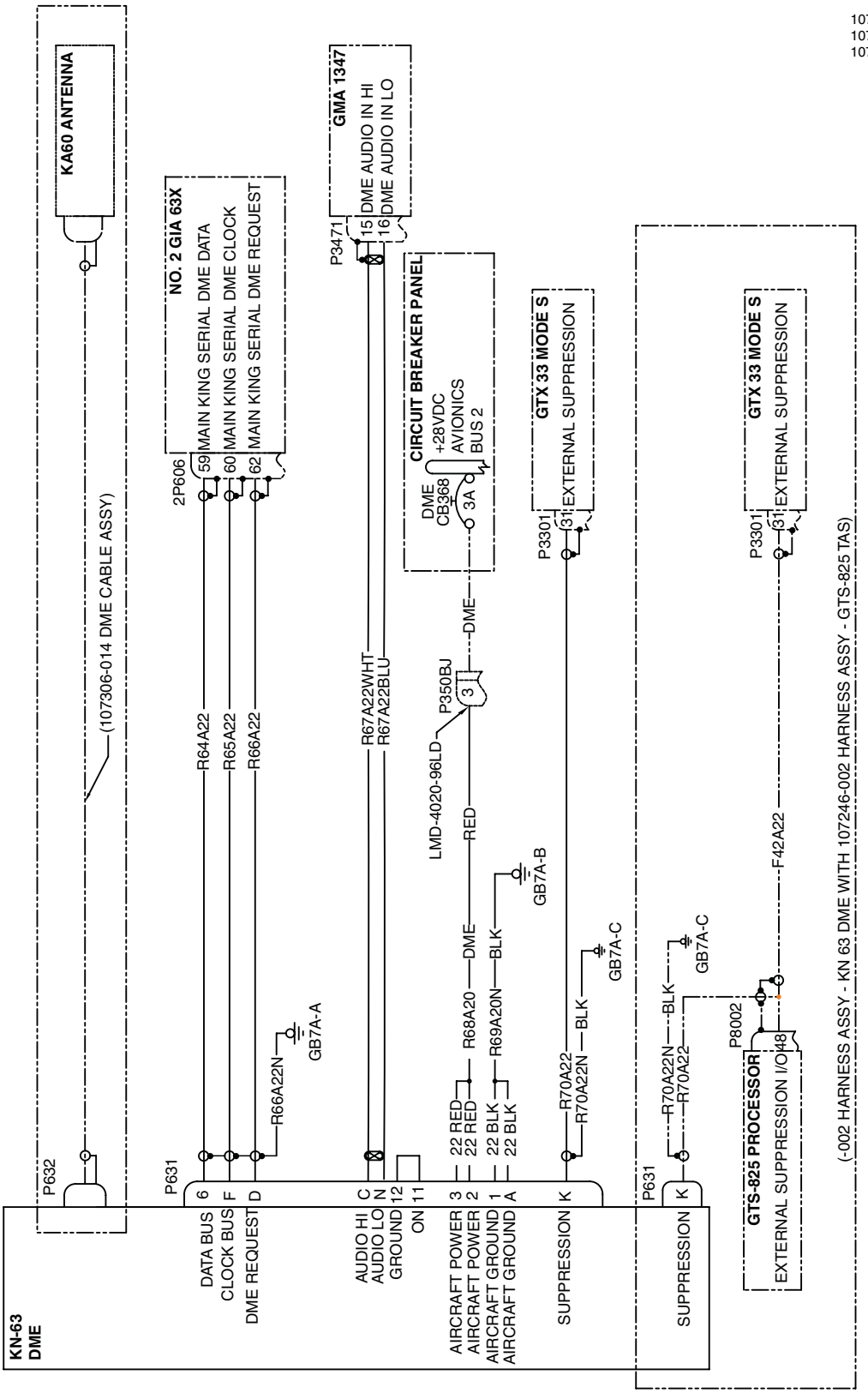
Traffic System / Transponder
Figure 11 (Sheet 2 of 3)

34F24A020 NEW 50.0



Traffic System / Transponder
Figure 11 (Sheet 3 of 3)

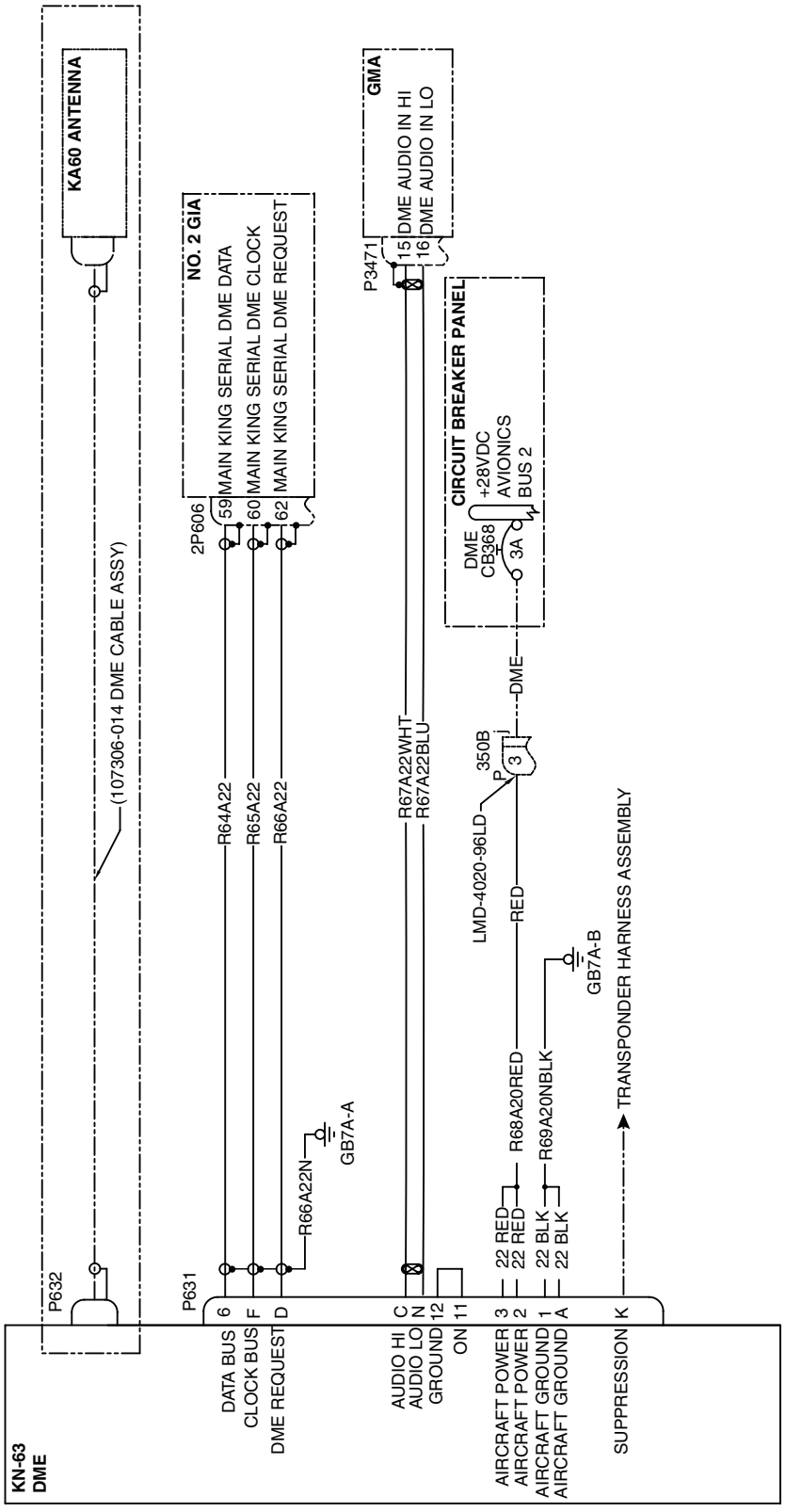
Effectivity
Seneca V
with Garmin G1000 NXi



Effectivity
Seneca V
with Garmin G1000 / GFC 700

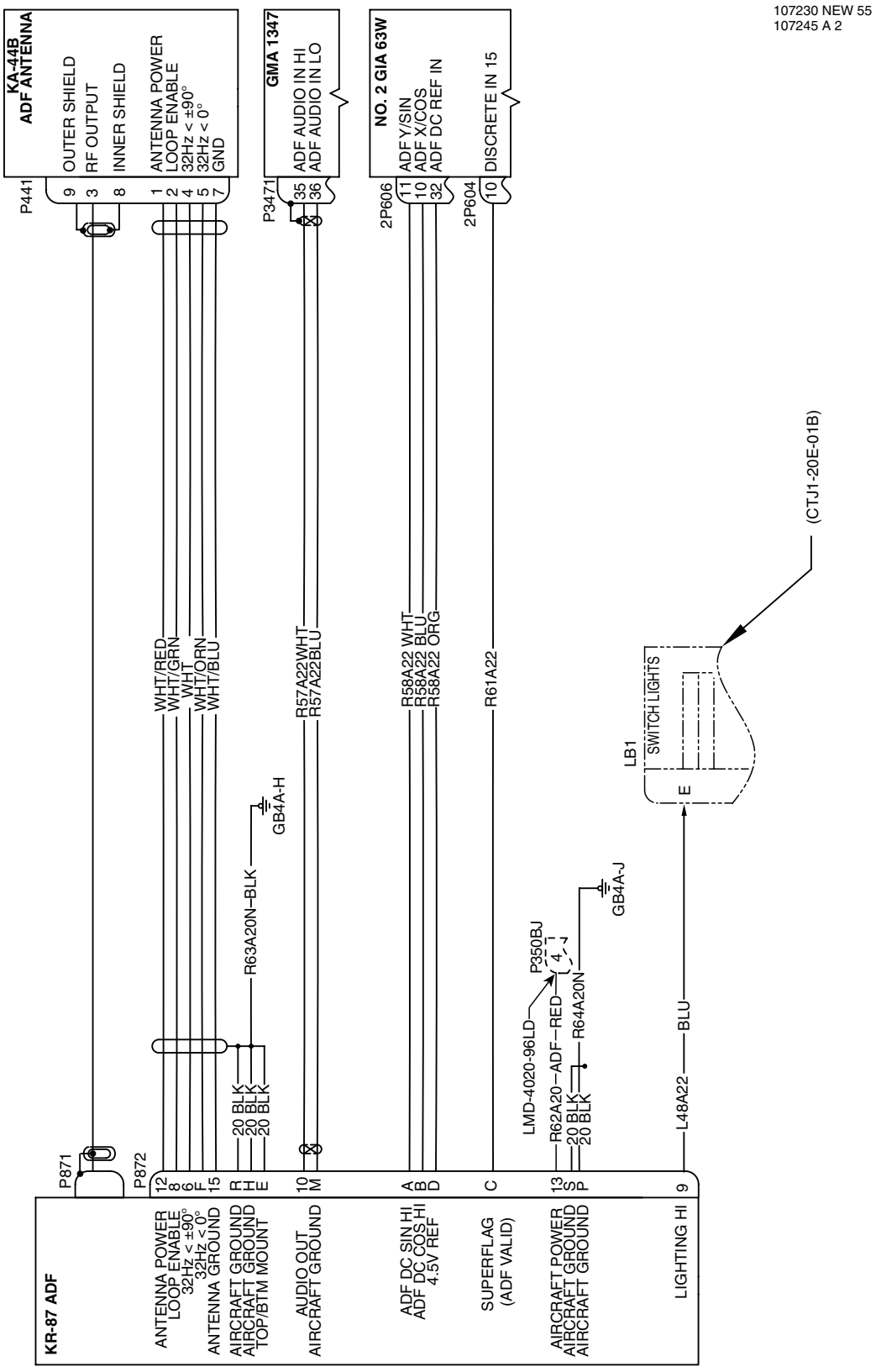
DME
Figure 12 (Sheet 1 of 2)

34F24A020 NEW 56.0
107244 A 3.0



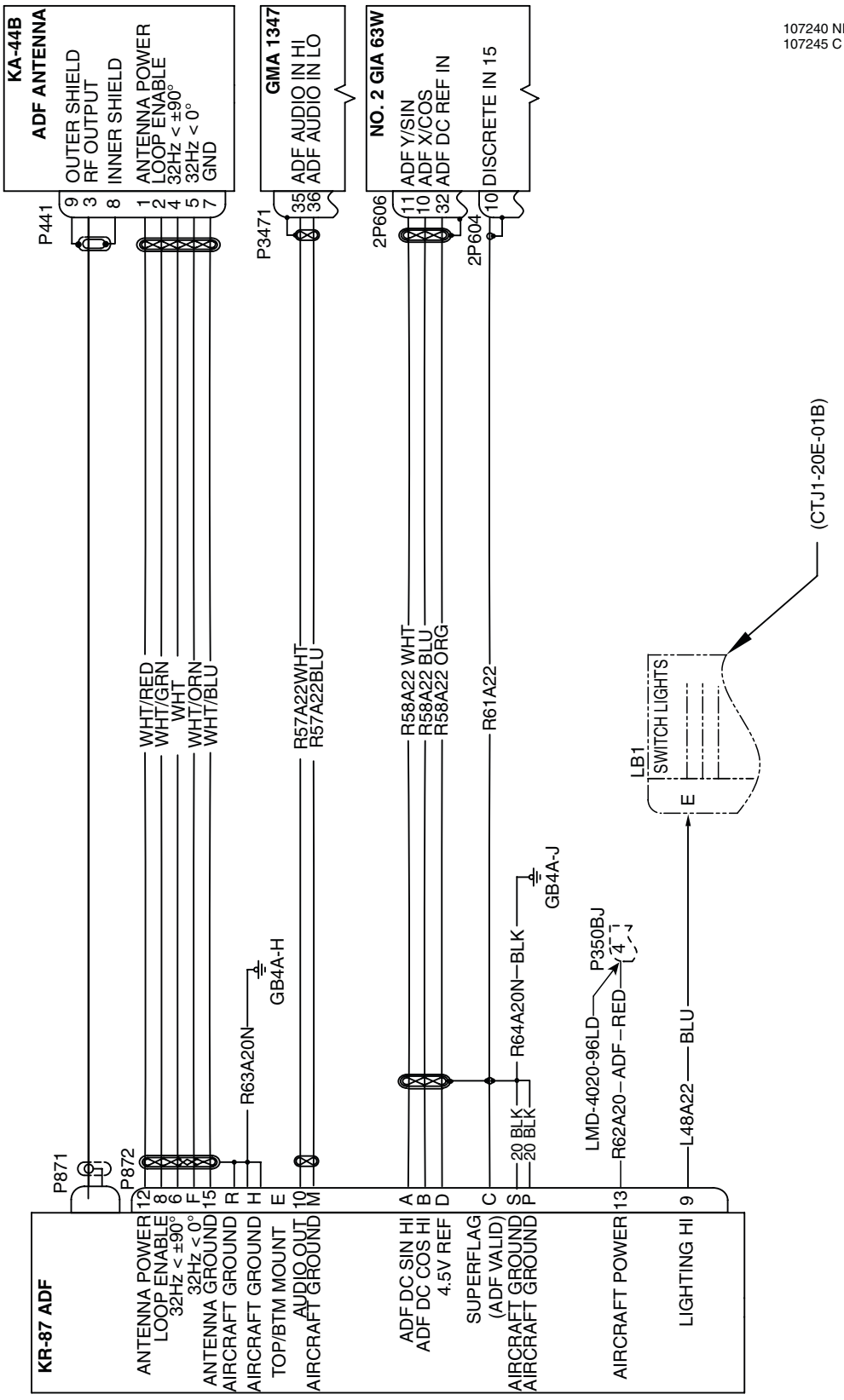
DME
Figure 12 (Sheet 2 of 2)

Effectivity
Seneca V
with Garmin G1000 NXi



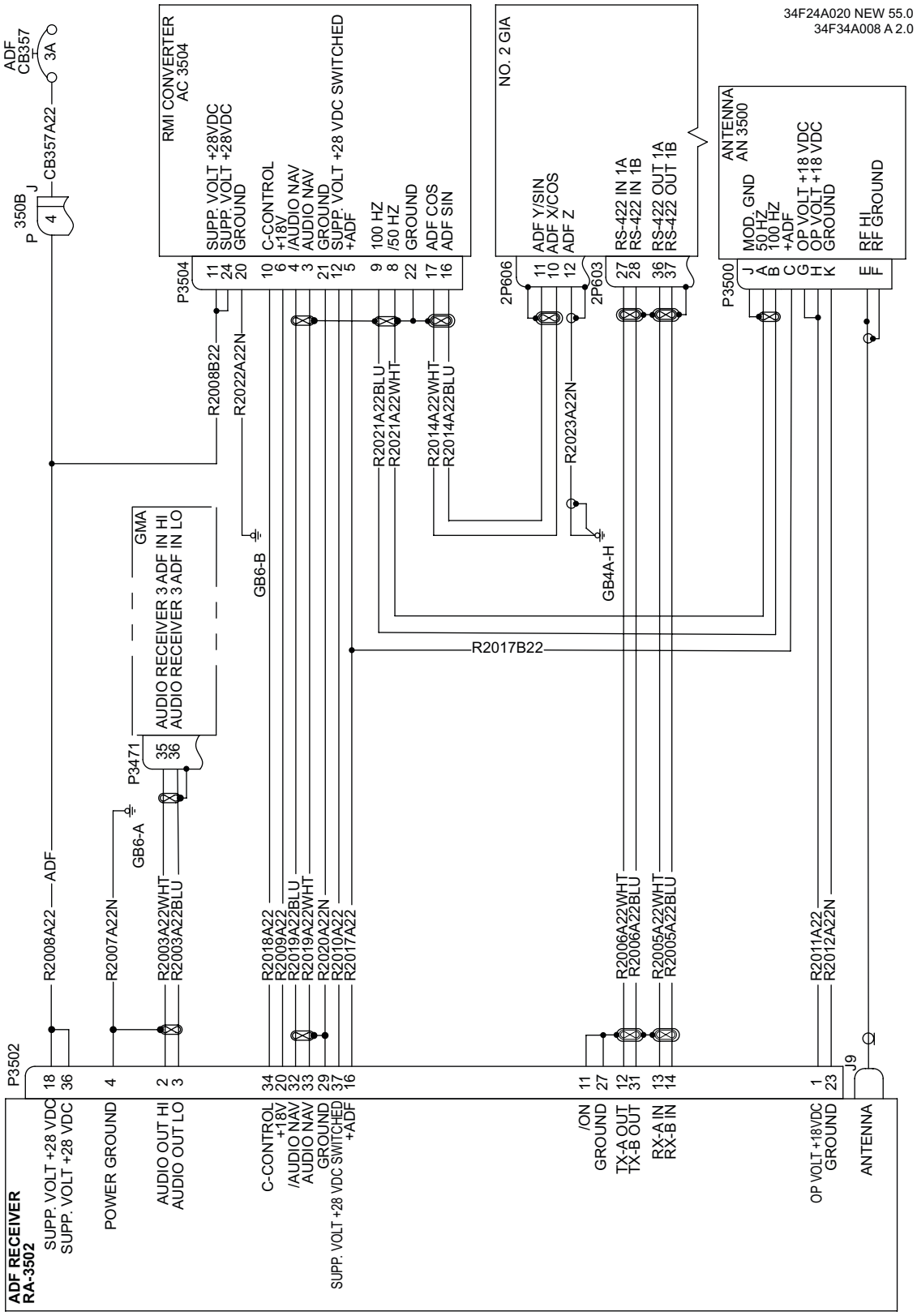
ADF
Figure 13 (Sheet 1 of 3)

Effectivity
Seneca V
with Garmin G1000



ADF
Figure 13 (Sheet 2 of 3)

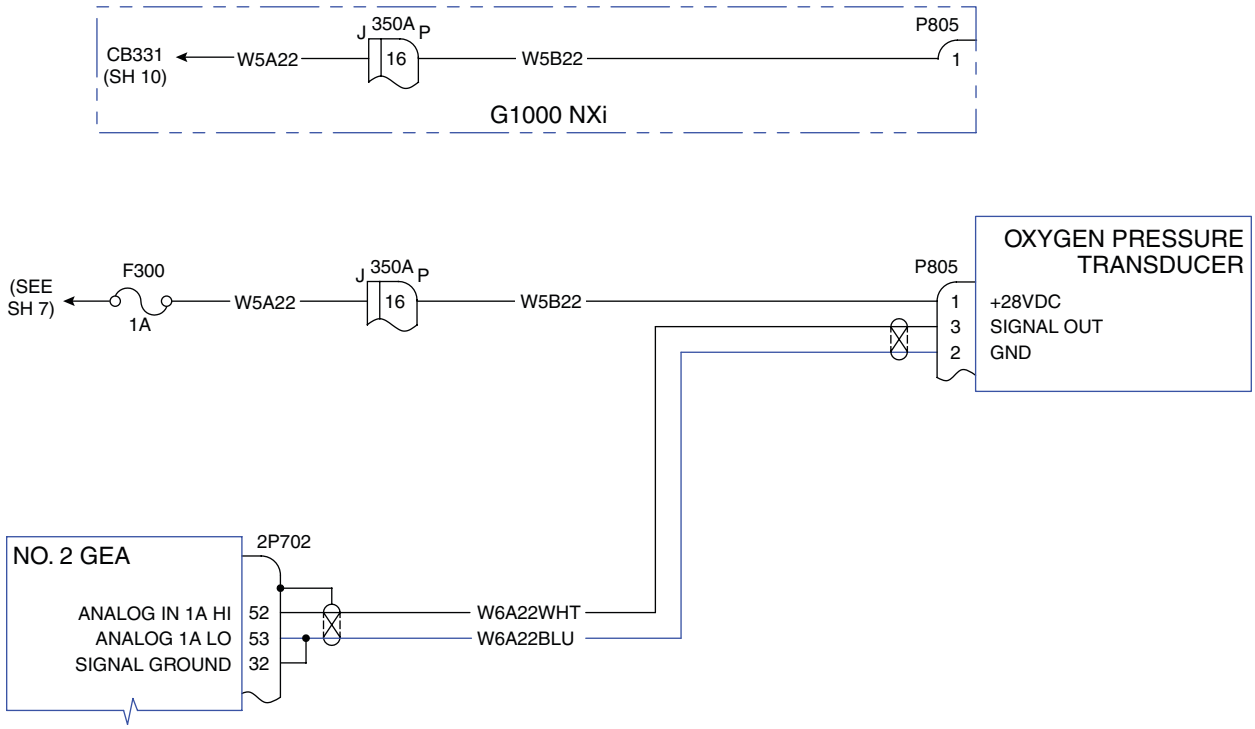
Effectivity
Seneca V
with Garmin G1000 and GFC 700



Effectivity
Seneca V
with Garmin G1000 NXi

ADF
Figure 13 (Sheet 3 of 3)

107230 NEW 24.0
107240 NEW 24.0
34F24A020 NEW 23.0

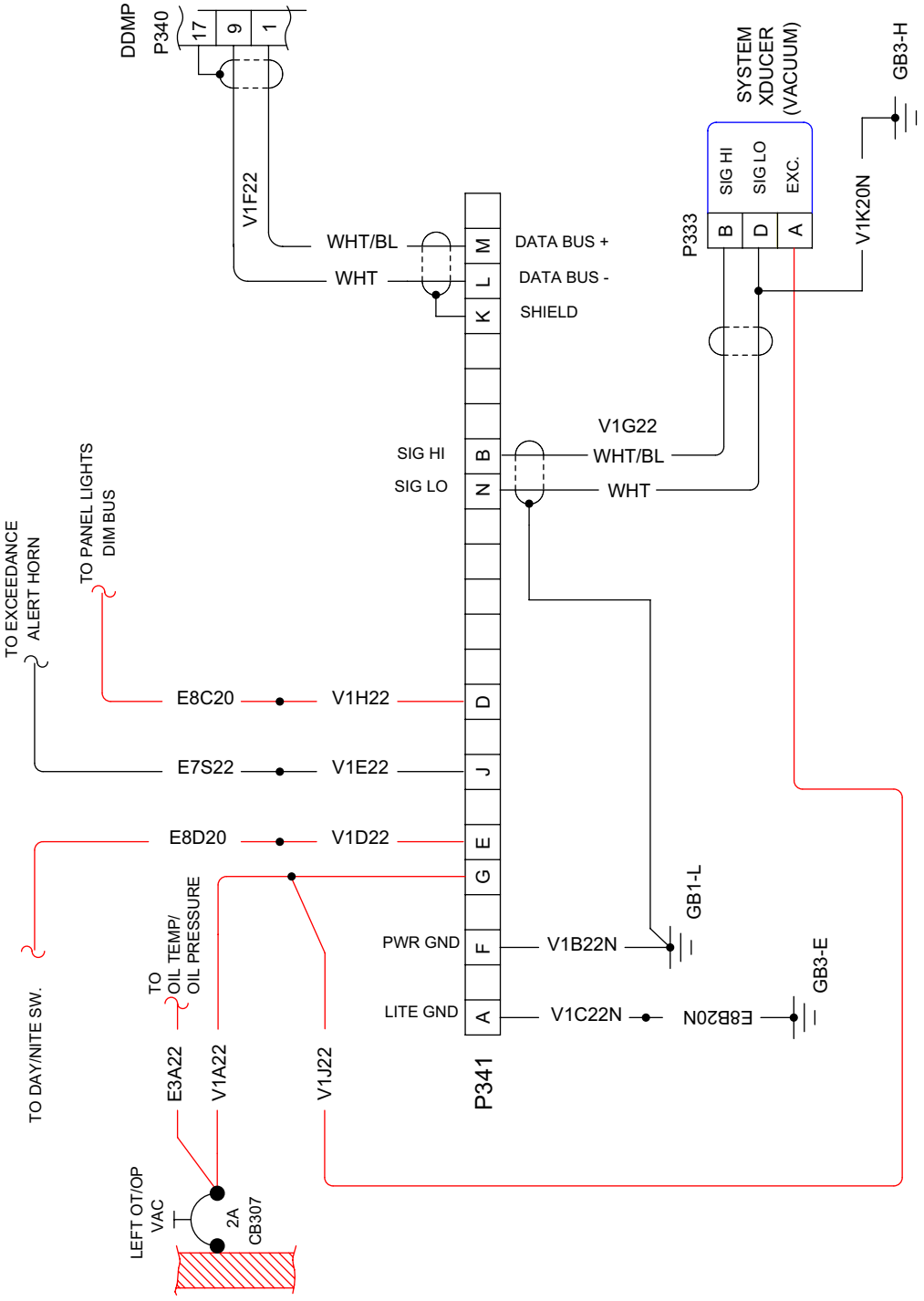


Oxygen Pressure Transducer
Figure 1

Effectivity
Seneca V
with Garmin G1000 / G1000 NXi

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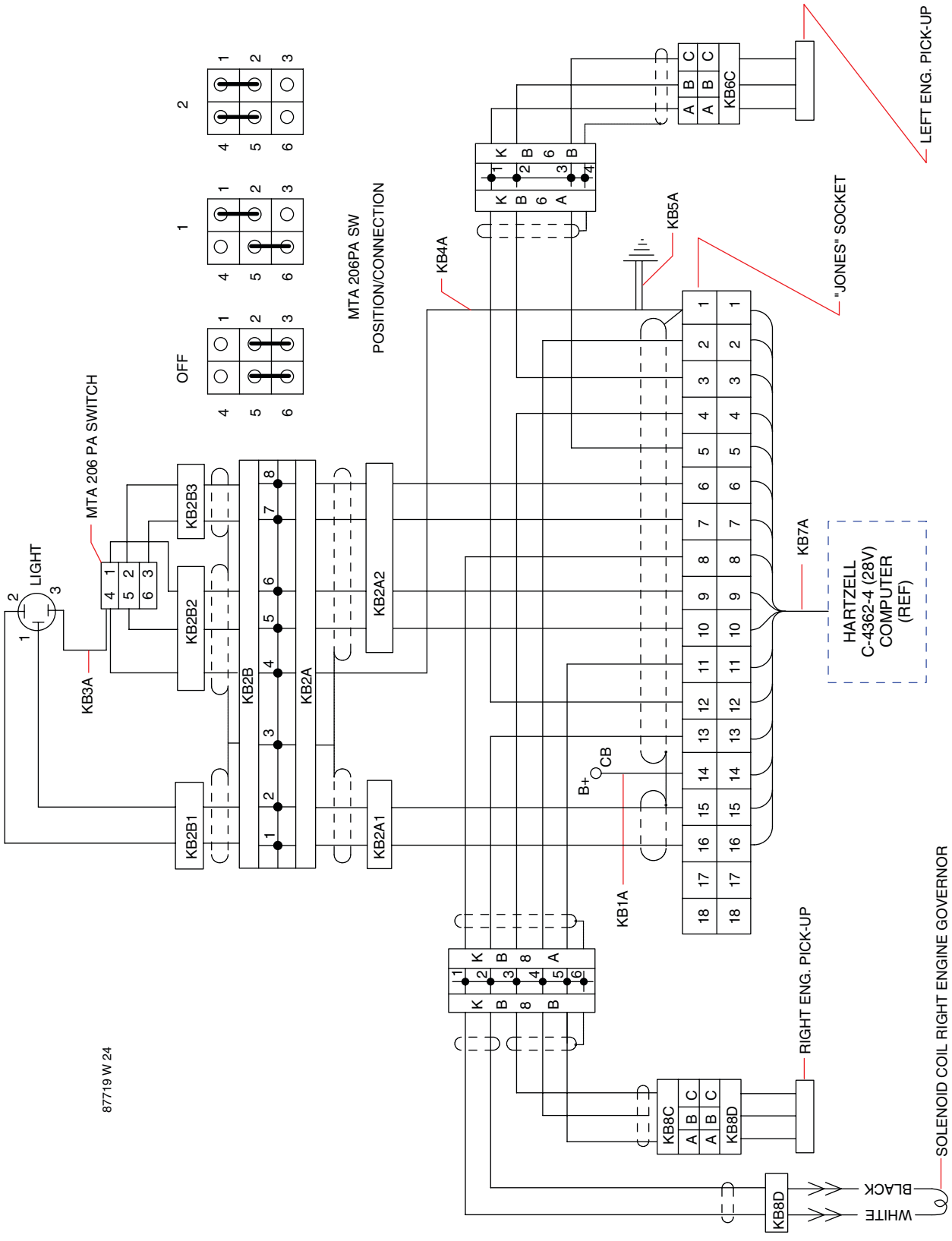
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101288 NEW/D 5.0
104229 NEW/C 5.0
104368 NEW/K 5.0



Vacuum Gauge
Figure 1

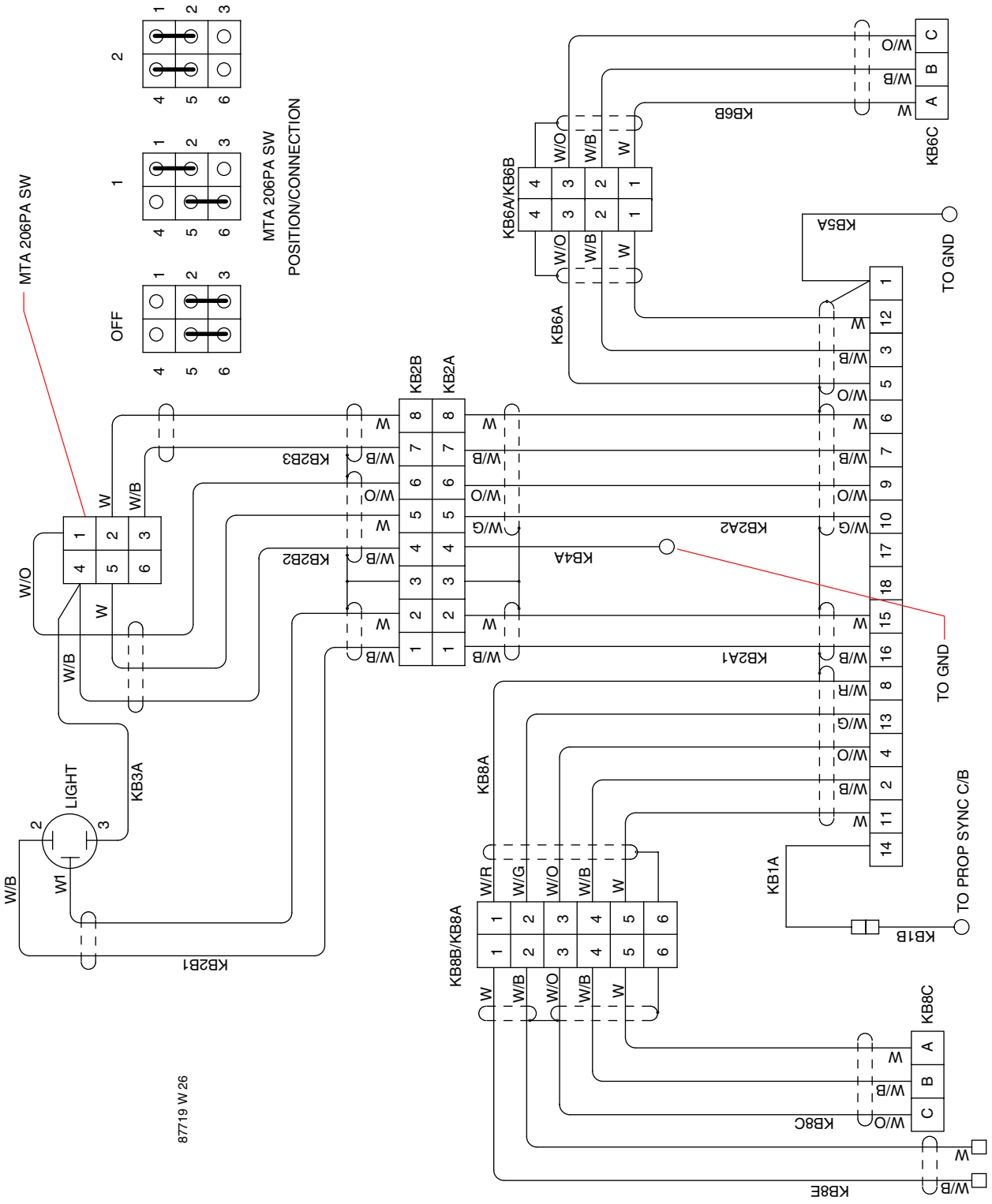
Effectivity
Seneca V

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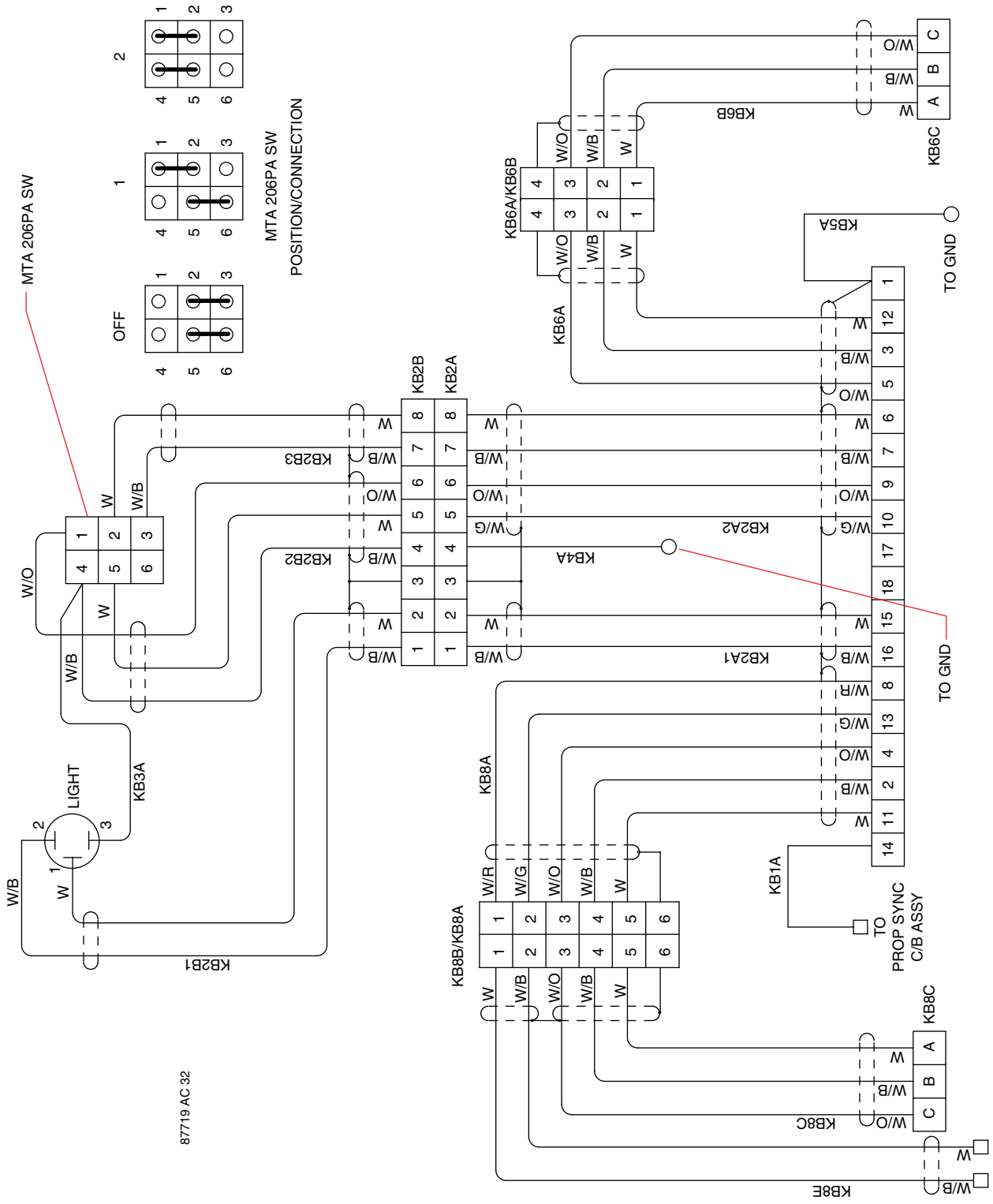
Propeller Synchrophaser
Figure 1 (Sheet 1 of 2)

Effectivity
Seneca IV



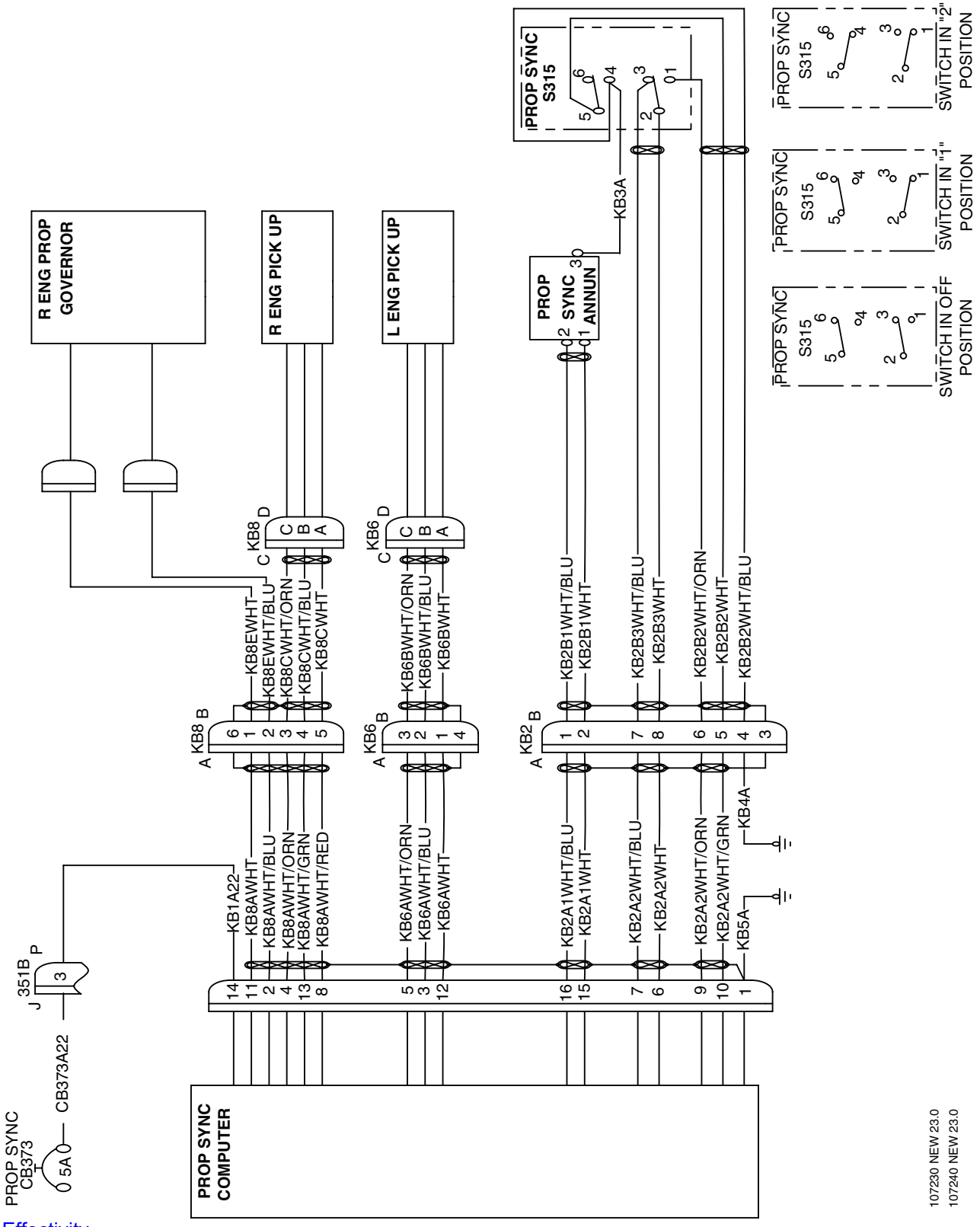
Effectivity
Seneca V

Propeller Synchrophaser
Figure 1 (Sheet 2 of 2)



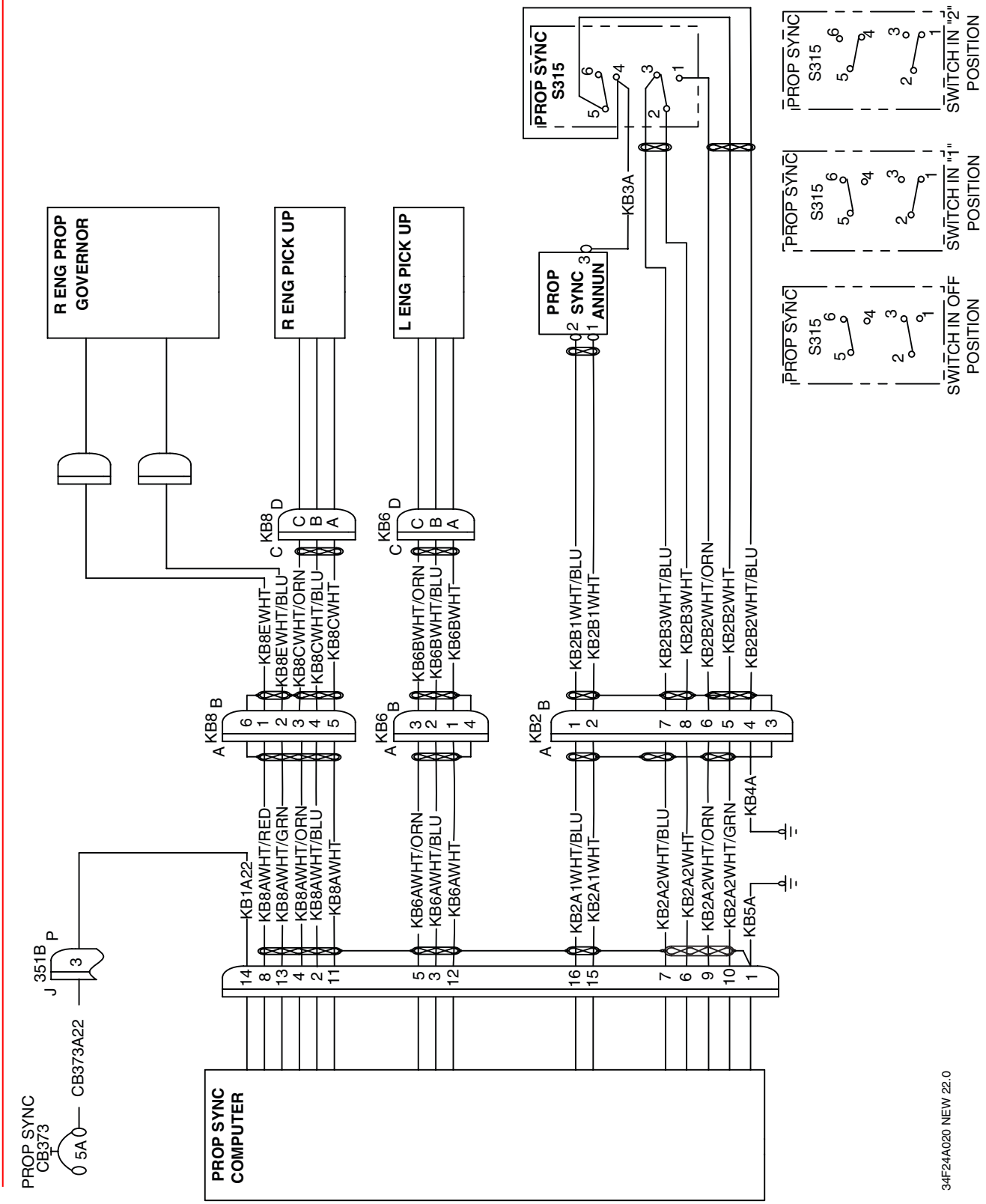
Propeller Synchrophaser
Figure 2 (Sheet 1 of 3)

Effectivity
Seneca V
with Garmin G1000



Effectivity
Seneca V
with Garmin G1000

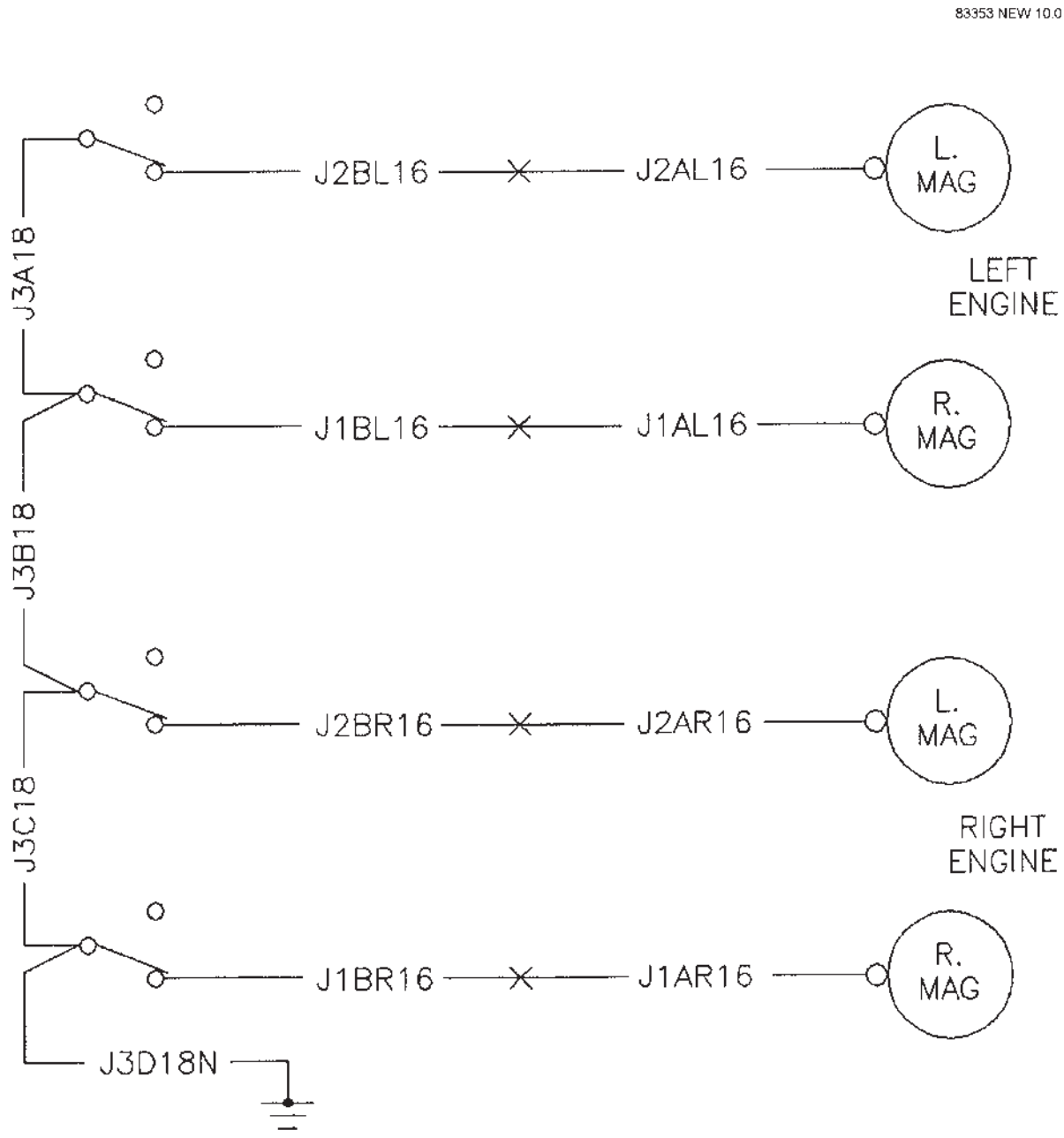
Propeller Synchrophaser
Figure 2 (Sheet 2 of 3)



Propeller Synchrophaser
Figure 2 (Sheet 3 of 3)

Effectivity
Seneca V
with Garmin G1000 NXi

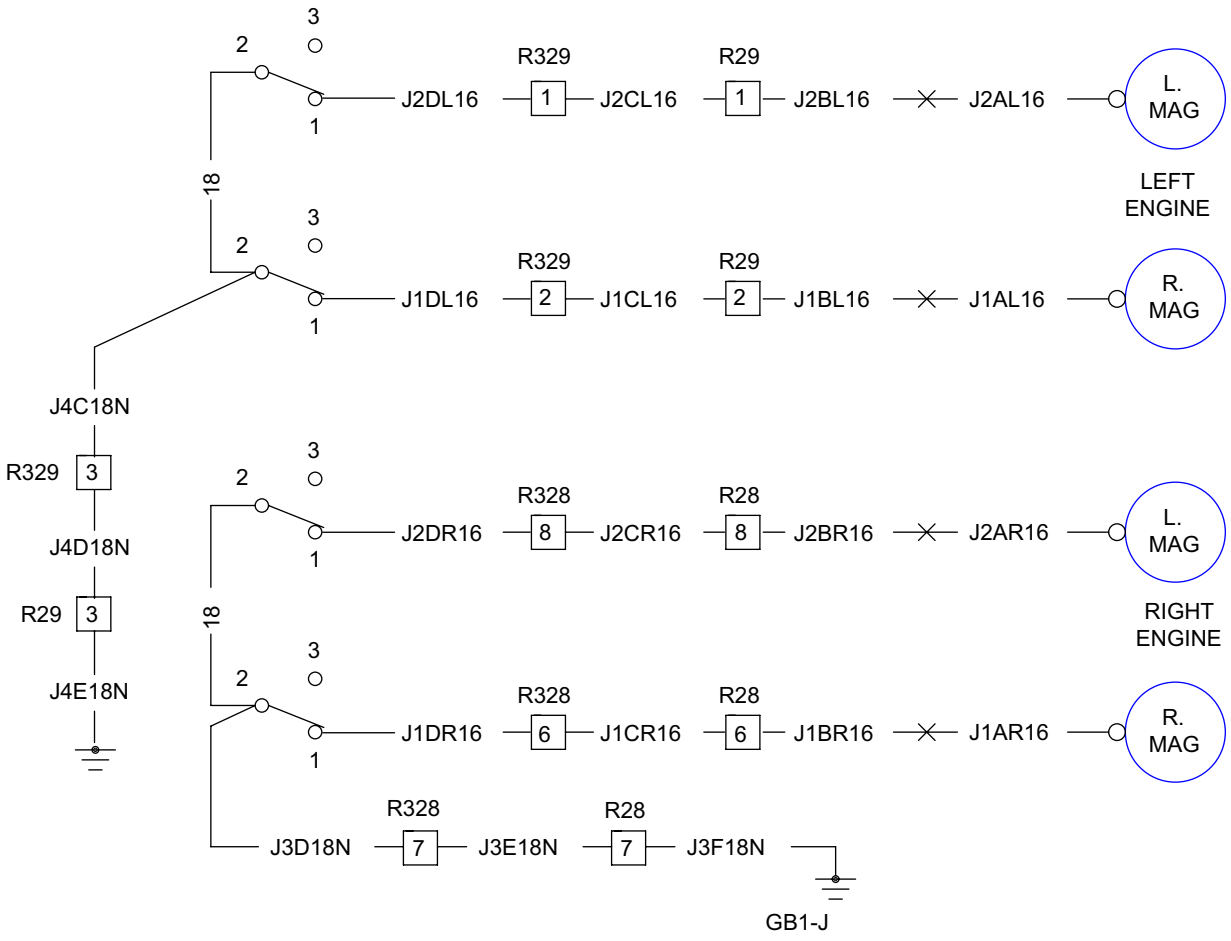
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Magnetos
Figure 1 (Sheet 1 of 4)

Effectivity
Seneca IV

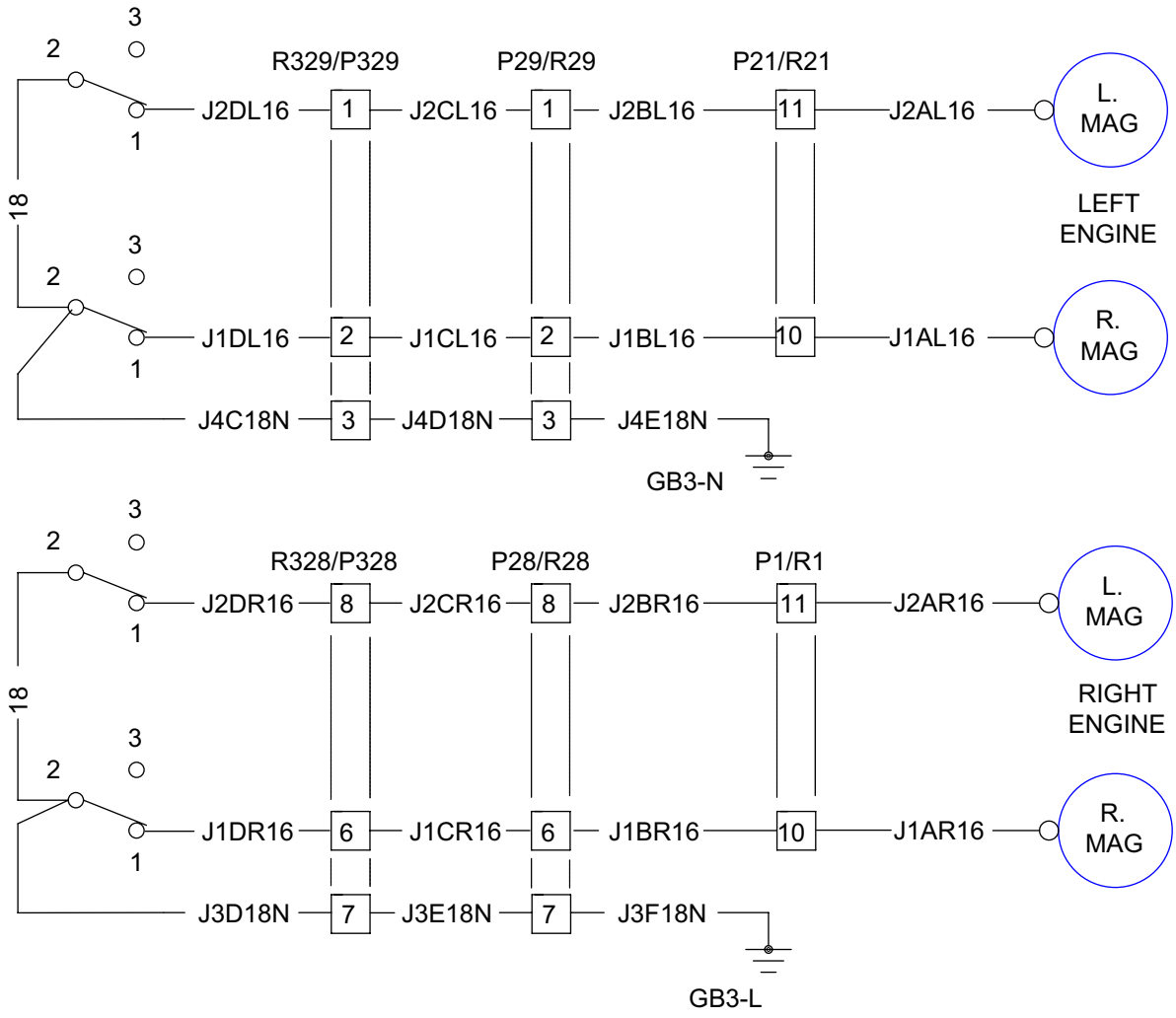
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Effectivity
3449001 thru 3449151

Magnetos
Figure 1 (Sheet 2 of 4)

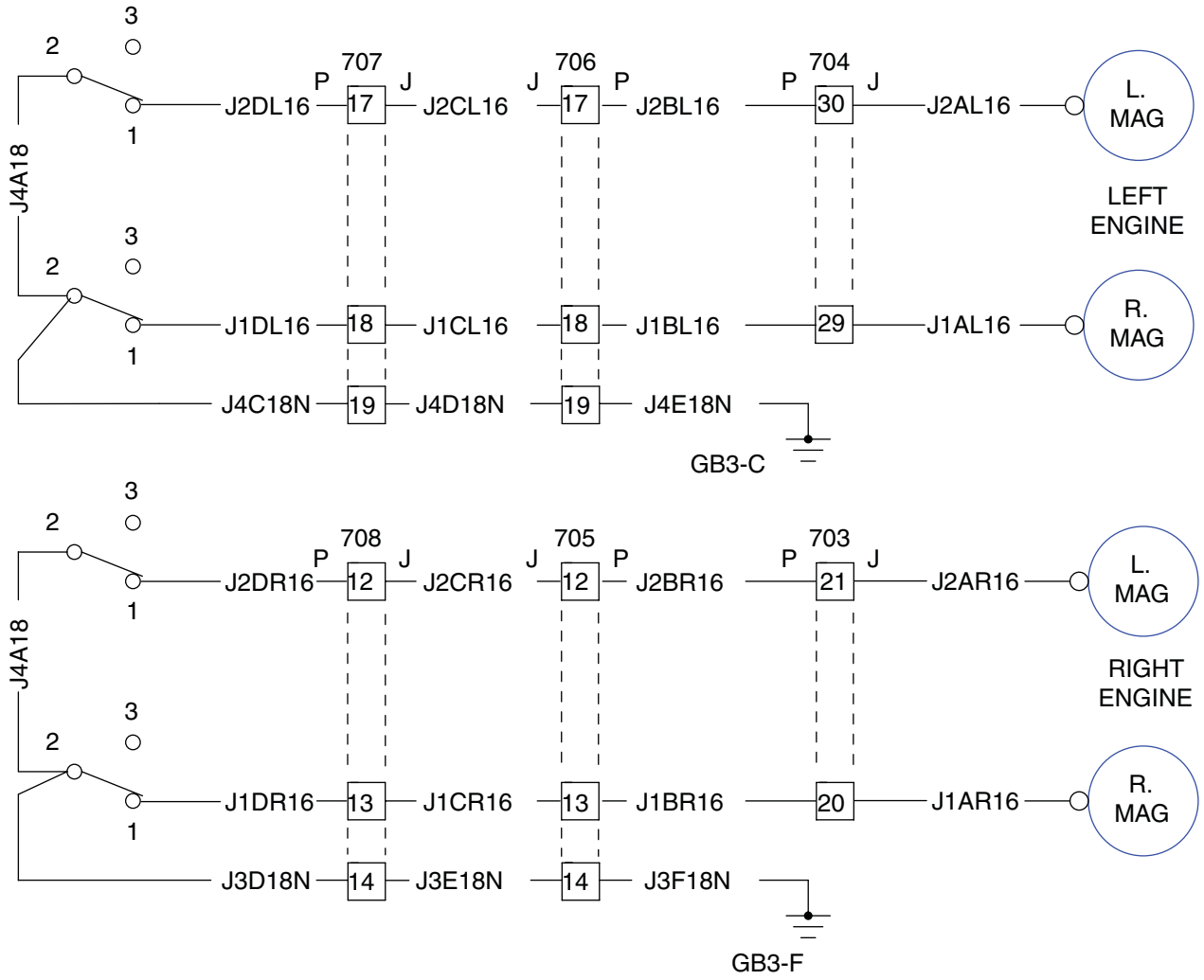
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104368 NEW/K 10.0



Magnetos
Figure 1 (Sheet 3 of 4)

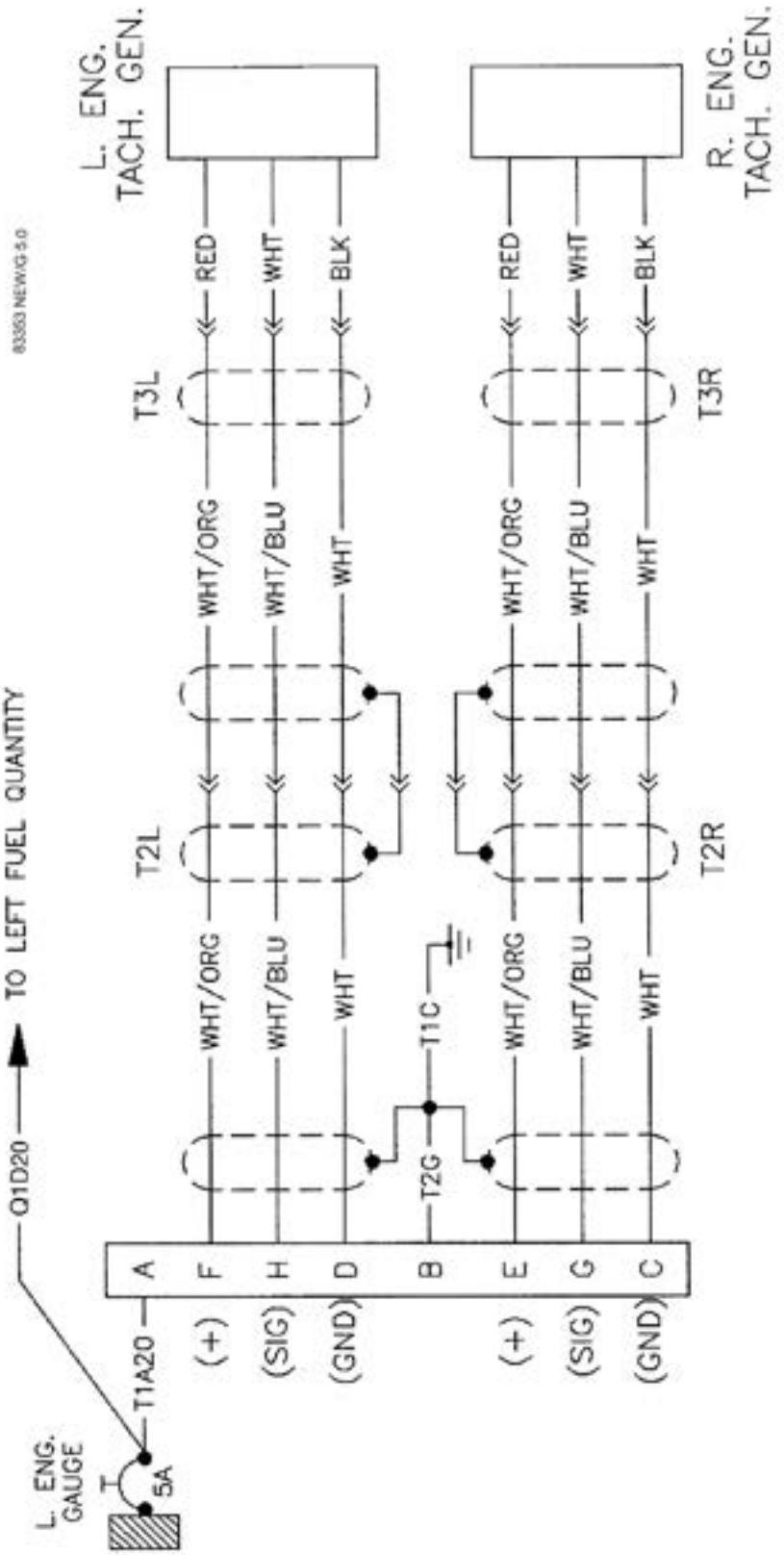
Effectivity
3449152 and up

101840 NEW 6.1



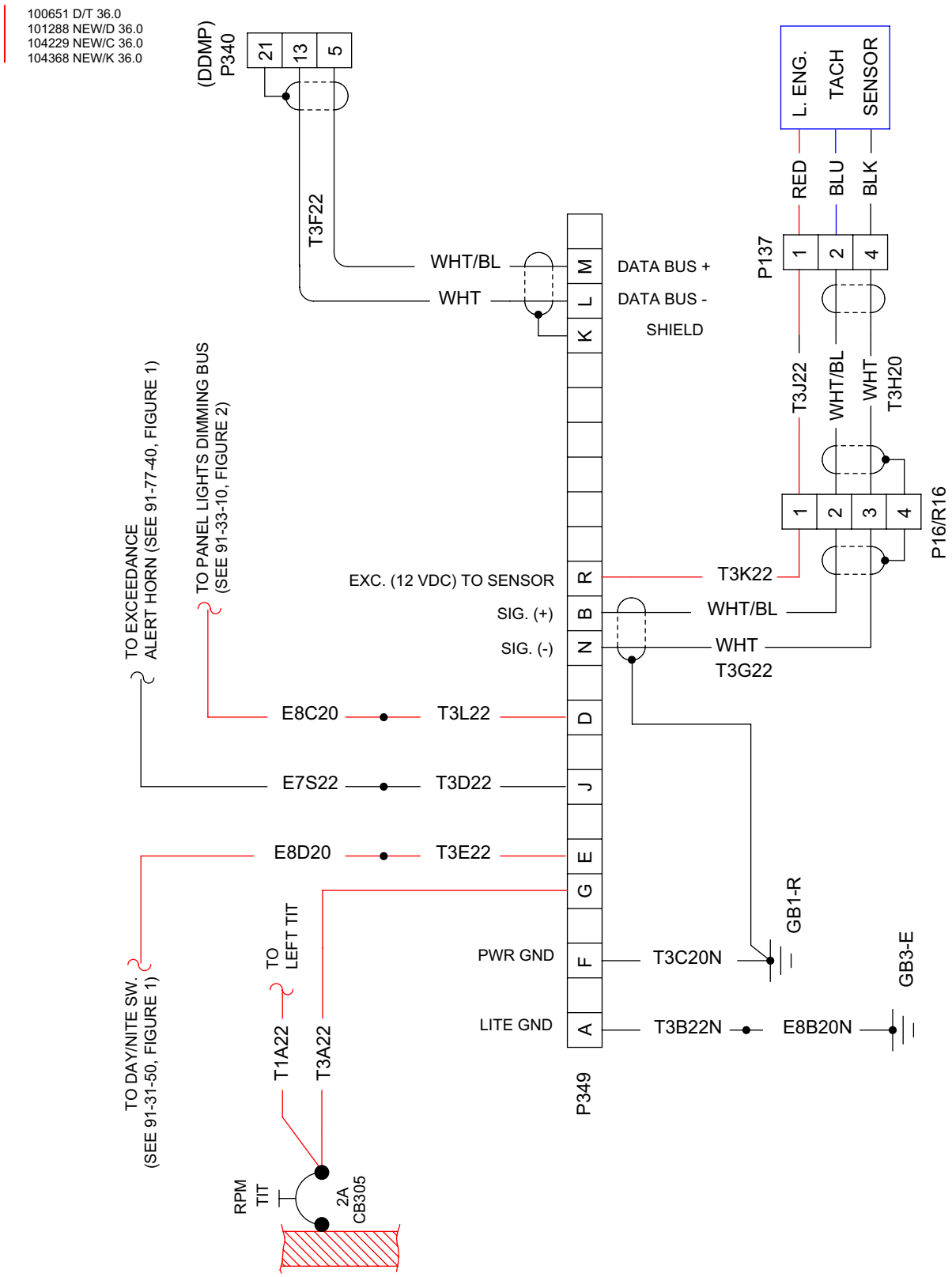
Effectivity
Seneca V
with Avidyne Entegra

Magnetos
Figure 1 (Sheet 4 of 4)



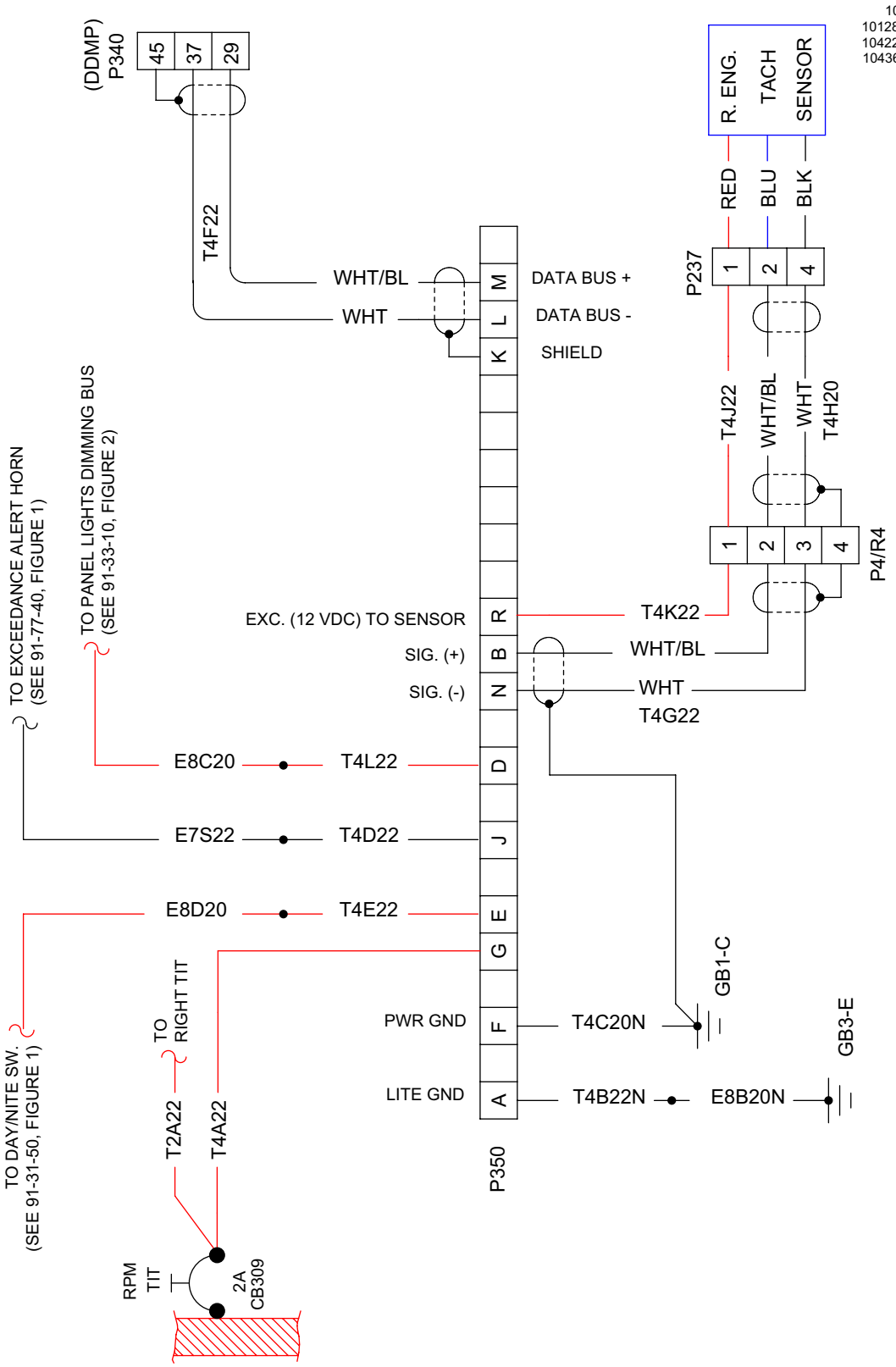
Engine Tachometer
Figure 1 (Sheet 1 of 3)

Effectivity
Seneca IV



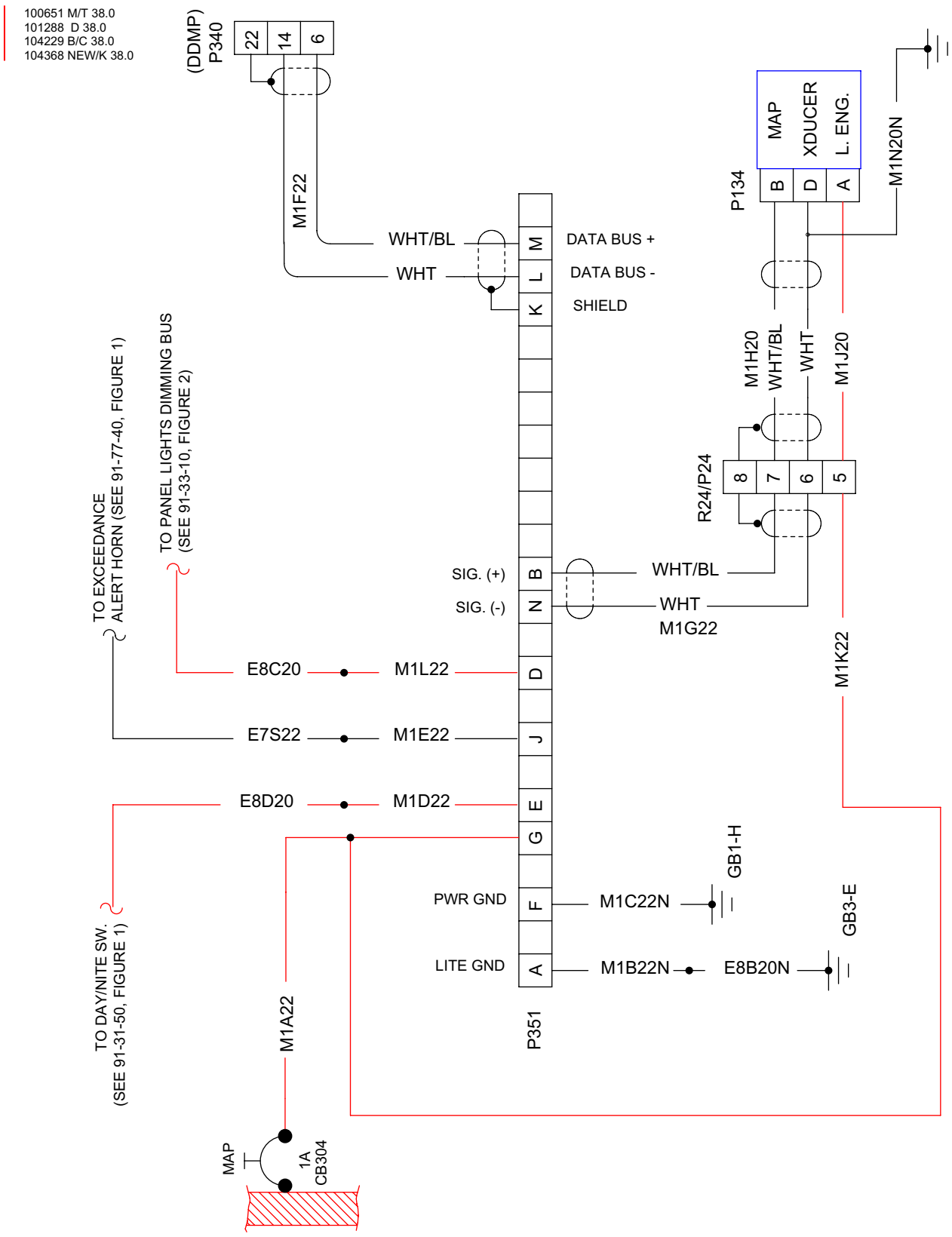
Effectivity
Seneca V - Left

Engine Tachometer
Figure 1 (Sheet 2 of 3)



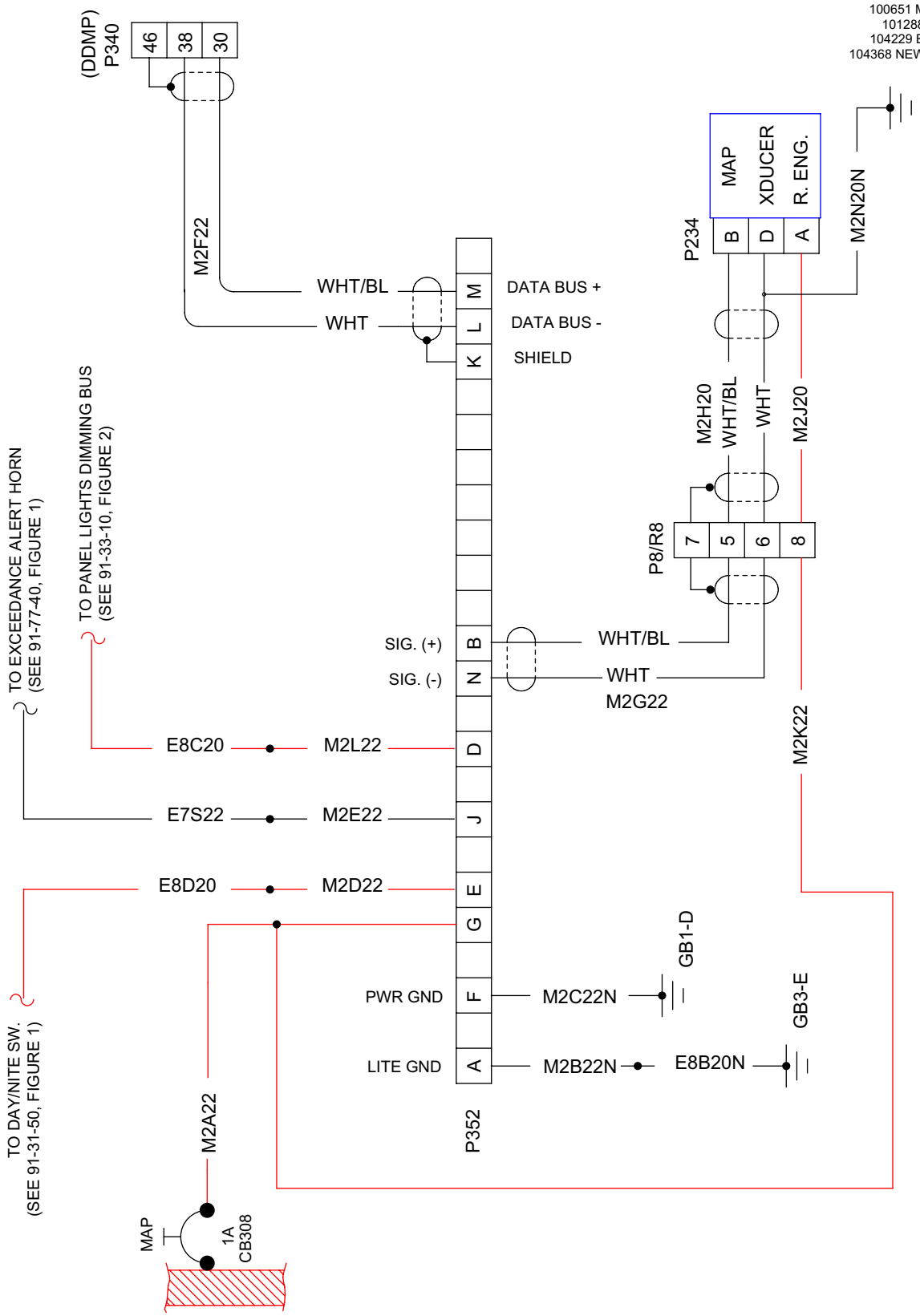
Engine Tachometer
Figure 1 (Sheet 3 of 3)

Effectivity
Right - Seneca V



Effectivity
Seneca V - Left

Engine Manifold Pressure
Figure 2 (Sheet 1 of 2)

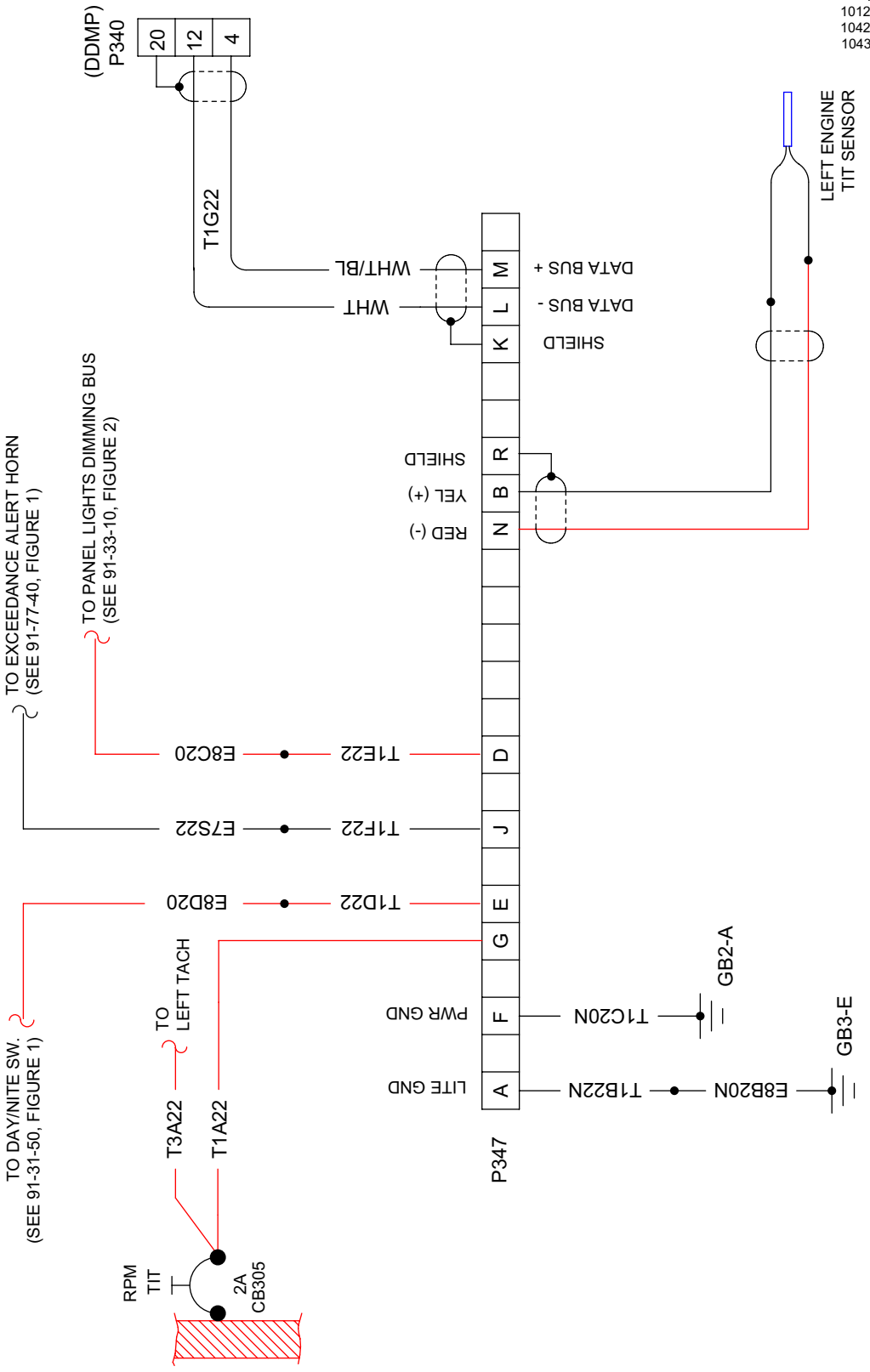


Engine Manifold Pressure
Figure 2 (Sheet 2 of 2)

Effectivity
Right - Seneca V

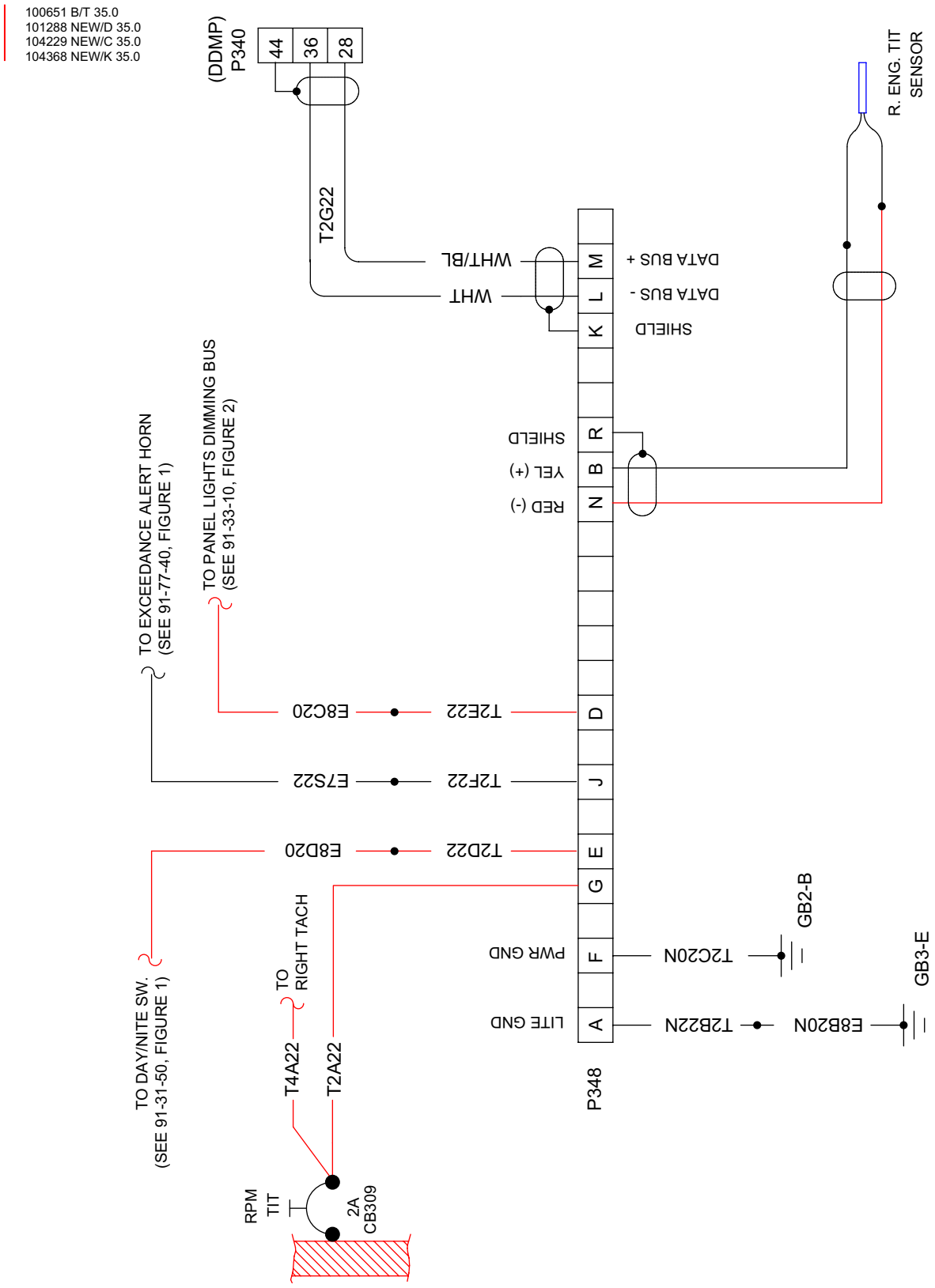
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100651 B/T 34.0
101288 NEW/D 34.0
104229 NEW/C 34.0
104368 NEW/K 34.0



Turbine Inlet Temperature
Figure 1 (Sheet 1 of 2)

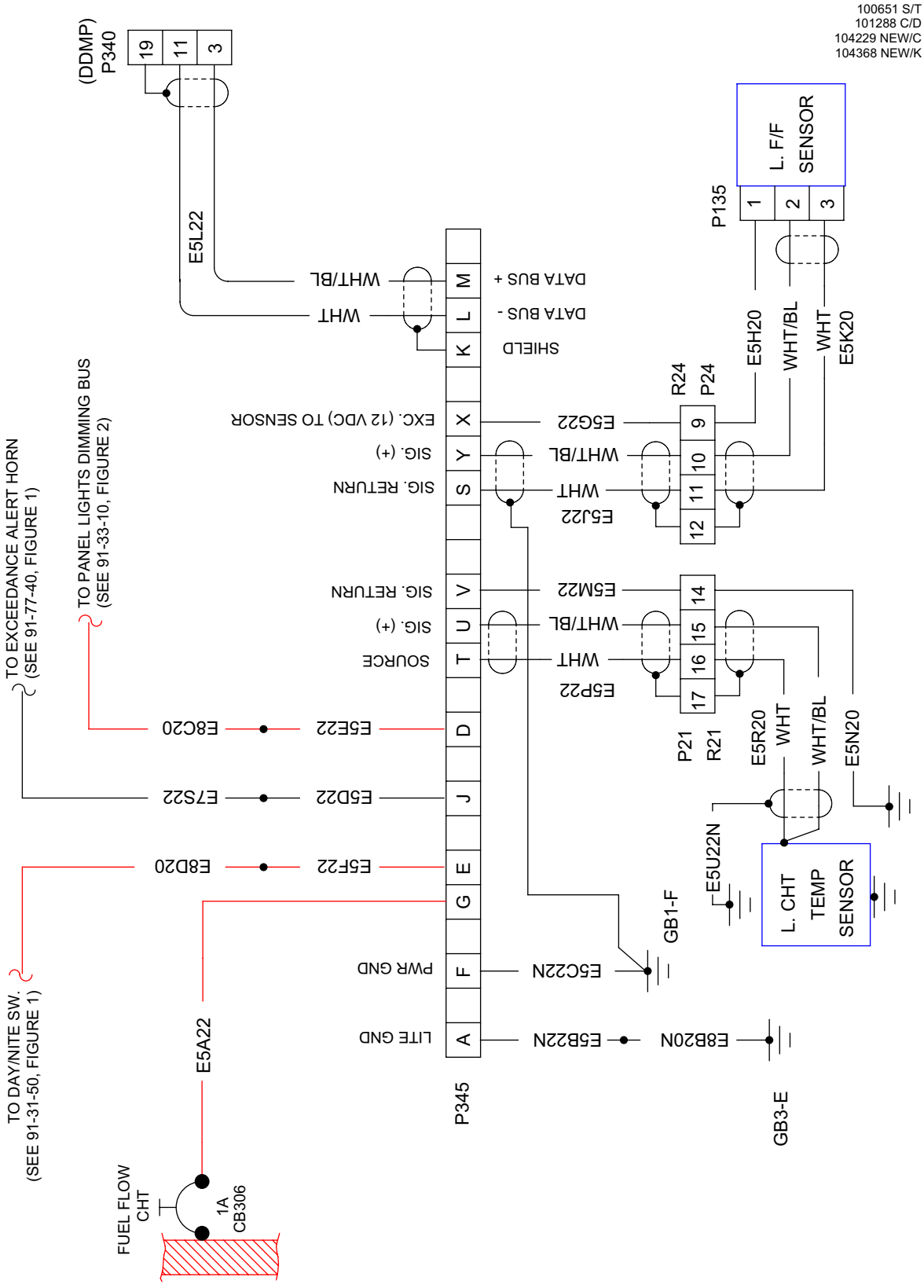
Effectivity
Seneca V - Left



Effectivity
Right - Seneca V

Turbine Inlet Temperature
Figure 1 (Sheet 2 of 2)

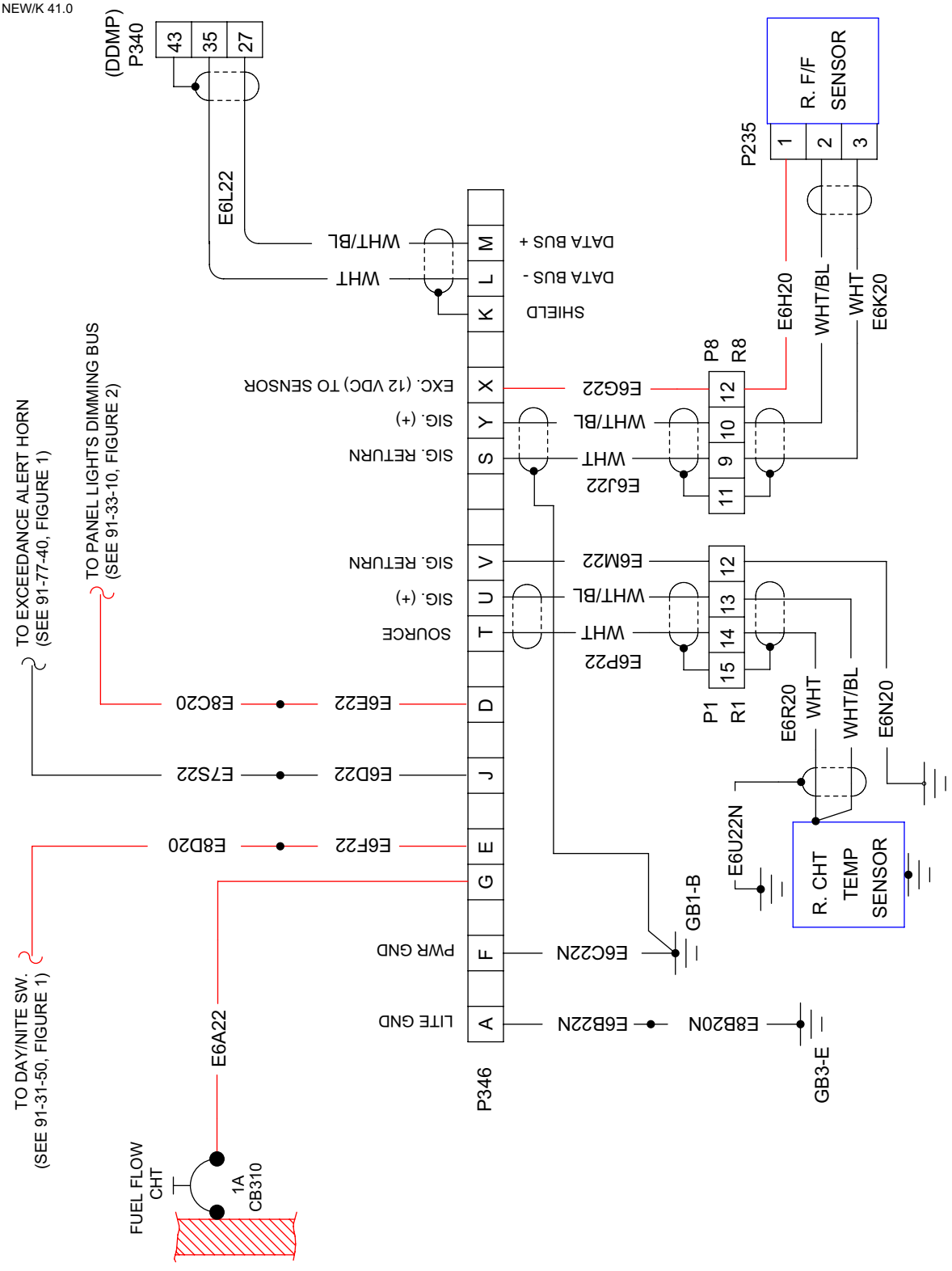
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101288 C/D 40.0
104229 NEW/C 40.0
104368 NEW/K 40.0



Engine Fuel Flow / Cylinder Head Temperature
Figure 2 (Sheet 1 of 2)

Effectivity
Seneca V - Left

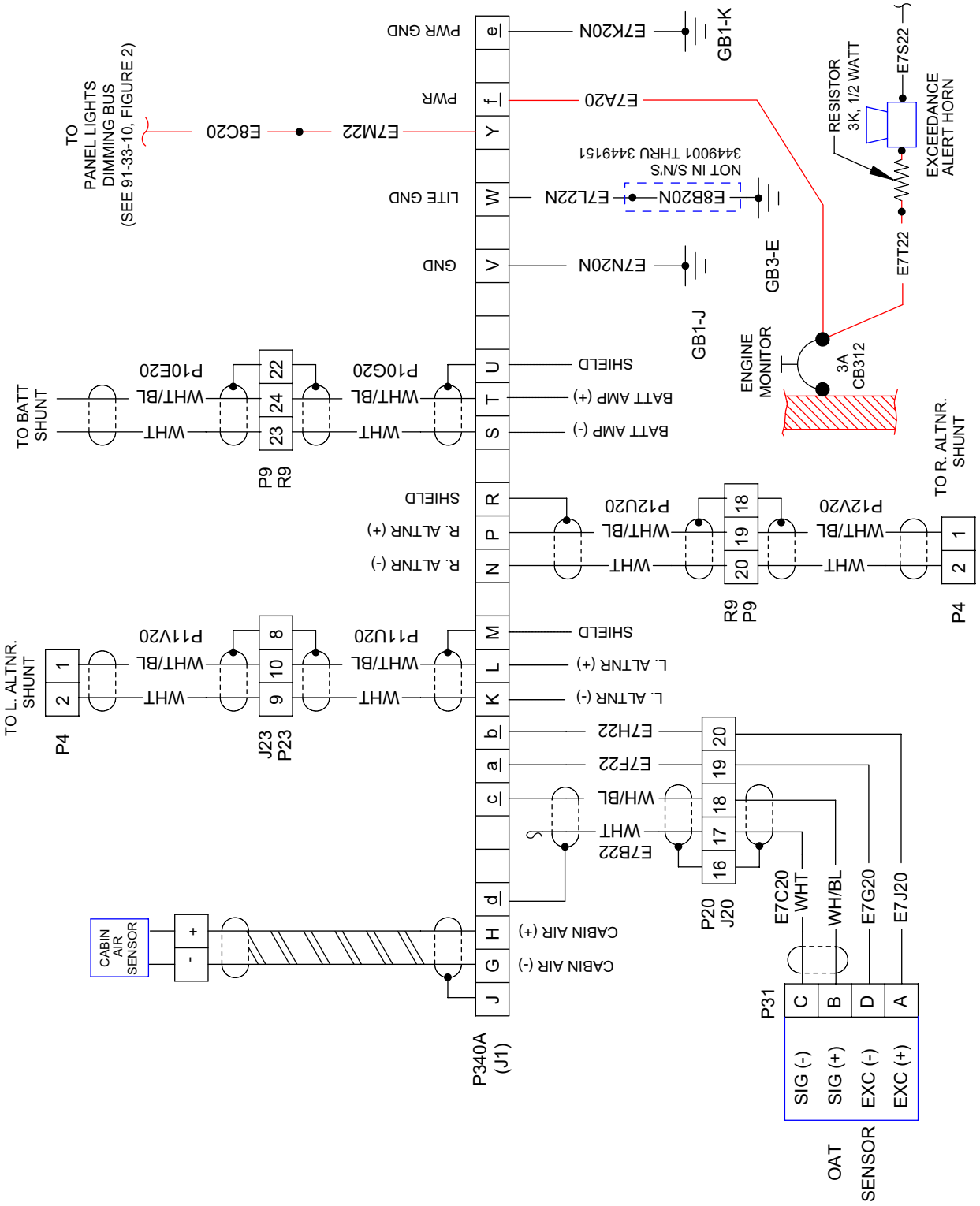
100651 S/T 41.0
101288 C/D 41.0
104229 NEW/C 41.0
104368 NEW/K 41.0



Effectivity
Right - Seneca V

Engine Fuel Flow / Cylinder Head Temperature
Figure 2 (Sheet 2 of 2)

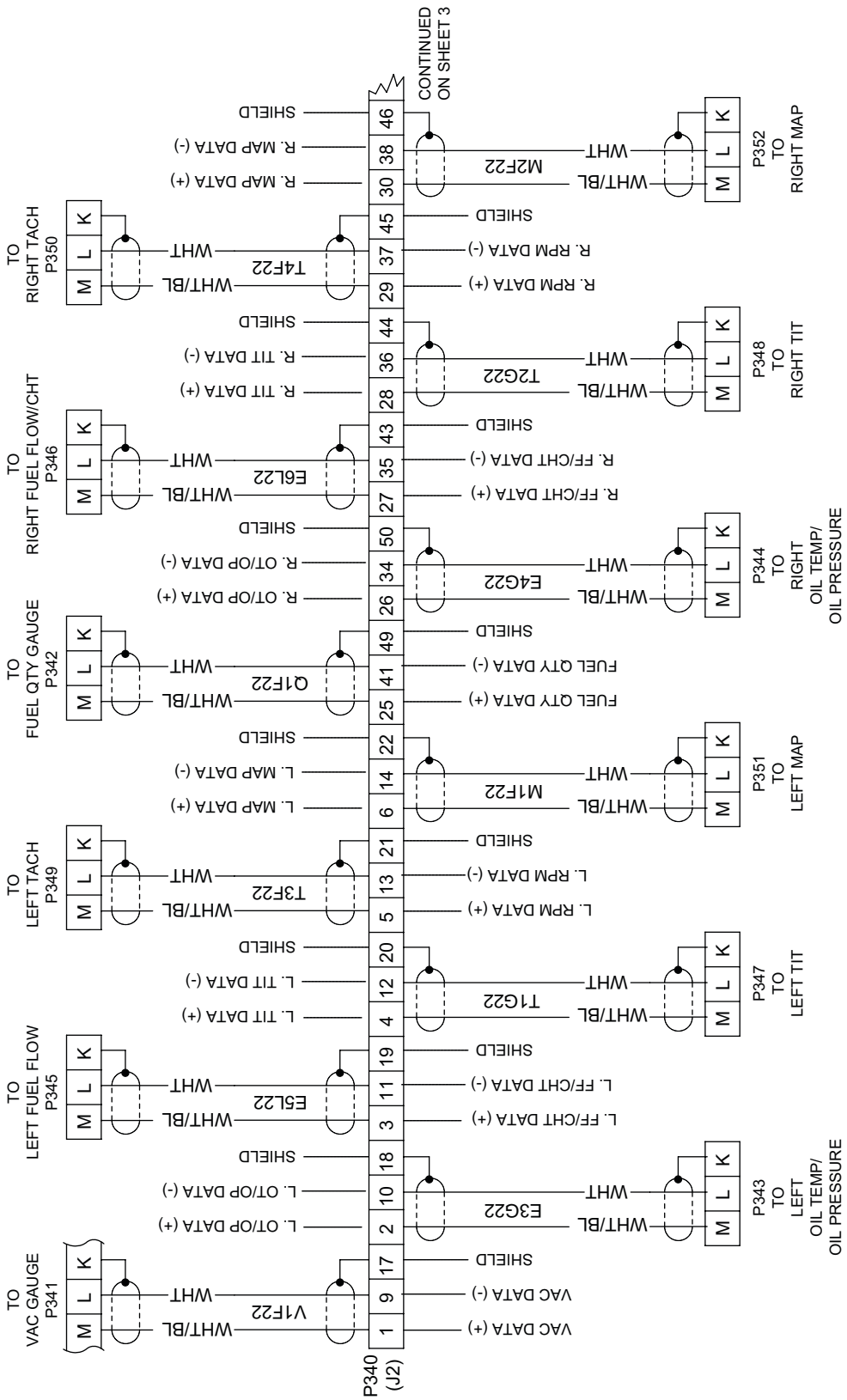
100651 G/T 6.0
101288 B/D 6.0
104229 NEW/C 6.0
104368 NEW/K 6.0



Engine Digital Display Monitoring Panel (DDMP)
Figure 1 (Sheet 1 of 3)

Effectivity
Seneca V

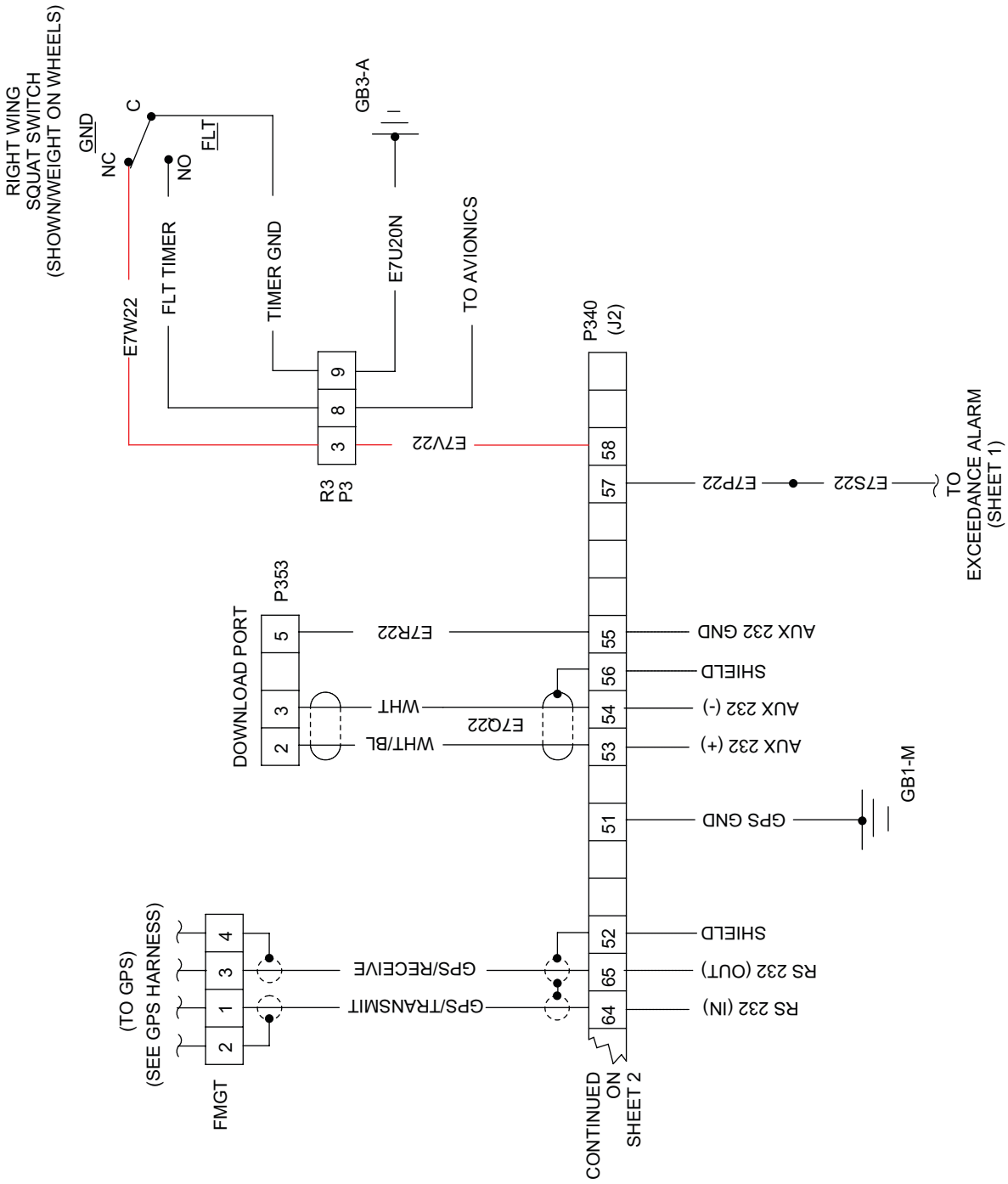
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101288 NEW/D 6.1
104229 NEW/C 6.1
104368 NEW/K 6.1



Effectivity
Seneca V

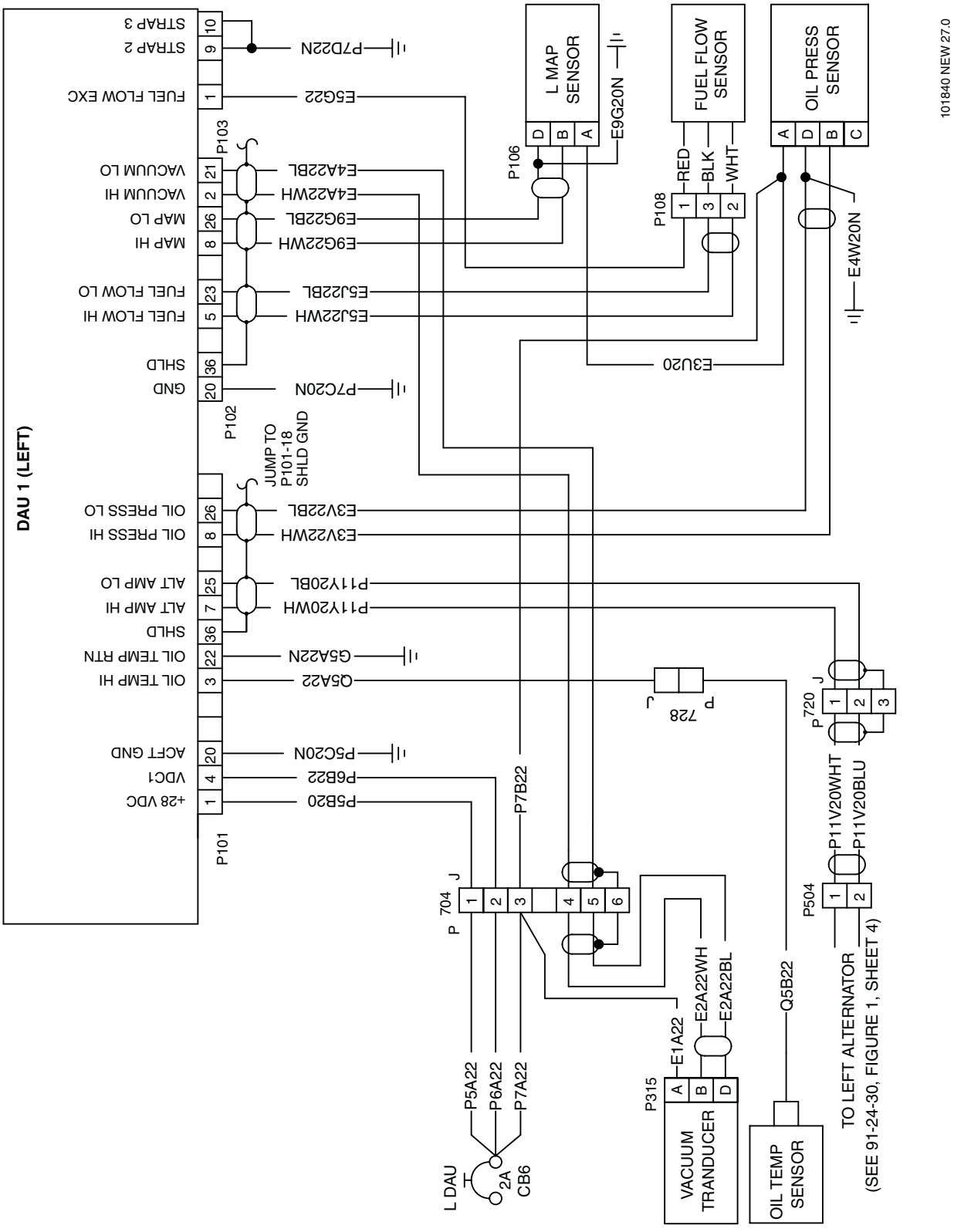
Engine Digital Display Monitoring Panel (DDMP)
Figure 1 (Sheet 2 of 3)

100651 D/T 6.2
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104229 NEW/C 6.2
104368 NEW/K 6.2



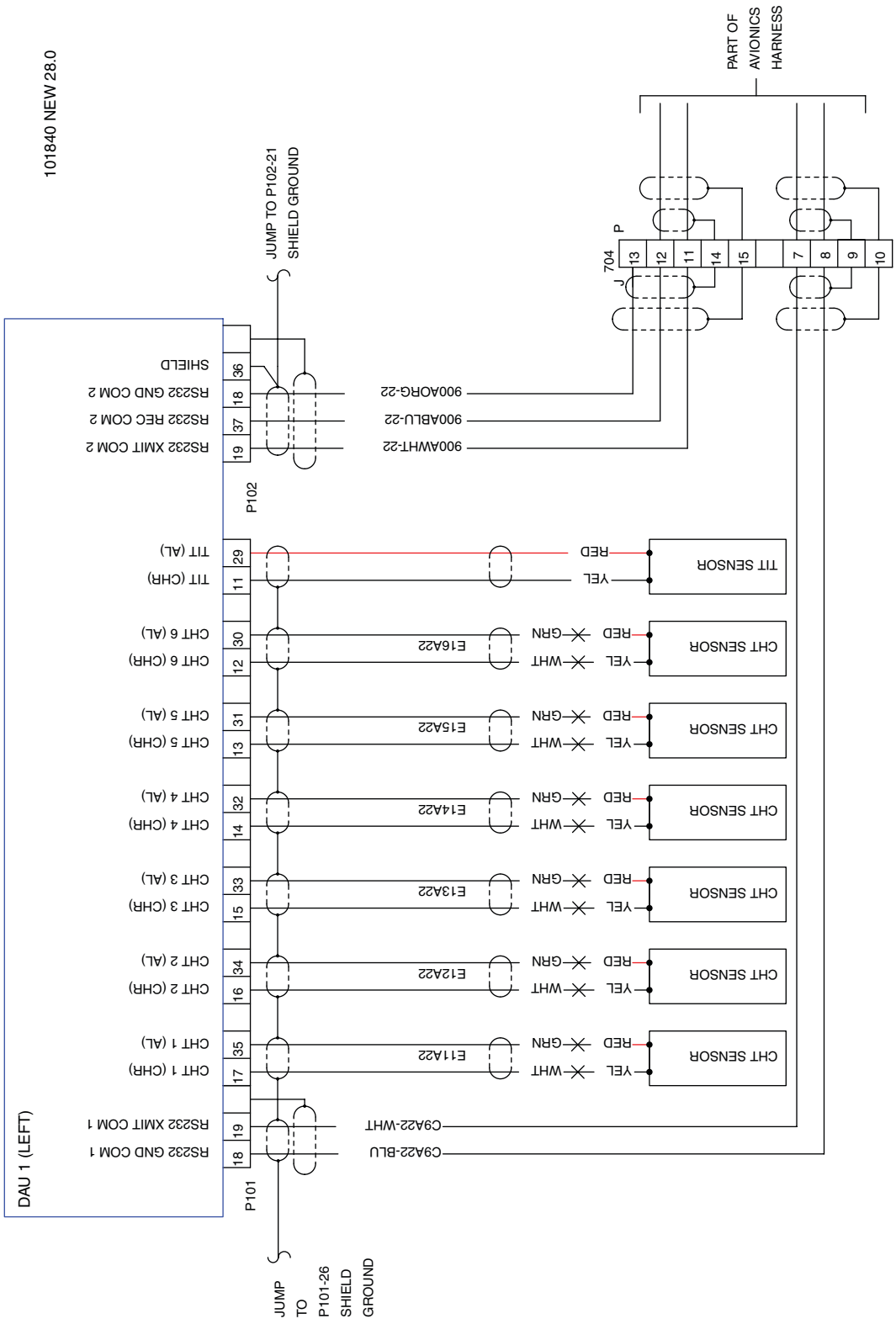
Engine Digital Display Monitoring Panel (DDMP)
Figure 1 (Sheet 3 of 3)

Effectivity
Seneca V



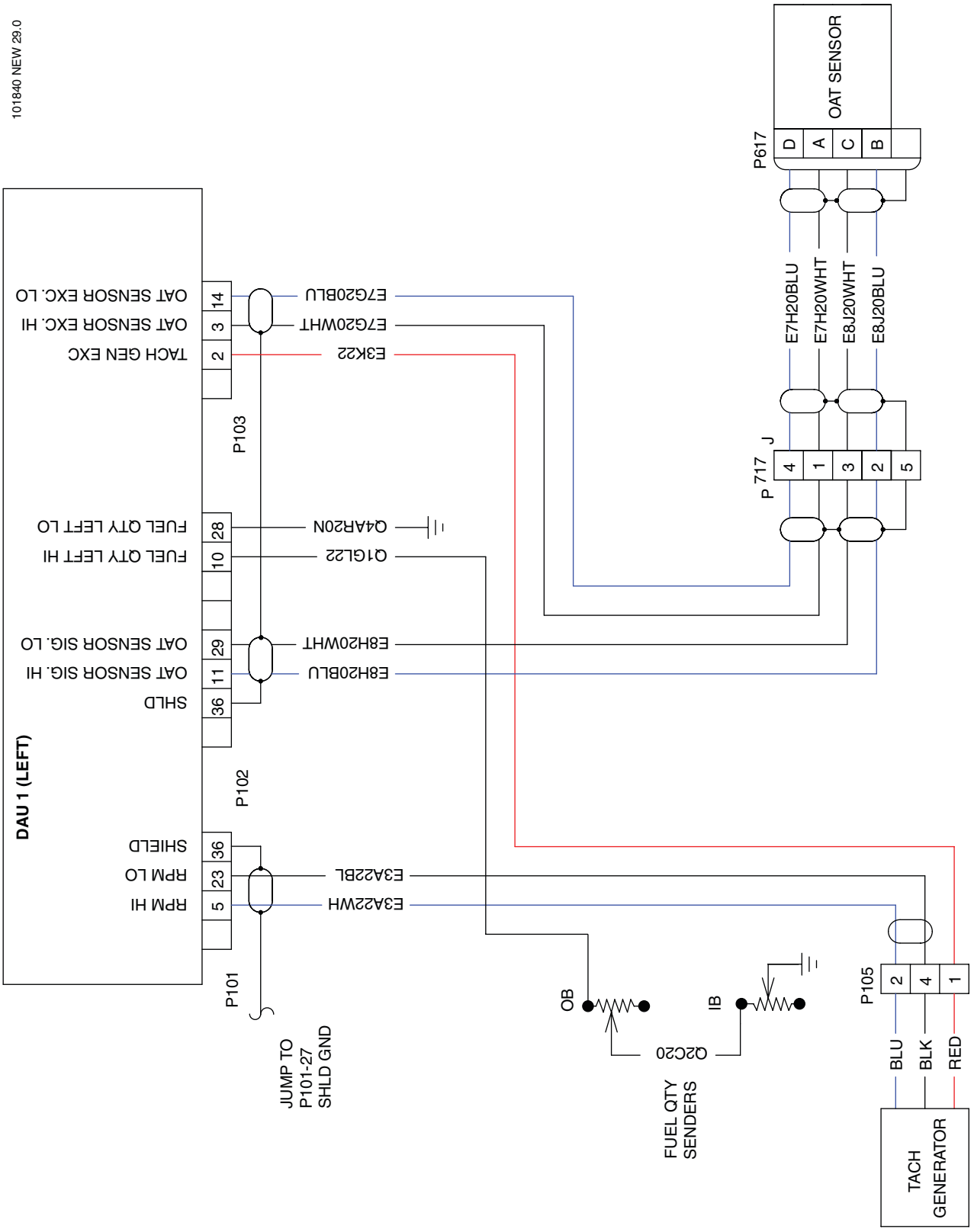
Effectivity
Seneca V
with Avidyne Entegra

Data Acquisition Units (DAU's)
Figure 2 (Sheet 1 of 6)



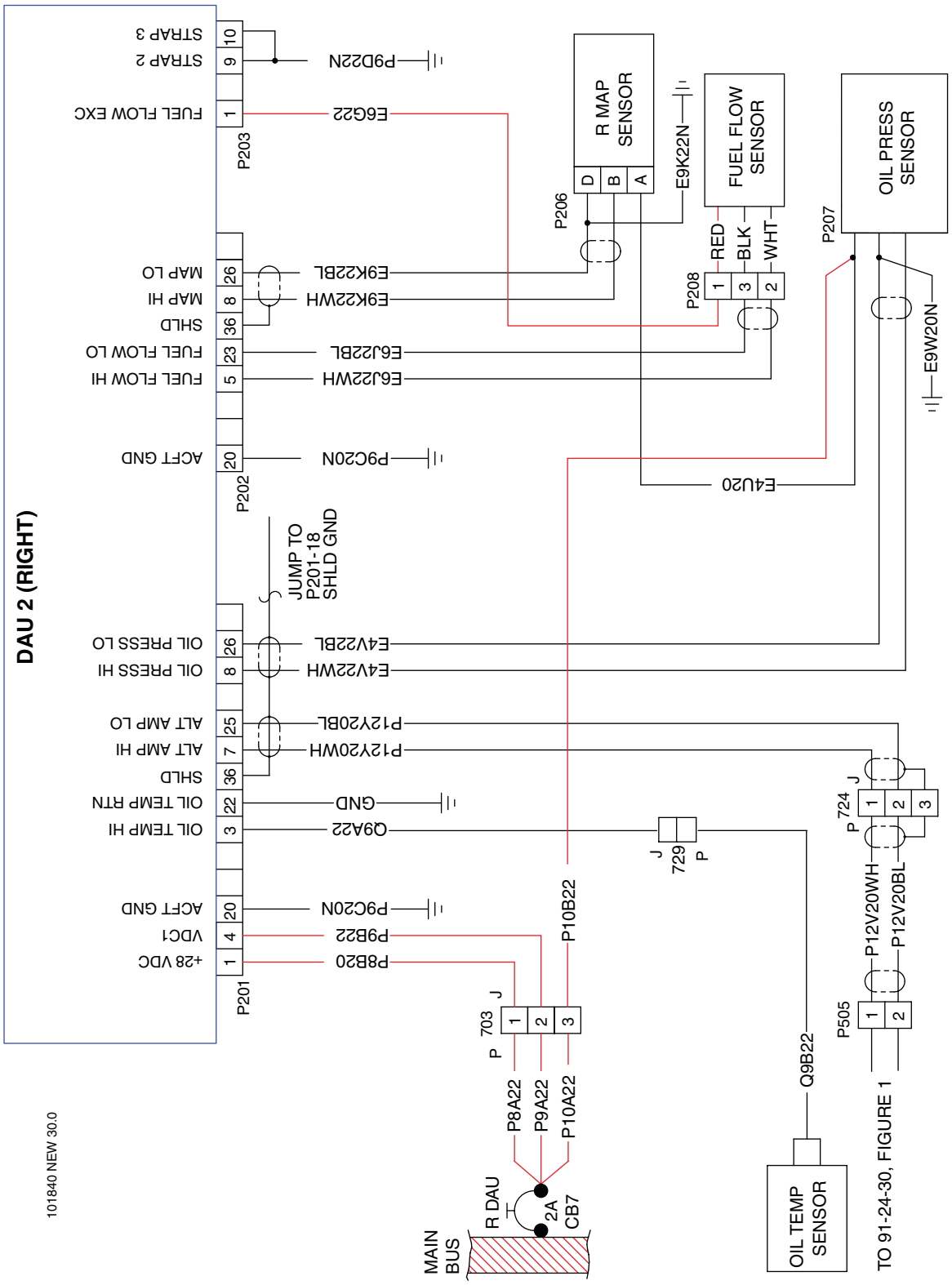
Data Acquisition Units (DAU's)
Figure 2 (Sheet 2 of 6)

Effectivity
Seneca V
with Avidyne Entegra



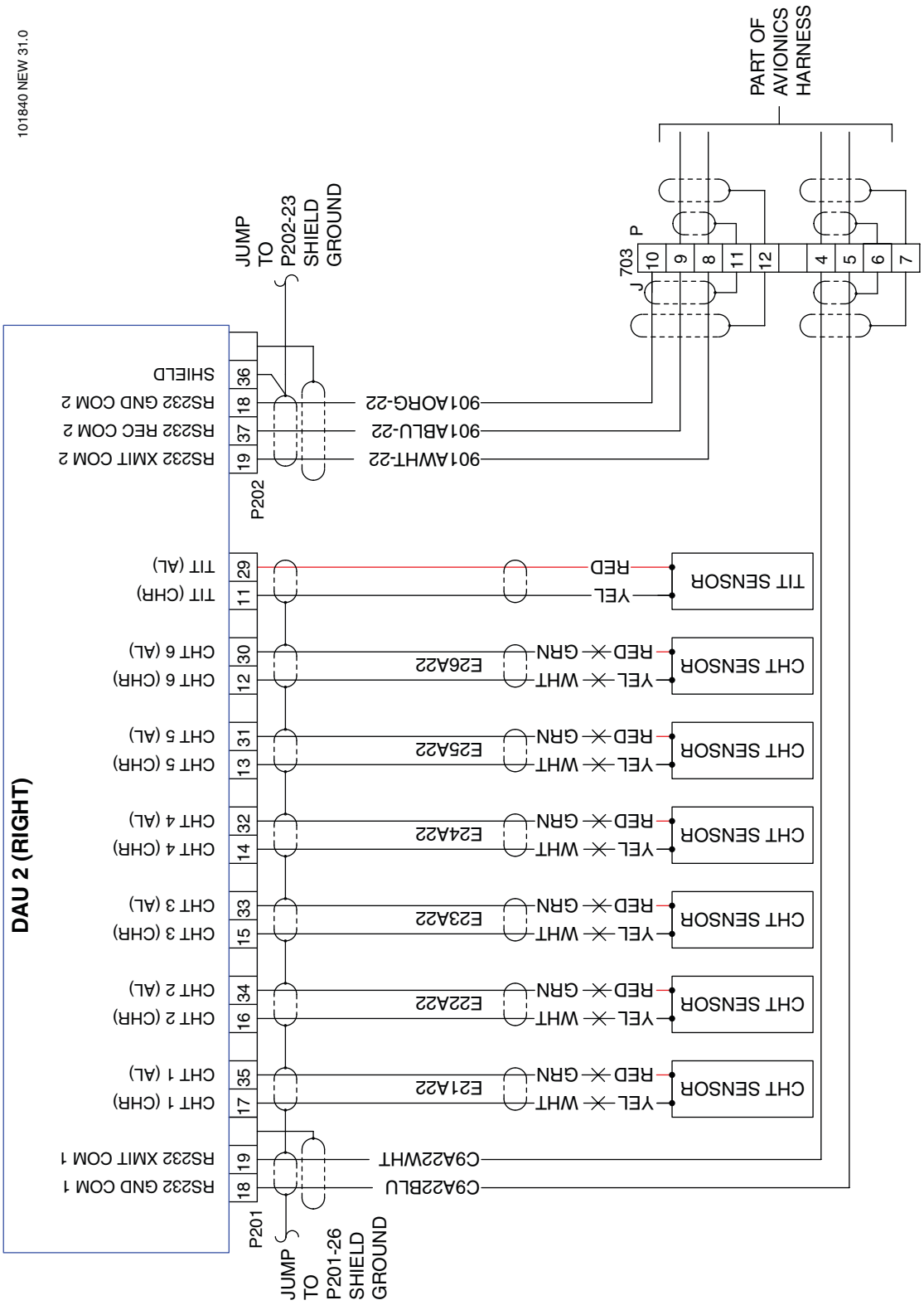
Effectivity
Seneca V
with Avidyne Entegra

Data Acquisition Units (DAU's)
Figure 2 (Sheet 3 of 6)



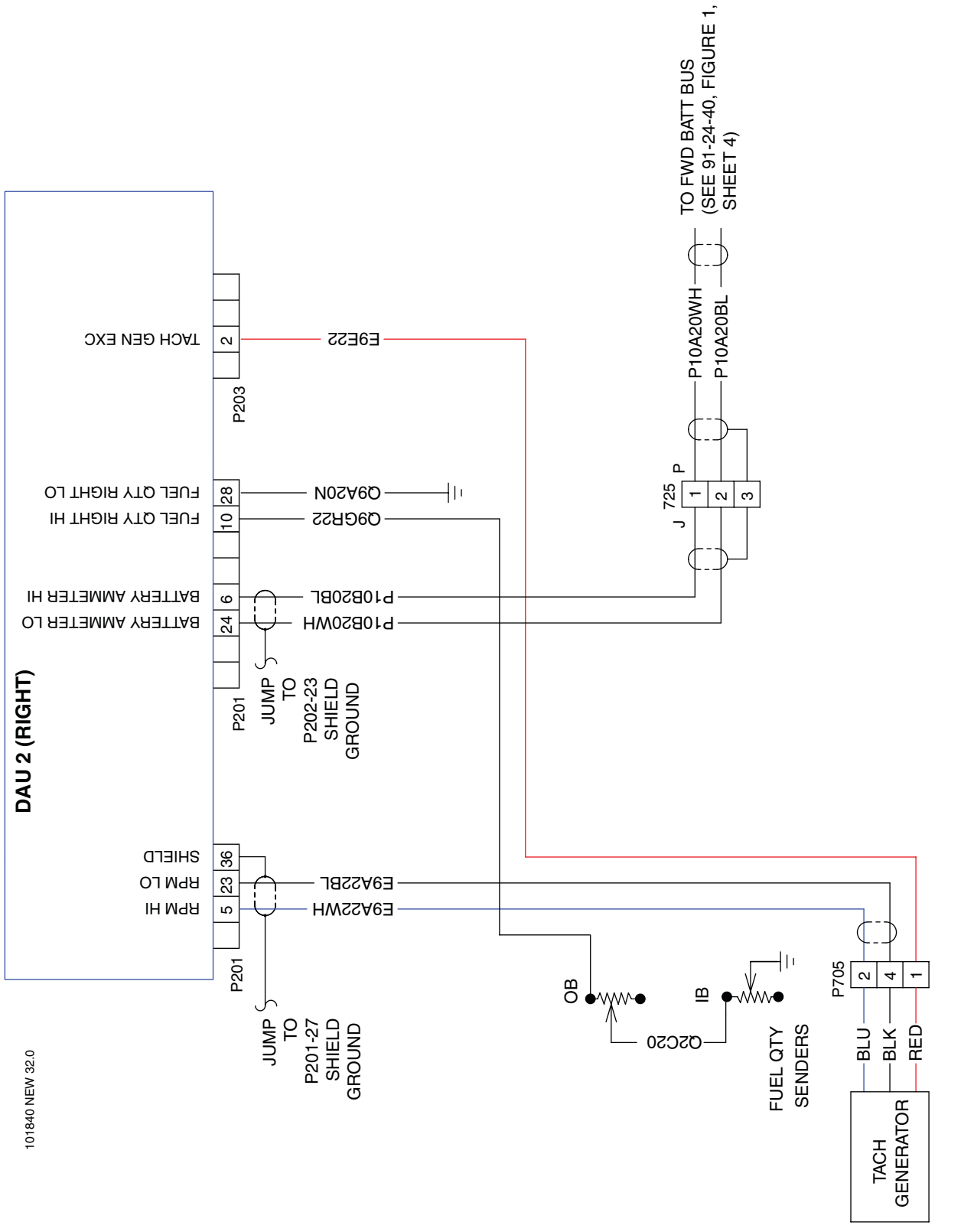
Data Acquisition Units (DAU's)
Figure 2 (Sheet 4 of 6)

Effectivity
Seneca V
with Avidyne Entegra



Effectivity
Seneca V
with Avidyne Entegra

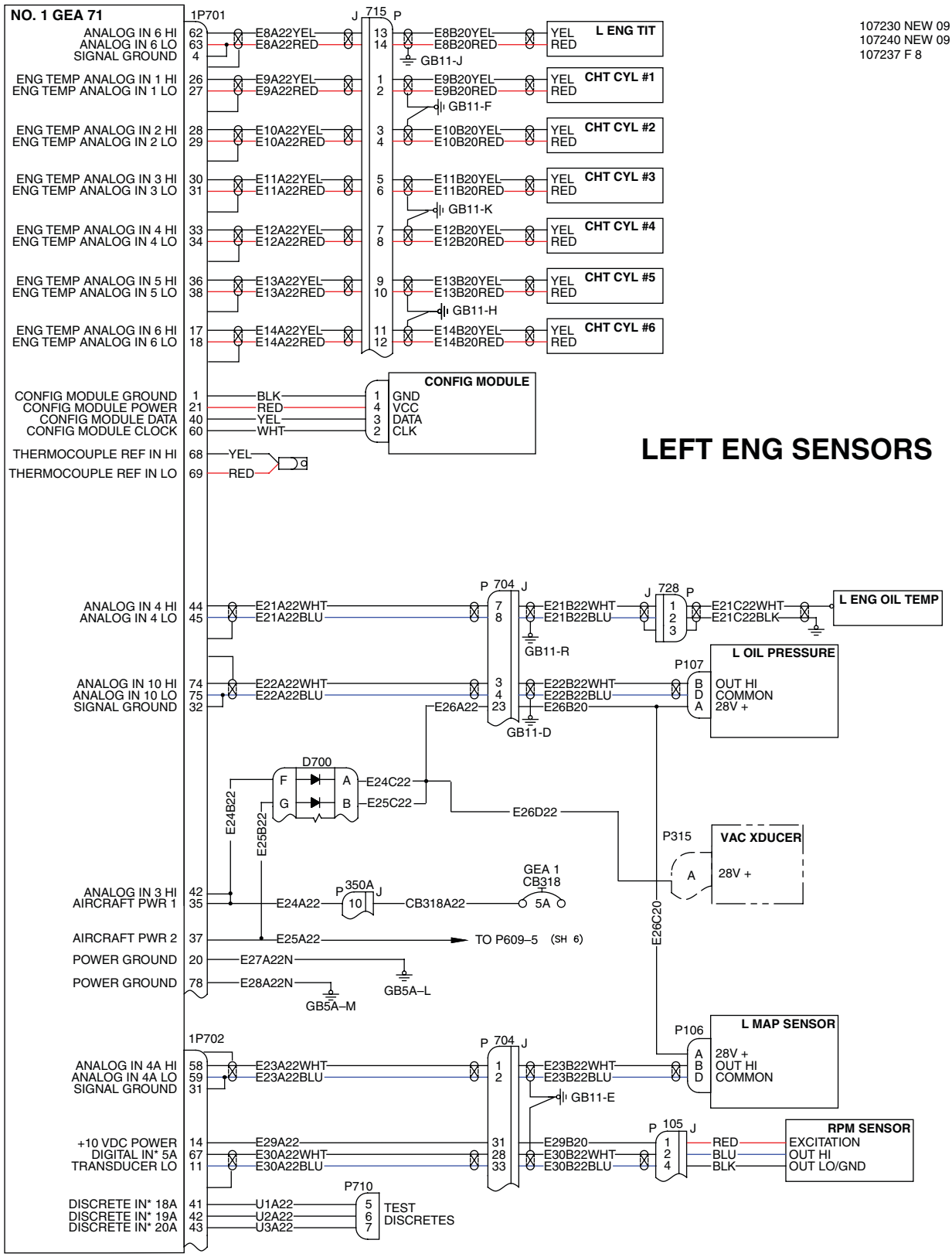
Data Acquisition Units (DAU's)
Figure 2 (Sheet 5 of 6)



Data Acquisition Units (DAU's)
Figure 2 (Sheet 6 of 6)

Effectivity
Seneca V
with Avidyne Entegra

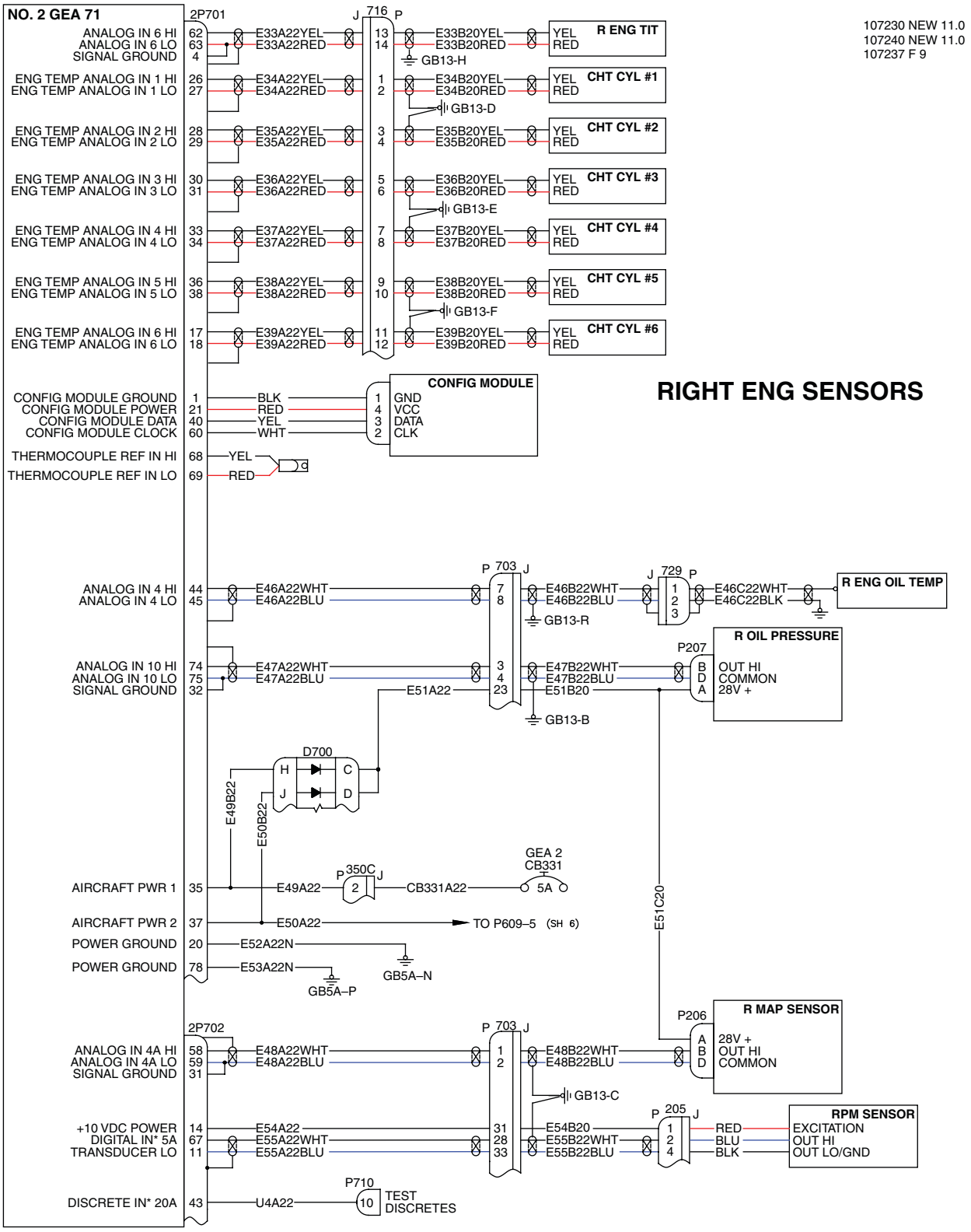
107230 NEW 09
107240 NEW 09
107237 F 8



Effectivity
Seneca V
with Garmin G1000

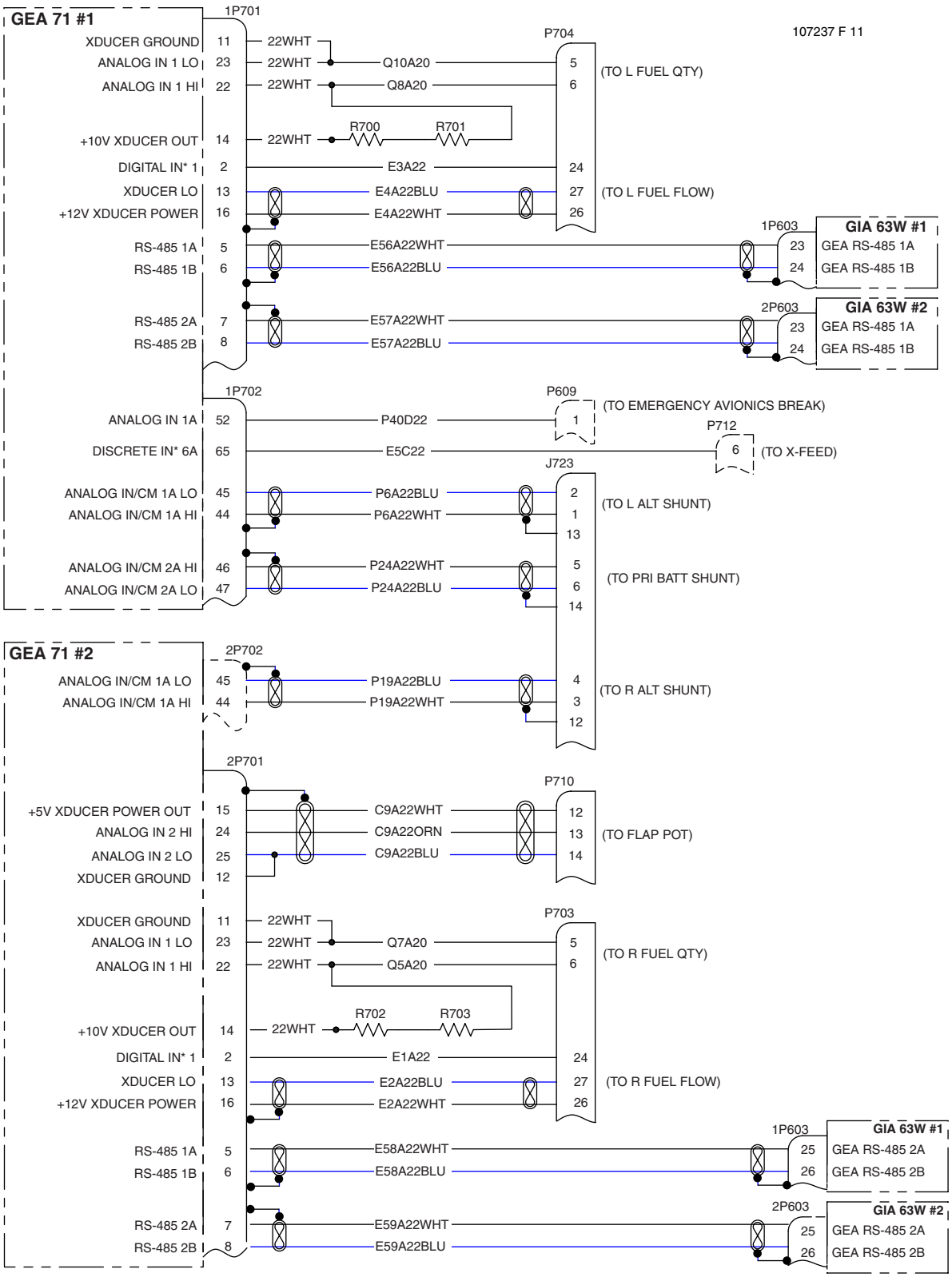
Engine Sensors
Figure 3 (Sheet 1 of 8)

107230 NEW 11.0
107240 NEW 11.0
107237 F 9



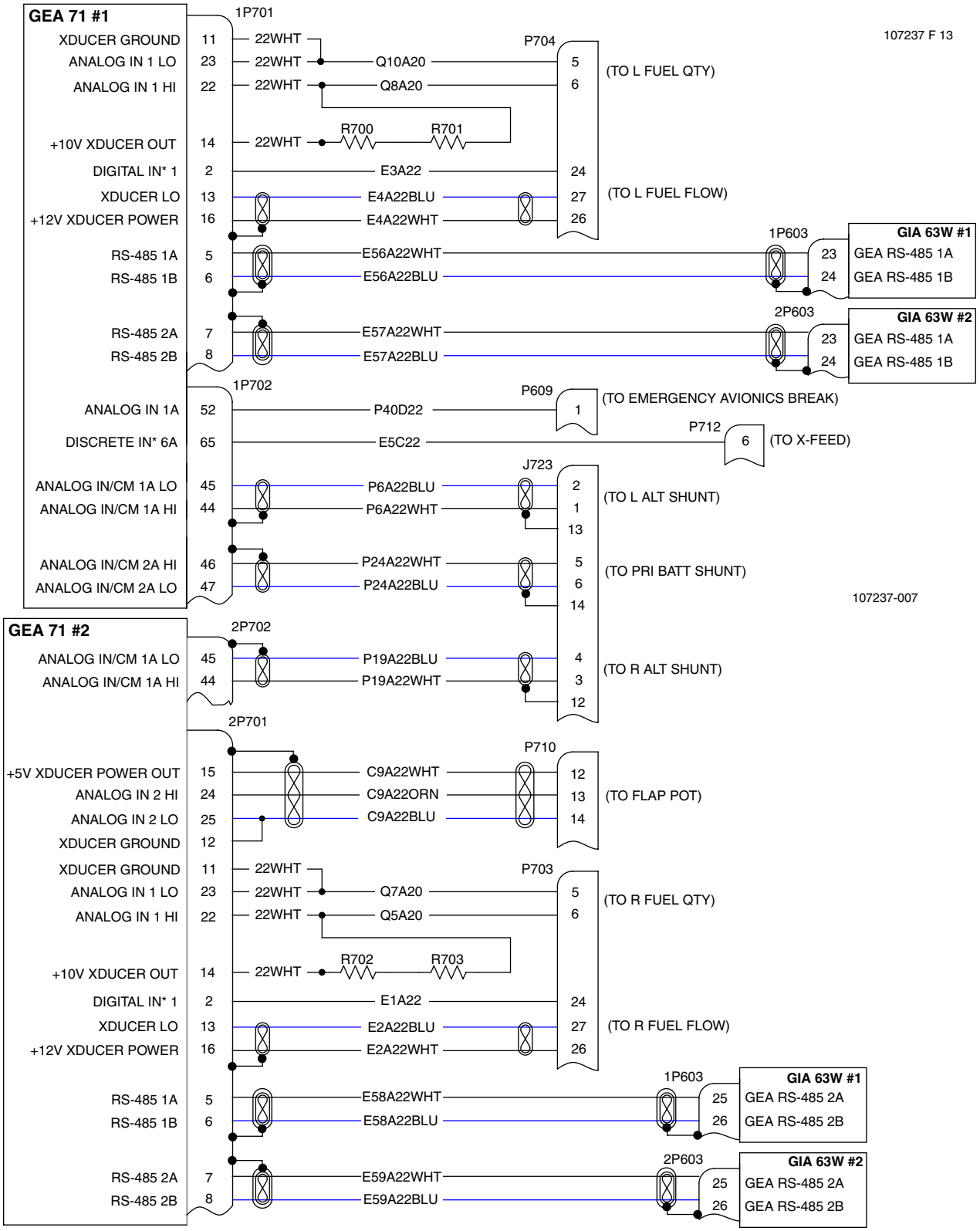
Engine Sensors
Figure 3 (Sheet 2 of 8)

Effectivity
Seneca V
with Garmin G1000



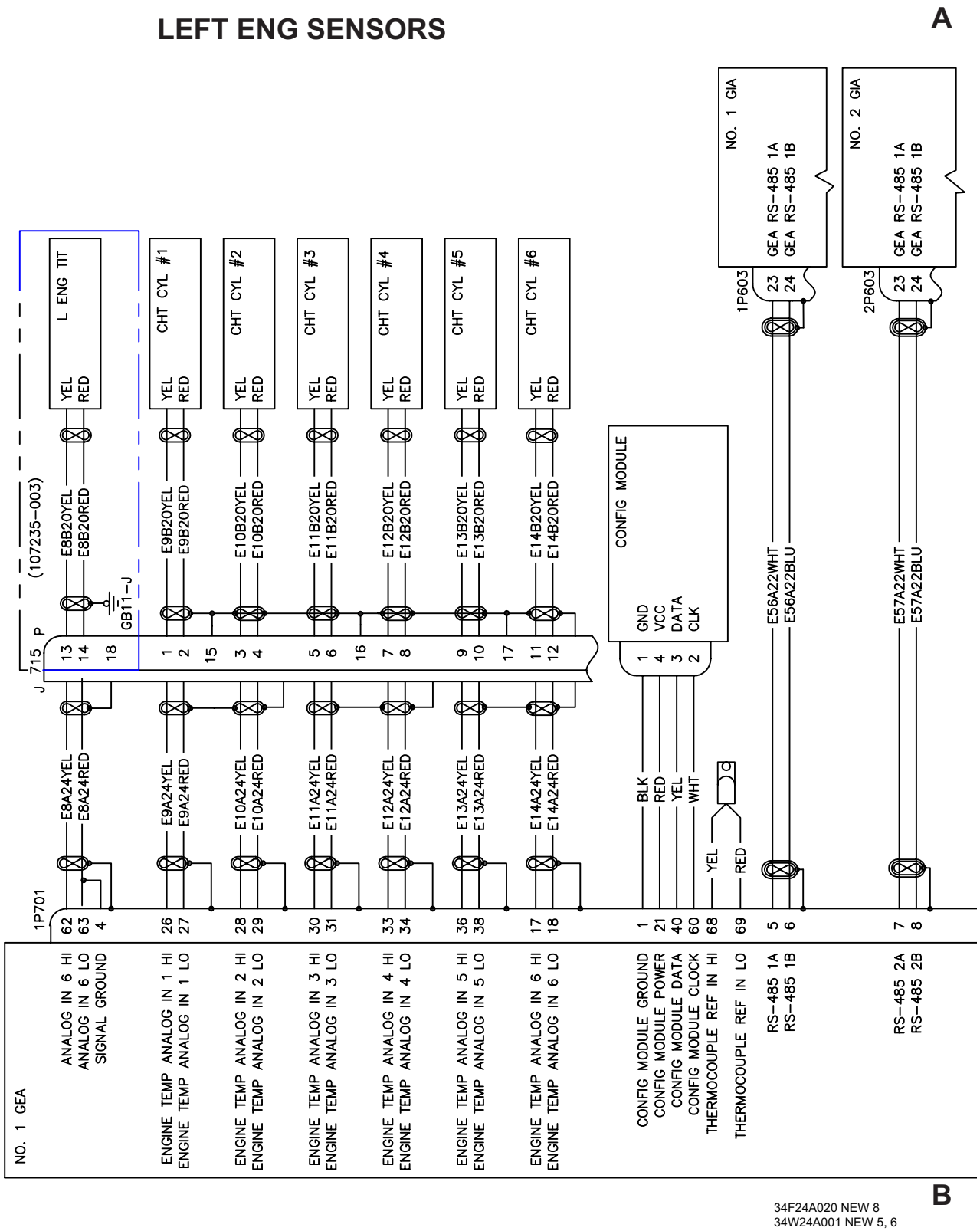
Effectivity
Seneca V
with Garmin G1000

Engine Sensors
Figure 3 (Sheet 3 of 8)



Engine Sensors
Figure 3 (Sheet 4 of 8)

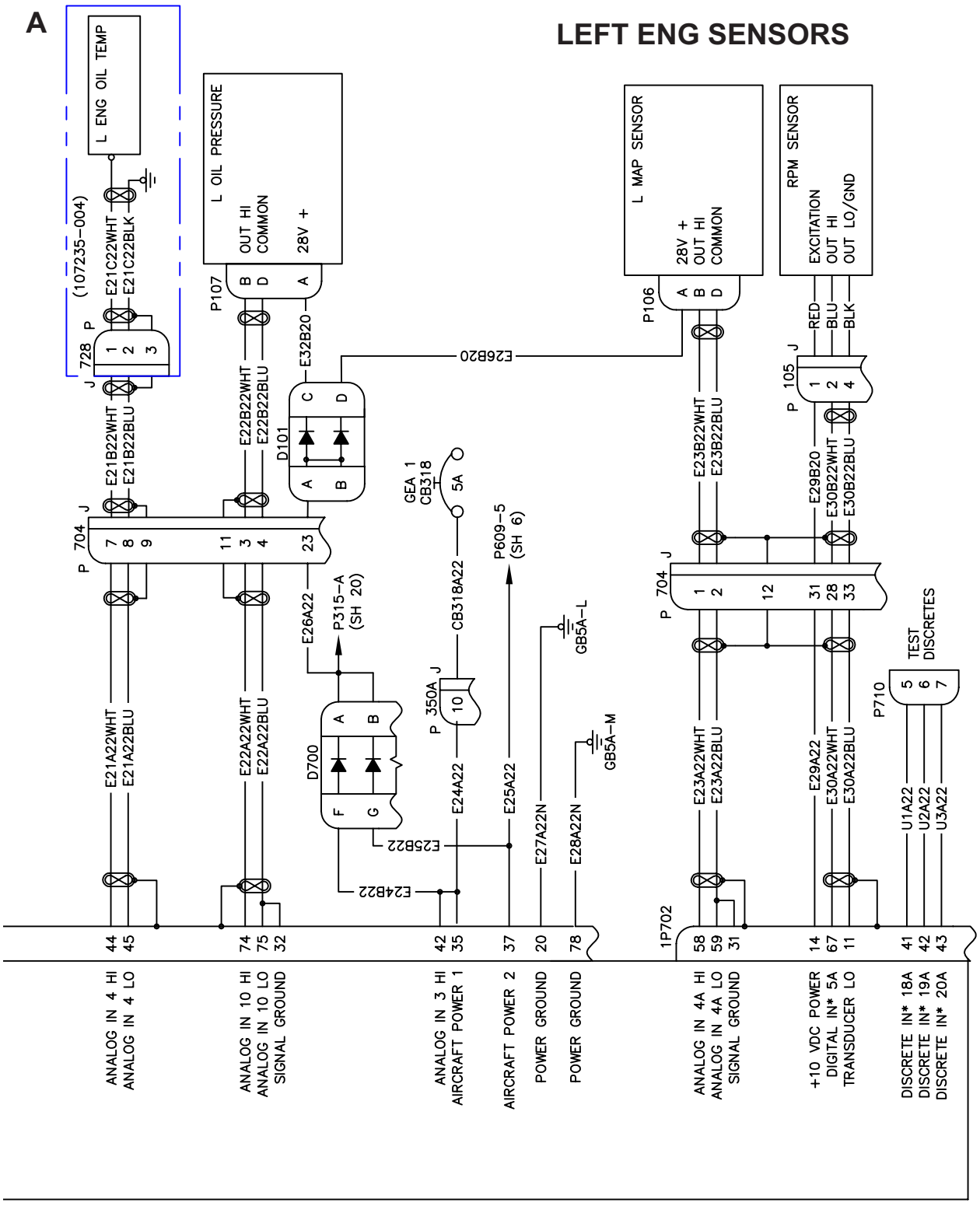
Effectivity
Seneca V
with Garmin GFC 700



34F24A020 NEW 8
34W24A001 NEW 5, 6

Effectivity
Seneca V
with Garmin G1000 NXi

Engine Sensors
Figure 3 (Sheet 5 of 8)

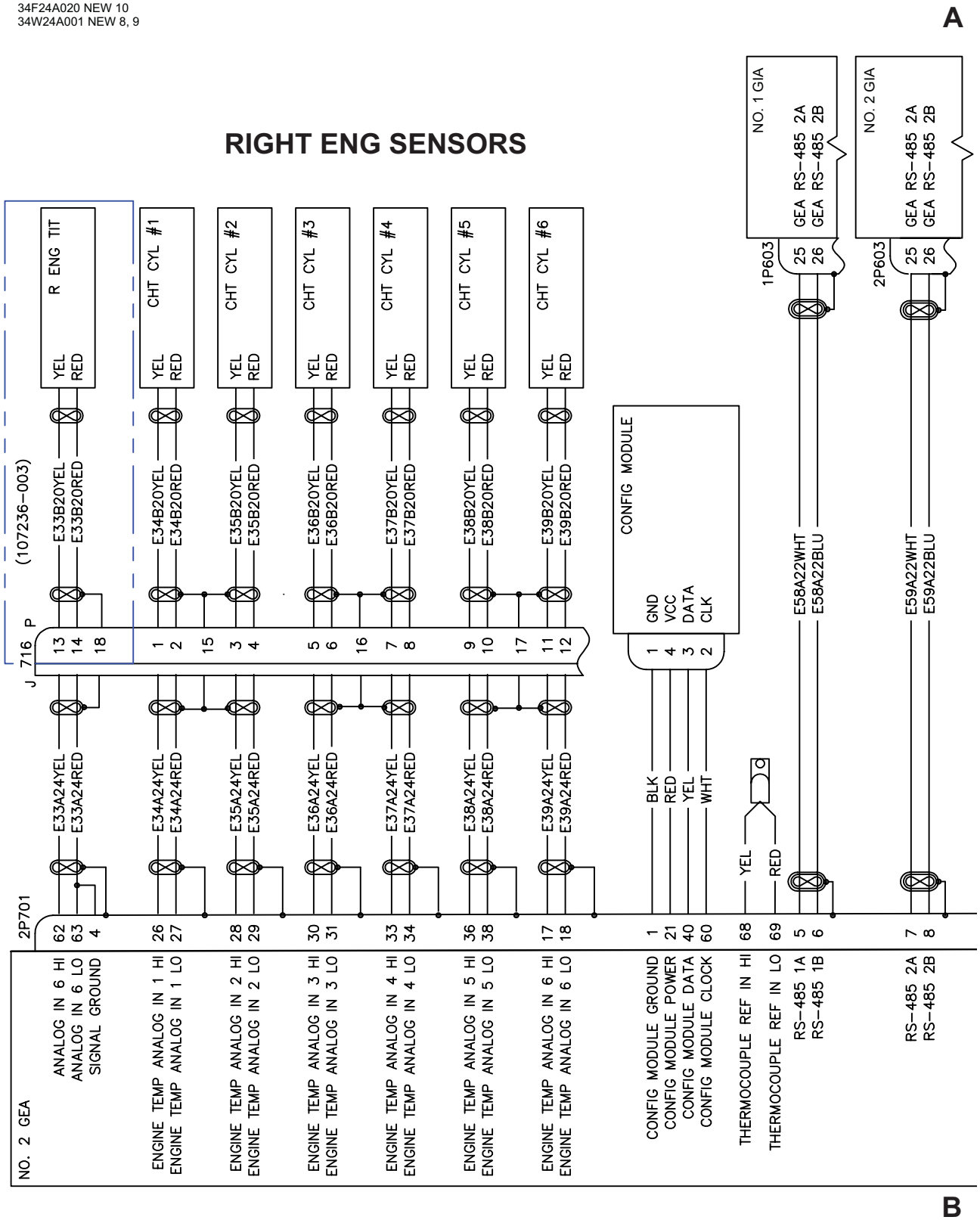


34F24A020 NEW 8
34W24A001 NEW 5, 6

Engine Sensors
Figure 3 (Sheet 6 of 8)

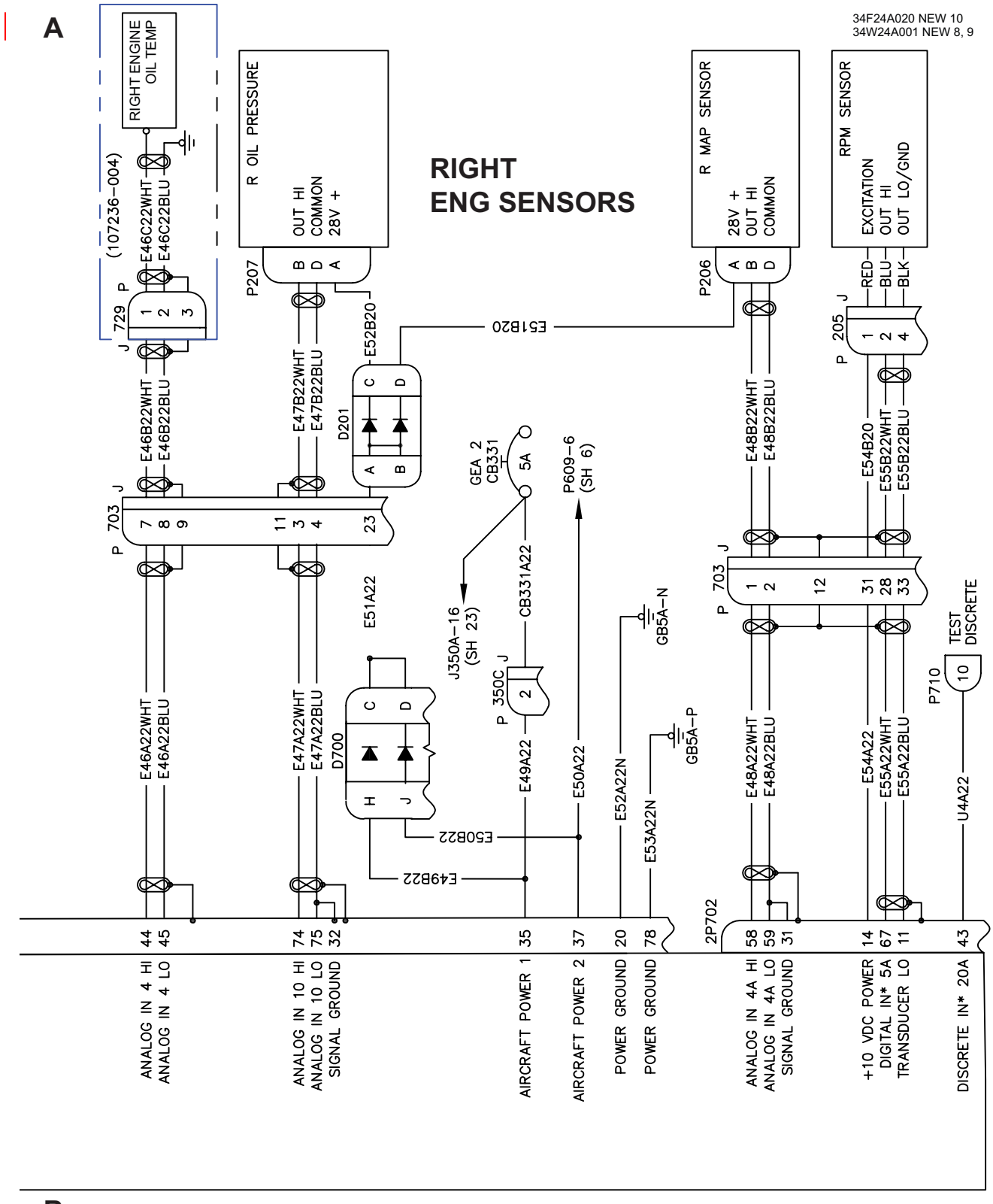
Effectivity
Seneca V
with Garmin G1000 NXi

34F24A020 NEW 10
34W24A001 NEW 8, 9



Effectivity
Seneca V
with Garmin G1000 NXi

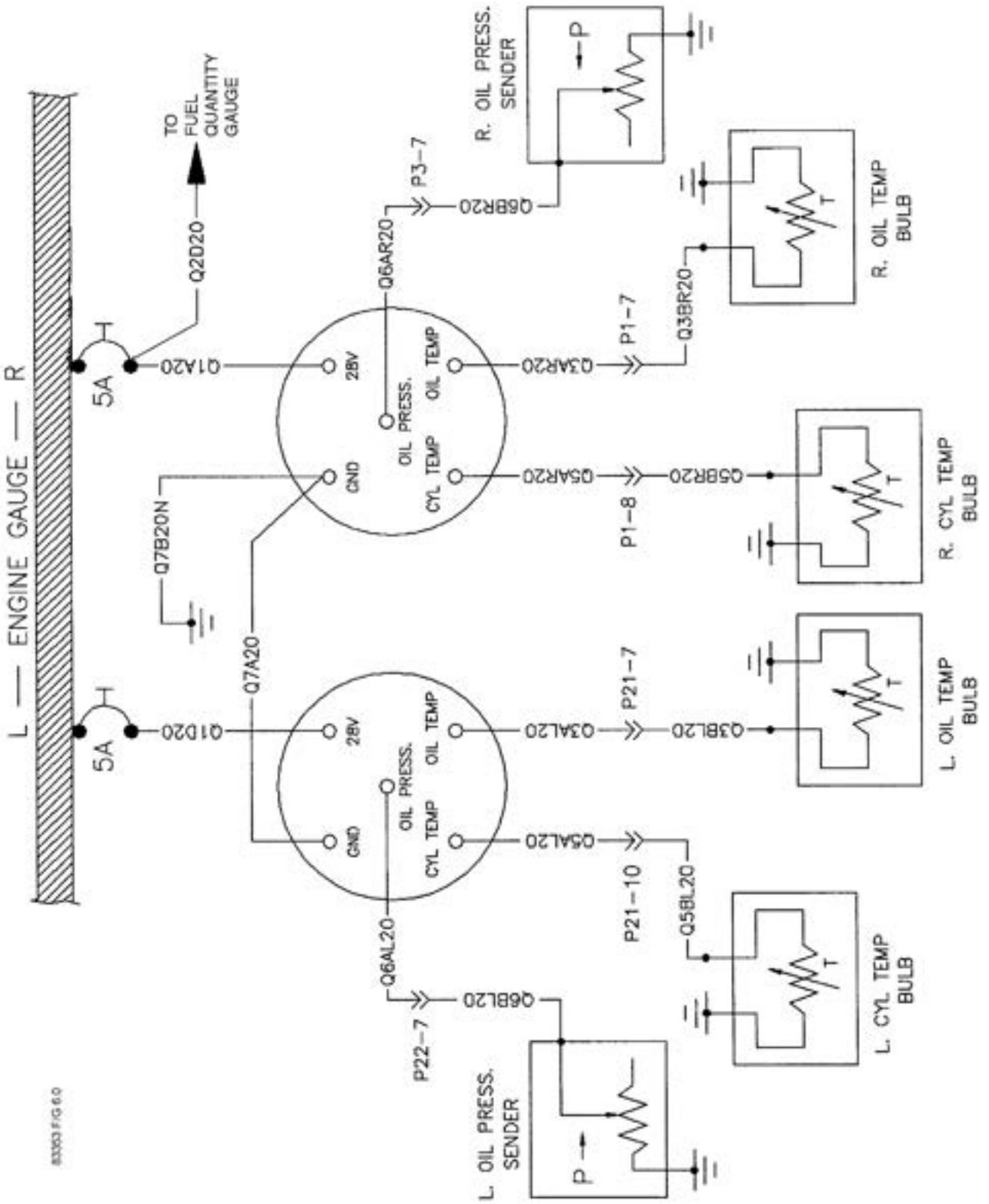
Engine Sensors
Figure 3 (Sheet 7 of 8)



Engine Sensors
Figure 3 (Sheet 8 of 8)

Effectivity
Seneca V
with Garmin G1000 NXi

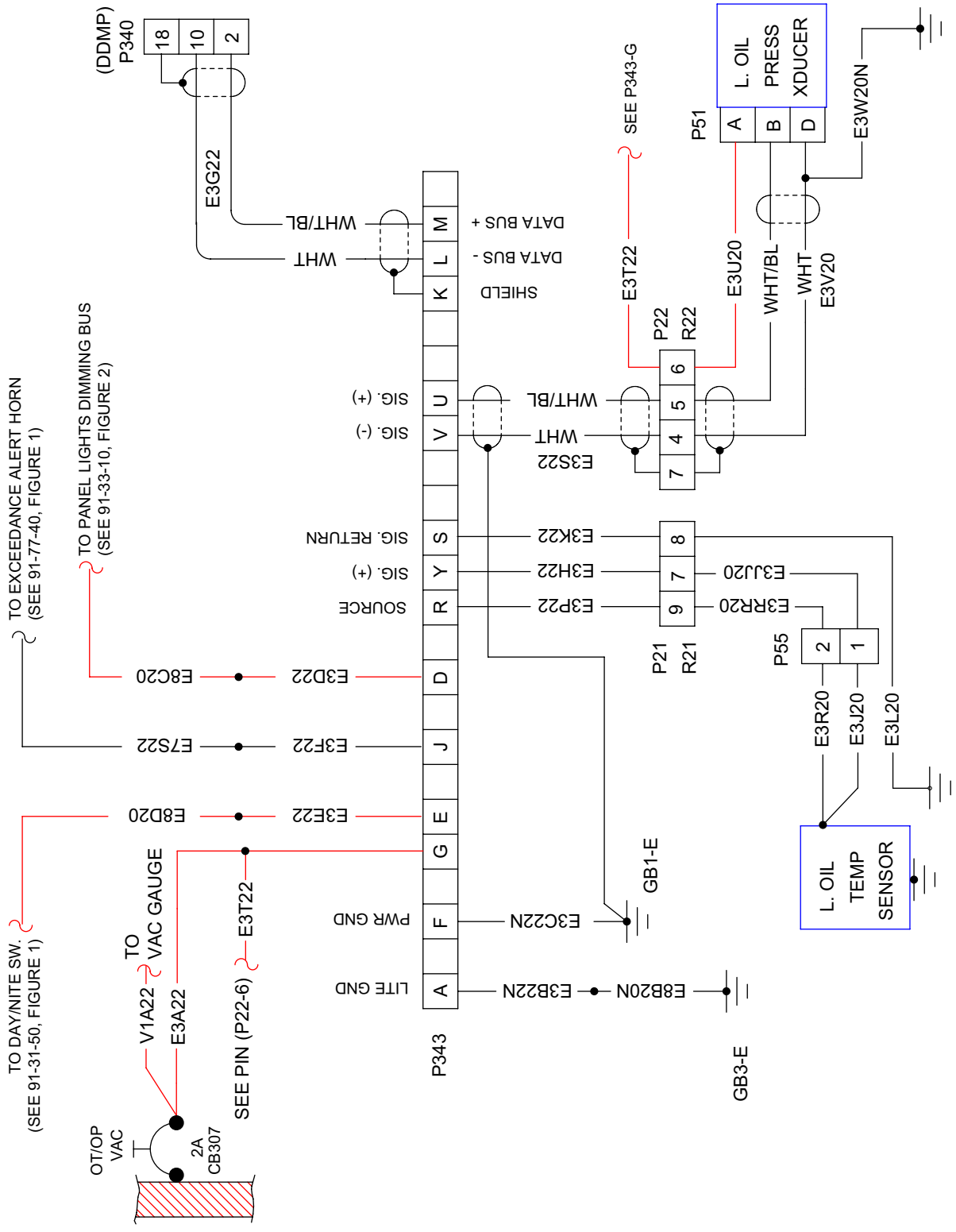
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Engine Gauges
Figure 1

Effectivity
Seneca IV

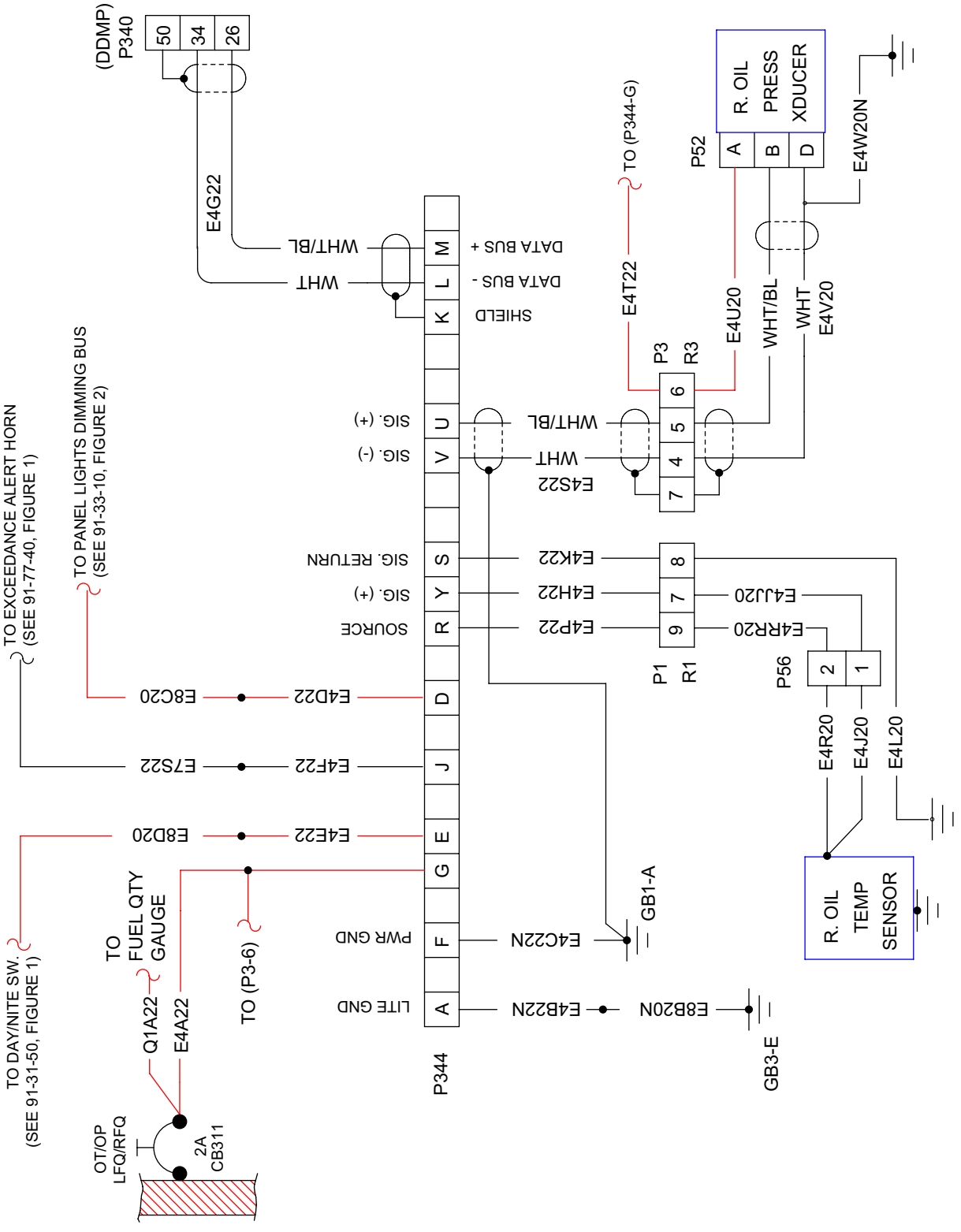
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104229 NEW/C 42.0
104368 NEW/K 42.0



Effectivity
Seneca V - Left

Engine Oil Temperature / Oil Pressure
Figure 2 (Sheet 1 of 2)

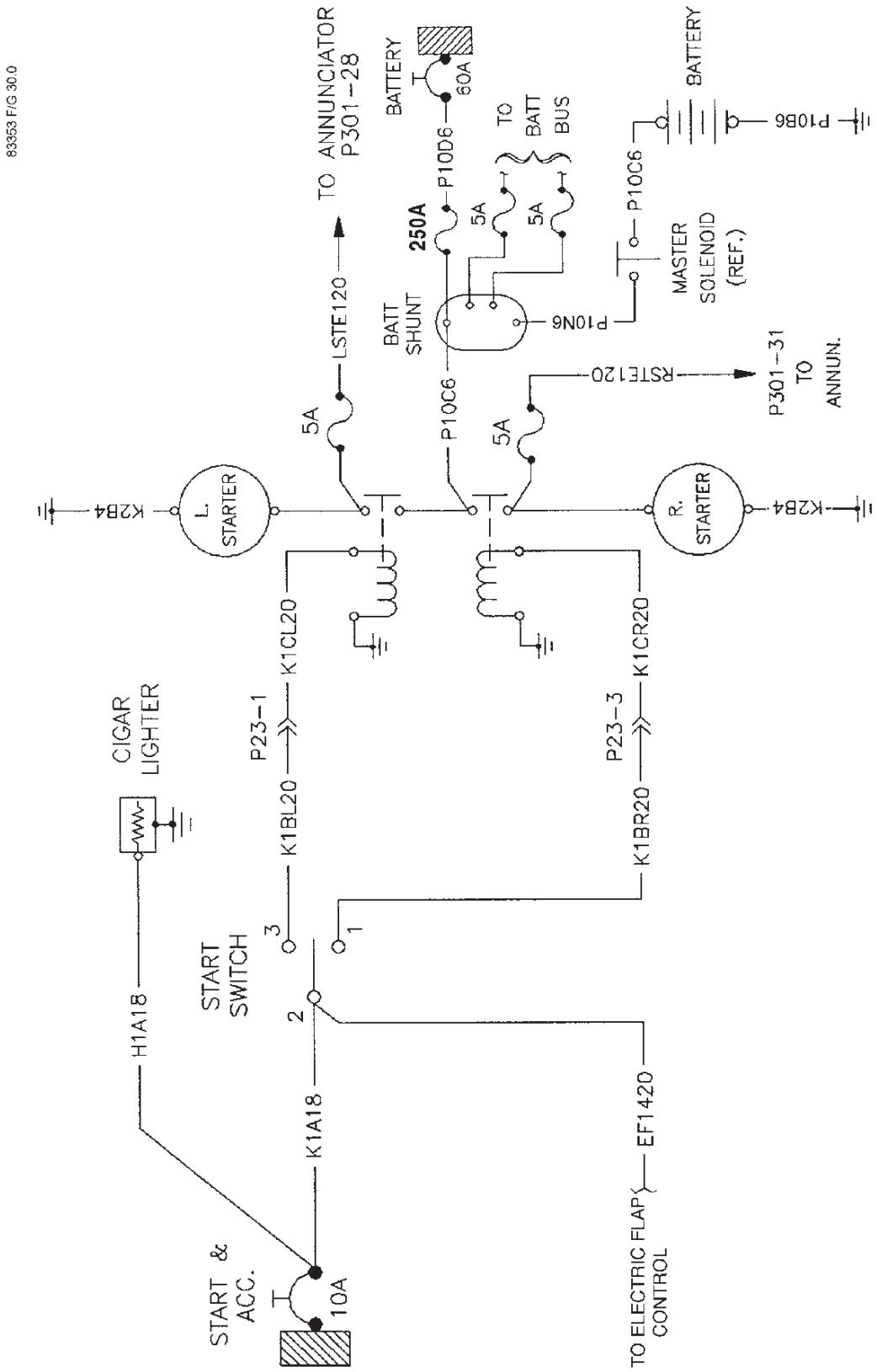
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101288 NEW/D 43.0
104229 NEW/C 43.0
104368 NEW/K 43.0



Engine Oil Temperature / Oil Pressure
Figure 2 (Sheet 2 of 2)

Effectivity
Right - Seneca V

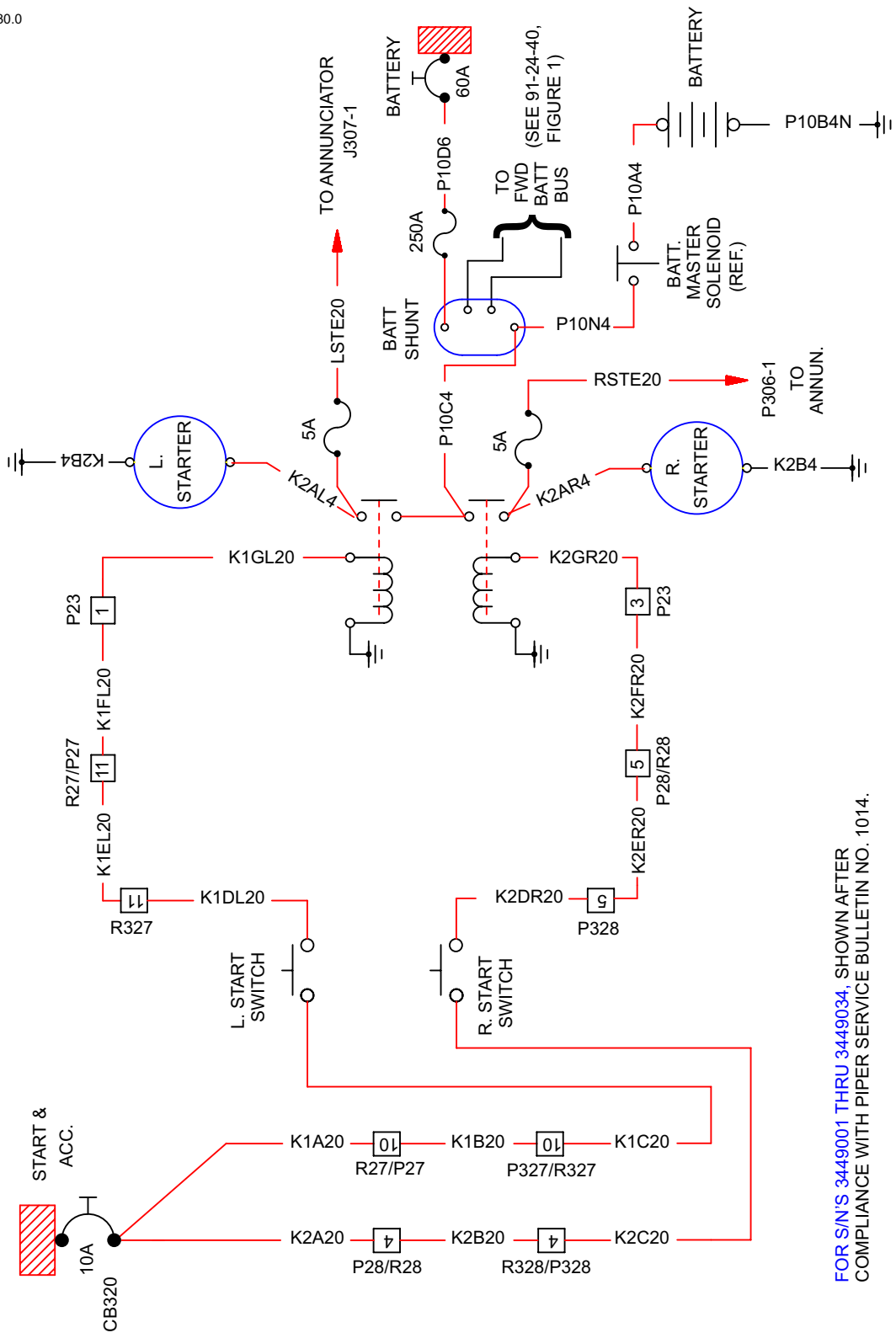
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Starter and Accessories
Figure 1 (Sheet 1 of 5)

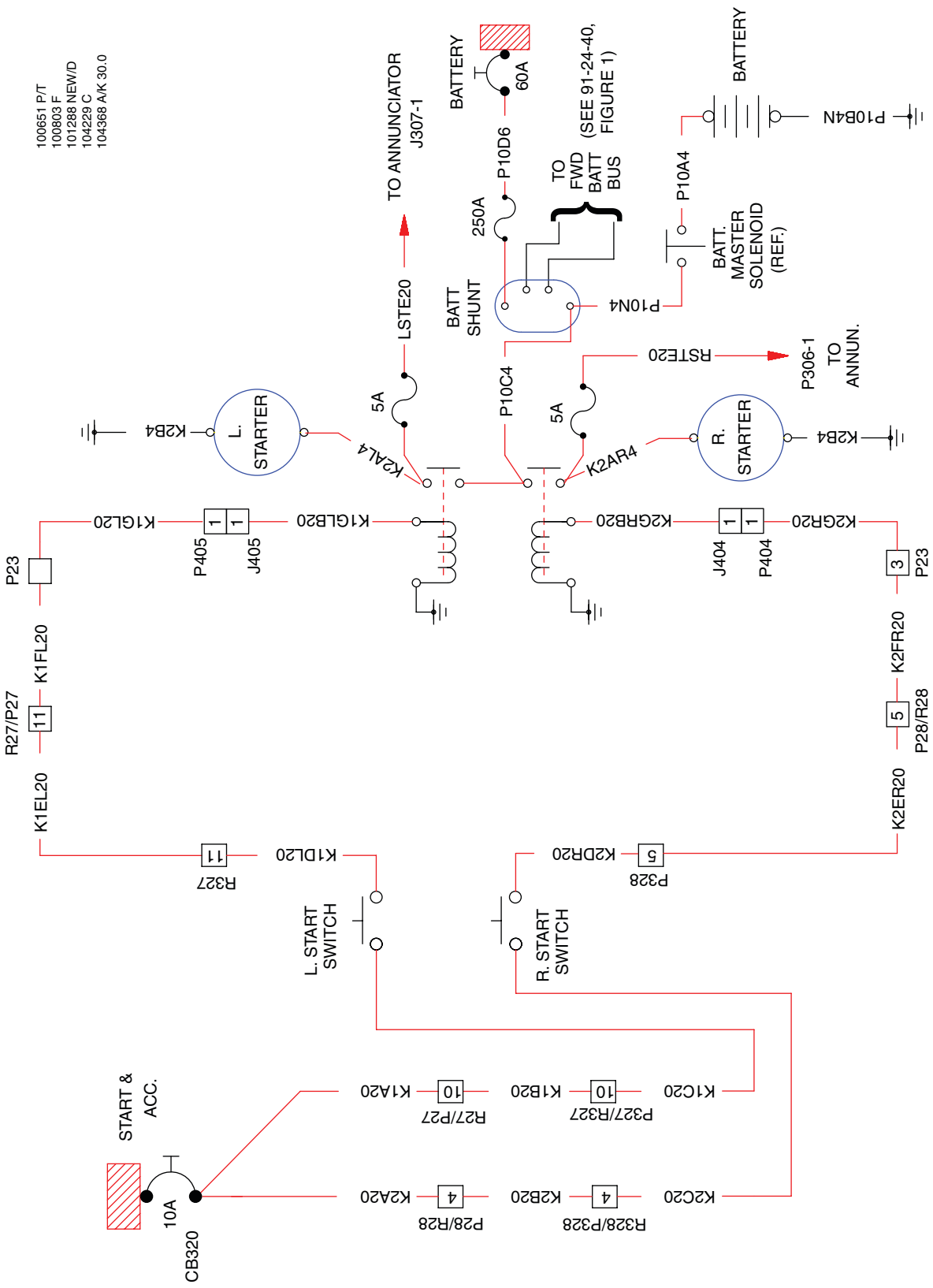
Effectivity
Seneca IV

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101288 NEW/D 30.0
104229 C 30.0
104368 A/K 30.0



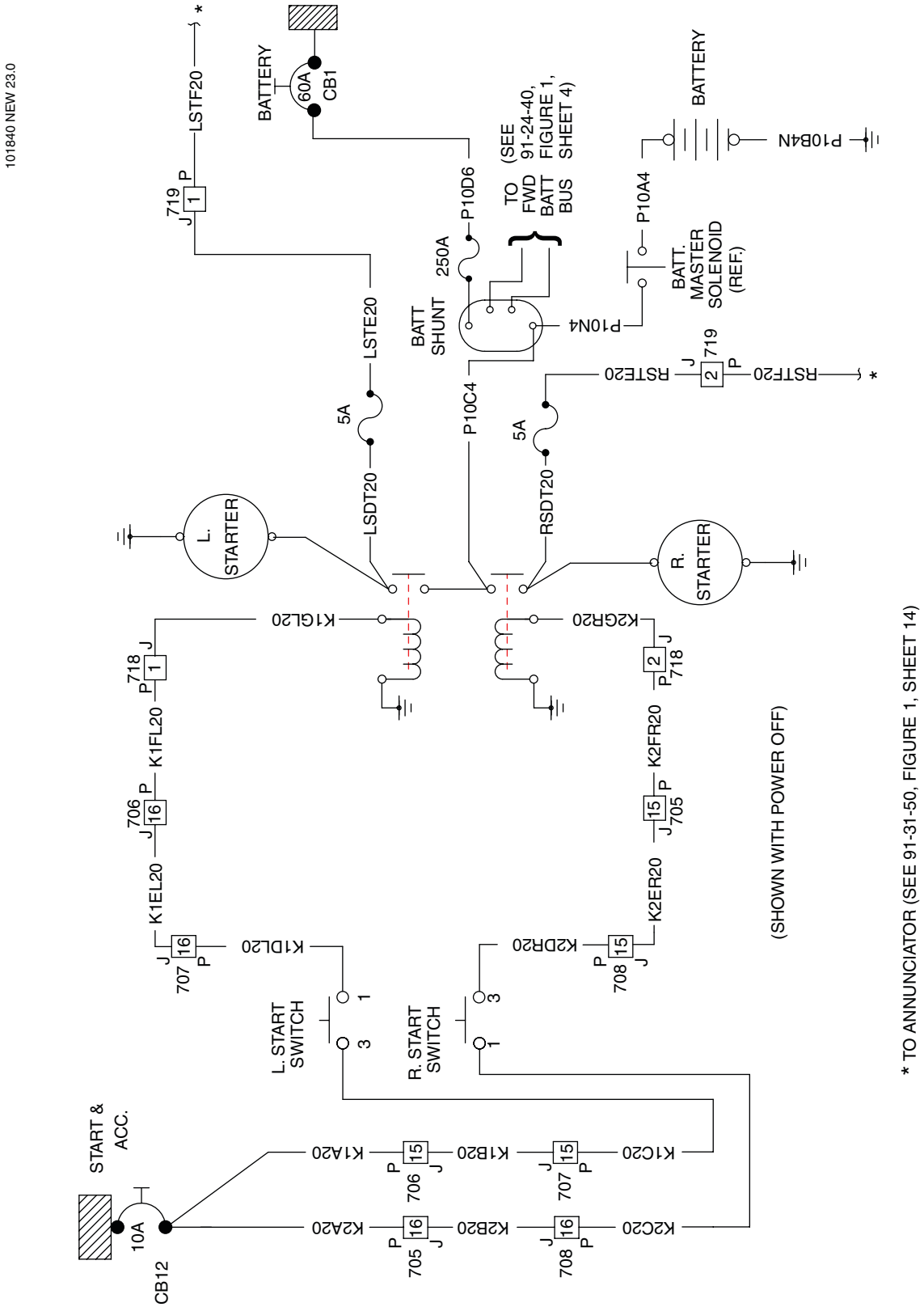
Effectivity
Seneca V
3449001 thru 3449405

Starter and Accessories
Figure 1 (Sheet 2 of 5)



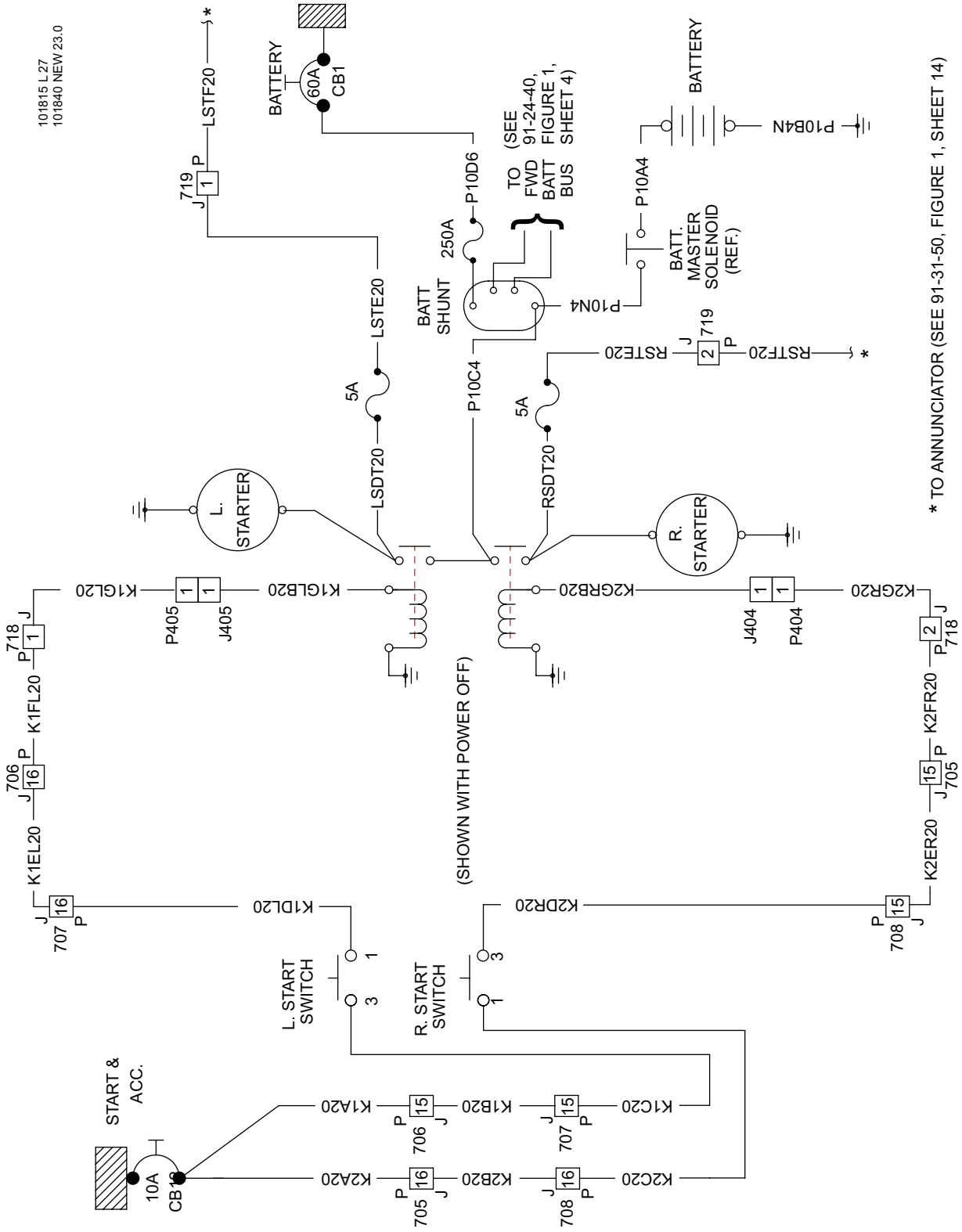
Starter and Accessories
Figure 1 (Sheet 3 of 5)

Effectivity
Seneca V
3449406 and up



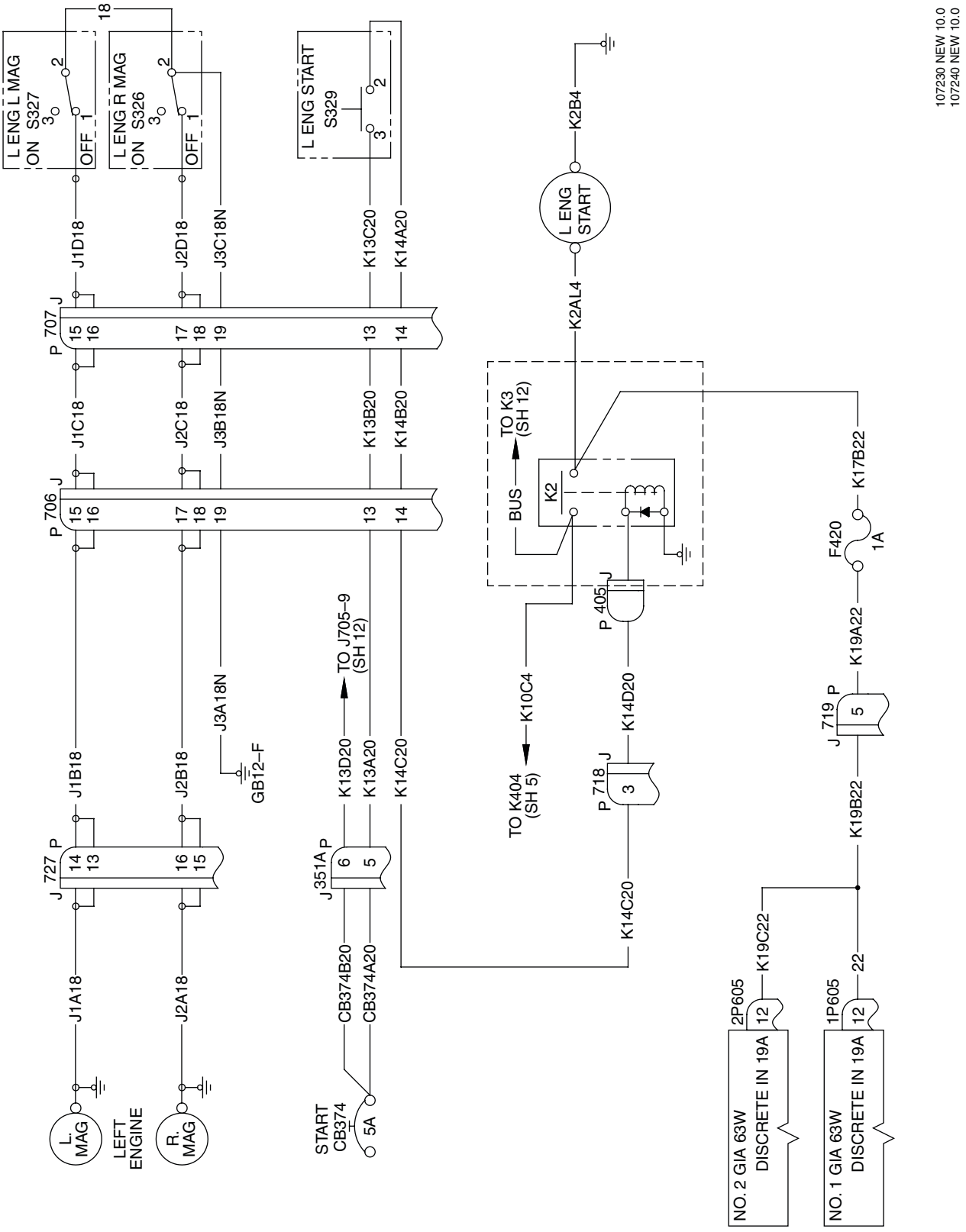
Effectivity
Seneca V
with Avidyne Entegra
3449311, 3449323,
3449326 thru 3449405

Starter and Accessories
Figure 1 (Sheet 4 of 5)



Starter and Accessories
Figure 1 (Sheet 5 of 5)

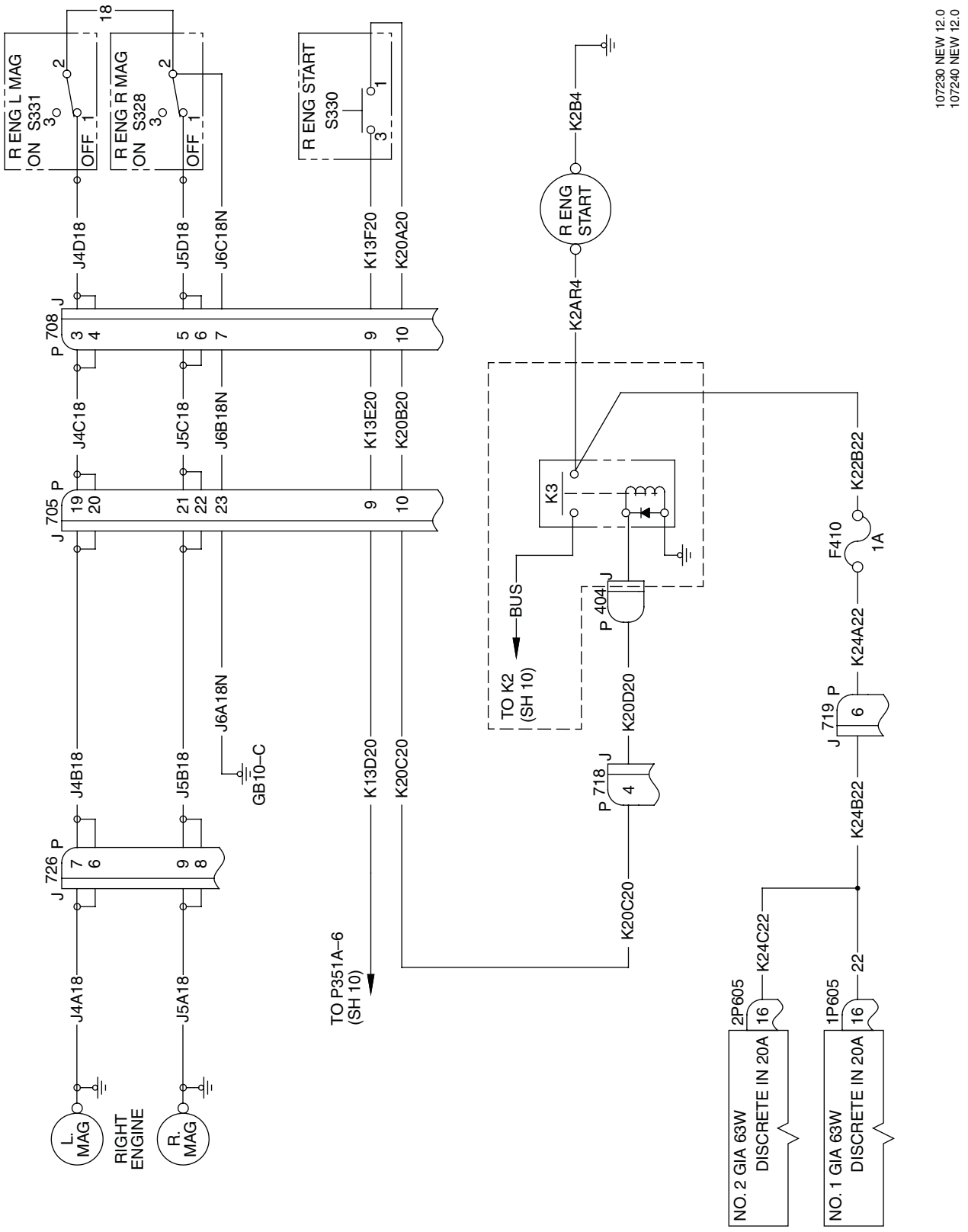
Effectivity
Seneca V
with Avidyne Entegra
3449406 and up



107230 NEW 10.0
107240 NEW 10.0

Effectivity
Seneca V - Left
with Garmin G1000

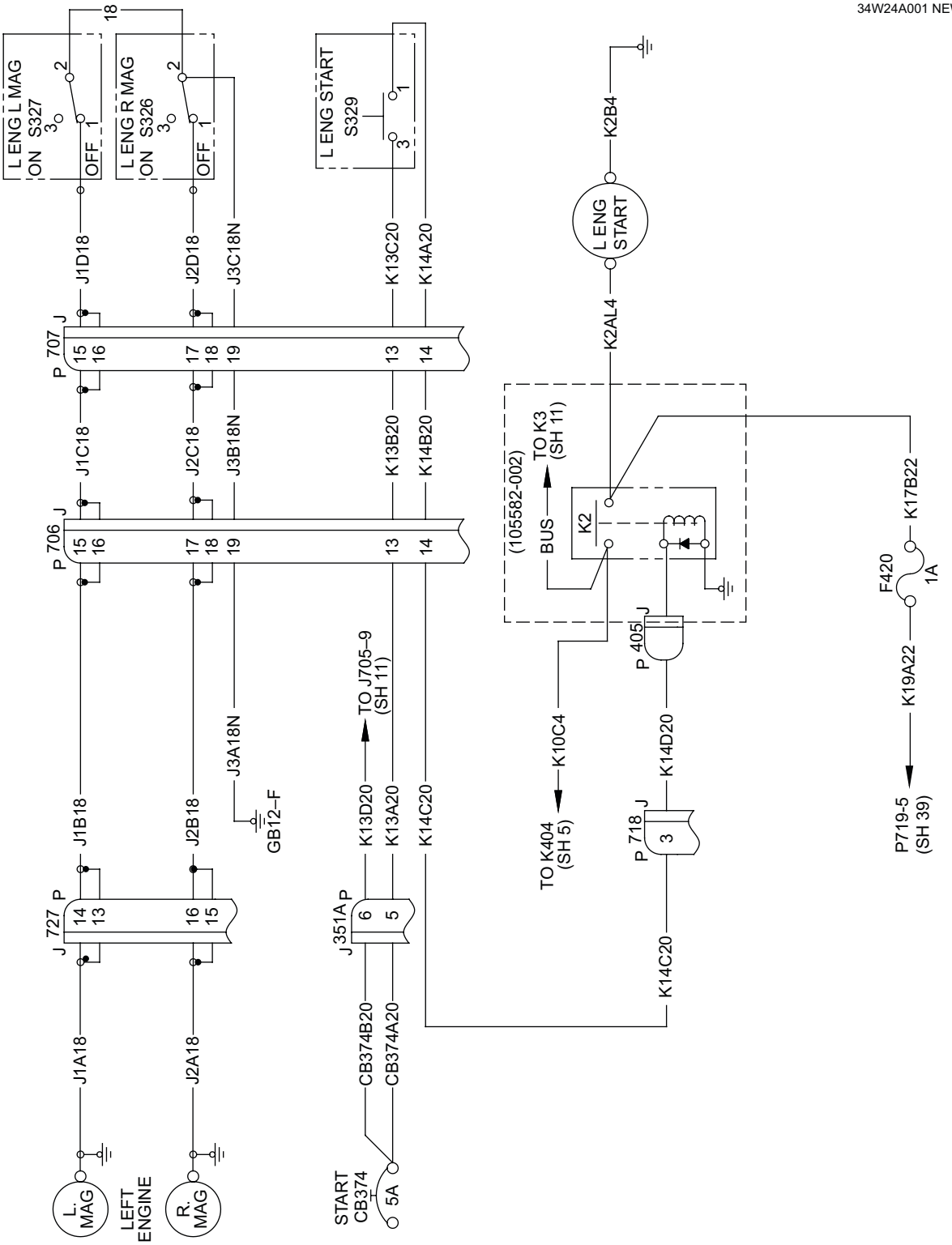
Engine Start / Ignition
Figure 2 (Sheet 1 of 4)



Engine Start / Ignition
Figure 2 (Sheet 2 of 4)

Effectivity
Right - Seneca V
with Garmin G1000

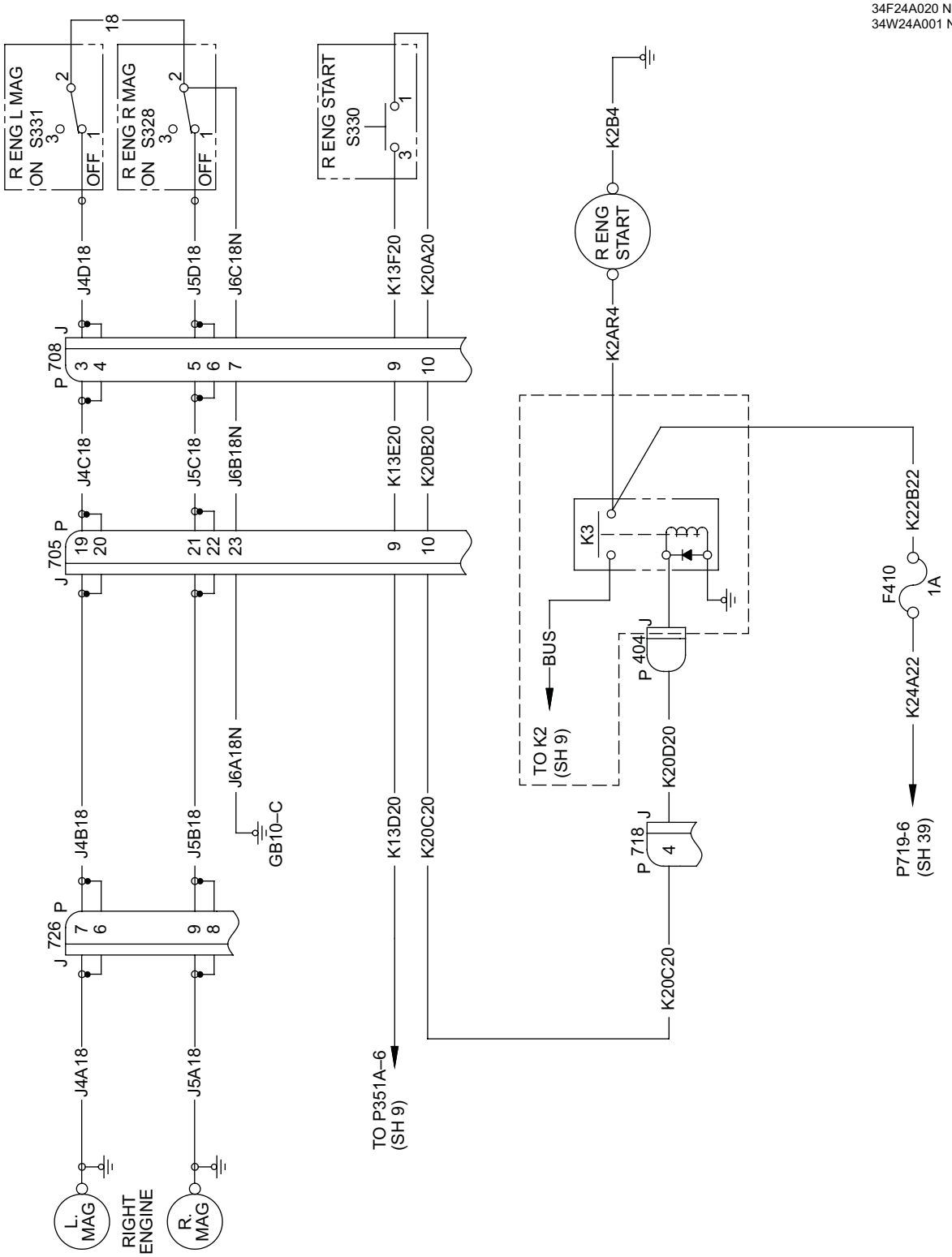
34F24A020 NEW 9
34W24A001 NEW 3



Effectivity
Seneca V - Left
with Garmin G1000 NXi

Engine Start / Ignition
Figure 2 (Sheet 3 of 4)

34F24A020 NEW 11
34W24A001 NEW 7



Engine Start / Ignition
Figure 2 (Sheet 4 of 4)

Effectivity
Right - Seneca V
with Garmin G1000 NXi

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WIRING

Airplanes equipped with Garmin GFC 700 Automatic Flight Control Systems have additional EMI shielding on some harness assemblies. If repair is required, see EMI Shielding in 20-97-00.

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APPENDIX

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GROUND TEST PROCEDURE

MEGGITT MAGIC AIR DATA & HEADING REFERENCE SYSTEM (ADAHRS) AND PITOT-STATIC SYSTEM

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APPENDIX 1

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
APP1 - List of Effective Pages	1	Sep 30/18			
	2	Sep 30/18			
APP1	1	Sep 30/18			
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Ground Test Procedure (Sheet 1 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

1. Reference Regulations and Guidance

Title 14 of the Code of Federal Regulations (14 CFR), Part 43 Appendix E, 23.1301(d), 23.1323(a)(c), 23.1325(b), 91.411, Advisory Circular AC 43-6B, and Advisory Circular AC 43.13-1B CHG 1 (or later) Chapter 12, Section 4.

2. Reference Documents

- Applicable Airplane Flight Manual (AFM)
- Applicable Airplane Maintenance Manual
- Applicable Meggitt Avionics/S-TEC MAGIC Pilot's Operating Handbook (POH)
- Meggitt Avionics/S-TEC Airspeed Configuration Data document (if applicable) for the airplane under test.

3. System Description

Each Meggitt MAGIC Electronic Flight Instrument System (EFIS) consists of the following components related to the Air Data and Pitot-Static systems:

- Primary Flight Display (PFD) with primary altimeter
- Navigation Display (ND) with reversionary altimeter
- Air Data/Attitude and Heading Reference System (ADAHRS)
- Standby Altimeter
- Standby Airspeed Indicator
- ATC Transponder (if installed)

Refer to the appropriate installation drawings or Instructions for Continued Airworthiness (ICA) document to determine the specific part numbers and locations of these components.

4. General

Purpose. This test procedure verifies the integrity and accuracy of the air data and pitot-static systems on the aircraft under test and shows that these systems comply with applicable airworthiness regulations. No bench testing or field repairs are included in this procedure. Any unsatisfactory findings should be repaired in accordance with other acceptable means.

Use this on-aircraft ground test procedure (GTP) to inspect and test each Meggitt Avionics/S-TEC MAGIC Air Data and Pitot-Static system, under the following conditions:

- In accordance with 24-month static system test; or
- To proof test static systems following any opening and closing of the pitot-static plumbing, except for the use of system drain and alternate static pressure valves (§ 23.1325(b)(2); or
- Following installation or maintenance on Air Data/Attitude and Heading Reference System (ADAHRS), Primary Flight Display (PFD), or Transponder Mode C interface.

5. Effectivity

This GTP is effective to aircraft with Meggitt Avionics/S-TEC MAGIC EFIS and pneumatic standby instruments.

6. Prior to testing

Conduct this GTP after the following has been accomplished:

- Continuity testing of new or altered electrical wiring
- Power and ground testing of new or altered electrical wiring
- Electrical bonding testing of new equipment
- Installation of equipment

- Required test equipment, technical data, and rated personnel are available to perform a static system check as required by § 91.411(a)(2).
- Calibrated pitot-static tester and transponder ramp tester is available to determine the pressure altitude transmitted by the transponder in accordance with § 91.217(b).
- Completion of appropriate EFIS Post-Installation GTP.

NOTE

This Pitot-Static GTP is to be accomplished after MAGIC EFIS Post-Installation GTP and after leveling of the ADAHRS, which may require that the pitot-static lines to be temporarily disconnected.

7. After testing

Ensure that an appropriately rated person returns the aircraft to service following the alteration or maintenance in accordance with § 43.9.

8. Standard atmospheric ambient conditions and tolerances

Unless otherwise specified, testing shall be performed during the following standard atmospheric ambient conditions:

- Temperature: 25° C ± 10° C
  Record temperature if outside of range: _____

Avoid calibrating air data systems at extreme temperatures.

9. Precautions

This section contains precautions that should be taken during tests to avoid damage to the sensitive instruments.

Damage can occur to instruments that are connected to both the static system and pitot system when only the static system is evacuated. The maximum design differential pressure of these instruments may be exceeded. This GTP requires that both the pitot and static ports be connected to the pitot-static tester.

Safeguards should be taken to prevent accidental disconnection of the test equipment plumbing from the aircraft or the test equipment while the static system is evacuated. The resultant sudden pressure change may damage both the test instruments and the aircraft instruments. The aircraft static system should be returned to ambient pressures before disconnecting static test equipment from the system

If static ports are blocked for the purpose of testing, use a long piece of brightly colored tape (red or orange) forming a streamer or similar method of warning be attached to the blockage device.

Damage may occur to other aircraft instruments if the altitude rate is changed faster than the limit of instruments.

10. Calibration of equipment

Conformity aircraft. Unless otherwise specified, test instruments used to calibrate or verify calibration in a certification project should be calibrated by an approved instrument repair inspector prior to FAA testing. Test instruments should be calibrated within 3 months of FAA testing. Reference AC 21-40, Section 8-4(h).

Follow-on installations and maintenance. In accordance with § 145.47 (b), the repair station shall ensure that all inspection and test equipment is tested at regular intervals to ensure correct calibration to a standard derived from the National Bureau of Standards or to a standard provided by the equipment manufacturer.

11. Ground Equipment

Use the following ground equipment for this GTP.

1. Ground Power Unit (GPU). Conduct this test procedure using a GPU that will provide sufficient regulated electrical power to the aircraft.
2. Pitot-Static tester. Use a calibrated pitot-static tester to simulate airspeed and altitudes on the aircraft. It is recommended to use a tester with a remote electronic control panel to facilitate the verification of the instruments in the aircraft.
3. ATC Transponder Ramp Tester. Use a calibrated ATC Transponder ramp tester to interrogate and verify proper Mode C (altitude) altitude reporting.
4. Barometer (pressurized aircraft only). A calibrated barometer may be used to determine the "station pressure" at the field where the aircraft will be tested. Station pressure will be used to calculate the MAX DIFF test altitude, which is the test altitude where the pitot-static system will be tested for leaks. (*Do not use altimeter setting barometric pressure for the MAX DIFF calculation.*)

CAUTION

Remove all tape and thoroughly clean ports to remove any adhesive resins after testing.

12. Ground Equipment ID

Record in Table I below the specific ground equipment used during this GTP:

Table I — Test Equipment

TEST EQUIPMENT	MANUFACTURER	MODEL	S/N	CAL DATE	CAL DUE
Ground Power Unit (GPU)					
Pitot-Static Tester					
ATC Transponder Tester					
Barometer					

13. Local information

Field elevation. Record the field elevation (H_a) where the aircraft is being tested, as a reference.

Field Elevation (H_a) _____ feet

14. Proof Test Parameters (Pressurized Aircraft Only)

MAX DIFF test altitude (computation). For pressurized aircraft only, determine the equivalent altitude of the maximum cabin pressure differential (MAX DIFF test altitude).

This is the test point at which the proof test will be accomplished. The test altitude will vary depending on the current station barometer and the cabin pressurization limit of the aircraft under test. The following steps are provided to compute the MAX DIFF test altitude:

Step 1 (Station Pressure): Obtain the current local barometric pressure (station pressure) within 25 miles. (*Note that the station barometer is the true barometric pressure, unlike altimeters settings, which are "reduced" to sea level pressure*):

Station Barometer (Inches of Mercury) _____ In Hg

Step 2 (Barometer Conversion): Convert the station barometric pressure to PSI, by dividing the barometric pressure by 2.036:

Station Barometer (Pounds per Square Inch) _____ PSI

Step 3: (Cabin Pressurization Limit). Obtain the approved maximum cabin differential pressure for the aircraft under test, which is usually found in the AFM Limitations:

Maximum Cabin Differential Pressure 5.5 PSI

Step 4 (MAX DIFF test pressure). Subtract Step 3 from Step 2 to determine the MAX DIFF test pressure:

MAX DIFF test pressure _____ PSI

Step 5 (MAX DIFF test altitude). Use Table IX at the end of this document to approximate the test pressure of Step 4 to an equivalent altitude, by selecting the next lower number in the PSI column and recording its corresponding altitude below:

Equivalent Altitude of Maximum Cabin Pressure Differential (MAX DIFF test altitude) _____ feet

EXAMPLE:

Step 1. Local station pressure: 25.39 Inches of Mercury

Step 2. PSI = $\frac{25.39}{2.036}$ = 12.47 PSI

Step 3: Cabin pressurization limit: 5.3 PSI

Step 4: 12.47 – 5.3 = 7.17 PSI

Step 5: 7.17 PSI = 19000 feet

The MAX DIFF test altitude of 19000 feet was estimated by rounding 7.17 PSI to the next lower value of 7.0412, as shown in Table IX at the end of this document.

MAX DIFF test altitude (**alternate method**). An alternate method to determine MAX DIFF test altitude on aircraft with an approved maximum differential cabin pressure indicator is to use the redline mark on this indicator. By this method, use the pitot-static tester to increase the altitude of the static system until the cabin differential pressure indicator reaches redline. Record the corresponding altitude as the MAX DIFF test altitude.

MAX DIFF tolerance. Calculate the MAX DIFF tolerance of 2% of MAX DIFF test altitude. Multiply MAX DIFF test altitude by **0.02** to obtain the tolerance. Insert this value into Table IV also.

MAX DIFF Tolerance = (0.02) x (MAX DIFF): ± _____ feet

15. Airspeed and Altitude Limits

Record the airspeed and altitude design limits of the aircraft under test, as specified in the appropriate Airplane Flight Manual (AFM) or Type Certificate Data Sheet (TCDS). Use these figures to determine the limits of testing. Do not exceed the airspeed limit by more than 10 knots or the altitude limit by more than 2500 feet.

Airspeed limit. Record the maximum operating airspeed of the aircraft under test. Also make a note of the maximum airspeed in the margin of Table VIII for the Indicated Airspeed Test.

Maximum Operating Airspeed: _____ * knots

Altitude limits. Record the maximum operating altitude of the aircraft under test. Also make a note of the maximum altitude in the margin of Table V for the Scale Error Test and Table VII for the Friction Test.

Maximum Operating Altitude: _____ ** feet

16. Hysteresis Test

Dive (hysteresis) test points. Calculate required hysteresis test points from the maximum operating altitude.

50% test point. Multiply the maximum operating altitude by **0.5**. From this value, select the closest test point from the altitudes listed in Table V as the 50% test point:

50 % test point: _____ feet

40% test point. Multiply the maximum operating altitude by **0.4**. From this value, select the closest test point from the altitudes listed in Table V as the 40% test point:

40 % test point: _____ feet

Record the 50% and 40% test points in the "*Calculated Test Point Altitude*" column of Table VI.

* 188 KIAS - PA-46-500TP
198 KIAS - PA-46-350P

** 30,000 FT - PA-46-500TP
25,000 FT - PA-46-350P

NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
1.	**Before Applying Electrical Power**				
2.	GROUND SAFETY INSPECTION Conduct a ground safety inspection in accordance with the appropriate Airplane Maintenance Manual prior to applying electrical power to the aircraft under test.	Check that master switches are OFF. Pull circuit breakers for engine start and fire control OUT.			
3.	Remove pitot-static covers (if installed)	Check that pitot-static ports are clear.			
4.	**Inspection**				
5.	*The following inspections are in compliance with § 23.1325(b)(1) and § 43 Appendix E(a).*				
6.	*Note: Whenever blockage of the static lines is suspected, they should be purged before the static pressure system test is performed. Purging may keep objects from entering the test equipment and instruments. Since purging applies positive pressure to lines, the following precautions should be taken:* *(1) Disconnect all instruments, sensors, and ADAHRS from the pitot-static lines.* *(2) Secure a cloth bag over the end of the lines to catch any debris.* *(3) Clean drains after purging as they can act as a sump for foreign material.*				
7.	Inspect and operate the pitot-static drains.	Verify drains operate normally and that positive drainage of moisture is provided in all pitot-static lines.			
8.	Inspect installation of the pitot-static lines.	Verify that chafing of the tubing and excessive distortion or restriction at bends in the tubing is avoided.			

Ground Test Procedure (Sheet 6 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
9.	Inspect the ports, tubing, accessories, and instruments connected to the static system to identify any parts that might be defective (e.g., broken "B" nuts, cracked flare sleeves, deteriorated flexible tubing, bad valves, etc.).	Verify that the materials used are durable, suitable for the purpose intended, and protected against corrosion.			
10.	Inspect static ports.	Verify static ports are free of debris and corrosion.			
		Verify that no alterations or deformations of the airframe surface have been made in the areas near the static ports.			
11.	Inspect alternate static ports.	Verify alternate static port is free of debris or corrosion.			
12.	Inspect pitot probes.	Verify inlet is free of debris or damage.			
		Verify that the pitot probe does not have any corrosion within ½-inch of probe tip.			
		Ensure that no alterations or deformations of the airframe surface have been made in the areas near the pitot probes.			
13.	Inspect transponder antennas. *Note: Oil, exhaust, and dirt buildup on a transponder antenna may attenuate and shift frequency out of tolerance.*	Verify transponder antennas are clean and not damaged.			
14.	**Set up ground equipment**				
15.	Set up Transponder ramp tester with the test antenna near the transponder antenna.				
16.	Connect Ground Power Unit to aircraft.				

Ground Test Procedure (Sheet 7 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
17.	**Apply electrical power to aircraft**				
18.	Activate ground power in accordance with the Airplane Maintenance Manual procedures and turn on electrical buses.				
19.	Turn on cabin lights to verify electrical power is applied to the aircraft.				
20.	Turn on EFIS power. *Check that PFD is not in reversionary mode.*	Verify normal PFD screen after PFD initialization screen. Verify ADAHRS initializes after normal countdown.			
21.	Turn on static port heat (if installed).	Verify either increased ammeter current or that static port becomes hot to the touch.			
22.	Turn off static port heat (if installed).	Verify either reduced ammeter current or that static port returns to ambient.			
23.	Turn on pitot heat.	Verify either increased ammeter current or that pitot probe becomes hot to the touch.			
24.	Turn off pitot heat.	Verify either reduced ammeter current or that pitot probe returns to ambient.			

Ground Test Procedure (Sheet 8 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
25.	**Alternate Static (if installed) Test at Field Elevation**				
26.	***Purpose.*** *This test will verify that the altimeters do not change more than 50 feet between normal and alternate static sources at field elevation, in compliance with § 23.1325(b)(3).* *✈ A functional flight check of the alternate static selector is recommended.*				
27.	Press **BARO** knob on PFD 1 and 2.	Verify PFD indicates STD.			
28.	Set BARO on standby altimeter to **29.92**.	Verify dial rotates normally.			
29.	Inspect the alternate static source selector.	Verify alternate static selector is clearly marked and can be switched freely between normal and alternate sources.			
30.	Switch alternate static to NORMAL.	Record results in the NORMAL row in Table II below.			
31.	Switch alternate static to ALTERNATE.	Record results in the Alternate row in Table II below.			
32.	Return alternate static to NORMAL.	Compute the difference between normal and alternate in Table II below. Verify that the altitude does not differ by more than 50 feet, IAW § 23.1325(b)(3).			

Table II — Alternate Static Test

Alternate Static Switch	Pilot's Altimeter (feet)	Copilot's Altimeter (feet)	Standby Altimeter (feet)	Tolerance (feet)	Pass/Fail
Normal					
Alternate					
Difference				±50	

NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
33.	**Barometric Scale Error Test**				
34.	***Purpose.*** *This test verifies the accuracy of the barometric scale on each altimeter, in compliance with § 43 Appendix E, Table IV.*				
35.	Set the BARO knob on each altimeter to standard pressure 29.92 in order to determine the pressure altitude (Pa) at field elevation.	Record the pressure altitude (Pa) in the first row of Table III below. Use this altitude as the reference point.			
36.	Adjust the BARO knob on each altimeter to the specified settings, record the new indicated altitudes, and compute the difference between the indicated altitudes from pressure altitude.	Verify that the change in altitude on each altimeter is within ±25 feet of the expected change from pressure altitude at field elevation.			

Table III — Pressure-Altitude Difference

BARO Setting	Expected Change from Pressure Altitude (Pa)	Tolerance	PFD 1 Altimeter (feet)		PFD 2 Altimeter (feet)		Standby Altimeter (feet)		Pass/Fail
			Indication	Altitude change from Pa	Indication	Altitude change from Pa	Indication	Altitude change from Pa	
29.92	Pressure Altitude (PA)								
29.50	-392	±25 ft							
29.00	-863	±25 ft							
28.50	-1,340	±25 ft							
28.10	-1,727	±25 ft							
30.50	+531	±25 ft							
30.90	+893	±25 ft							
30.99	+974	±25 ft							

37.	Press **BARO** knob on PFD 1 and 2.	Verify PFD indicates STD.			
38.	Reset BARO on standby altimeter to **29.92**.				

Ground Test Procedure (Sheet 10 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
39.	**Pitot-Static Tester Set Up**				
	Test Points. *Connect the pitot-static test equipment directly to the pitot-static ports on the aircraft, if practicable. Otherwise, connect to a static system drain or tee connection and seal off the static ports. If the test equipment is connected to the static system at any point other than the static port, it should be made at a point where the connection may be readily inspected for system integrity after the system is returned to its normal configuration.* ***Seals.*** *Block unconnected ports with brightly colored tape long enough to have a streamer.* **Note:** Remove all static port seals after completion of this test procedure.				
40.	***Separation.*** *If the aircraft under test has more than one pitot-static system, test each system separately to ensure their independence and that the leak rate for each is within tolerance.* ***Vibration.*** *Unless otherwise specified, each test may be conducted with vibration applied to pneumatic standby instruments.* **Note:** **When applying vibration, lightly tapping the standby instrument face with a finger should suffice.** **DO NOT install a shaker or vibrator on the instrument panel.** **Caution:** Secure pitot-static test connections carefully. Avoid circumstances that would result in a test line disconnecting unintentionally while test pressure is being applied.				

Ground Test Procedure (Sheet 11 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
41.	**Proof Test**				
	Purpose. *This test will verify the integrity of the pitot-static system(s), including the lines and components, in compliance with § 23.1325(b)(2) and § 91.411(a)(2). Procedures are included for both pressurized and non-pressurized aircraft, with the allowable tolerance of each type shown. Use one or the other as appropriate for the aircraft under test.* ***Method.*** *Each pitot-static system shall be tested separately to verify that it can hold the test pressure for 1 minute without support from the vacuum pump in the tester. The pitot and static ports of each system shall be connected to the pitot-static tester to verify both lines.*				
42.	**Note:** On aircraft with multiple pitot systems using a single static system, connect all pitot systems to the pitot-static tester to avoid damage to instruments. **Caution:** Monitor all air data instruments during testing to ensure that none are operated beyond limitations.	*All instruments in the aircraft under test shall be connected to their normal lines to assure that no leaks occur within the instruments or their connections.* *The primary altimeter in the aircraft under test or that in the pitot-static tester may be used as a vacuum gauge for the proof test.* *Use Table IV to record readings, calculations, and pass/fail results.*			

Ground Test Procedure (Sheet 12 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
43.	Unpressurized Airplanes. With the pitot-static tester connected to each pitot-static system separately, simulate a test altitude of 1,000 feet above field elevation at 0 knots airspeed. Upon reaching the test altitude, record the indicated altitude displayed on the specified altimeter as the *Initial Altitude* in Table IV. *After 1 minute* without additional vacuum being applied to the aircraft, record the second altitude from the same altimeter.	Compute the *Altitude Loss* in Table IV by subtracting the *Initial Altitude* from the *Altitude After 1 Minute*. Verify that the *Altitude Loss* is less than the unpressurized proof test tolerance of ±100 feet.			
44.	Pressurized Airplanes. With the pitot-static tester connected to each system separately, simulate MAX DIFF test altitude, as computed in Section 14: ▶_____ feet Upon reaching the MAX DIFF altitude, record the indicated altitude displayed on the specified altimeter as the *Initial Altitude* in Table IV. *After 1 minute* without additional vacuum being applied to the aircraft, record the second altitude from the same altimeter.	Compute the *Altitude Loss* in Table IV by subtracting the *Initial Altitude* from the *Altitude After 1 Minute*. Verify that the *Altitude Loss* is less than the Pressurized MAX DIFF Tolerance (2% of MAX DIFF altitude).			
45.	Return pitot-static tester to atmospheric conditions on conclusion of proof test.				

Table IV — Proof Test

Parameter	Pilot's Altimeter (feet)	Copilot's Altimeter (feet)	Standby Altimeter (feet)	Unpressurized Proof Test Tolerance(feet)	Pressurized MAX DIFF Tolerance(feet)	Pass/Fail
1st reading → *Initial Altitude*						
2nd reading → *Altitude After 1 Minute*						
Calculate → 1st – 2nd = *Altitude Loss*				±100	▶ ±	

Ground Test Procedure (Sheet 13 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
46.	**Scale Error and Altitude Reporting Test**				
47.	***Purpose.*** *This test will verify the accuracy of the altimeters and altitude-reporting devices during a 4000--6000 FPM climb between test points, in compliance with Part 43 Appendix E, Table I and § 91.217(b).* ***Method.*** *Each static system may be tested separately or together. The pitot and static ports of each system shall be connected to the pitot-static tester.* **Caution:** Monitor all air data instruments during testing to ensure that none are operated beyond limitations. *All instruments in the aircraft under test shall be connected to their normal lines.* *Each altimeter in the aircraft under test will be compared to the altitude of the tester at test point altitudes, as specified in Table V.* *Use Table V to record readings, calculations, and pass/fail results.*				
48.	Set pitot-static tester altimeter to 29.92 and airspeed to 0.				
49.	Press the BARO knob on PFD 1 and 2 for STD (29.92) setting.				
50.	Set the BARO on standby altimeter to 29.92.				
51.	Set up the transponder ramp tester to interrogate the transponder(s) on the aircraft.				
52.	Turn on transponder 1 in Mode 3/A with altitude (Mode C) off.	Verify on the ramp tester that the transponder is NOT transmitting altitude information.			
53.	Turn on altitude (Mode C) on transponder 1.	Verify proper altitude pulse train or decoded altitude on the ramp tester.			

Ground Test Procedure (Sheet 14 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
54.	Turn on transponder 2 (if installed) in Mode 3/A with altitude (Mode C) off.	Verify on the ramp tester that the transponder is NOT transmitting altitude information.			
55.	Turn on altitude (Mode C) on transponder 2 (if installed).	Verify proper altitude pulse train or decoded altitude on the ramp tester.			
56.	Continue to interrogate each transponder with the ramp tester through the remainder of the scale test, up to the maximum altitude.	Record the altitude reporting of each transponder at each test point in Table V and verify that the reported altitude is within MODE C tolerance of ±125 feet.			
57.	*Use the pitot-static tester to apply suction to the static system in accordance with the test points listed in Table V up to the maximum operating altitude of ▶ ** feet for the aircraft. NOTE: Do not exceed the maximum altitude of the aircraft by more than 2500 feet.*				
58.	Adjust the pitot-static test set vertical speed to climb at 4000–6000 FPM between test points. Stop when the tester reaches each test point and record the indicated altitude on each altimeter.	Verify that the indicated altitude on each altimeter is within the SCALE tolerance specified in Table V.			
59.	Turn off transponder(s) after reaching maximum altitude.				
60.	Turn off transponder ramp tester after reaching maximum altitude.				
61.	End of Scale Error and Altitude Reporting Test. *Begin the hysteresis (dive) test from the maximum altitude within 15 minutes of reaching the maximum altitude.*				

** 30,000 FT - PA-46-500TP
25,000 FT - PA-46-350P

Ground Test Procedure (Sheet 15 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

Table V — Scale Error and Altitude Reporting Test (Climb at 4000–6000 FPM)

TEST POINTS Test Set Altitude	Pilot PFD Altimeter	Copilot PFD Altimeter (if installed)	Standby Altimeter (if installed)	SCALE Tolerance ±(feet)	Transponder1 Mode C Altitude (if installed)	Transponder2 Mode C Altitude (if installed)	MODE C Tolerance ±(feet)	Pass/Fail
-1,000				±20			±125	
0				±20			±125	
500				±20			±125	
1,000				±20			±125	
1,500				±25			±125	
2,000				±30			±125	
3,000				±30			±125	
4,000				±35			±125	
6,000				±40			±125	
8,000				±60			±125	
10,000				±80			±125	
12,000				±90			±125	
14,000				±100			±125	
16,000				±110			±125	
18,000				±120			±125	
20,000				±130			±125	
22,000				±140			±125	
25,000				±155			±125	
30,000				±180			±125	
35,000				±205			±125	
40,000				±230			±125	
45,000				±255			±125	
50,000				±280			±125	

NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
62.	**Hysteresis (Dive) and After Affect Test**				
63.	***Purpose.*** *This test will verify the accuracy of the altimeters after a simulated dive from maximum altitude, to 50% of maximum, to 40%, to Atmospheric Pressure, in compliance with § 43 Appendix E, Table II.* *!! Begin dive test at maximum altitude within 15 minutes of reaching maximum altitude.* *Use Table VI to record readings, calculations, and pass/fail results.*				
64.	*Descend to 50% test point.* Adjust test set to descend to the 50% test point at greater than or equal to 6000 FPM until within 3000 feet. Approach at 3000 FPM. Hold at test point for 5 minutes before taking a reading.	Verify indicated altitude is within ±75 feet of the value recorded in Table V at the same altitude after holding at the 50% test point for 5 minutes.			
65.	*Descend to 40% test point.* Adjust test set to descend to the 40% test point at greater than or equal to 6000 FPM until within 3000 feet. Approach at 3000 FPM. Hold at test point for 1 minute before taking a reading.	Verify indicated altitude is within ±75 feet of the value recorded in Table V at the same altitude after holding at the 40% test point for 5 minutes.			
66.	Descend to test point. Adjust test set to descend to Atmospheric Pressure at greater than or equal to 6000 FPM until within 3000 feet. Approach at 3000 FPM and hold.	Within 5 minutes of reaching Atmospheric Pressure, verify indicated altitude is within ±30 feet of the pitot-static test set.			

Table VI — Hysteresis Test (dive from MAX ALT at greater than or equal to 6000 FPM)

TEST POINT	Calculated Test Point Altitude (from Section 16)	Pilot PFD Altimeter	Copilot PFD Altimeter (if installed)	Standby Altimeter (if installed)	Tolerance (±feet) from the value recorded in Table V	Pass/Fail
50% test point	▲				±75 after 5 min	
40% test point	▲				±75 after 1 min	
Atmospheric Pressure	▲				±30 within 5 min	

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NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
67.	**Friction Test (750 FPM Climb)**				
68.	***Purpose.*** *This test will verify the accuracy of the altimeters during a 750 FPM climb, in compliance with § 43 Appendix E, Table III.* ***Method.*** *Each static system shall be tested separately. Both the pitot and static ports of each system shall be connected to the pitot-static tester.* *All instruments in the aircraft under test shall be connected to their normal lines.* *Each altimeter in the aircraft under test will be compared to the altitude of the tester at test point altitudes, as specified in Table VII.* *Use Table VII to record all readings, calculations, and pass/fail results.*				
69.	Set pitot-static tester altimeter to 29.92 and airspeed to 0.				
70.	Press the BARO knob on PFD 1 and 2 for STD (29.92) setting.				
71.	Set the BARO on standby altimeter to 29.92.				
72.	*Use the pitot-static tester to apply suction to the static system in accordance with the test points listed in Table VII up to the maximum operating altitude of ►_____ feet for the aircraft. NOTE: Do not exceed the maximum altitude of the aircraft by more than 2500 feet.*				
73.	Adjust the pitot-static test set vertical speed to climb at 750 FPM between test points. Stop when the tester reaches each test point and record the indicated altitude on each altimeter.	Verify that the indicated altitude on each altimeter is within the FRICTION tolerance specified in Table VII.			
74.	End of Friction Test Adjust pitot-static tester to descend to atmospheric conditions.				

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Table VII — Friction Test (Climb at 750 FPM)

TEST POINTS Test Set Altitude	Pilot PFD Altimeter	Copilot PFD Altimeter (if installed)	Standby Altimeter (if installed)	FRICTION Tolerance ±(feet)			Pass/Fail
1,000				±70			
2,000				±70			
3,000				±70			
6,000				±70			
10,000				±80			
16,000				±90			
20,000				±100			
25,000				±120			
30,000				±140			
35,000				±160			
40,000				±180			
50,000				±250			

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NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
75.	**Airspeed Test**				
76.	***Purpose.*** *This test will verify the accuracy of the airspeed indicators at pressure altitude, in compliance with § 23.1323(a).* ***Method.*** *Each pitot-static system may be tested separately or together. The pitot and static ports of each system shall be connected to the pitot-static tester.* *All instruments in the aircraft under test shall be connected to their normal lines.* *Each airspeed indicator in the aircraft under test will be compared to the airspeed Indicator of the tester at various test point speeds, as specified in Table VIII.* *Use Table VIII to record readings, calculations, and pass/fail results.*				
77.	Set pitot-static tester altimeter to 29.92.				
78.	Press the BARO knob on PFD 1 and 2 for STD (29.92) setting.				
79.	Set the BARO on standby altimeter to 29.92.				
80.	*Use the pitot-static tester to apply pressure to the pitot system in accordance with the test points listed in Table VIII up to the maximum operating speed of ▶ _____ knots for the aircraft. NOTE: Do not exceed the maximum airspeed of the aircraft by more than 10 knots.*				
81.	Adjust the pitot-static test set to attain the speeds specified in Table VIII. Stop when the tester reaches each test point and record the indicated airspeed.	Verify that the indicated airspeed on each indicator is within the tolerance specified in Table VIII.			
82.		Verify redline displayed at the maximum airspeed per AFM at ▶ _____ kts			
83.		Verify overspeed alerter (if installed) activates within ±2 knots of redline.			
84.	Decrease pitot-static tester to atmospheric conditions.				

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Table VIII — Indicated Airspeed (at field elevation)

Test Set Indicated Airspeed	Pilot PFD Indicated Airspeed	Copilot PFD Indicated Airspeed	Standby Airspeed Indicator	Tolerance ±KIAS	Pass	Fail
40				±5		
50				±5		
60				±5		
70				±5		
80				±5		
90				±5		
100				±5		
110				±5		
120				±5		
130				±5		
140				±5		
150				±5		
160				±5		
170				±5		
180				±6		
190				±6		
200				±6		
210				±6		
220				±7		
230				±7		
240				±7		
250				±7		
260				±8		

NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
85.	**Shut Down**				
86.	Select EFIS off.				
87.	Shut down electrical power.				
88.	Select cabin lights off.				
89.	Disconnect Ground Power Unit from aircraft.				
90.	Disconnect pitot-static tester from aircraft.				

Note:
Remove any seals or adhesive from pitot-static ports upon completion of this test procedure.

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Table IX — Pressure-to-Altitude Conversion Chart

Altitude (ft)	Pressure (PSI)	Altitude (ft)	Pressure (PSI)	Altitude (ft)	Pressure (PSI)	Altitude (ft)	Pressure (PSI)
-1000	15.2348	17500	7.4915	36000	3.2966	54500	1.3549
-500	14.9634	18000	7.3389	36500	3.2183	55000	1.3227
0	14.6959	18500	7.1888	37000	3.1419	55500	1.2913
500	14.4323	19000	7.0412	37500	3.0673	56000	1.2606
1000	14.1726	19500	6.8961	38000	2.9945	56500	1.2307
1500	13.9166	20000	6.7534	38500	2.9234	57000	1.2015
2000	13.6644	20500	6.6131	39000	2.854	57500	1.173
2500	13.4159	21000	6.4752	39500	2.7862	58000	1.1451
3000	13.1711	21500	6.3397	40000	2.72	58500	1.1179
3500	12.9299	22000	6.2064	40500	2.6554	59000	1.0914
4000	12.6923	22500	6.0754	41000	2.5924	59500	1.0655
4500	12.4583	23000	5.9466	41500	2.5308	60000	1.0402
5000	12.2277	23500	5.8201	42000	2.4707		
5500	12.0007	24000	5.6958	42500	2.4121		
6000	11.777	24500	5.5736	43000	2.3548		
6500	11.5567	25000	5.4536	43500	2.2989		
7000	11.3398	25500	5.3356	44000	2.2443		
7500	11.1262	26000	5.2197	44500	2.191		
8000	10.9159	26500	5.1059	45000	2.139		
8500	10.7088	27000	4.9941	45500	2.0882		
9000	10.5049	27500	4.8843	46000	2.0386		
9500	10.3041	28000	4.7764	46500	1.9902		
10000	10.1065	28500	4.6705	47000	1.9429		
10500	9.9119	29000	4.5665	47500	1.8968		
11000	9.7204	29500	4.4644	48000	1.8518		
11500	9.5319	30000	4.3641	48500	1.8078		
12000	9.3464	30500	4.2657	49000	1.7649		
12500	9.1638	31000	4.1691	49500	1.723		
13000	8.9841	31500	4.0742	50000	1.682		
13500	8.8072	32000	3.9811	50500	1.6421		
14000	8.6332	32500	3.8898	51000	1.6031		
14500	8.462	33000	3.8001	51500	1.565		
15000	8.2935	33500	3.7121	52000	1.5279		
15500	8.1278	34000	3.6258	52500	1.4916		
16000	7.9648	34500	3.5411	53000	1.4562		
16500	7.8044	35000	3.458	53500	1.4216		
17000	7.6466	35500	3.3765	54000	1.3878		

Refer to Table IV GEOPOTENTIAL ALTITUDE, ENGLISH UNITS, contained in the document titled "U.S. Standard Atmosphere, 1976 (Stock No. 003-017-00323-0)

17. Ground Test Witness

I certify that I have performed or witnessed this Ground Test Procedure on the following aircraft:

TEST LOCATION			
TYPE AIRCRAFT	SERIAL No.	REGISTRATION No.	MODE S ADDRESS (if applicable).
WITNESS NAME	SIGNATURE		DATE

Comments:

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