



NATIONAL TRANSPORTATION SAFETY BOARD
Office of Aviation Safety
Washington, D.C. 20594

September 21, 2021

POWERPLANT GROUP CHAIRMAN'S FACTUAL REPORT

NTSB No: ENG21LA013

A. INCIDENT

Location: ONTARIO, CALIFORNIA

Date: January 29, 2021

Time: 10:49 pacific standard time

Aircraft: Aerotransportes Mas de Carga, S.A. de C.V (Mas Air Cargo), Boeing 767-338F, registration number N363CM, Flight MMA6853

B. POWERPLANT GROUP MEMBERS

Safety Board Group Chairman: Jean-Pierre Scarfo
Powerplant Lead Engineer
Washington D.C.

General Electric Members: Dale Denis
Product Safety Engineer
Commercial Flight Safety
Cincinnati, Ohio

James Easton
Product Safety Engineer
Commercial Flight Safety
Cincinnati, Ohio

**Aerotransportes Mas de Carga, S.A. de C.V.
Members:** Adrian Acosta
Manager of Safety, QA & AvSec
Mexico City, Mexico

Diego Moreno
Powerplant Engineer
Mexico City, Mexico

Boeing Members:

Brett Bennett
Boeing Air Safety Investigation
Seattle, Washington

Michael Germani
Propulsion Safety
Seattle, Washington

Federal Aviation Administration Members:

Patrick D. Lusch
Air Accident Investigator
Washington D.C.

Herman Mak
Engine Directorate Powerplants
Burlington, Massachusetts

C. SUMMARY

On January 29, 2021, about 10:49 pacific standard time, a Boeing 767-300, registration number N363CM, operated by Aerotransportes Mas de Carga, S.A. de C.V. as flight MMA6853, and powered by two General Electric (GE) CF6-80C2-B6 turbofan engines, experienced a right (No. 2) engine fire during initial climb right after takeoff from Los Angeles International Airport (LAX), Los Angeles, California. The flightcrew reported feeling vibrations in the airplane as the landing gear was cycled up, so a second cycling of the gear was attempted, and the vibration continued. Shortly after that, the flightcrew detected a strong odor consistent with burning and a right engine fire warning message displayed on the engine indicating and crew alerting system (EICAS) display. The flightcrew declared an emergency, performed the Quick Reference Handbook (QRH) engine fire procedures, which included shutting down the affected engine and discharging 1 fire suppression bottle, and diverted to the Ontario International Airport (ONT), Ontario, California for an uneventful single engine overweight landing with no reported injuries to any of the flightcrew members. The incident flight was a 14 *Code of Federal Regulations (CFR)* Part 129 cargo flight from LAX to the Mexico City International Airport (MEX), Mexico City, Mexico.

A Powerplant Group was formed and comprised of members from GE, Boeing, Federal Aviation Administration (FAA), and the National Transportation Safety Board (NTSB), and Aerotransportes Mas de Carga, S.A. de C.V. (referred to for the remainder of this report as Mas Air Cargo). Assisting in the on-scene part of the investigation were personnel from for Jett Pro Line Maintenance stationed at ONT and ONT operations center. The NTSB did not travel to ONT to examine and document the airplane and engine damage. Instead, Jett Pro and ONT operations center personnel provided initial pictures and descriptions of the damage. On January 30, 2021, the day following the event, Boeing field representatives arrived on-site to photo document the damage along with providing written field notes. Several days later, a GE representative arrived on-site to assist in the removal of the engine and securing it into a transportation stand for shipping to the GE Evendale, Ohio facility for examining and disassembly.

The on-scene examination of the airplane revealed thermal distress and fire damage to the inside of the right engine core cowl and thrust reverser. The entire outer portion of the right engine turbine exhaust nozzle was missing as was the aft portion of the inner sleeve; the forward portion of the inner sleeve remained attached to the engine. The right engine strut/pylon outdoor skirt in-line with the aft edge of what remained of the turbine exhaust nozzle exhibited gouging and scratch marks. The airplane exhibited some impact marks and dings on the underside of the right wing and flaps, on the vertical and horizontal stabilizer, and on the right cargo door; however, some of the impacts appeared old and it could not be confirmed if all the damage observed was from this incident.

The on-scene examination of the No. 2 engine revealed no signs of engine uncontainment but did show signs of low-grade thermal distress comprised of sooting and melted, consumed, and damaged cushion clamps, fire detector loop isolators, and electrical wire outer sheathing. Looking at the front of the engine, no damage was observed to the fan blades and the fan rotated freely by hand; the low pressure turbine rotated along with the fan. Looking through the engine exhaust: 1) all the low pressure turbine stage 5 blade roots were still installed in the disk and all the blades exhibited a combination of airfoil transverse fractures at various lengths and airfoil impact damage, tears, gouges, and missing airfoil material, 2) all the low pressure turbine stage 5 blade outer shroud segments were still present and exhibited a combination of gouging, missing material, heavy rub, and the honeycomb was worn down to the backing strip, 3) low pressure turbine stage 5 nozzles segments were present and exhibited a combination of trailing edge airfoil impact damage, gouges, tears, and missing material, and 4) the turbine rear frame exhibited a combination of multiple cracks, tears, openings, and punctures holes; no signs of low pressure turbine debris penetrating through the turbine rear frame skin was observed. The oil level in

the oil tank was low and debris was noted on the magnetic chip detector. When the integrated drive generator was removed in preparation for transportation of the engine, a fractured oil supply line that runs from the aft side of accessory gearbox behind the integrated drive generator was observed. The No. 2 engine, engine serial number 695440, was removed and shipped to GE for disassembly and examination.

The Powerplant Group comprised of members from GE, FAA, Mas Air Cargo, and the NTSB convened at the GE Evendale facility from February 22-26, 2021, to perform a detailed examination of the incident engine. The pre-disassembly/induction borescope inspection found the following: 1) no damage was noted to the combustor section, 2) all the high pressure turbine stages 1 blades were present, and no impact damage was noted but many exhibited thermal distress consistent with mid-span core burn-through, 3) first indications of event related turbine damage or distress was to the leading edge of the low pressure turbine stage 1 blades, and 4) damage was observed throughout all five stages of the low pressure turbine rotor.

Each stage of the low pressure turbine exhibited combinations of the following damage to various degrees: 1) significant amount of leading and trailing edge blade and nozzle segment airfoil impact damage, material loss, and thermal distress except for the stage 1 nozzle segments that only exhibited minor thermal distress and minor trailing edge impact damage, 2) all blade roots were installed in their respective disk and all were fractured transversely across the airfoil, 3) blade outer shroud segments were gouged, the honeycomb worn down to the backing strip, and were thermally distressed, 4) disk rub and contact marks, 5) the rotating interstage air seal knife edges contact rub and material loss, 6) stationary honeycomb interstage air seal land trenching, gouging, material loss, and attachment bolt contact rub, and 7) consistent throughout each stage was evidence of contact between rotating and stationary hardware.

With the impact damage, contact rub, and thermal distress observed to each of the low pressure turbine stages, a few items were of particular interest. One stage 5 nozzle segment, labelled as the No. 3 segment, was missing two complete adjacent airfoils labeled airfoil Nos. 5 and 6; each stage 5 nozzle segment is comprised of 6 individual airfoils. No other nozzle segments from any other low pressure turbine stage were missing airfoils. Although all the low pressure turbine blades were fractured transversely across the airfoil at various lengths, none were fractured at or close to their platform and all the blade roots remained in the disk. The low pressure turbine case exhibited numerous outward impacts, bulges, tears, and a small hole in line with the stage 4 blades; none appeared to be a penetration hole where low pressure turbine passed through it.

GE evaluated of the material condition of the low pressure turbine hardware, the turbine rear frame, and the fractured oil supply line at their metallurgical laboratory in Evendale, Ohio while Boeing evaluated the material condition of the exhaust hardware, which included the turbine exhaust sleeve and the exhaust plug, at their Boeing Equipment Quality Analysis laboratory in Seattle Washington.

The GE metallurgical analysis of the cracks and tears in the turbine rear frame and the turbine exhaust were due to overstress; no anomalies or pre-existing defects were noted that would have contributed to their condition. The fracture surfaces of the oil supply tube exhibited plastic deformation and a planar fracture consistent with secondary high amplitude fatigue/cyclic tensile loading. None of the low pressure turbine blade fracture surfaces showed signs of fatigue but features consistent with an overload failure. Signs consistent with high cycle fatigue growth were found on the fracture surfaces of airfoil Nos. 5 and 6 of the low pressure turbine stage 5 nozzle segment No. 3; the fatigue cracks propagated from both the leading and trailing edges of the missing airfoils. In the interpretable areas (not damaged area), GE found no material anomalies in the base material, aside from the areas of intergranular oxidation.

Low pressure turbine stage 5 nozzle segment No. 3, part number 9367M85P10, serial number HCM21194, had an extensive overhaul repair conducted in 2014 and at the time of the event had been in operational service for 10,232 hours and 4,068 cycles since the overhaul repair was performed. Due to the leading and trailing edge impact damage on airfoils Nos. 5 and 6 of the low pressure turbine stage 5 nozzle segment No. 3 at the fracture locations, no determination could be positively made on whether a repair had been previously performed in that area, so no assessment of the repair conformity was made.

The Boeing metallurgical analysis of the turbine exhaust sleeve revealed that all the fractures were composed entirely of ductile separation fracture mode, consistent with a single event, and no other anomalies were observed contributing to the fractures.

GE performed a rotor imbalance and deflection analysis to determine the amount of radial deflection each stage of the low pressure turbine would be anticipated to experience based on the condition of the hardware as documented during the engine exam. Since the exact condition of the low pressure turbine hardware for any point in time during the event was not known and the vibration monitoring system only registers a maximum value of 5 cockpit units, the analysis does not provide the exact vibration levels or deflections at the time of the event or during the engine deceleration and may not represent the worst-case or maximum rotor imbalance or deflection (hence the highest engine vibration level) that the engine experienced during the incident flight. However, the analysis did provide a general assessment of the vibration levels and the amount of deflection the low pressure turbine was expected or predicted to experience after the event as the engine decelerated. The results of the analysis showed that the amount of expected low pressure turbine rotor radial deflection was several times greater than the nominal radial blade running clearances at takeoff for each of the low pressure turbine rotor stages. The observed gouging and heavy wear down to the backing strip of the honeycomb on the blade outer shroud segments for each stage of the low pressure turbine was consistent with the loss of blade clearance as predicted by the rotor imbalance and deflection analysis. The analysis also predicted that the low pressure turbine stage 5 would be the stage that experienced the most deflection and that the deflection would be the greater not at the takeoff/climb low pressure turbine rotor speed but as it decelerates from the accumulation of damage and the pilot's action to shut down the engine.

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TABLE OF ACRONYMS

°C	Degree Celsius	IGO	Intergranular Oxidation
°F	Degree Fahrenheit	IPC	Illustrated Parts Catalog
A/T	Auto Throttle	ITT	Interstage Temperature
AC	Advisory Circular	LAX	Los Angeles International Airport
ADH	Activated Diffusion Healing	lbs./hr.	Pounds per Hour
AFM	Aircraft Flight Manual	LE	Leading Edge
AGB	Accessory Gearbox	LLP	Life Limited Parts
ALF	Aft Looking Forward	LPC	Low Pressure Compressor
AMM	Aircraft Maintenance Manual	LPT	Low Pressure Turbine
ARFF	Aircraft Rescue and Fire Fighting	MAS AIR CARGO	Aerotransportes Mas de Carga, S.A. de C.V.
ATL	Hartsfield-Jackson Atlanta International Airport	MEC	Main Engine Control
AVM	Airborne Vibration Monitoring	MEX	Mexico City International Airport
BR&T	Boeing Research and Technology	N1	Low Pressure Rotor Rotation Speed in %
BSI	Borescope Inspection	N2	Low Pressure Rotor Rotation Speed in %
CDP	Compressor Discharge Pressure	NTSB	National Transportation Safety Board
CDU	Center Display Unit	OD	Outer Diameter
CFR	Code of Federal Regulations	ONT	Ontario International Airport
CRF	Compressor Rear Frame	PBM	Powerplant Buildup Manual
CSLSV	Cycles Since Last Shop Visit	PMC	Power Management Control
CSN	Cycles Since New	PN	Part Number
CU	Cockpit Unit	POM	Polyoxymethylene
DAL	Delta Air Lines	PSI	Pounds per Square Inch
DER	Designated Engineering Representative	QRH	Quick Reference Handbook
DFDAC	Digital Flight Data Acquisition Unit	RPM	Revolution per minute
EICAS	Engine Indicating and Crew Alerting System	RSS	Repair Source Substantiation
EM	Engine Manual	SEM	Scanning Electron Microscope
ENG	Engine	SN	Serial Number
EQA	Equipment Quality Analysis	STC	Supplemental Type Certificate
ESM	Engine Shop Manual	T/O	Takeoff
ESN	Engine Serial Number	TBC	Thermal Barrier Coating
FAA	Federal Aviation Administration	TCDS	Type Certificate Data Sheet
FCOM	Flight Crews Operations Manual	TE	Trailing Edge
FCTM	Flight Crew Training Manual	TEC	Turbine Exhaust Case
FDAU	Flight Data Acquisition Unit	TECHOPS	Technical Operations
FDR	Flight Data Recorder	TLA	Technical License Agreement
GE	General Electric	TR	Thrust Reverser
HCF	High Cycle Fatigue	TRA	Throttle Resolver Angle
HP	High Pressure	TRF	Turbine Rear Frame
HPC	High Pressure Compressor	TSLSV	Time Since Last Shop Visit
HPT	High Pressure Turbine	TSN	Time Since New
ID	Inner Diameter	WOW	Weight on Wheels
IDG	Integrated Drive Generator	XRF	X-ray Fluorescence

D. DETAILS OF THE INVESTIGATION

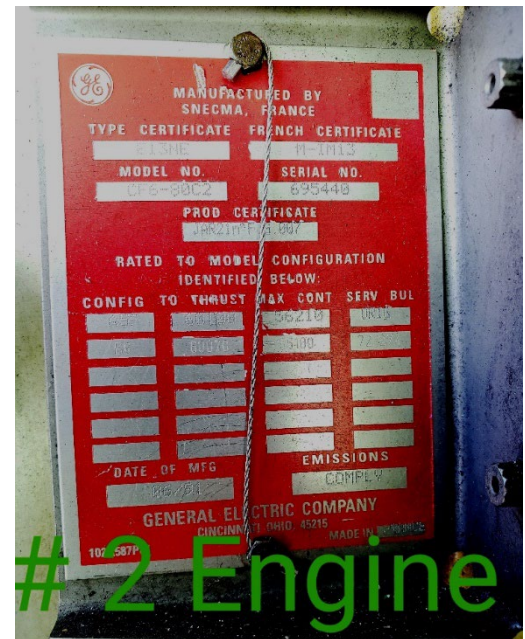
1.0 ENGINE AND AIRPLANE INFORMATION

1.1 AIRPLANE INFORMATION

The incident airplane was a Boeing 767-300ER BDSF, registration number N363CM, line number 319, serial number (SN) 24853, block number VN086, produced in 1990, operated by Mas Air Cargo, and owned by Cargo Aircraft Management; the original operator of the airplane was Qantas.

1.2 RIGHT (No.2) ENGINE HISTORY

The No. 2 engine installed on the incident airplane, N363CM, was a GE CF6-80C2-B6 turbofan engine, engine serial number (ESN) 695440 (**PHOTO 1**). At the time of the engine fire event, ESN 695440 had accumulated 54,187 hours time since new (TSN), 20,656 cycles since new (CSN), 1,649 hours time since last shop visit (TSLSV), and 669 cycles since last shop visit (CSLSV).



**PHOTO 1: No. 2 ENGINE DATA PLATE
ESN 695440**

PHOTO COURTESY OF JETT PRO

The last shop visit occurred in August-December 2019 at the Delta Airlines (DAL) TechOps Center located at the Hartsfield-Jackson Atlanta International Airport (ATL), Atlanta, Georgia. According to DAL maintenance shop report for ESN 695440, the engine was removed from position 1 from airplane registration N364CM on August 3, 2019, and sent to Delta TechOps for repair due to life limited parts expiration.¹ At the time the engine was removed from N364CM, the engine had accumulated 52,538 hours TSN, 19,987 CSN, 8,588 hours TSLSV, and 3,399 CSLSV. A post-test cell run borescope inspection found foreign object damage (FOD) to several LPT stage 1 blades that were beyond allowable limits. The engine was returned to the shop, the LPT was removed and disassembled, and re-inspection of the LPT identified two LPT stage 1 blades that needed to be replaced and FOD damage on the LPT stage 1 disk that

1. Approving Civil Aviation Authority/Country: FAA/United States		2. AUTHORIZED RELEASE CERTIFICATE FAA Form 8130-3, AIRWORTHINESS APPROVAL TAG		3. Form Tracking Number: D282201912162030	
4. Organization Name and Address: Delta TechOps 1775 MH Jackson Service Rd., Atlanta, Georgia, USA 30354; Repair Station Certificate DALR026A				5. Work Order/Contract/Invoice No. 501664809	
6. Item:	7. Description:	8. Part Number:	9. Quantity:	10. Serial Number:	11. Status/Work:
1	Engine	CF6-80C2B6	1	695440	Repaired
12. Remarks: Full release: ETOP's approved in accordance with Boeing CMP Document D6T11604 Rev AE. Complied with repairs as per workscope 58332-04 rev J 12/02/2019. Complied with repairs and tests in accordance with BOEING CF6-80C2 Engine Manual (GEK92451), OEM manual with Delta Supplemental info, Release 88.05 pub 38622 10/17/2019. Complied with long term preservation, due Dec 14, 2020. Complied with post test borescope, no major findings. TT 52/538/19 TC 19,987					
Certifies that the work specified in block 11/12 was carried out in accordance with EASA Part-145 and in respect to that work the component is considered ready for release to service under EASA Part-145 Approval Number: EASA.145.4380.					
13a. Certifies that the items identified above were manufactured in conformity to: ☐ Approved design data and are in a condition for safe operation. ☐ Non-approved design data specified in Block 12.			14a. ☒ 14 CFR 43.9 Return to Service ☒ Other regulation specified in Block 12 Certifies that unless otherwise specified in Block 12, the work identified in Block 11 and described in Block 12 was accomplished in accordance with Title 14, Code of Federal Regulations, part 43 and in respect to that work, the items are approved for return to service.		
13b. Authorized Signature:		13c. Approval/Authorization No.:		14b. Authorized Signature:	
				DALR026A	
13d. Name (Typed or Printed):		13e. Date (dd/mm/yyyy):		14d. Name (Typed or Printed):	
				16/DEC/2019	
User/Installer Responsibilities					
It is important to understand that the existence of this document alone does not automatically constitute authority to install the aircraft engine/propeller/article. Where the user/installer performs work in accordance with the national regulations of an airworthiness authority different than the airworthiness authority of the country specified in Block 1, it is essential that the user/installer ensures that his/her airworthiness authority accepts engine/propeller/article from the airworthiness authority of the country specified in Block 1. Statements in Blocks 13a and 14a do not constitute installation certification. In all cases, aircraft maintenance records must contain an installation certification issued in accordance with the national regulations by the user/installer before the aircraft may be flown.					

PHOTO 2: ESN 695440 FAA 8130 FORM DATED DECEMBER 16, 2019

¹ The following LLP were replaced during the last shop visit: Fan Disk, Fan Mid Shaft, HPC stages 2 and 10 disks, HPC stage 11-14 spool, HPC compressor discharge pressure (CDP) rotating seal, HPT stage 1 disk, LPT stages 1 – 5 disks, and the LPT shaft.

required replacement. The engine was reassembled, a test cell run was performed, and the post-test cell run borescope inspection found no internal damage. On December 16, 2019, DAL issued FAA form 8130-3 Airworthiness Approval Tag Authorization Release Certificate (**PHOTO 2**). After the repair, the engine was delivered to Mas Air Cargo on July 21, 2020, when it was installed on the right hand (No. 2) position of airplane N363CM.

1.3 ENGINE DESCRIPTION

The incident airplane was powered by two GE CF6-80C2-B6 turbofan engines (**FIGURES 1 and 2**). The CF6-80C2-B6 turbofan engine is a dual-rotor, axial-flow high bypass turbofan that features an integrated front fan and 3-stage low pressure compressor (LPC) (this is also referred to as the booster) driven by a 5-stage low pressure turbine (LPT), an annular combustor, and a 14-stage high pressure compressor (HPC) driven by a 2-stage high pressure turbine (HPT). The engine is equipped with a Woodward Governor main engine control (MEC) and a GE Power Management Control (PMC) unit. Stations are identified in numeric sequence from front to rear (**FIGURE 3**).

According to the engine's FAA Type Certificate Data Sheet (TCDS) E13NE, Revision 26, dated September 11, 2014, the engine has a maximum takeoff (5 minutes) at sea level static thrust rating of 60,070 pounds, flat-rated² to 86°F (30°C) and a maximum continuous at sea level static thrust rating of 56,100 pounds flat-rated to 77°F (25°C). For the CF6-80C2 (all models), the maximum low pressure rotor speed (N1) is 3,854 revolutions per minute (rpm), which equals 117.5% N1 and maximum high pressure rotor speed (N2) is 11,055 rpm, which equals 112.5% N2. The CF6-80C2-B6 TCDS was certificated on February 20, 1987, and based on the engine serial number³, the engine was manufactured by SNECMA (currently SAFRAN Aircraft Engines) under a production agreement.

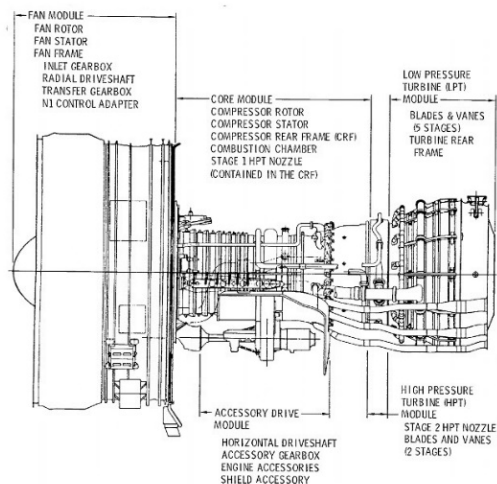


FIGURE 1: ENGINE MODULES

FIGURE COURTESY OF GE

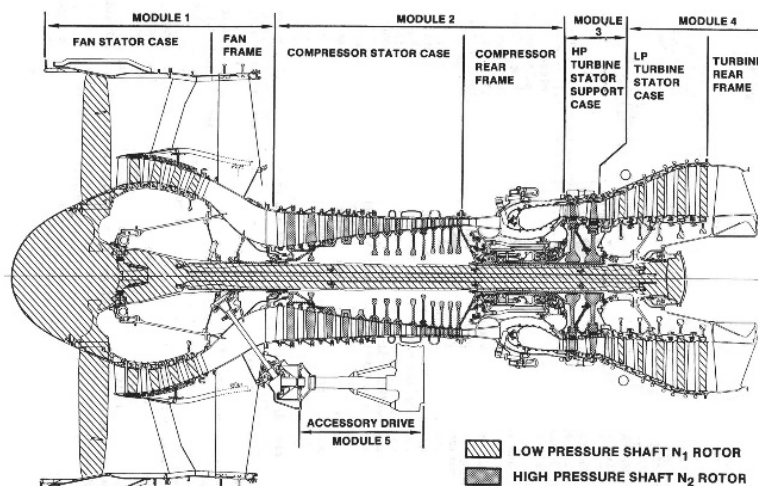


FIGURE 2: CROSS-SECTION OF ENGINE

FIGURE COURTESY OF GE

² Flat-rated to a specific temperature indicates that the engine will be capable of attaining the rated thrust level up to the specified inlet temperature.

³ Engines produced by SNECMA (SAFRAN Aircraft Engines) under a production agreement are designed as serial numbers 695- for CF6-80C2 PMC engines.

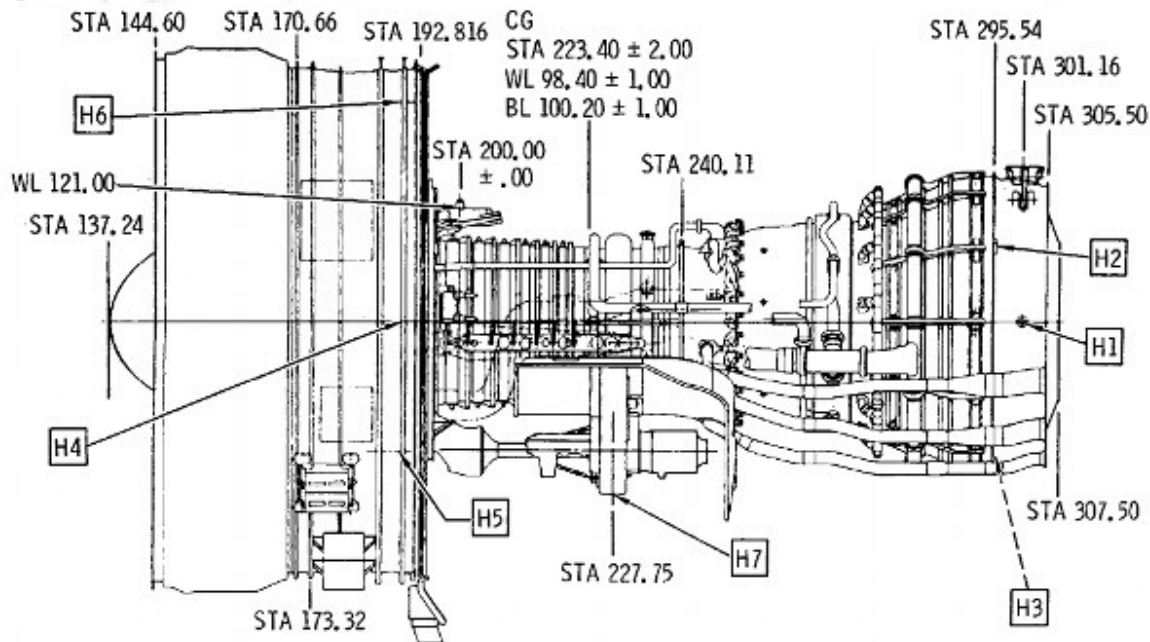


FIGURE 3: ENGINE STATION IDENTIFICATION

FIGURE COURTESY OF GE

All directional references to front and rear; right and left; top and bottom; and clockwise and counterclockwise are made aft looking forward (ALF) as is the convention. All numbering is in the circumferential direction starting with the No. 1 position at the 12:00 o'clock position or immediately clockwise from the 12:00 o'clock position and progressing sequentially clockwise ALF. The direction of rotation of the engine rotors is clockwise ALF.

1.4 DESCRIPTION OF ENGINE NACELLE

The engine inlet cowl (also known as the inlet duct), fan cowl, reverser-fan duct cowl (consisting of the thrust reverser (TR) and fan duct cowl), core cowl, and exhaust nozzle, comprise the boundaries of the engine nacelle and consist of fixed and hinged components. The fixed components include the inlet cowl, and exhaust nozzle; while the hinged components include the fan cowl, fan reverser, and core cowl (**FIGURE 4**). The left TR half incorporates a ventilation air outlet panel at the 6:00 o'clock position used for vent flow of the nacelle and is also used in the event of an over-pressurization event. The left TR half incorporates the integrated drive generator (IDG) access panel and is not intended to be used for an over-pressurization event. The inlet duct is bolted to the engine fan case and the exhaust nozzle to the turbine rear frame (TRF). The fan cowl, reverser-fan duct cowl (subsequently referred to simply as the TR), and core cowl are each in two halves, hinged at either side of the strut, and joined by latch hooks on the bottom centerline. The TR inner barrel inner skin is coated with MA25S red/orange-colored thermal coating⁴ for fire protection. The inlet, fan cowl, and core cowl are supplied by Collins Aerospace, the TR is supplied by Middle River Aircraft Systems (MRAS)⁵, the exhaust hardware is supplied by Magellan Aerospace, and the engine is supplied by GE. Boeing is responsible for the integration of the individual components and installation in the airplane.

⁴ MA25S is a Polyoxymethylene (POM) thermoplastic that has high heat resistance and is produced under the trade name Duracon®. The 'S' in MA25S stands for standard grade.

⁵ MRAS was acquired by ST Engineering in 2019.

Although the turbine exhaust sleeve and the exhaust plug (exhaust plug is also commonly referred to as the exhaust centerbody) are attached to the engine they are both Boeing supplied components. The turbine exhaust sleeve is attached to the outer rear flange of the engine's turbine rear frame (TRF) and the exhaust plug is bolted to the TRF hub aft flange. The turbine exhaust nozzle is comprised of an inner and outer sleeve, an aft ring at the location where the two skins meet in the back and a seal assembly that seals against the core cowl. The exhaust components provide a smooth exit path for turbine exhaust. The exhaust sleeve and plug together form a convergent-divergent nozzle that aids in producing thrust.

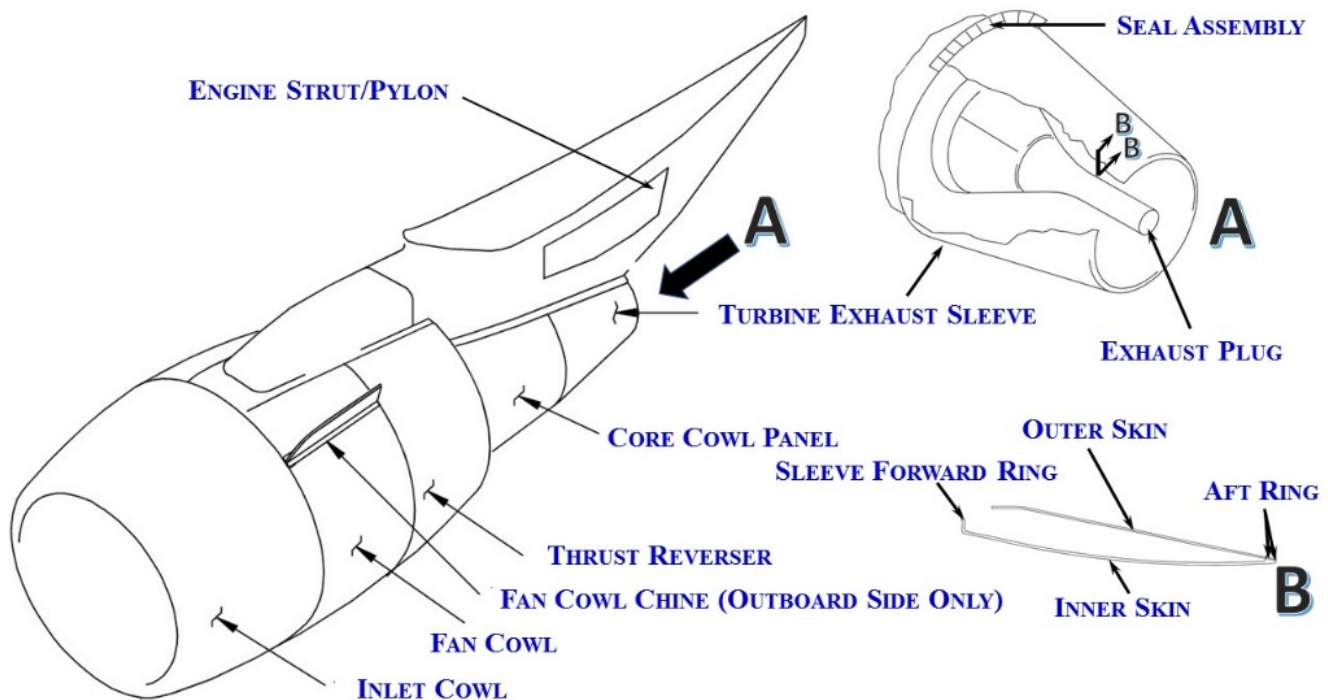


FIGURE 4: BOEING 767 NACELLE NOMENCLATURE AND LOCATION

FIGURE COURTESY OF BOEING

2.0 INITIAL ON-SITE EVALUATION OF THE NO. 2 ENGINE AT ONT

The NTSB did not travel to ONT for an on-site examination and evaluation of the event airplane and engine damage. The following written documentation in this section was provided by Boeing field representatives and from Jett Pro mechanics providing engine and airplane maintenance support for Mas Air Cargo. Photos in this section are from the Boeing field representative, a GE field representative who supported the effort to remove the engine from the airplane and secure it in a transportation stand, and from Jett Pro and Ontario Airport Operations personnel supporting the on-scene part of the investigation.

2.1 AIRPLANE DAMAGE ASSESSMENT

2.1.1 Nacelle Damage

The fan cowl, thrust reverser cowls, and the core cowls were closed, fully latched, and secure; the pressure relief door⁶ on the outboard core cowl was open and still attached (**PHOTO 3**), and the IDG access door was closed. When all the cowls were unlatched, all the latch hardware - latch hooks and latch keepers - appeared to be intact and in good condition (**PHOTO 4**). No debris exit penetrations holes or burn-through holes were observed on any of the cowls.

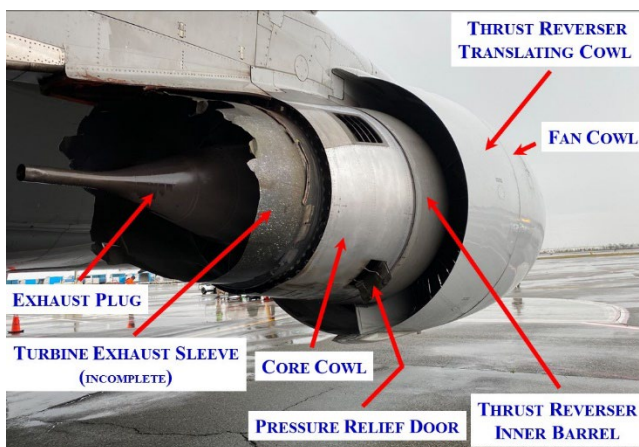


PHOTO 3: INITIAL ON-SCENE OF NO. 2 ENGINE
PHOTO COURTESY OF JETT PRO

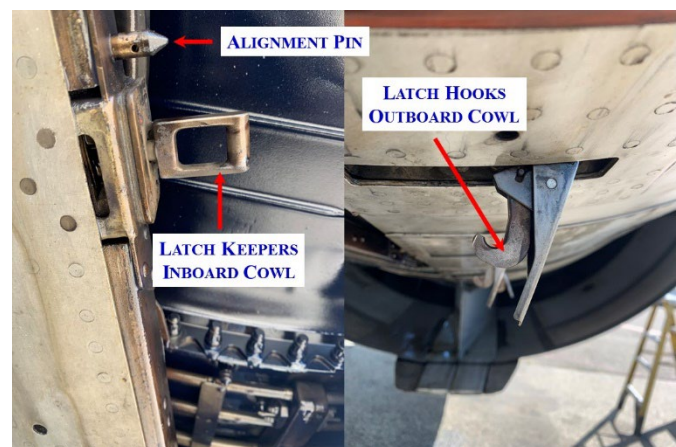


PHOTO 4: CORE COWL LATCHING MECHANISM
PHOTO COURTESY OF BOEING

The TR exhibited thermal distress to the inner barrel inner skin MA25S red/orange-colored thermal coating on both the inboard (**PHOTO 5**) and outboard (**PHOTO 6**) halves; the thermal distress was localized towards the rear of the TR and consisted of discoloring and flaking/blistering of the coating (**PHOTO 5** – yellow outline). Debris was found in the frame of the IDG access door (**PHOTO 7**).

⁶ The core cowl pressure relief door is located on the right cowl only.

**PHOTO 5: LEFT TR DISTRESS**

PHOTOS COURTESY OF BOEING

**PHOTO 6: RIGHT TR DISTRESS**

PHOTO COURTESY OF BOEING

**PHOTO 7: DEBRIS COLLECTED IN THE TR**

PHOTOS COURTESY OF JETT PRO

The left core cowl was unable to be opened fully; the hold-open rod was found barely attached to the cowl structure at the receiver end of the rod and detached at the cowl structure bracket end (**PHOTO 8**). Significant damage was observed that included cracked circumferential frames, missing pieces of the rear flange, missing and melted sections of the rear circumferential channel just forward of the core cowl aft hat sections, blistered and peeled primer paint, and missing sections of the rear flange (**PHOTO 9**). Some debris was found in the leading edge frame channels.

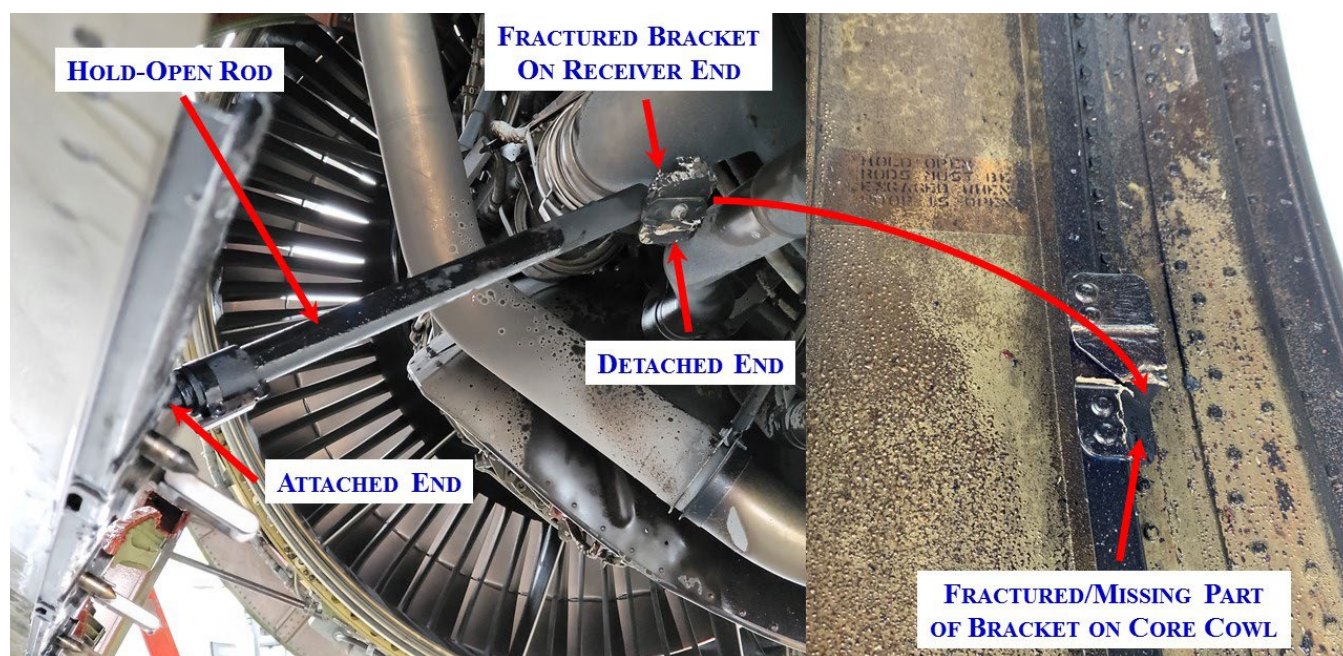


PHOTO 8: FRACTURE LEFT CORE COWL HOLD-OPEN ROD BRACKET

PHOTOS COURTESY OF BOEING

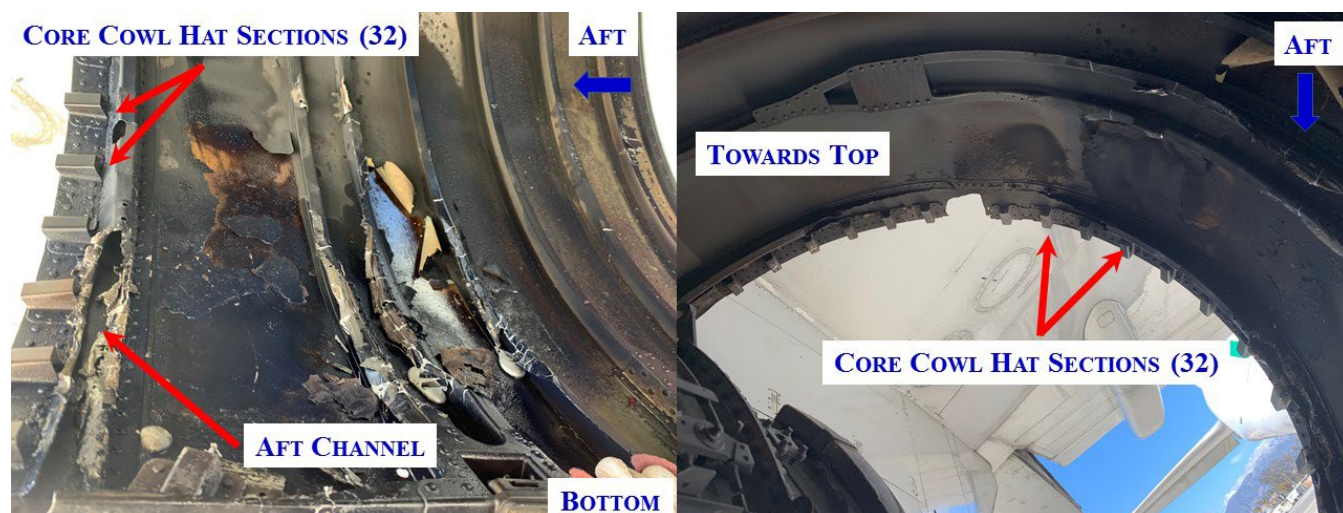


PHOTO 9: LEFT CORE COWL THERMAL DISTRESS

PHOTOS COURTESY OF BOEING

The right core cowl was opened fully using the hold-open rod; the hold-open rod was still attached and secured to the outboard cowl, but the bracket was fractured (**PHOTO 10**). Similar thermal distress that was also observed on the left core cowl was also observed on the right core cowl that included cracked frames, blistered primer paint, and eroded and missing aft channel near the hat sections (**PHOTO 11**). Due to distortion, the core cowls were not able to fully close after they were initially opened.

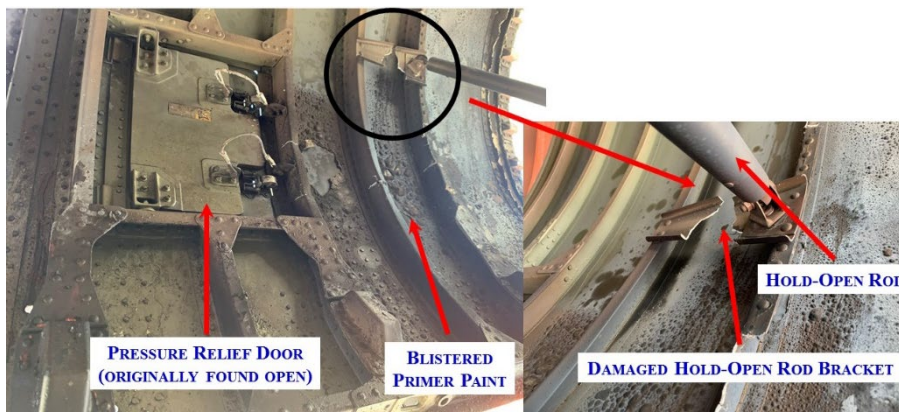


PHOTO 10: FRACTURE HOLD-OPEN DOOR BRACKET

PHOTOS COURTESY OF BOEING



**PHOTO 11: AFT END
THERMAL DISTRESS**

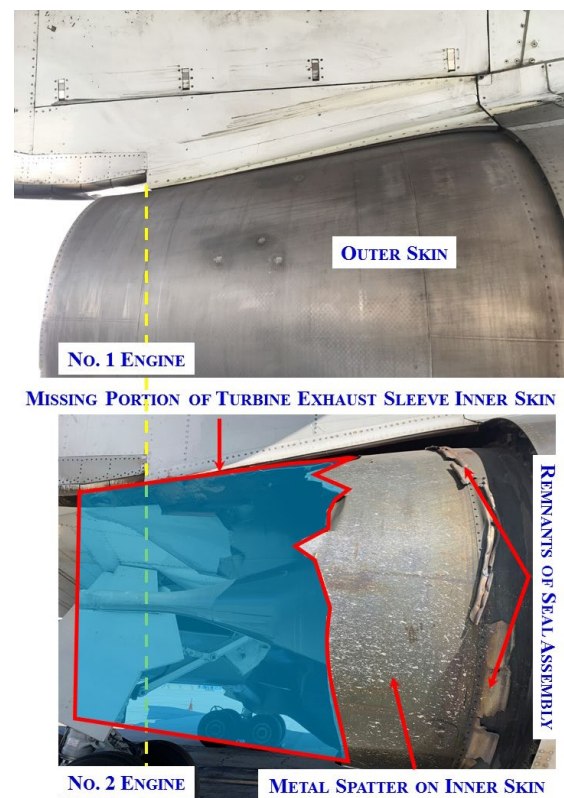
PHOTO COURTESY OF BOEING

The entire outer portion of the double walled turbine exhaust nozzle assembly was missing. What remained was the forward half of the turbine exhaust inner sleeve which remained attached to the TRF; the remnants of the inner sleeve were torn, impact marks were observed on the outer surface, and oil was puddled on the inside inner surface at the 6:00 o'clock position (**PHOTO 12**). Remnants of damaged and torn seal assembly remained; the exhaust plug exhibited several impact dents along its length and no holes were noted.

The engine strut/pylon outdoor skirt, in-line with the aft edge of what remained of the turbine exhaust nozzle, exhibited gouging and scratch marks.

2.1.2 Airframe Damage

The right cargo door exhibited impact damage and a gouge/hole; the inner door frame and inner cargo door wall showed no signs of penetration into the frame or cargo compartment (**PHOTO 13**). The underside of the right wing and the flap fairing exhibited several dings and scrape marks and as well as the flap directly behind the No. 2 engine; the flap had small metallic piece embedded in the outer skin. Some of the impact marks appeared old and it could not be confirmed if all the damage observed was from this event. The vertical stabilizer exhibited a gouge but could not confirm if the damage was from this event (**PHOTO 14**). The right horizontal stabilizer exhibited small dings near the root and on the leading edge (LE) about mid span.



**PHOTO 12: SIDE-BY-SIDE COMPARISON
OF TURBINE EXHAUST NOZZLE**

PHOTOS COURTESY OF FAA

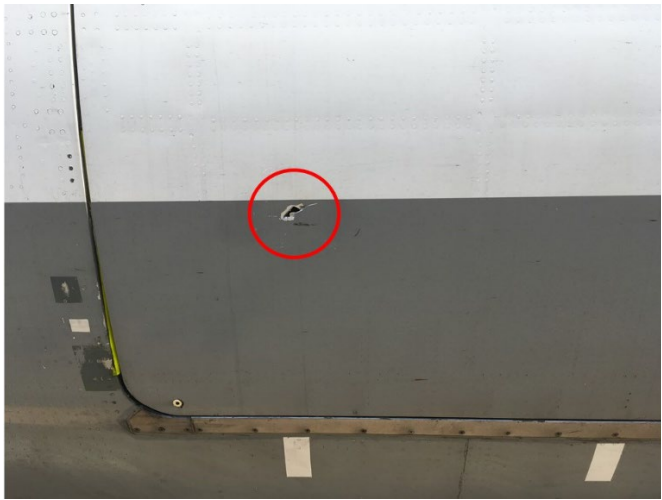
**PHOTO 13: RIGHT CARGO DOOR IMPACT**

PHOTO COURTESY OF BOEING

**PHOTO 14: VERTICAL STABILIZER DAMAGE**

PHOTO COURTESY OF BOEING

2.2 NO. 2 ENGINE – SERIAL NUMBER 695440 – DAMAGE ASSESSMENT

The fan was rotated by hand and the LPT rotor rotated along with it; the low rotor system rotated smoothly. No damage was observed to the fan blades; they all appeared in good condition.

With the TR and the core cowls open, the outside of the engine was exposed for examination. The oil tank sight gage, which is located on the right side of the forward fan case, no signs of oil, indicating that the oil level was below the minimum dispatch level of 23 quarts (**PHOTO 15**). The lube and scavenge pump assembly is located on the forward side of the accessory gearbox at about the 6:30 o'clock position. A magnetic chip detector is installed in the common scavenge discharge flow line from the lube and scavenge pump assembly; the chip detector is located on the left side of the engine. The magnetic chip detector was removed, and debris was noted on the detector (**PHOTO 16**). The magnetic chip detector was reinstalled for transportation. In preparation for transportation of the engine, the IDG was removed from the aft side of the accessory gearbox. A fractured oil supply (pressure) line, which runs from the aft side of accessory gearbox (AGB) behind the IDG at the 7:00 o'clock position, was noted (**PHOTO 17**).

**PHOTO 15: OIL TANK SIGHT GAGE**

PHOTO COURTESY OF JETT PRO

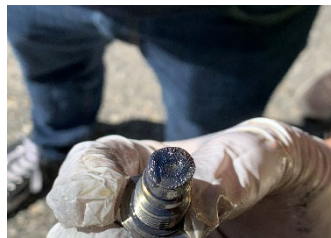
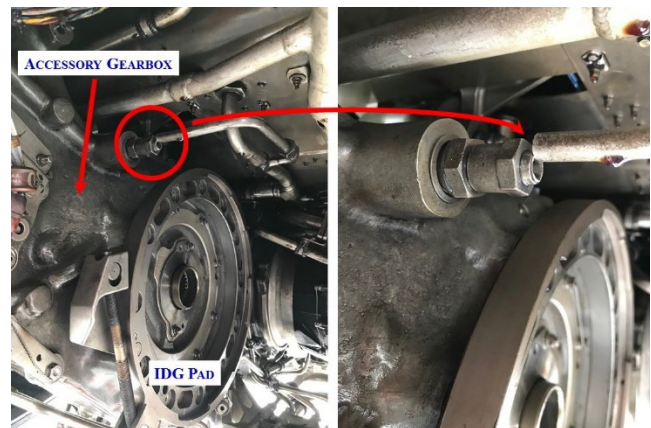
**PHOTO 16: MAGNETIC CHIP DETECTOR**

PHOTO COURTESY OF BOEING

**PHOTO 17: FRACTURE OIL LINE**

PHOTOS COURTESY OF GE

Looking through both the turbine exhaust sleeve and the engine's TRF, the LPT stage 5 blades were visible; all the blade roots were still installed in the disk and all the blades exhibited a combination of airfoil transverse fractures at various lengths and airfoil impact damage, tears, gouges, and missing airfoil material. All the LPT stage 5 blade outer shroud segments were still present and installed in the LPT stator case and exhibited a combination of gouging, missing material, heavy rub, and the honeycomb was worn down to the backing strip. Several metal pieces were found resting on the LPT stage 5 blade shroud segments at the 6:00 o'clock position; the material was consistent with LPT rotor and stator material. The LPT stage 5 nozzles segments were also visible, and they were all present and exhibited a combination of trailing edge (TE) airfoil impact damage, gouges, tears, and missing material (**PHOTO 18**).

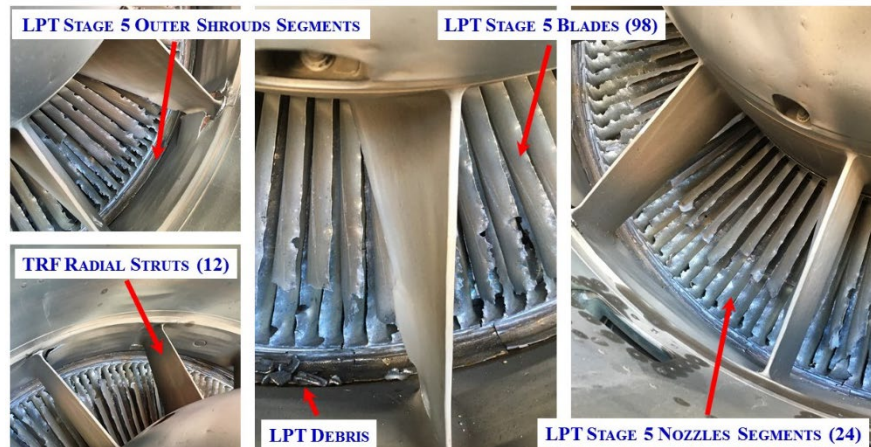


PHOTO 18: LPT STAGE 5 DAMAGE

PHOTOS COURTESY OF BOEING

With the engine removed and placed in a shipping stand, a hole with debris stuck in it was observed at about the 10:30 o'clock position in the LPT stator case; the location of the hole appears to be in-line with the LPT stage 4 rotor (**PHOTO 19**). No signs of any LPT debris penetrating through the LPT stator case out skin was observed.

The damage to the TRF was comprised of cracks around the outer skin bosses, radial struts that had pushed through the outer skin creating gaps, outer skin tears, and the radial struts themselves exhibited leading edge damage (**PHOTO 20**). No signs of any LPT debris penetrating through the TRF out skin; however, molten metal splatter deposits were observed on what remained of the turbine exhaust nozzle inner skin wall (See **PHOTO 12** lower picture) as well as on the TRF outer surface at the locations where the TRF skin was pushed outwards at the 4:00 o'clock and at the 9:00 o'clock positions where the ground handling/shipping bosses were located. The low pressure (LP) recoup manifold, located at the 9:00 o'clock position, was covered in metal spatter in-line axially with the opening in the case at the 9:00 o'clock ground handle boss. The TRF exhibited the following cracks or openings: 1) a crack at the ground handling boss at the 3:00 o'clock position, 2) the case skin pushed outwards creating an opening at the 4:00 o'clock position, 3) a crack at the No. 6 bearing oil supply boss at the 5:00 o'clock position, 4) the case skin was pushed outwards creating a gap at 8:00 o'clock position, and 5) the case skin was pushed outwards creating a gap at 9:00 o'clock position at the ground handling hole where metal spatter was noted on the LP recoup manifold.

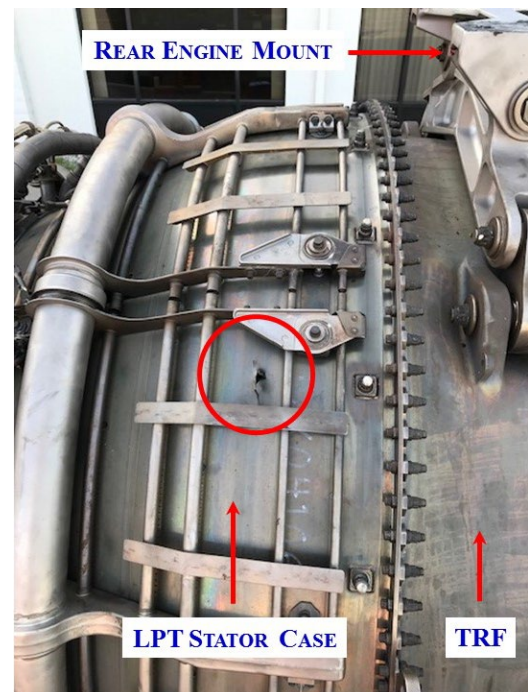


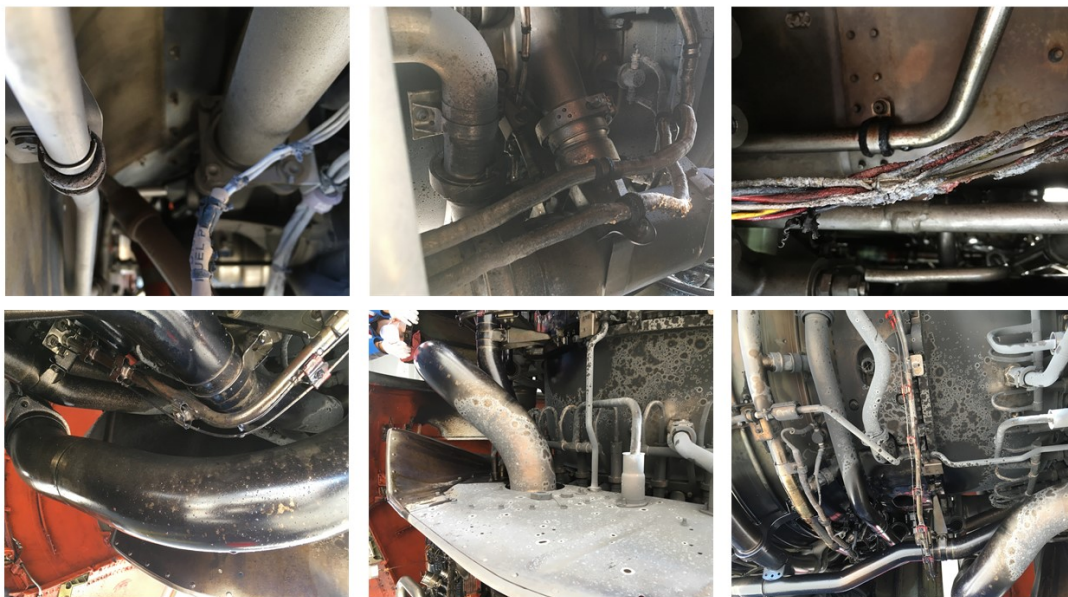
PHOTO 19: LPT STATOR CASE HOLE

PHOTO COURTESY OF GE

**PHOTO 20: TURBINE REAR FRAME DAMAGE**

PHOTOS COURTESY OF BOEING

The outside of the engine exhibited low grade thermal distress comprised of sooting and melted, consumed, and damaged cushion clamps, fire detector loop isolators, and electrical wire outer sheathing (**PHOTO 21**). All the fire loops appeared to be intact and continuous, the only distress noted was to some of the fire loop isolators. Thermal distress was only noted above and aft of the engine heat shield; no thermal distress was noted under the heat shield in what is designated a fire zone. The components under the heat shield appeared wetted coin the area between the 4:00 and 7:00 o'clock positions. None of the oil or fuel lines exhibited any thermal distress or damage except for the oil line coming from the accessory gearbox which was previously mentioned (**PHOTO 22**).

**PHOTO 21: FIRE DAMAGE LOCATED ABOVE AND AFT OF HEAT SHIELD**

PHOTOS COURTESY OF BOEING

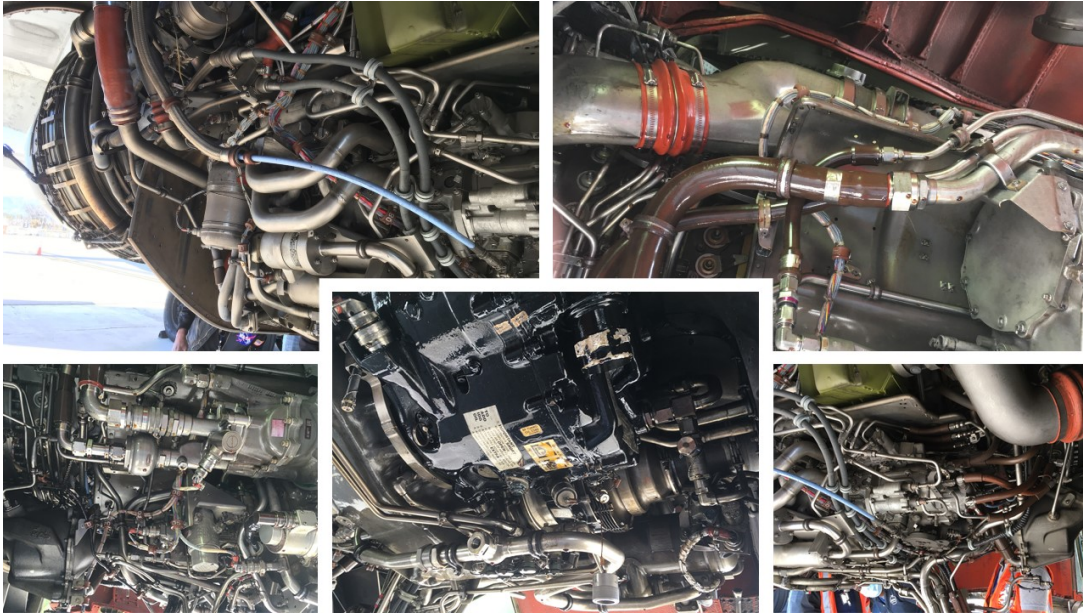


PHOTO 22: NO FIRE DAMAGE UNDERNEATH THE HEAT SHIELD

PHOTOS COURTESY OF BOEING

After the on-scene examination was completed, the engine was dropped from the airplane and placed in a transportation stand. A GE field representative aided and provided guidance on how to properly transport the engine based on the damage observed to the TRF. The TRF is equipped with two ground handling rails/sockets located at the 3:00 and 9:00 o'clock positions with holes for mount pins. Due to the extensive damage to the TRF, additional support was provided underneath the engine to ensure that the engine could be transported safely (**PHOTO 23**).

2.3 COCKPIT

According to Jett Prop maintenance personnel, the No. 2 engine fire handle, located in the cockpit overhead panel, was found rotated to the left indicating that one of the two fire suppression bottles had been discharged.



PHOTO 23: ENGINE NO. 2 READY FOR TRANSPORT

PHOTOS COURTESY OF GE

3.0 DETAILED EXAMINATION OF THE NO. 2 ENGINE – ESN 695440

The No. 2 engine was removed from the airplane and transported to the GE Evendale facility in Cincinnati, Ohio where it was quarantined until the Powerplant Group arrived.

3.1 AS-RECEIVED AT THE GE FACILITY

When the engine arrived at the GE facility, the engine was placed in a secure quarantined area and the NTSB authorized GE to remove several external components/parts for the engine to be hung in a gantry (**PHOTOS 24** and **25**).

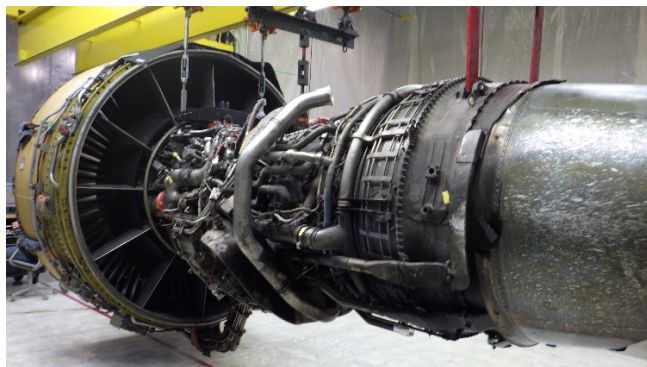


PHOTO 24: LEFT SIDE OF ENGINE

PHOTO COURTESY OF GE

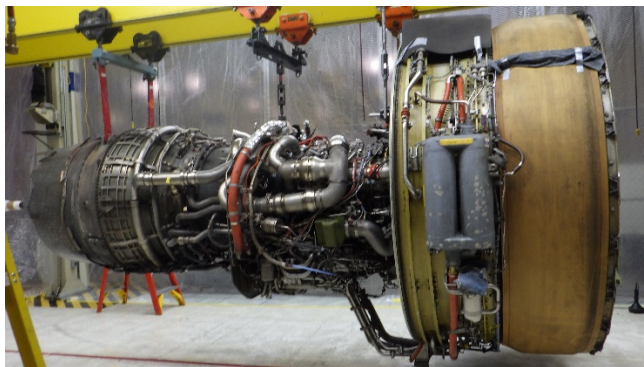


PHOTO 25: RIGHT SIDE OF ENGINE

PHOTO COURTESY OF GE

The NTSB authorized GE to perform a preliminary incoming borescope inspection (BSI) prior to the arrival of the Powerplant Group. The BSI was started in the HPT rotor section and no visible impact damage was observed; however, many HPT stage 1 blades exhibited thermal distress consistent with mid-span core burn-through. From there, BSIs were performed working aft (**PHOTO 26**). The first indication of event-related turbine damage or distress was to the leading edge of the LPT stage 1 blades (**PHOTO 27** bottom right corner photo). Damage was observed throughout all five stages of the LPT rotor; the damage included fractured blades at various lengths, missing blade and vane airfoil material, and LE and TE blade and vane airfoil impact damage. The LP rotor system (Fan-LPT, also commonly referred to as the N1 rotor) rotated smoothly during the borescope inspection. The combustion section also underwent a BSI as part of the Powerplant Group engine examination and no damage was observed.



PHOTO 26: NO HPT ROTOR SECTION DAMAGE

PHOTO COURTESY OF GE



PHOTO 27: LPT STAGE 1 BLADE DAMAGE

PHOTO COURTESY OF GE

The oil and fuel drain lines located below the bifurcation panel (**PHOTO 28**, yellow box) towards the 6:00 o'clock position were wetted; however, no visible damage was observed on the fuel and oil tubes, lines, and hoses or to the electrical cables and brackets. As noted during the on-scene examination of the engine, the oil tank sight gage indicated that the oil tank was empty. The oil drain line at the lowest point on the engine (**PHOTO 28**, green circle) was disconnected and oil samples were collected. The main fuel supply line was intact and the orange thermal sleeve exhibited minor thermal distress at its lowest point near the 4:00 o'clock position.

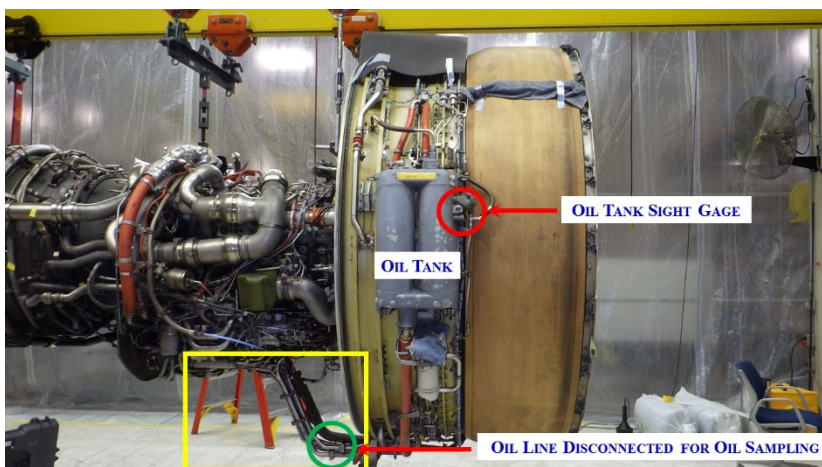


PHOTO 28: OIL SAMPLE LOCATION

PHOTO COURTESY OF GE

The AGB assembly is installed on the bottom of the engine and is attached to the compressor stator case. The transverse link attached to the AGB mount lug located at the 7:00 o'clock position towards the center AGB housing was found separated (**PHOTO 29**).

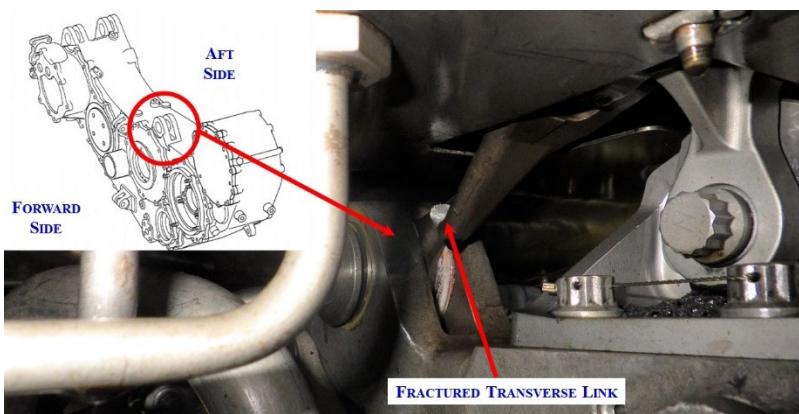


PHOTO 29: FRACTURED TRANSVERSE LINK

PHOTO COURTESY OF GE

A crack was observed around the 9:00 o'clock position in the cooling manifold (**PHOTO 30**).

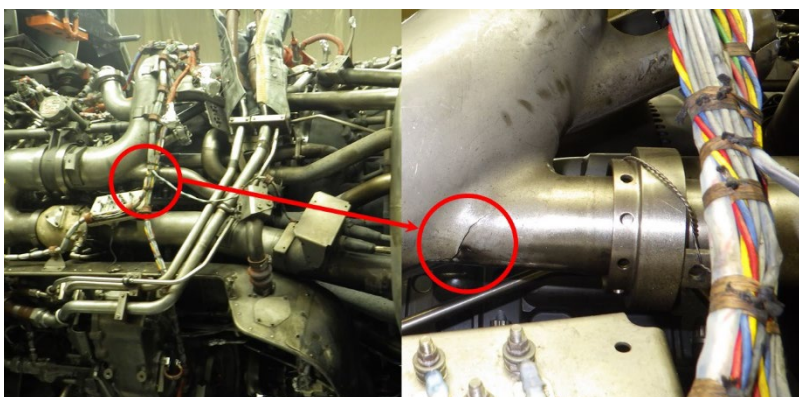


PHOTO 30: CRACKING COOLING MANIFOLD

PHOTO COURTESY OF GE

3.2 POWERPLANT GROUP EXAMINATION AND DISASSEMBLY

The Powerplant Group, comprised of members from GE, FAA, Mas Air Cargo, and the NTSB, convened at the GE Evendale facility from February 22-26, 2021, to perform a detailed examination of the incident engine.

3.2.1 Fuel Leak Check

Prior to disassembling the engine, a leak check of the fuel system forward of the heat shield (heat exchanger, fuel pump, MEC, etc.) was performed; no fuel leak check was performed on the fuel manifolds or fuel nozzles since no indication of fuel wetting of the compressor rear frame was noted. Fuel was first drained from the fuel system and the fuel supply lines to the left and right fuel manifold rings were disconnected from the common fuel supply T-fitting forward of the heat shield. The left fuel manifold supply fitting was capped, and a nitrogen tank was attached to the right fuel manifold supply fitting (**PHOTO 31**). Nitrogen was slowly introduced into the fuel system and after fifty (50) pounds per square inch (psi) was applied the supply valve was closed; the needle on the pressure gage of the nitrogen tank stayed steady and did not drop. The supply pressure was then increased to 80 psi and again the supply valve was closed and the needle on the pressure gage did not drop, consistent with no leak in the fuel system forward of the left and right manifold fuel supply T-fitting.

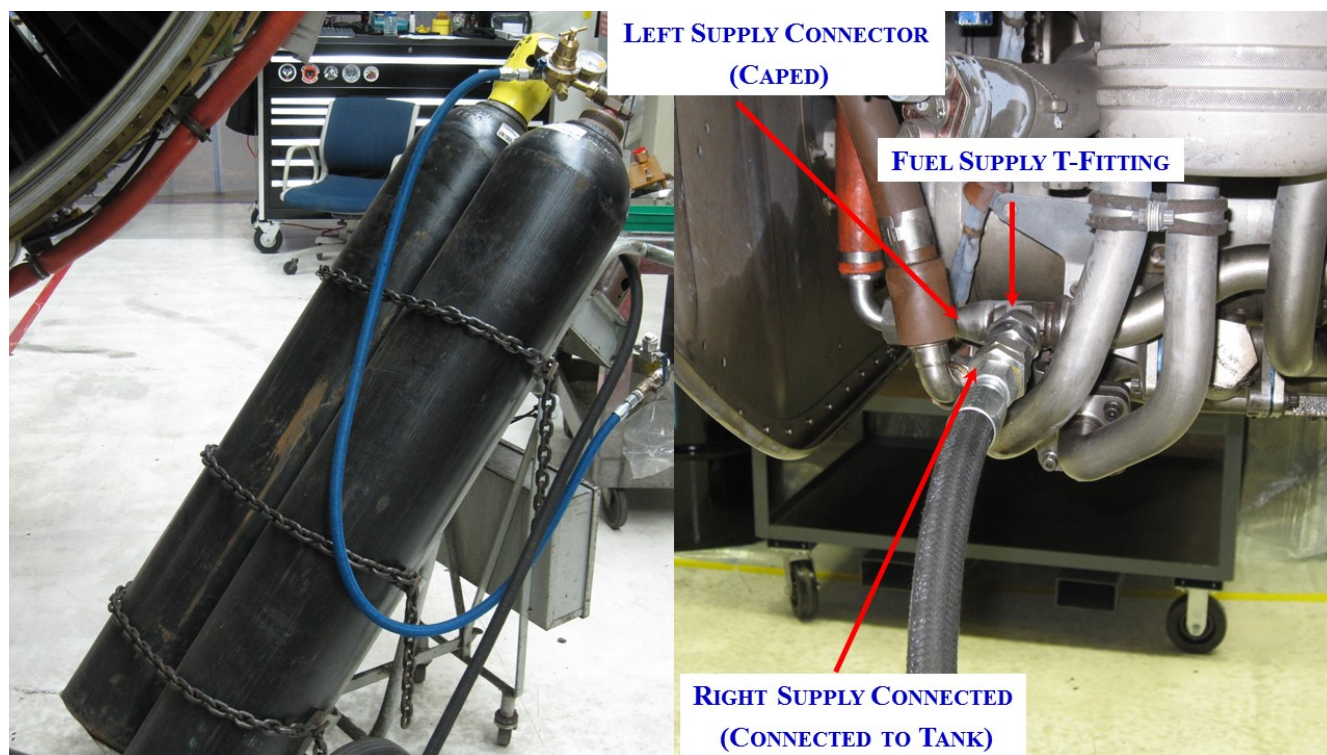


PHOTO 31: FUEL LEAK CHECK SETUP

3.2.2 Turbine Exhaust Nozzle and Exhaust Plug Removal

Three bolts that secure the turbine exhaust sleeve to the engine's TRF were fractured/sheared and missing their bolt head; the shank of the bolts remained. The sheared bolts were located as follows: one fractured bolt was located at the 4:00 o'clock position in the vicinity of the TRF opening and the other two were located at the 5:00 o'clock position in the vicinity of the TRF No. 6 bearing oil supply boss crack. The turbine exhaust sleeve was removed and several axial length measurements of the inner skin from the front flange were taken to document that amount of inner skin that remained and those measurements were as follows: at the 12:00 o'clock position it measured 38.5-inches in length, at the 3:00 o'clock position it measured 26.5-inches in length, at the 6:00 o'clock position it measured 21.25-inches in length, and at the 9:00 o'clock position it measured 39-inches in length. The largest amount of skin material that remained was around the 12:00 o'clock positions and the high point measured 46-inches in length. All the bolts that secure the exhaust plug to the TRF were intact and in place. The turbine exhaust sleeve and the exhaust plug were shipped to Boeing for further analysis (See Section 10).

3.2.3 No. 6 Bearing and "D" Sump Area

The exhaust plug was removed exposing the aft end of the "D" sump and the No. 6 bearing housing where vent air is discharged from the engine. The discharge air cavity was sooted, oil wetted and stained, exhibited black particles consistent with coke formation, the clamps and brackets that secure of the oil tubes were present and intact, and the oil tubes (supply and scavenge) appeared intact and undamaged. The oil tubes clamps exhibited minor thermal distress and the thermal insulation shield was intact and blackened. According to GE, that amount of oil, sooting, coking seen in this compartment was greater than what is usually seen on a normal in-service engine; however, it was consistent with their experience when a LPT failure occurs. The insulation shield, the No. 6/"D" sump rear cover and No. 6 bearing oil slinger were removed to expose the No. 6 bearing/"D" sump area; all the nuts and bolts that secure the rear bearing cover to the oil slinger were intact and tight. All the No. 6 bearing rollers elements were present, intact, and appeared shiny and oil wetted (**PHOTO 32**).

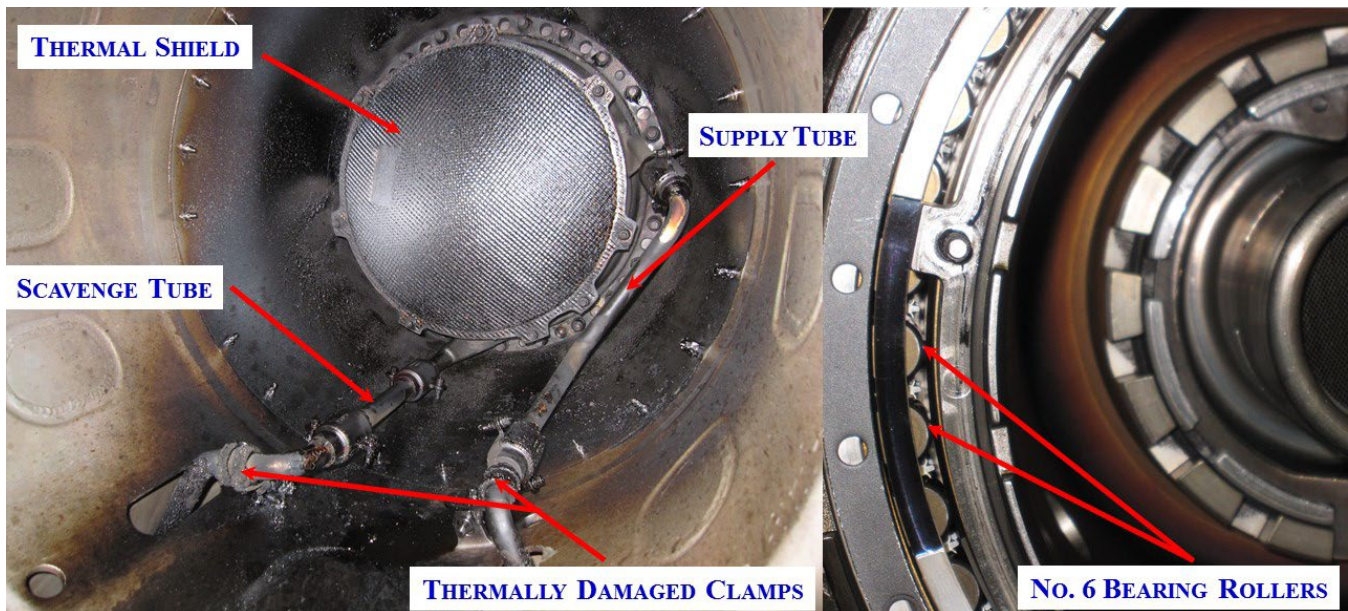


PHOTO 32: NO. 6 BEARING COMPARTMENT/"D" SUMP CONDITION – OIL AND COKE NOTED

3.2.4 High Pressure Turbine Module

The LPT module was separated from the rest of the engine, exposing the aft end of the HPT assembly. **FIGURE 5** is a cross-sectional view of the HPT rotor assembly and **PHOTO 33** is representative of the HPT stage 2 damage observed. The HPT rotor spun freely by hand. All the HPT stage 2 blade outer shroud segments, which are part of HPT stage 2 nozzle assembly were present, intact, and exhibited minor contact rub. All the HPT stage 2 blades were present and intact; no airfoil damage was noted, and all the blades exhibited minor tip roll-over with the material rolled over in the direction opposite of rotation. The HPT stage 2 blade “angel wing” aft seal land (the seal lands that protrude away from both the forward and aft sides of the LPT blade shanks are commonly referred to as an “angel wing” seal as a description of their geometry and shape) exhibited circumferential scoring, bluing, and metal transfer on both the outboard and inboard surfaces 360° circumferentially around. The HPT stage 2 blade retainer plate was intact, and the outboard lip exhibited rub and material roll-over. The HPT rotor aft rotating air seal is comprised of a single seal that seals with the pressure balance seal land on the LPT stator assembly and a three knife-edge seal that seals with the forward rotating honeycomb seal attached to the LPT shaft. All the tips of the seals exhibited contact rub and tip roll-over; the pressure balance seal appeared bluish. No damage to the HPT stage 2 disk was observed. Looking forward through the HPT stage 2 blades, the HPT stage 2 nozzle segments were visible; no TE damage was observed. No disassembly of the HPT section was performed.

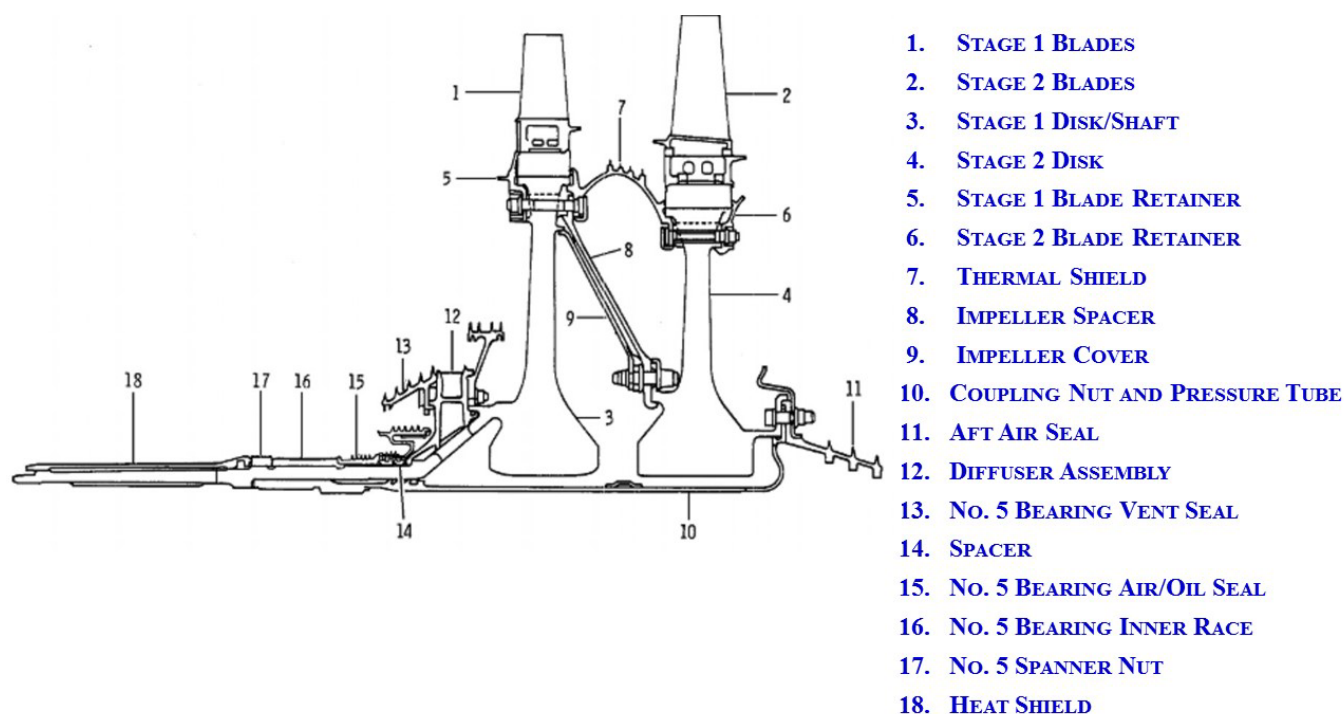


FIGURE 5: HPT ROTOR ASSEMBLY AND NOMENCLATURE

FIGURE COURTESY OF GE

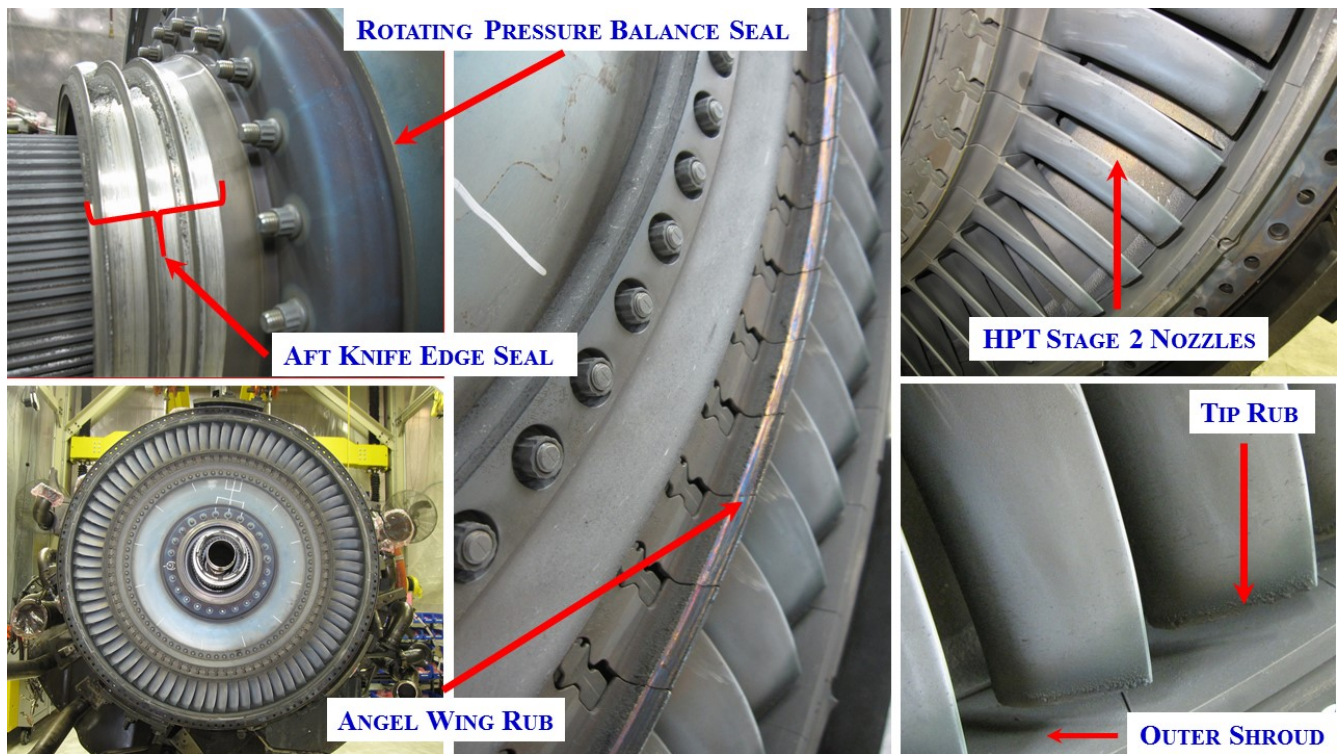


PHOTO 33: HPT ROTOR DAMAGE

3.2.5 Low Pressure Turbine Module – Turbine Rear Frame and No. 6 Bearing

The LPT module was disassembled aft moving forward as it is typically disassembled. **FIGURES 6** and **7** are cross-sectional views of the LPT rotor and stator assembly. The TRF was removed from the stator case and all the struts were sequentially numbered clockwise ALF 1 to 12 starting with the No. 1 strut at the 12:00 o'clock position. All the struts exhibited metal splatter and minor impact damage. The No. 4 strut, located at the 3:00 o'clock position, exhibited an outer boss crack at the ground handling hole. The No. 5 strut, located at the 4:00 o'clock position, pushed through the skin creating an opening. The No. 6 strut, located at the 5:00 o'clock position, exhibited a crack at the oil supply boss. The Nos. 9 and 10 struts located at the 8:00 o'clock and 9:00 o'clock positions respectively pushed through the skin creating an opening (**PHOTO 34**). The No. 6 bearing rotated smoothly by hand and appeared undamaged and in good condition.

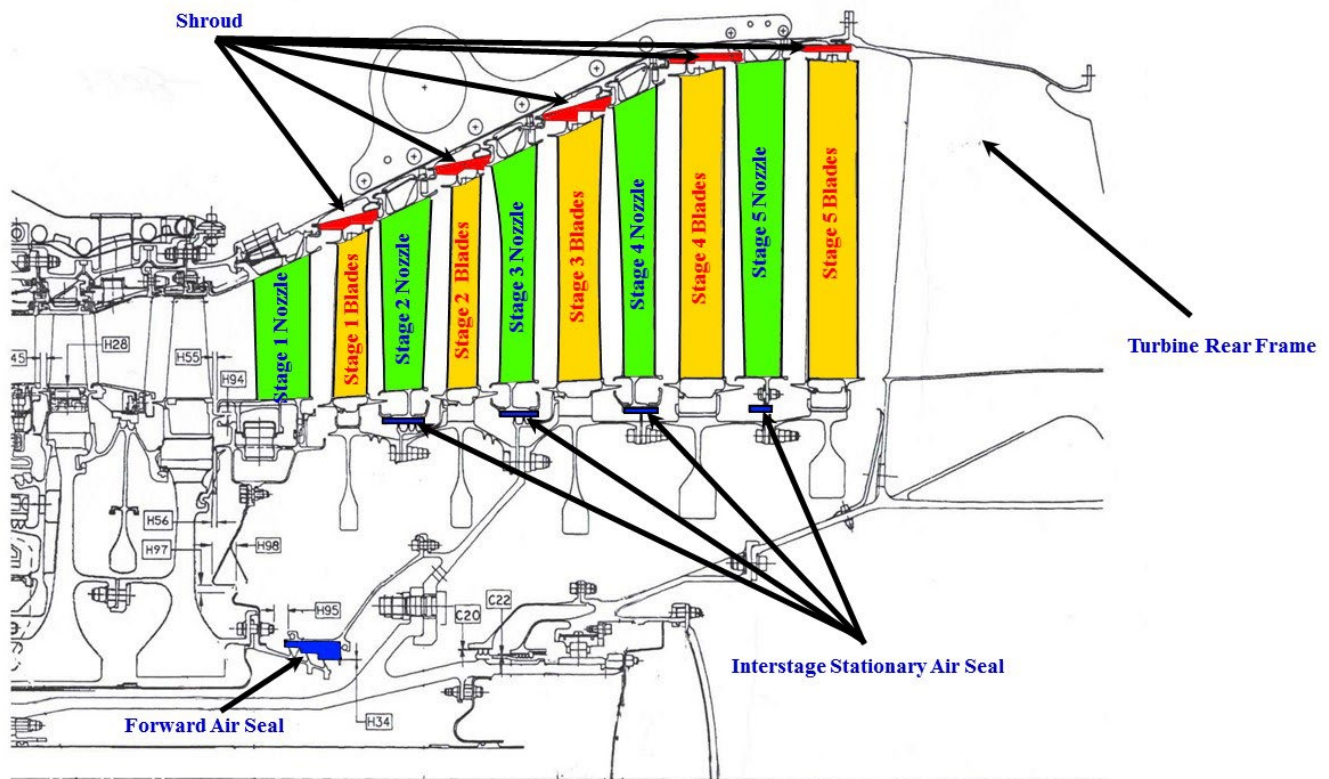


FIGURE 6: LPT ROTOR AND STATOR ASSEMBLY CROSS-SECTION AND NOMENCLATURE – GENERAL
FIGURE COURTESY OF GE

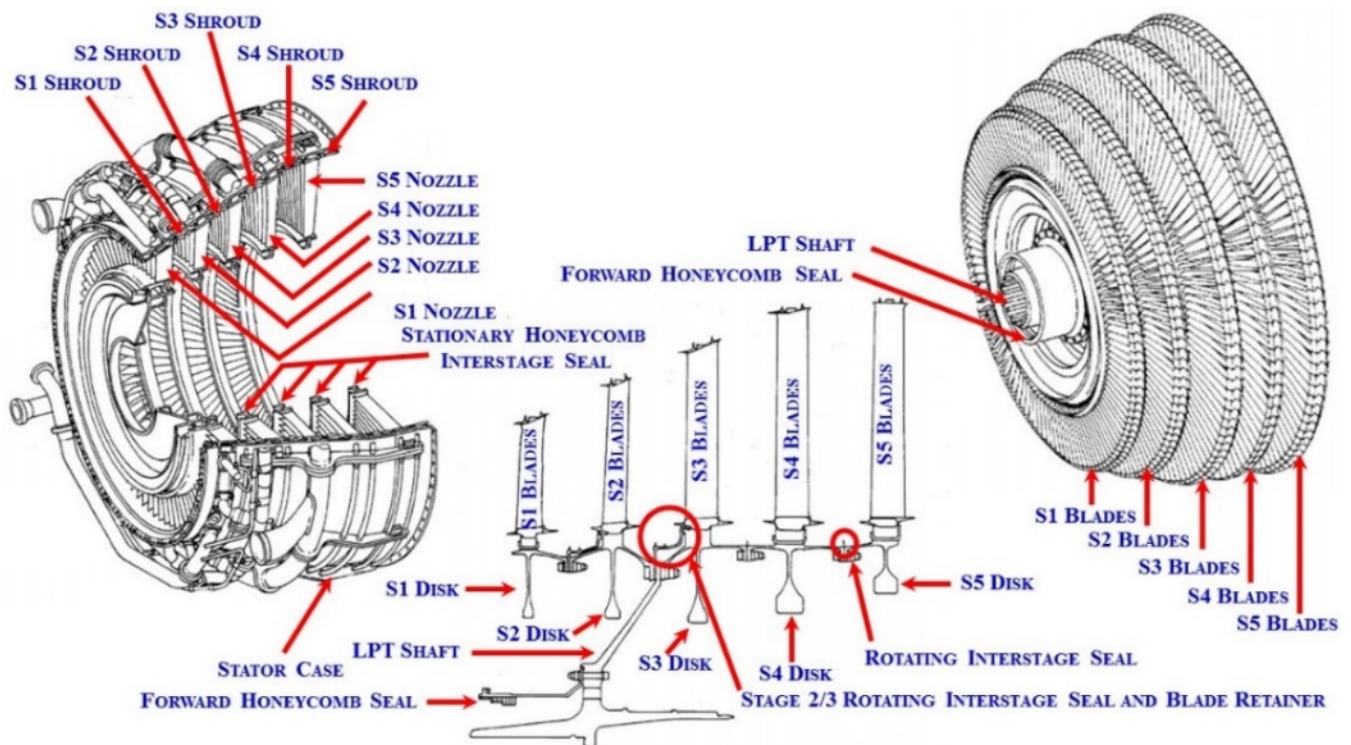


FIGURE 7: LPT ROTOR AND STATOR ASSEMBLY CROSS-SECTION AND NOMENCLATURE - GENERAL
FIGURE COURTESY OF GE



PHOTO 34: TRF CASE OPENINGS AT STRUTS NO. 5, 9, AND 10

3.2.6 Low Pressure Turbine Module – Stage 5

All the LPT stage 5 blade roots were still installed in the disk and all the blades were fractured transversely across the airfoil at various lengths and exhibited a combination of damage consisting of metal splatter, LE and TE impact damage and gouging, missing airfoil material, and the airfoils outer edges were bluish consistent with heat distress. The forward angel wing air seal land of each blade was battered and exhibited circumferential rub along the front edge. In an arc of consecutive blades, about 25-inches in length, the forward angel wing seal land was almost entirely missing; in this same 25-inch arc, the blade's LE shank-to-airfoil transition radius exhibited significant rub (**PHOTO 35** upper left photo).

The aft side of the LPT stage 5 disk exhibited what appeared to be light scuff/impact marks on the disk web, on the blade fir trees, and on the blade retainers; however, these scuff marks were smooth to the touch and no raised metal was felt. On the front side of the disk, coincident with the 25-inch arc of the blades missing their forward angel seal land, the outer diameter front flange surface of the disk exhibited a band of rotational scoring coincident with contact with the stage 4/5 interstage stationary honeycomb seal.

All the stage 5 blade outer shroud segments were sequentially numbered clockwise ALF 1 to 24 starting clockwise from the 12:00 o'clock position and were still installed in the stator case except for one segment located about the 4:30 o'clock position; this would have been blade outer shroud segment No. 8 (**PHOTO 35** upper right photo). Those blade outer shroud segments that remained were all heavily battered and exhibited a combination of damage consisting wear of the honeycomb seal down to the backing strip, torn/missing material, distortion, and some were worn completely through. Blade outer shroud segment No. 24 located one spot counterclockwise from the 12:00 o'clock position had become partially dislodged from the stator case and was curled and rolled into the gas path.

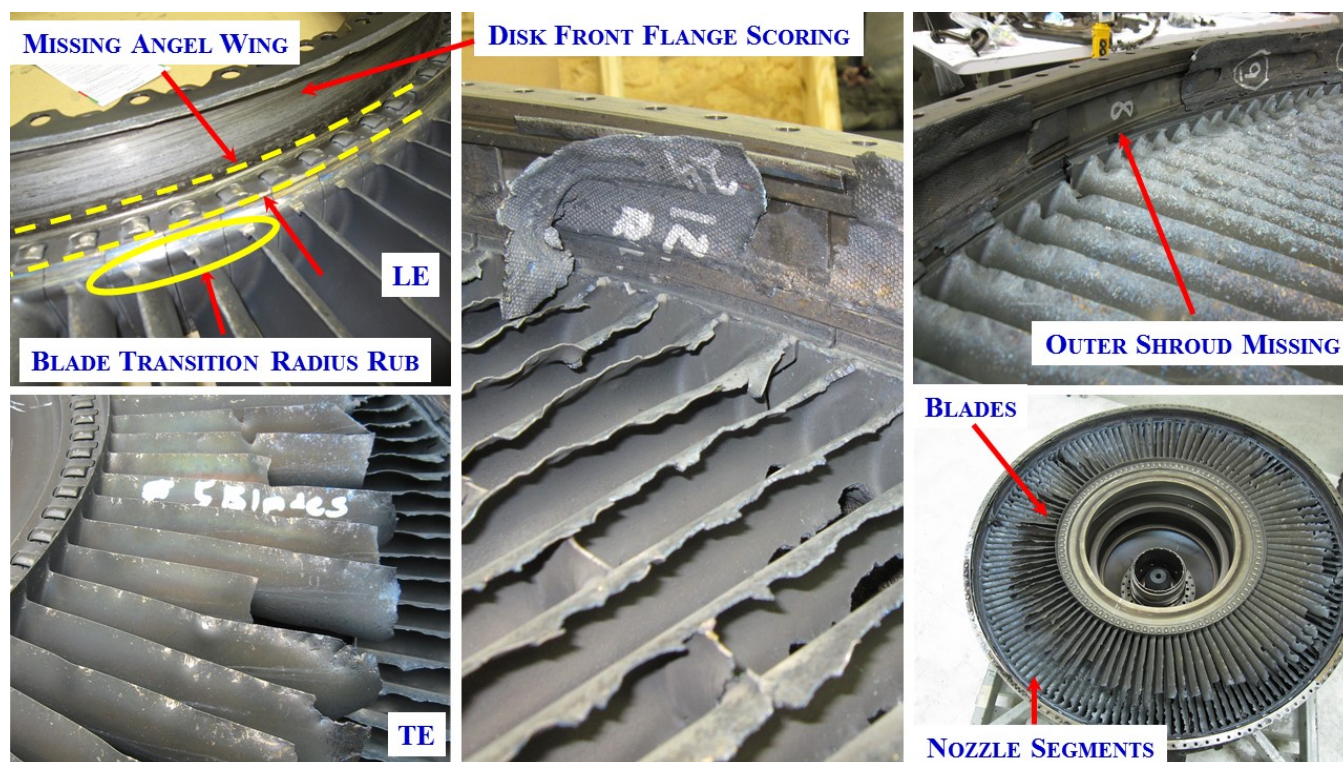


PHOTO 35: LPT STAGE 5 DISK, BLADE, AND BLADE SHROUD DAMAGE

The stage 5 nozzle segments are comprised of 6 airfoils with an outer and inner shroud. The outer shroud incorporates a hook flange to secure it to the stator case while the inner shroud incorporates an attachment flanged used to secure the interstage honeycomb static land; each stage of nozzle segments has this same design. Like the blade outer shroud segments, the nozzles were sequentially numbered clockwise ALF 1 to 24. All the nozzle segments were present, and one segment located about the 1:30 o'clock position, the No. 3 nozzle segment, was missing two complete adjacent airfoils, airfoils Nos. 5 and 6 (**PHOTO 36** upper left photo). Nozzle segment No. 9 had partially disengaged on one end from the stator case hook flange that coincided with the missing No. 8 shroud segment. All the nozzle segments exhibited a combination of damage consisting of gouging, missing airfoil material, missing platform TE material, tearing, bluing consistent with heat distress, and the inner platform TE flattened/pushed forward; the greatest amount of flattening of the TE platform appeared around the 9:00 o'clock position to nozzle segments Nos. 18 to 20. The airfoil damage was primarily focused on the outer platform. The airfoil TEs were folded over/pushed forward closing the exit area. Although all the TE edges exhibited damage, two distinct and separate groups of consecutive nozzle segments exhibited a pattern of outer airfoil damage and material loss greater than the surrounding nozzles. One section was centered around the 9:00 o'clock position, nozzle segments Nos. 17 - 21 and the other section was between 12:00 and 1:00 o'clock positions, nozzle segments Nos. 1 - 4 (**PHOTO 36** upper right photo). The airfoil LEs were folded over/pushed aft closing the entrance area.

The stage 4/5 stationary interstage honeycomb seals are bolted to a radial flange below the inner platform. All the tops of bolt heads of the stage 4/5 interstage honeycomb seal were machined consistent with contact with the forward edge of the stage 5 blade LE angel wing seal land. The No. 3 (1:00 o'clock position) and the No. 11 (5:00 o'clock position) nozzles segments stage 4/5 interstage honeycomb seal attachment bolts were completely missing (bolt head and shank) and No. 23 (11:00 o'clock position) the stage 4/5 interstage honeycomb seal attachment bolt was missing the bolt head, but

the shank was still present (**PHOTO 36** lower right photo). All the stage 4/5 interstage honeycomb seal segments exhibited gouging, wear down to the backing strip, and trenching of the honeycomb seal consistent with contact with stage 4/5 rotating knife edge interstage seal that is sandwiched between the LPT stage 4 and stage 5 disks; a section of stage 4/5 interstage honeycomb seal centered around the 9:00 o'clock position attached to nozzle segments Nos. 16 to 19 were circumferentially machined and forward half of the seal was missing (**PHOTO 36** lower left photo). The rotating knife edge seal was intact and exhibited contact rub.

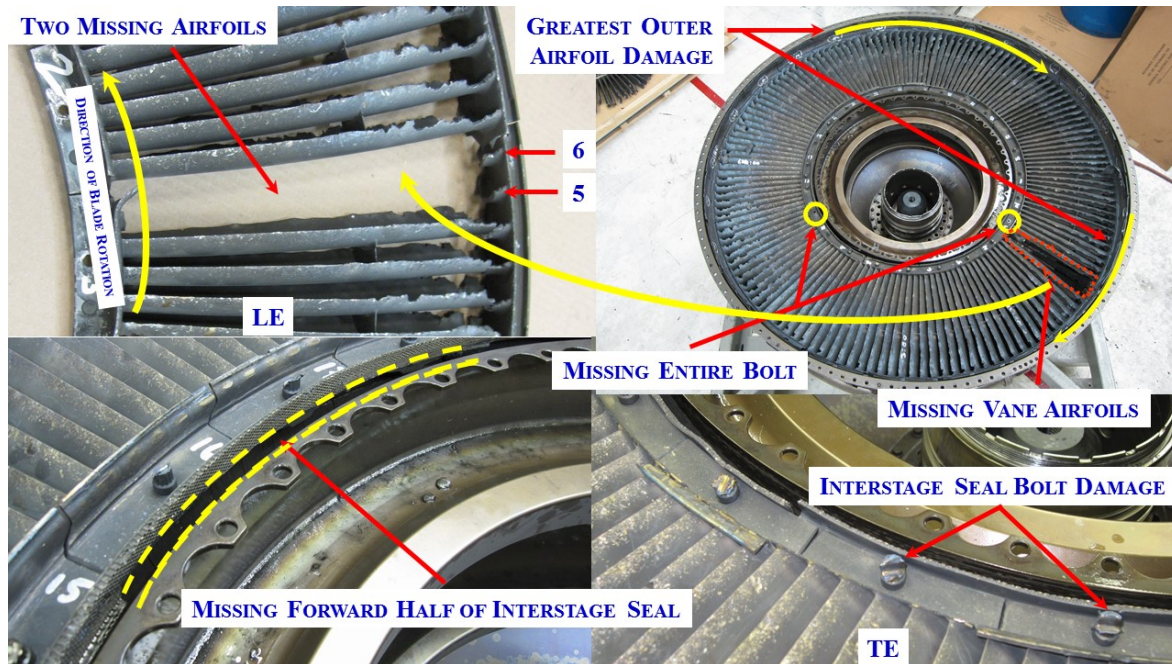


PHOTO 36: LPT STAGE 5 NOZZLE SEGMENT AND INTERSTAGE HONEYCOMB SEAL DAMAGE

3.2.7 Low Pressure Turbine Module – Stage 4

All the LPT stage 4 blade roots were still installed in the disk and all the blades were fractured transversely across the airfoil at various lengths and exhibited similar damage to the LPT stage 5 blades; however, the amount of damage appeared visually to be more extensive than what was observed to the LPT stage 5 blades. Many airfoil fragments collected in this stage at the 6:00 o'clock position (**PHOTO 37** upper left photo). In an arc of consecutive blades, about 22-inches in length, the forward angel wing seal land was almost entirely missing; in this 22-inch arc, the blade's LE shank-to-airfoil transition radiused exhibited significant rub (**PHOTO 37** right photo). This was the same type and roughly the same amount of damage that was observed on the forward angel wing on the LPT stage 5 blades.

The front side of the LPT stage 4 disk outer diameter front flange exhibited a band of rotation scoring marks coincident with contact with the stage 3/4 interstage stationary honeycomb seal (**PHOTO 37** right photo); this is similar to the front flange damage observed in the LPT stage 5 disk.

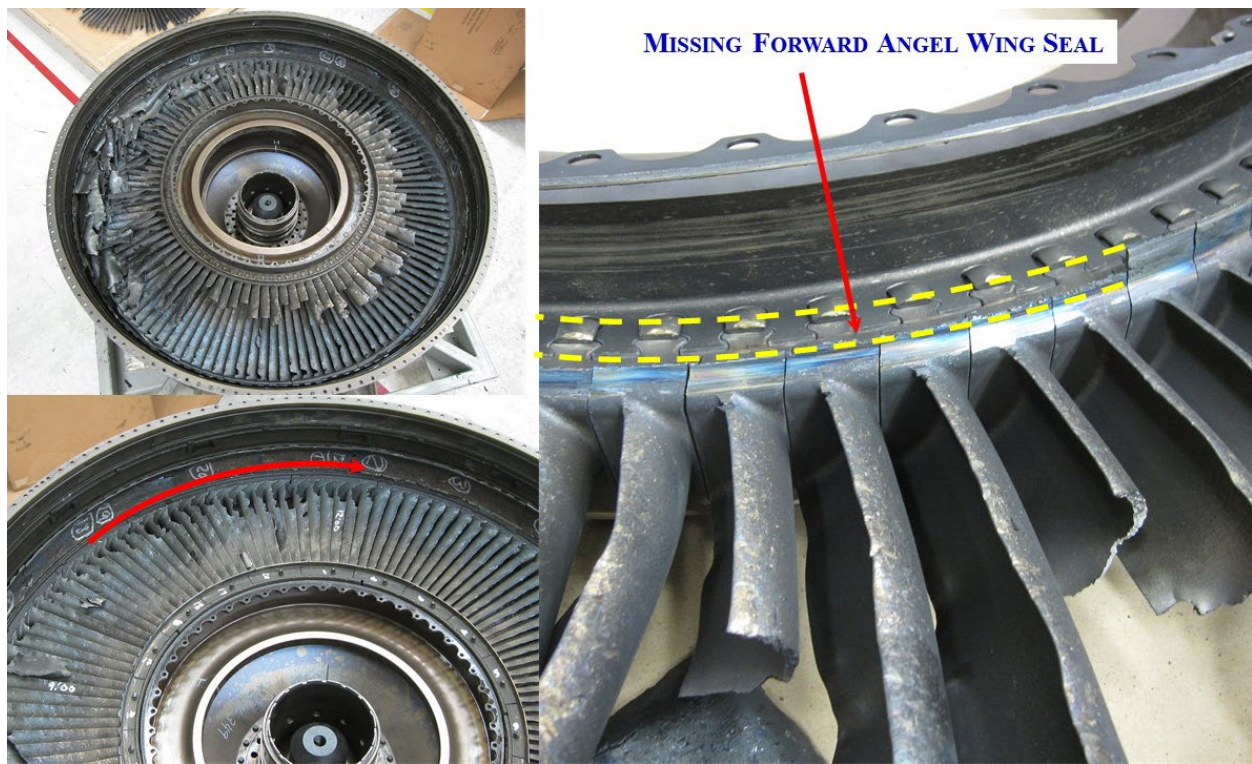


PHOTO 37: LPT STAGE 4 BLADE AND BLADE SHROUD SEGMENT DAMAGE

All the stage 4 blade outer shroud segments again were sequentially numbered ALF clockwise 1 to 20 starting slight counterclockwise from the 12:00 o'clock position. All the blade outer shroud segments were present, still secured to the stator case rail by the front "C" channel but disengaged from the case at the aft hooks (**PHOTO 38**, upper left photo), the aft hooks were collapsed upon themselves, and the honeycomb was worn down to the backing strip. The most significant rub and bluing consistent with heat distress was noted between blade outer shroud segments Nos. 14 to 3 (about the 7:00 to 2:00 o'clock position) (See **PHOTO 37** lower left photo) and from the Nos. 8 to 10 (about the 8:00 to 10:00 o'clock position). Blade outer shroud segments Nos. 2 and 15 exhibited a rub through, but no distortion or breach of the case was noted in these areas. Blade outer shroud segments Nos. 17 and 18 (about the 10:00 o'clock position) were also rubbed through and a small outward bulge in the stator case was noted but no breach. Blade outer shroud segment No. 19 (about the 10:30 o'clock position) exhibited a rub through and a corresponding breach of the stator case. Near the blade outer shroud segment No. 19 rub through, the case exhibited a 2.5-inch circumferential bulge and tear with a corresponding 0.5-inch hole; the hole appeared not be a penetration hole (**PHOTO 38** right photo). With all the blade outer shroud segments removed, 12 total impact bents/breaches were documented in-line radially with the stage 4 blade running path; the only breach observed was the one previously documented about the 10:30 o'clock position. All 12 bents/breaches were in an arc from about the 8:00 to 2:00 o'clock position, except for one (**PHOTO 38** upper left photo).

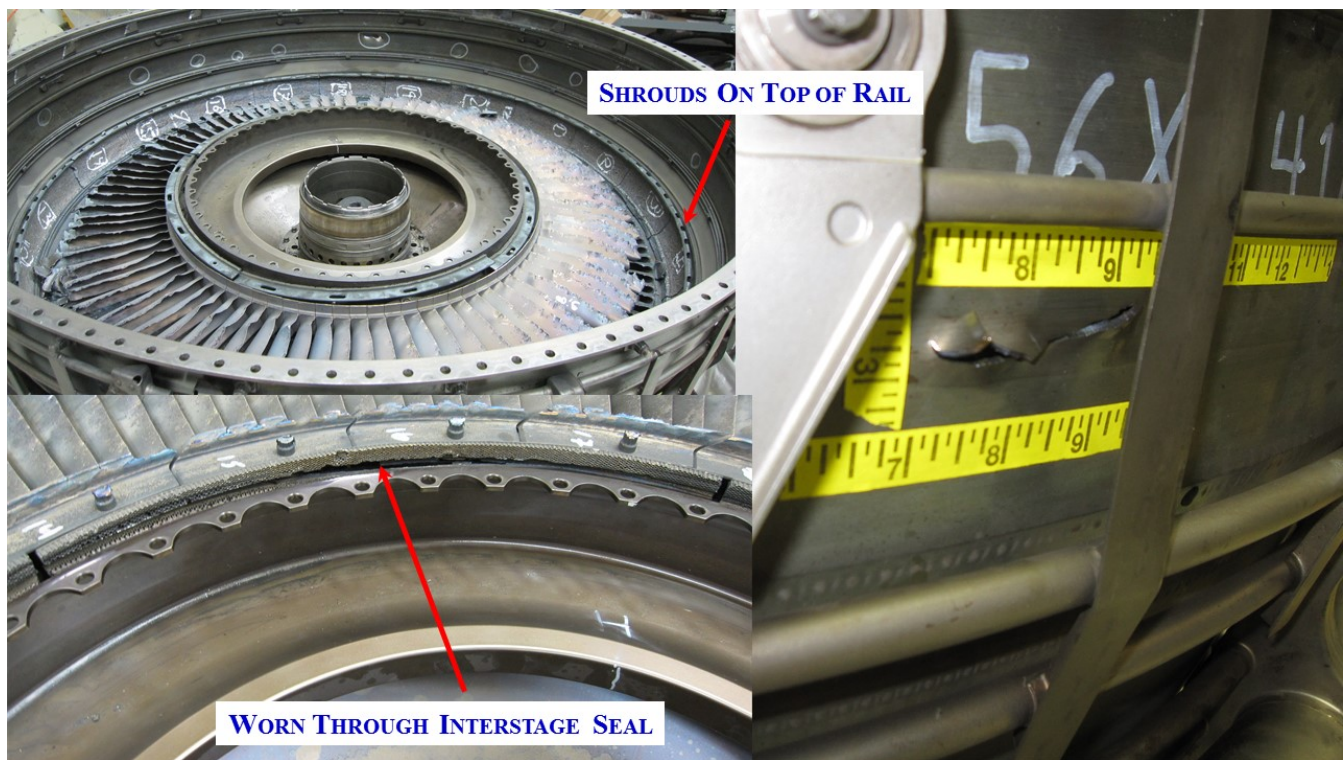


PHOTO 38: LPT STAGE 3/4 INTERSTAGE DAMAGE AND STATOR CASE DAMAGE

All the stage 4 nozzle segments were sequentially numbered ALF clockwise 1 to 20 starting counterclockwise from the 12:00 o'clock position. All the stage 4 nozzle segments were intact with no missing airfoils. Similar airfoil and inner and platform damage were noted to what was observed on the LPT stage 5 nozzles with the airfoil damage again more focused towards the outer shroud; all the shroud aft edges were completely worn away. Two sections of nozzle segments exhibited extensive airfoil damage and considerable airfoil missing material; those segments were Nos. 18 to 1 (about 10:00 to 12:00 o'clock position) (See **PHOTO 37** lower left photo) and Nos. 14 to 16 (about 8:00 to 9:00 o'clock position). The inner aft platform was heavily worn, heat distress, and missing material from Nos. 14 to 17 (about 7:30 to 10:00 o'clock position) (**PHOTO 38** lower left photo).

All the tops of bolt heads of the stage 3/4 interstage honeycomb seal were machined/cut consistent with contact with the forward edge of stage 4 blade LE angel wing seal land; none were missing as-seen in the LPT stage 5. All the stage 3/4 stationary interstage honeycomb seal segments exhibited 360° circumferential rub and trenching like what was observed in the LPT stage 5 consistent with contact with stage 3/4 rotating knife edge interstage seal that is sandwiched between the LPT stage 3 and stage 4 disks. The greatest amount of stage 3/4 interstage honeycomb seal damage was to nozzle segments Nos. 14 to 17 collocated with the damage observed on nozzle segment inner platform.

3.2.8 Low Pressure Turbine Module – Stage 3

All the LPT stage 3 blade roots were still installed in the disk and exhibited similar impact damage and had a bluish appearance consistent with thermal distress; the major differences in the condition of the LPT stage 3 blades from the LPT stages 4 and 5 blades were: 1) stages 4 and 5 blades were fractured transversely across the airfoil at various lengths while the stage 3 blades were all fractured almost uniformly about 80% span except for one fractured at about 40% span and 2) the TEs were much more jagged in appearance. Examination of the LE edges of the blades revealed a similar consecutive

blade 22-inch arc where the LE angel wing seal land was almost entirely missing (a clear step change was observed), like what was observed on the previous LPT rotor stages (**PHOTO 39** lower left photo).

All the stage 3 blade outer shroud segments again were sequentially numbered ALF clockwise 1 to 20 starting slight counterclockwise from the 12:00 o'clock position. All the blade outer shroud segments were present, still secured to the stator case rail by the front "C" channel but disengaged from the case at the aft hook (**PHOTO 39**, lower center photo), the aft hooks were collapsed upon themselves, and the honeycomb was worn down to the backing strip; however, unlike the previous stages, no blade shroud segments were rubbed through. There were two blade outer shroud segments, No. 9 about the 5:00 o'clock position, and the No. 14 about the 8:00 o'clock position that exhibited ripped and pushed-over material. With all the blade shroud segments removed, numerous superficial stator case impact marks were noted in-line with the stage 3 blade running position; however, none distorted the case, and the impacts could not be felt from the outside (**PHOTO 39** lower center photo).

The rotating stage 2/3 knife edge seal and stage 3 blade retainer (this is a single part is sandwiched between the stage 2 and 3 disks along LPT shaft that performs both functions) exhibited rub on both knife edge seals (**PHOTO 39** left photo). The rear knife edge seal exhibited significantly more rub and material loss (height of the knife edge seal was significantly reduced) than the forward knife edge seal. The greatest amount of material loss was the rear knife edge seal height from the 4:00 to 8:00 o'clock position. The outer diameter aft of the rear knifed edge seal exhibited circumferential scoring from the 4:30 to 8:00 o'clock position and the rear flange front face exhibited scoring from the 4:30 to 9:30 o'clock position. Circumferential scoring was also observed between the forward and rear knife edge seal from the 4:00 to 6:00 o'clock position. The LPT shaft was in good condition.

All the stage 3 nozzle segments were sequentially numbered ALF clockwise 1 to 20 starting counterclockwise from the 12:00 o'clock position. All the stage 3 nozzle segments were intact with no missing airfoils like was what observed for the stage 4 nozzles. The TE edges exhibited impact damage and material loss while the LE exhibited minor impact damage and material loss in comparison with the stages 4 and 5 nozzle segments. Many of the bolt heads of the stage 2/3 interstage honeycomb seal were machined consistent with contact with the forward edge of stage 4 blade LE angel wing seal land; none were missing like in the LPT stage 5. The stage 2/3 interstage honeycomb seals were present and exhibited appreciable gouging and trenching of the honeycomb from about the 3:00 to 10:00 o'clock positions. The most significant damage was centered around the 9:00 o'clock position seals Nos. 14 to 17 (**PHOTO 39** upper center photo) and at the 3:00 o'clock position seals Nos. 4 to 9 where the honeycomb was trenched down to the backing strip and cutting the backing strip.

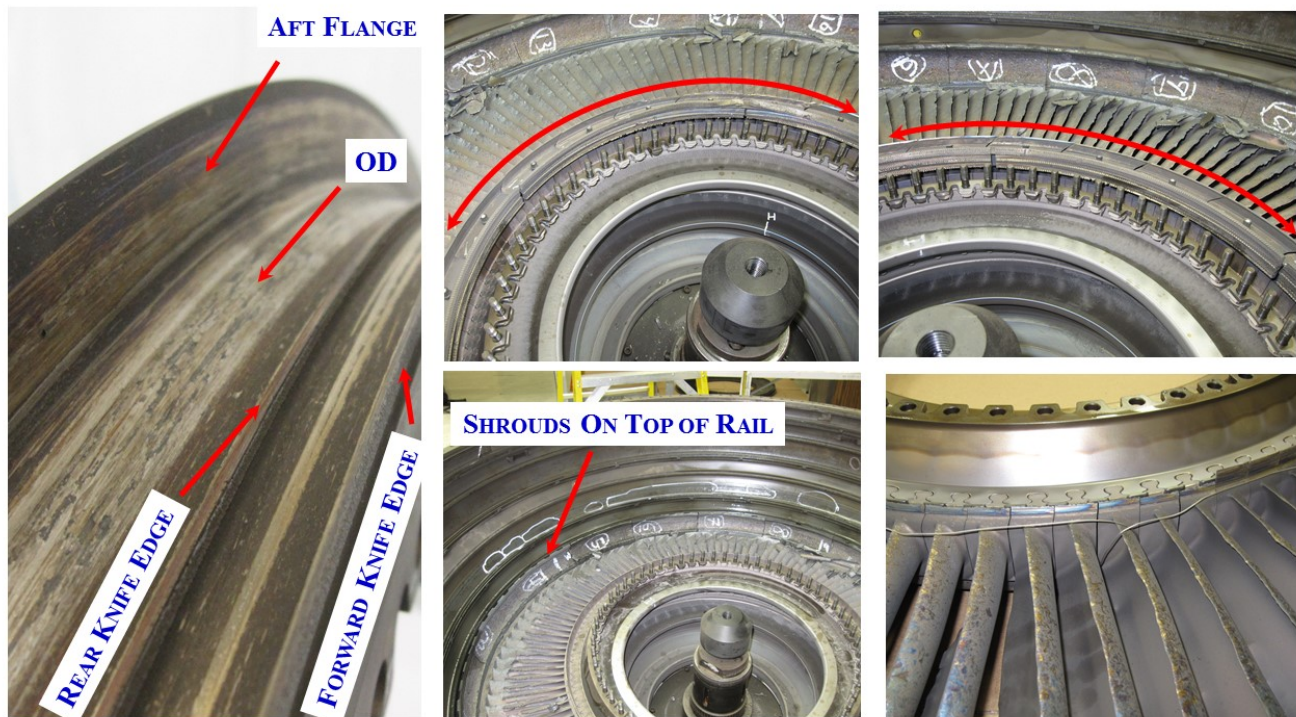


PHOTO 39: LPT STAGE 3 DAMAGE

3.2.9 Low Pressure Turbine Module – Stage 2

All the LPT stage 2 blade roots were still installed in the disk, fractured transversely across the airfoil at various lengths, and exhibited airfoil impact damage and a bluish appearance consistent with thermal distress. A consecutive 180° circumferential arc of blades were all fractured at about the same span or just over half span (**PHOTO 40**, right top photo red line); the other consecutive 180° circumferential arc of blades were fractured at various lengths in a serpentine pattern (**PHOTO 40**, right top photo yellow line). The airfoil LEs and TEs did not exhibit much impact damage or material loss. With the stage 1/2 rotating knife edge interstage seal unbolted from the stage 2 disk to facilitate the removal of the stage 2 disk, many of the blades slide out of their slots. A GE metallurgist did a visual review of all the blade fracture surfaces (did not exam loose airfoil fragments debris) and did not find any evidence of a preexisting fatigue crack. Examination of the disk found it to be in good condition with no scoring of the forward or aft outer diameter (OD) flanges.

The rotating stage 1/2 knife edge seal and stage 2 blade retainer exhibited rub on both knife edge seals. The rear knife edge seal exhibited more rub and material loss than the forward knife edge seal, not as much as was observed on the rear knife edge seal of the stage 2/3 knife edge seal and stage 3 blade retainer. The greatest amount of material loss of the rear knife edge seal height was from the 9:00 to 12:00 o'clock position. The outer diameter aft of the rear knife edge seal and the rear flange front face both exhibited circumferential 360° scoring. Circumferential scoring was also observed between the forward and rear knife edge seal from the 10:00 to 11:00 o'clock positions (**PHOTO 40**, lower right photo).

All the stage 2 blade outer shroud segments were sequentially numbered ALF clockwise 1 to 16 starting counterclockwise from the 12:00 o'clock position. All the blade outer shroud segments were present, still secured to the stator case rail by the front "C" channel but disengaged from

the case at the aft hooks (**PHOTO 40**, lower left photo), the aft flange hooks were collapsed, upon themselves, and the honeycomb was worn down to the backing strip; none was worn though.

All the stage 2 nozzle segments were sequentially numbered ALF clockwise 1 to 16 starting counterclockwise from the 12:00 o'clock position. All the stage 2 nozzle segments were intact with no missing airfoils. The TEs exhibited minor impact marks and material loss, less than what was seen on the downstream stages. The impact damage was focused mainly on the outer shroud. The LEs exhibited little in the way of impact damage; however, about a dozen sporadic airfoils exhibit chunks of missing material (**PHOTO 40**, top left photo).

All stage 1/2 interstage honeycomb seal segments exhibited circumferential rub and some amount of trenching with greatest amount of damage centered around the 9:00 o'clock position. Interstage segments Nos. 12 and 13 were cut in half and forward edge was missing consistent with being cut by the knife edge from the rotating stage 1/2 knife edge seal and stage 2 blade retainer. Interstage segments No. 11 and 12 were cut in a similar fashion but remained together (**PHOTO 40**, lower left photo).



PHOTO 40: LPT STAGE 2 DAMAGE

3.2.10 Low Pressure Turbine Module – Stage 1

All the LPT stage 1 blades were still installed in the disk, fractured transversely across the airfoil at essentially the same length about 70%-80% span except for two that were fractured well below 50% span (**PHOTO 41**). All the blades exhibited airfoil TE impact damage and a bluish appearance consistent with thermal distress. The airfoil LE did not exhibit much airfoil damage; however, the forward angel wing seal land was machined and missing about 90% around the circumference; there were only two sections about 6-inches circumferentially around that remained. The rear angel wing seal land exhibited some minor circumferential scoring but was otherwise intact. A GE metallurgist did a

visual review of all the blade fracture surfaces (did not exam loose airfoil fragments debris) and did not find any evidence of a preexisting fatigue cracks.

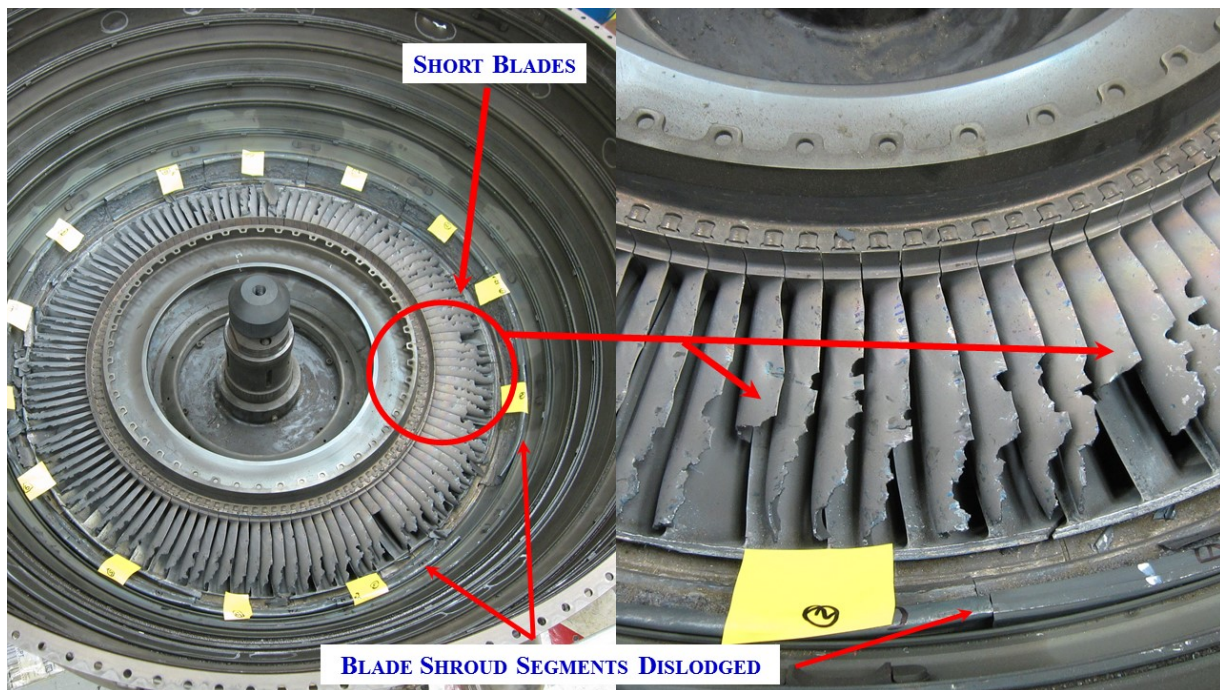


PHOTO 41: LPT STAGE 1 BLADE DAMAGE

The disk forward flange exhibited continuous circumferential scoring around 80% on the ID surface; about 20% of the scoring resulted in a cut through and slight separation of that flange (**PHOTO 42**).

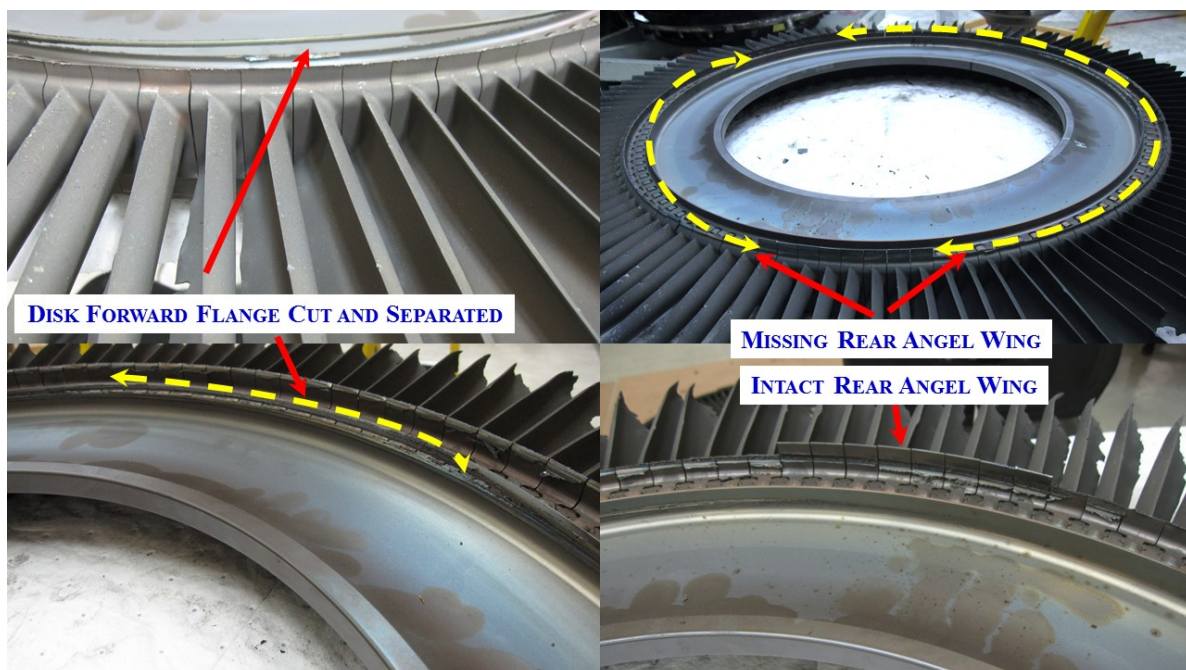


PHOTO 42: LPT STAGE AFT ANGEL WING DAMAGE AND DISK DAMAGE

All the stage 1 blade outer shroud segments were sequentially numbered ALF clockwise 1 to 13 starting counterclockwise from the 12:00 o'clock position. All the blade shroud segments were present, battered, gouged, and the honeycomb was worn down to the backing strip. Blade shroud segments Nos. 3, 4, 5, and 6 had rotated circumferentially within the case in the direction of blade rotation and became wedge underneath blade shroud segment No. 7 leaving a small gap where the stator case skin was exposed, and no shroud segment was present. The blade shroud segments were secured in the LPT case rail by the front "C" channel only and were disengaged from the LPT case at the aft hooks (See **PHOTO 41** and **PHOTO 43**).

All the stage 1 nozzle segments were sequentially numbered ALF clockwise 1 to 13 starting counterclockwise from the 12:00 o'clock position. All the stage 1 nozzle segments were present, no airfoils were missing, and very few minor TE impacts and missing material were noted. The TE inner shroud platform aft face exhibited rub, heat distress, and small pieces of the platform were missing. The TE outer shroud platform was more damaged and the nozzle segments Nos. 3 to 5 from the 2:00 to 4:00 o'clock position were missing the rear platform. The stage 1 nozzles are coated with a white thermal barrier coating (TBC) applied on the outboard portion of the airfoil. Only remnants of the TBC were observed on suction side of segments Nos. 6 and 7. The coating was only on the suction side edge none were observed on the pressure side (**PHOTO 43**). The LE of the stage 1 nozzles exhibited no impact damage or heat distress and all the interstage temperature (ITT) probes were intact, whole, and undamaged (**PHOTO 44**); one vane exhibited a tight LE pressure side crack that measured about 0.7-inches.

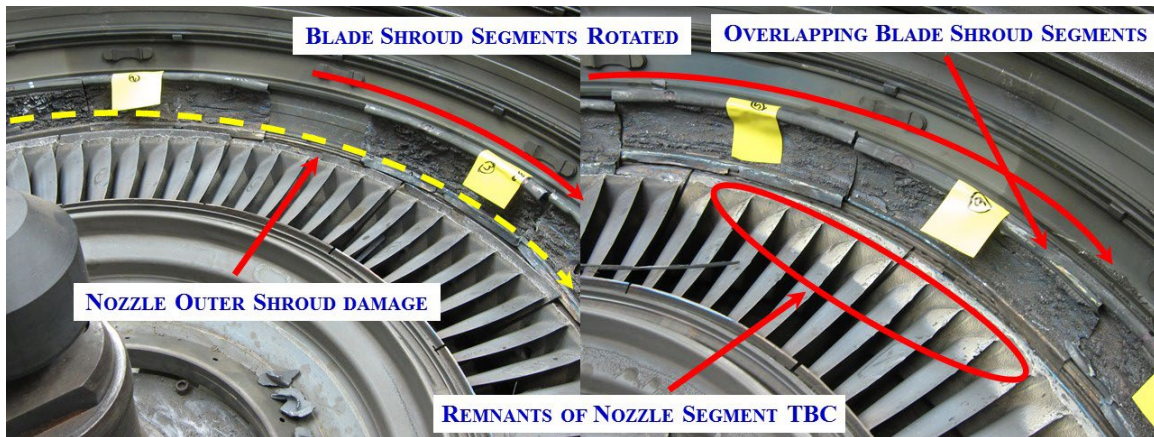


PHOTO 43: LPT STAGE 1 BLADE SHROUD AND NOZZLE SEGMENT DAMAGE



PHOTO 44: LPT STAGE 1 NOZZLE SEGMENTS – LE FRONT VIEW

3.0 METALLURGICAL EXAMINATION OF LPT PARTS AND FRACTURED OIL LINE

All the LPT hardware was sent to the GE metallurgical laboratory in Evendale, OH for analysis; their findings are provided in [ATTACHMENT 1](#) titled ‘*Mas Air Cargo ESN 695440 Failure Analysis Findings & Root Cause Summary*’, dated June 4, 2021. See [ATTACHMENT 1](#) for completed details of the metallurgical examination.

Examination of all the LPT blades showed signs consistent with overstress and the TRF struts showed bulk deformation and the fracture surfaces exhibited a rough appearance consistent with overstress. As previously stated, none of the fractured LPT blades were at or near the blade platform. The only LPT part that exhibited signs of fatigue was the stage 5 nozzle segment No. 3 that was completely missing two adjacent airfoils (airfoils Nos. 5 and 6) (See [PHOTO 36](#) upper right and left photo). Even though most of the fracture surface of what remained of airfoils Nos. 5 and 6 were heavily damaged due to post-event impact damage, binocular examination of the remnants revealed regions consistent with fatigue on both the outboard and inboard fracture surfaces. The outboard fracture surfaces on both the Nos. 5 and 6 airfoils showed cracking extending from the LE and TE edges towards the middle ([PHOTOS 45 and 46](#)). Under higher magnification using a scanning electron microscope (SEM), the fracture surface showed signs consistent with high cycle fatigue (HCF) growth from both the LEs and TEs ([PHOTO 47](#); picture taken from [ATTACHMENT 1](#)). In the interpretable areas (not damaged area), GE found no material anomalies in the base material, aside from the areas of intergranular oxidation (IGO). Due to the LE and TE impact damage on airfoils Nos. 5 and 6 at the fracture locations, no determination could be positively made on whether a repair had been previously performed in that area, so no assessment as the repair conformity was made.

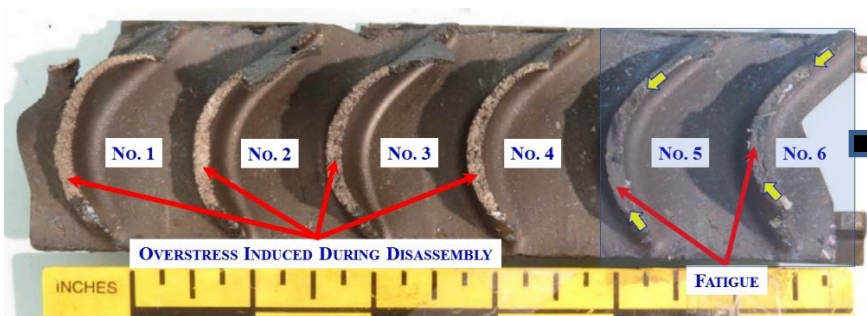


PHOTO 45: STAGE 5 NOZZLE OUTER SHROUD AIRFOIL FATIGUE FRACTURES – TOP VIEW

PHOTO COURTESY OF GE

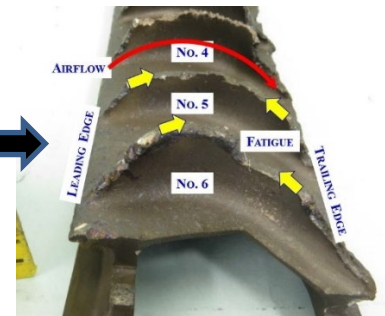


PHOTO 46: FATIGUE FRACTURES – AIRFOIL VIEW

PHOTO COURTESY OF GE

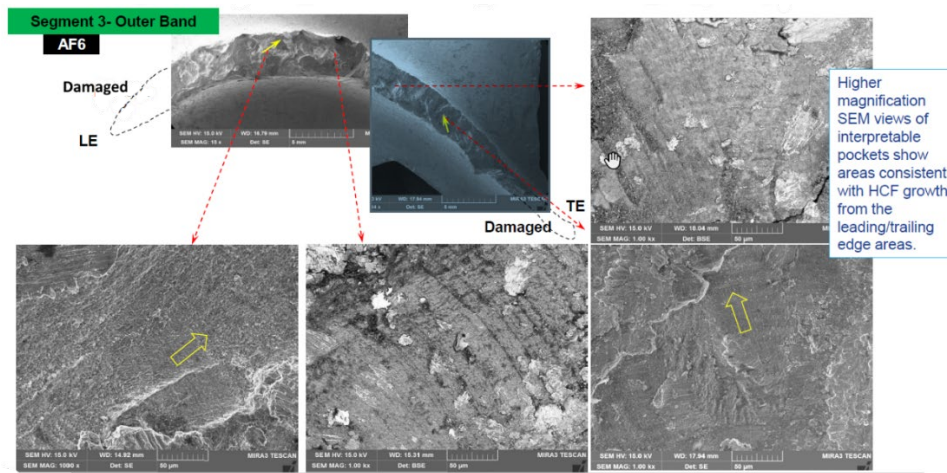


PHOTO 47: SEM VIEW OF FATIGUE ON STAGE 5 NOZZLE SEGMENT NO. 3, AIRFOIL 6

PHOTO COURTESY OF GE

Examination of the cracks and tears around the TRF bosses (See **PICTURES 20** and **34**) showed signs of bulk deformation and rough fracture features consistent with overstress.

The fractured oil supply (pressure) line (See **PHOTO 17**) fracture surfaces exhibited plastic deformation and a planar fracture consistent with secondary high amplitude fatigue/cyclic tensile loading (**PHOTO 48**; picture taken from **ATTACHMENT 1**).

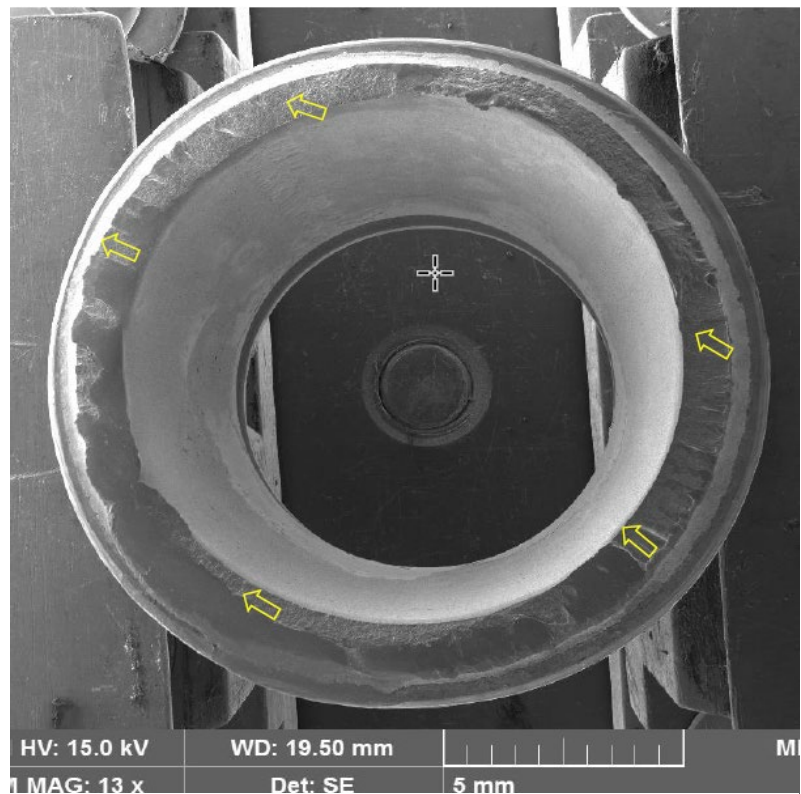


PHOTO 48: SEM VIEW OF FATIGUE ON OIL SUPPLY TUBE

PHOTO COURTESY OF GE

4.0 LOW PRESSURE TURBINE REPAIR HISTORY

The engine was removed and sent to DAL TechOps in August 2019 for maintenance; the reason for removal was Life Limited Parts (LLP) expiration. During the August-December 2019 shop visit all the LPT rotor disks and the LPT shaft were replaced due to limited time left on the life limit of the parts and the TRF underwent heavy maintenance to incorporate service bulletin (SB) 72-1259 revision 2 oil supply tube improvement.

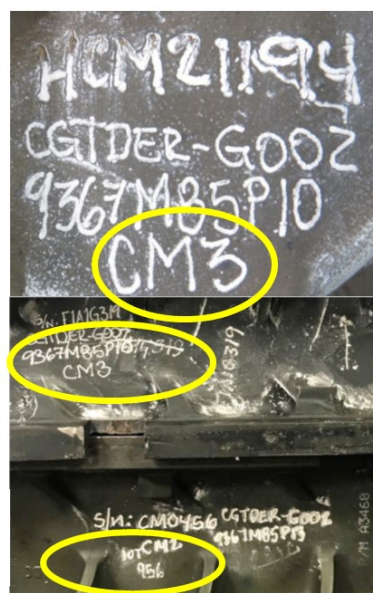


STAGE 5 NOZZLE SEGMENTS NOS. 2-4

STAGE 5 NOZZLE
SEGMENT NO. 3
REPAIR MARKS

STAGE 5 NOZZLE
SEGMENT NO. 2
REPAIR MARKS

STAGE 5 NOZZLE
SEGMENT NO. 4
REPAIR MARKS



STAGE 5 NOZZLE SEGMENT REPAIR MARKING

PHOTO 49: PART MAKING FOR LPT STAGE 5 NOZZLE SEGMENTS NOS. 2-4

PHOTO COURTESY OF GE

According to the 2014 DAL maintenance records provided, the LPT stage 5 nozzle segment PN 9367M85P10, SN HCM21194 was repaired by Chromalloy Southwest, FAA repair station number CZ7Y045N in Mexicali Baja California, Mexico; Chromalloy issued FAA form 8130 Airworthiness Approval Tag indicating that it was repaired in accordance with DAL 01 Repair, CM3 on October 24, 2014 (FIGURE 9). The DAL 01 Repair document specifies that “CHROMALLOY SOUTHWEST is to repair nozzles in accordance with their procedure identified as 93-CLA-002-17.”; the repair is titled “Full Repair in Conjunction with Compatible Alloy Braze Repair of CF6-80C2 Stage 2-5 LPT Nozzle Segments”.

1. Approving Civil Aviation Authority/Country: FAA/United States		2. AUTHORIZED RELEASE CERTIFICATE FAA Form 8130-3, AIRWORTHINESS APPROVAL TAG		3. Form Tracking Number: REF. No F44122 778602-000	
4. Organization Name and Address: CHROMALLOY		Chromalloy S.A. de C.V. GALAXIA 91 21210 MEXICALI BAJA CALIFORNIA MEXICO FAA REPAIR STATION CZ7Y045N		5. Work Order/Contract/Invoice: 4700680768 JOB ORDER #: C028953	
6. Item:	7. Description:	8. Part Number:	9. Quantity:	10. Serial Number:	11. Status/Work:
1	CF6-80C2 LPT VANE STAGE 5	9367M85P10	1	HCM21194	OVERHAULED
12. Remarks: Inspect to: OEM WITH DELTA SUPPLEMENTAL INFORMATION CF6-80C2 Engine Manual (GEK92451) 72-96-02 Rel. 80-00 PUB. 38622 Repaired to: DAL Repair 01 Repair CM3 ESN 695446					
13a. Certifies the items identified above were manufactured in conformity to: ☐ Approved design data and are in condition for safe operation. ☐ Non-approved design data specified in block 12.			14a. ☒ 14 CFR 43.9 Return to Service ☐ Other regulations specified in Block 12. Certifies that unless specified in block 12, the work identified in block 11 and described in block 12 was accomplished in accordance with Title 14, Code of Federal Regulations, Part 43 and in respect to that work, the items are approved for return to service		
13b. Authorized Signature:		13c. Approval/Authorization No.:		14b. Authorized Signature:	
13d. Name (Typed or printed):		13e. Date (dd/mm/yyyy):		14c. Approval/Certificate No.: CZ7Y045N	
				14d. Date (dd/mm/yyyy): 24/Oct/2014	
User/Installer Responsibilities It is important to understand that the existence of this document alone does not automatically constitute authority to install the aircraft engine/propeller/article. Where the user / installer performs work in accordance with the national regulations of an airworthiness authority different than the airworthiness authority of the country specified in Block 1, it is essential that the user/installer ensures that his/her airworthiness authority accepts aircraft engine(s)/propeller(s)/article(s) from the airworthiness authority of the country specified in block 1. Statements in block 13a and 14a do not constitute installation certification. In all cases, aircraft maintenance records must contain an installation certification issued in accordance with the national regulations by the user/installer before the aircraft may be flown.					

FIGURE 9: LPT STAGE 5 NOZZLE SEGMENT PN 9367M85P10,
SN HCM21194 REPAIR AND AIRWORTHINESS TAG

DAL provided a copy of FAA Designated Engineering Representative (DER) APPROVED REPAIR Control Number 93-CLA-002-17. This repair applies to LPT stages nozzle segments Nos. 2-5 of a certain PNs. According to the summary section of Control Number 93-CLA-002-17, the repair “can be summarized as: receive customer vane and record markings, visually inspect for reparability, clean, remove external coating, ultrasonic inspect airfoil wall thickness, notch and rout cracks, TIG weld,

fluoride ion clean, high vacuum cleaning, braze, vacuum brazing, belt sand, final machining, EDM, final FPI, final blend, reapply external coating, heat treat, final dimensional inspect, final inspect.”

The PN of the fatigued LPT stage 5 nozzle segment, PN9367M85P10, is authorized to have this repair performed. Chromalloy confirmed the CGTDER-G002 and the CM3 markings indicate the DER repair 93-CLA-002-17. At the time of the October 2014 shop visit at DAL, the engine had accumulated 43,950 hours TSN and 16,588 CSN; thus, the LPT stage 5 nozzle segment PN 9367M85P10, SN HCM21194 had been in service for 10,232 hours and 4,068 cycles since the overhaul repair performed by Chromalloy in 2014.

5.0 LOW PRESSURE TURBINE IMBALANCE AND ROTOR DEFLECTION CALCULATIONS

GE performed a LPT rotor imbalance and deflection computation based on the as-received post-event condition of the LPT. It should be noted that this may not represent the worst-case rotor imbalance or deflection, or the imbalance and deflection that may have occurred in flight as the exact condition of the LPT hardware at any point during the flight was unknown and the airplane, which is equipped with an engine vibration monitoring system, only measures vibration up to a maximum of 5 cockpit scaler units⁷. Any vibration values higher than 5 cockpit units (CU) are recorded on the flight data recorder (FDR) are capped to 5 CUs. See Section 7.0 in this report for more details on the vibration levels recorded during the flight.

To support the LPT imbalance calculation, the lengths of the blades that remained installed in the disk from each stage, along with its location, were measured and used as inputs into the static imbalance calculations. All the airfoil debris that was collected was not used for the imbalance calculation; it was only used as part of the metallurgical evaluation. For the imbalance load and deflection calculation, a N1 rotor speed of 3,450 N1 was used which corresponds to the N1 rotor speed shortly after the vibrations reached 5 CUs (See [TABLE 2](#) for event timeline). [TABLE 1](#) provides predicted deflections for each stage at the given N1 rotor speed of 3,450 while [CHART 1](#) shows the predicted deflections just for the LPT stage 5 at various rotor speeds as the rotor decelerates. According to GE, the LPT nominal radial rotor running blade clearances at takeoff is 0.080-0.120-inches; the rotor deflection values calculated for each stage were several times greater than the maximum blade clearance, thus heavy blade tip and outer shroud contact would be anticipated ([ATTACHMENT 1](#), pages 37-40).

TABLE 1: LPT IMBALANCE AND DEFLECTION CALCULATIONS	
	Deflection (mils) ⁸
Stage 1 LPT rotor	191
Stage 2 LPT rotor	234
Stage 3 LPT rotor	288
Stage 4 LPT rotor	369
Stage 5 LPT rotor	431

⁷ A CU is a dimensionless scaler unit, a magnitude with no unit of measure attached.

⁸ A mil is a thousandth of an inch. 1 mil = 0.001-inch

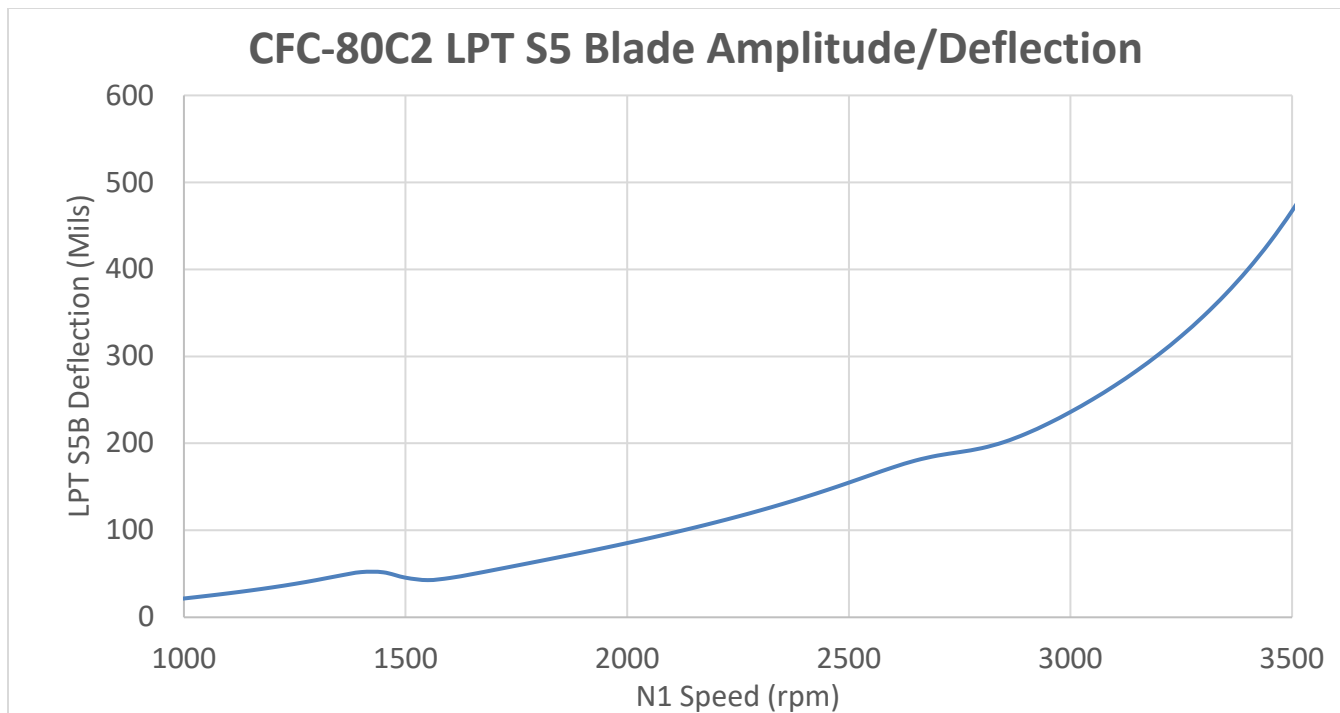


CHART 1: LPT STAGE 5 AMPLITUDE AND DEFLECTION CURVES FOR VARIOUS N1 ROTOR SPEEDS

CHART COURTESY OF GE

6.0 AIRBORNE VIBRATION MONITORING DATA

The airplane was equipped with an airborne vibration monitoring (AVM) system that continuously monitors the engine vibration levels and provides an early warning of engine malfunction to permit corrective actions to be taken before extensive damage results. The system consists of two accelerometers (vibration pickups) for each engine; one is mounted on the No. 1 bearing housing in oil sump “A” with its external lead connector on the No. 8 fan strut and the other is mounted on the forward side of the flange between the compressor rear frame (CRF) and the compressor stator case at the 11:00 o’clock position; the vibration sensor on the CRF is also commonly referred to LPT vibration sensor. As part of the imbalance calculation work that GE performed (See Section 5.0), GE calculated deflection values that the CFR/LPT vibration sensor would have experienced during this event. Based on their calculations, the largest rotor deflections were not at the event rotor speed of roughly 3,450 N1 rpm but as the engine decelerates and passed through about 2,700 N1 rpm which is about 82% N1 speed ([CHART 2](#)). The amount of deflect calculated for the LPT event was, according to GE, comparable to the deflection data for a CF6-80C2 engine fan blade out test and is well above the AVM limit of 5 CUs.

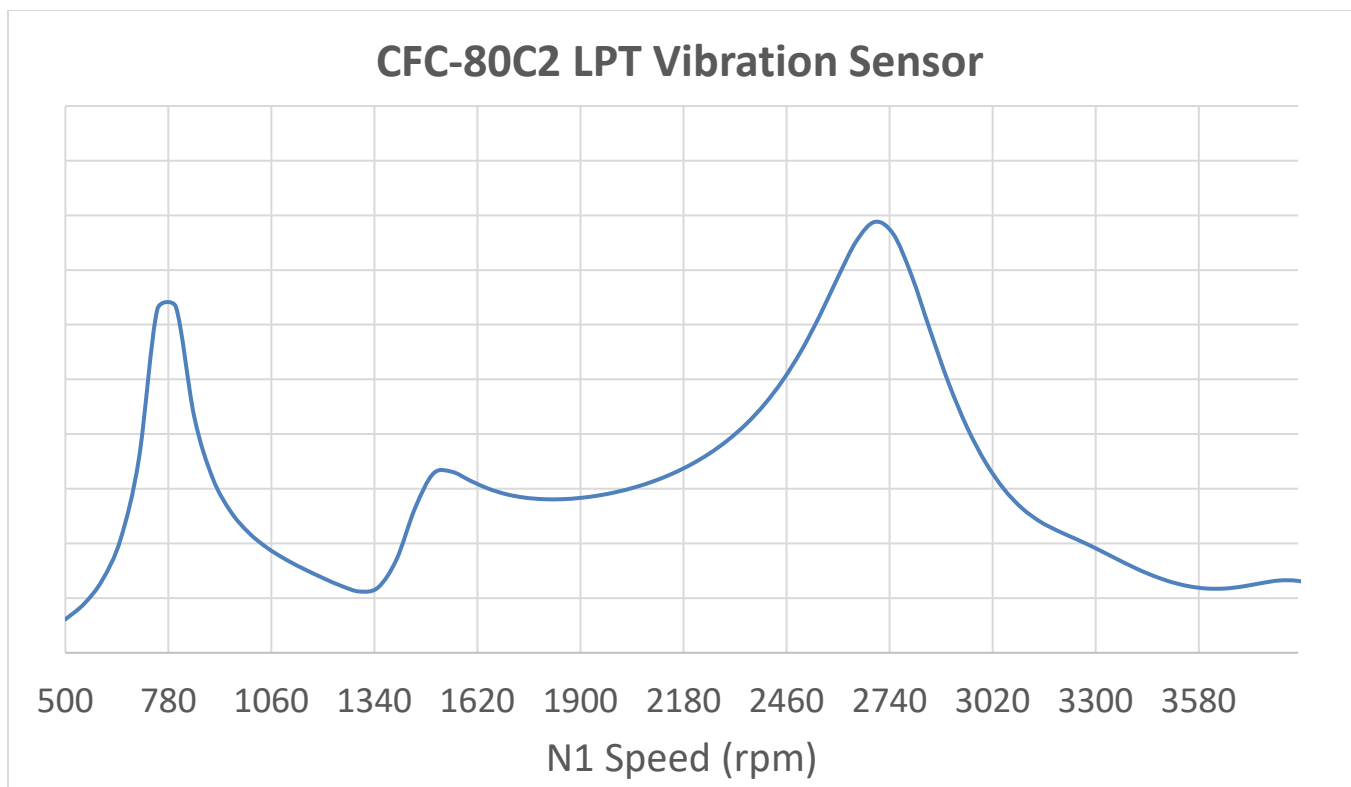


CHART 2: CFR/LPT VIBRATION SENSOR DEFLECTION CALCULATION VARIOUS N1 ROTOR SPEEDS
 CHART COURTESY OF GE

With the engine operating, the engine accelerometers generate signals proportional to engine motion in a radial direction. These signals are received by the AVM (also referred to as the engine vibration monitor unit (EVMU)) where they are sent to the Engine Indicating and Crew Alerting System (EICAS) computer. Signals from the EICAS computer are then simultaneously sent to the two EICAS center display units (CDU), referred to as the right and left display (although they are mounted in the flight compartment vertically) where vibration displacement or velocity is displayed and to the Flight Data Acquisition Unit (FDAU). In the CF6-80C2 engine installation, vibration is conveyed in terms of displacement and is displayed in terms of CUs. The EICAS system is an automatic engine monitoring and centralized alerting system that displays for engine parameters and crew alerts and provides status information and ground maintenance data. Data from FDAU is then sent to the FDR but before the data is sent, the FDAU encoded and converted the signals suitable for the FDR to receive, record, and store (**FIGURE 10**). According to the Boeing Aircraft Data Frame for 767-300, Appendix D, and the Supplemental Type Certificate (STC) for the Teledyne Digital Flight Data Acquisition Unit (DFDAC), the source of the AVM vibration values, and the AVM alert Yellow/White for both Nos. 1 and 2 engines are from the EICAS.

Within the AVM is a signal conditioner that receives two accelerometer signals from each engine along with the N1 and N2 speed sensors. The AVM signal conditioner is programmed to determine the N1 rotor vibration from the fan sensor and N2 rotor vibration from the core sensor for the HPT and LPT. The Boeing 767/CF6-80C2 combination does not provide an engine vibration alert annunciation or indication in the flight deck for the flight crew even though the FDR does record a “AVM ALERT” (See Section 7.0). The vibration source with the high vibration level for each engine is displayed on the secondary EICAS display along with the location of the source.

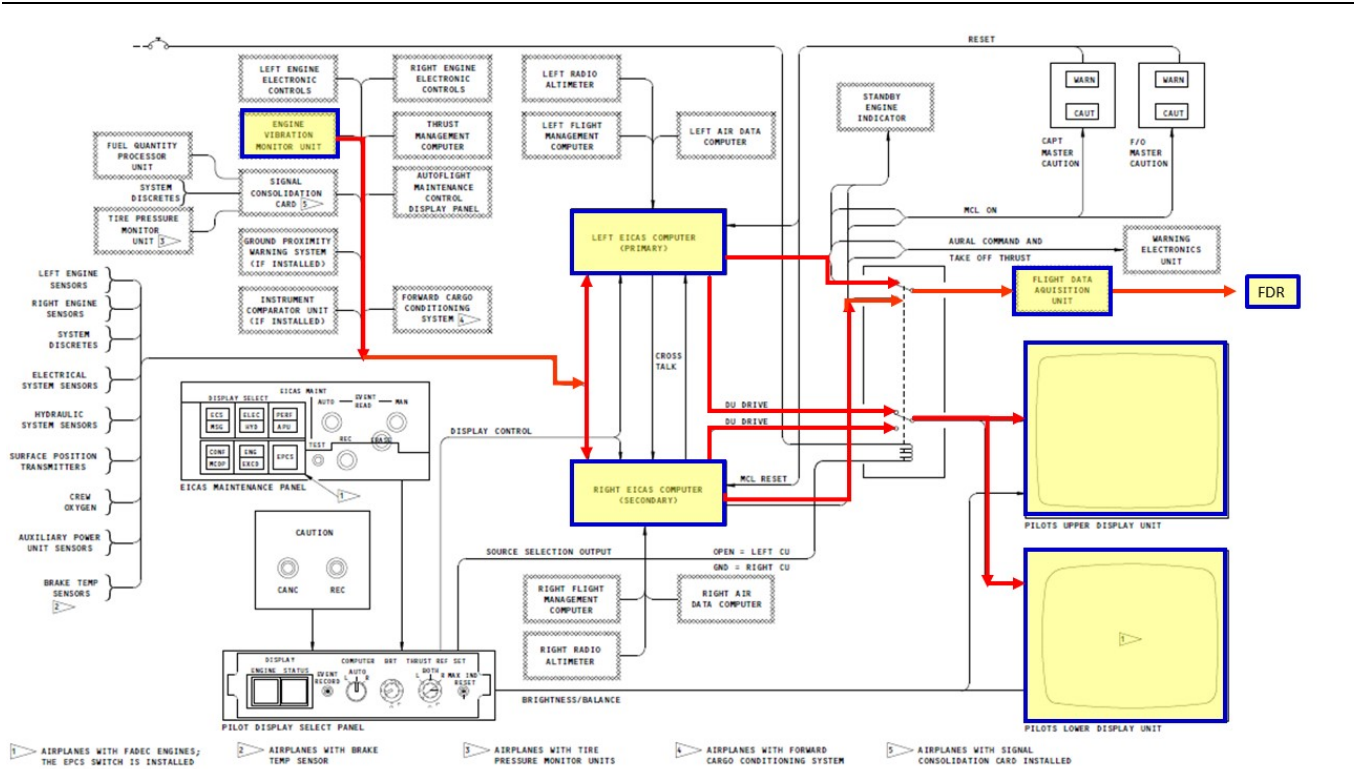


FIGURE 10: EICAS SIMPLIFIED BLOCK DIAGRAM

FIGURE COURTESY OF BOEING - ©BOEING USED WITH PERMISSION

7.0 FLIGHT DATA RECORDER DATA

The FDR was sent to the NTSB's Headquarters in Washington DC and was readout by the Vehicle Recorder Group. A sequence of events timeline was created based on data from the FDR for the incident flight. It should be noted that parameters are sampled at different rates based on the need for the fidelity of data and are recorded at different times. For example, the fire warning, master caution, engine fuel cutoff, airplane airspeed, airplane altitude, engine vibration, engine fan speed (N1), engine core speed (N2), engine exhaust gas temperature (EGT), engine fuel flow (FF), throttle resolver angle (TLR) and auto throttle engagement/designment are all sampled every second; the engine fire handle (extinguishing switch) is sampled every 4 seconds; and the weight of wheels is sampled 4 times every second. None of these parameters are recorded at the exact same time but are staggered timewise from one another and even parameters that are recorded in the same second may not have occurred at the same time as the seconds are further divided into fractions of a second. Times are rounded to the nearest whole second for simplicity. See **FIGURE 9** and **TABLE 2** for the timeline and see the FDR Group Chairman's Report for this investigation for additional data. Time T = 0 is when the fire warning is annunciated; negative time is time prior to the fire annunciation and positive time is time after the fire annunciation.

There are three engine vibration recorded parameters for this engine/airframe (Boeing 767/GE CF6-80C2) combination: Fan, LPT and N2 (core). Review of the FDR revealed that the ENG2 Filter1 Vib (scaler number) corresponded to the Fan, ENG2 Filter2 Vib (scaler number) corresponded to the LPT, and ENG2 Filter3 Vib (scaler number) corresponded to the N2. In addition to the specific engine vibration filters listed above, the FDR also provided three additional vibration parameters: 1) a ENG2 Vib value (scaler number) that corresponds to one of the three specific vibration filter values (scaler number), 2) a ENG2 Vib Indent (word value – Fan, LPT, or N2) that corresponds to one of the three specific vibration filter values, and 3) AVM Alert ENG2 (word value – ON or OFF). Which ENG Filter(X) Vib value

corresponded to which vibration location (Fan, LPT, or N2) was determined by comparing the ENG2 Filter(X) Vib scaler values with the scaler value recorded for the generic ENG2 Vib and its corresponding ENG2 Vib Indent.

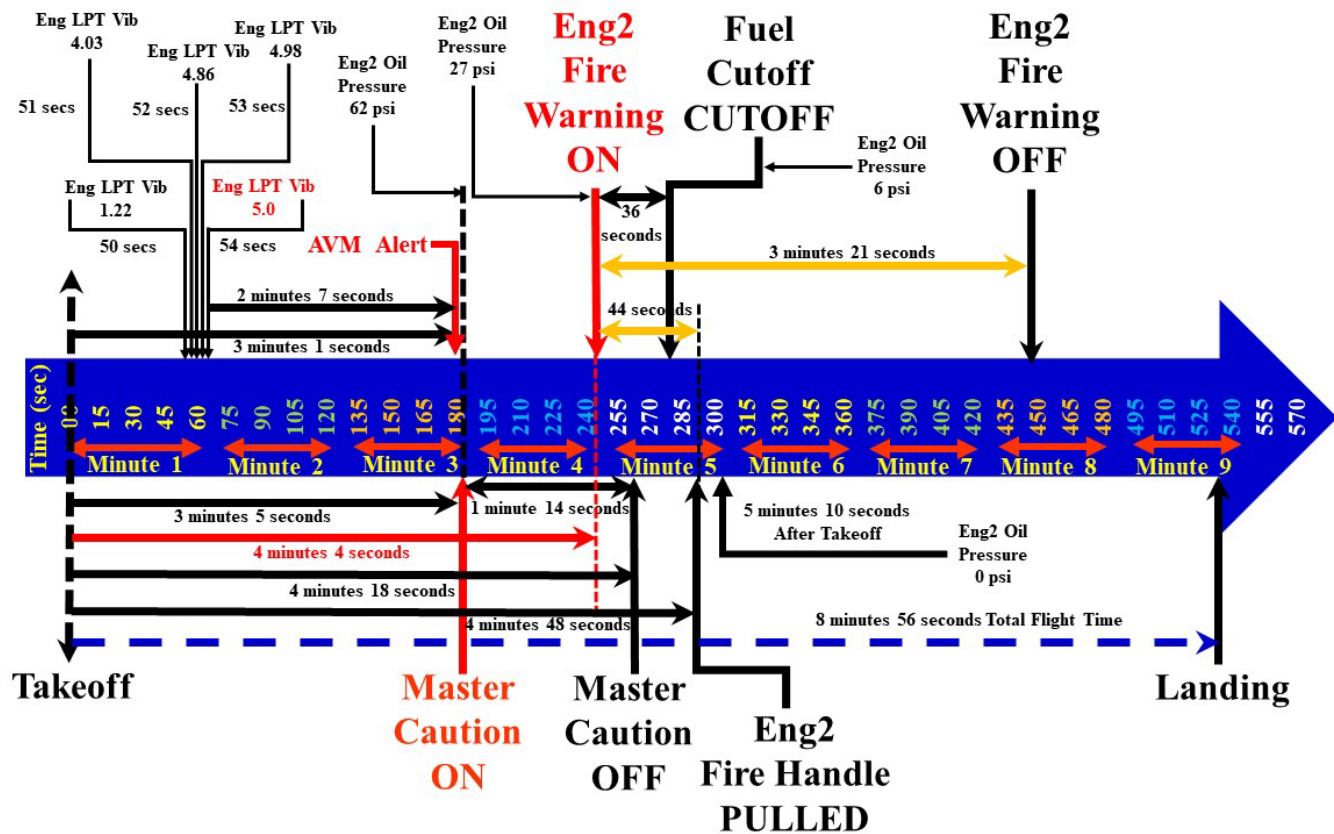


FIGURE 11: FLIGHT AND FIRE INCIDENT TIMELINE

TABLE 2: INCIDENT TIMELINE		All values relate to the No. 2 engine unless otherwise noted	
FDR data (minutes/seconds)			
T = -244s (-4 min 4 secs) 12:45:33 UTC	T = 0 is when the fire warning is illuminated Airplane Takeoff (T/O) – Weight on Wheels (WOW) – AIR MODE		
T = - 195s -3min 15 sec 12:46:23 UTC 50 sec after T/O	Airspeed 200 knots Airplane climbing though 1,584 feet Fuel Cutoff - Run	Vib Filter1 = 0.69 units Vib Filter2 = 0.22 units Vib Filter3 = 1.27 units Eng Vib = 1.28 units ⁹ (N2) AVM Alert OFF	
T = - 194s -3min 14 sec 12:46:24 UTC 51 sec after T/O	Airspeed 202 knots Airplane climbing though 1,607 feet Fuel Cutoff - Run	ENG2 N1 at 97.13% ENG2 N2 at 104.38% ENG2 TRA at 113.73° Fuel Flow at 18,400 lbs./hr. ¹⁰ EGT at 828°C	Vib Filter1 = 4.01 units Vib Filter2 = 4.03 units Vib Filter3 = 1.23 units Eng Vib at 4.25 units (LPT) AVM Alert OFF
T = - 193s -3min 13 sec 12:46:25 UTC 52 sec after T/O	Airspeed 202 knots Airplane climbing though 1,632 feet Fuel Cutoff - Run	ENG2 N1 at 97.38% ENG2 N2 at 104.13% ENG2 TRA at 113.55° Fuel Flow at 18,624 lbs./hr. EGT at 836°C	Vib Filter1 = 4.81 units Vib Filter2 = 4.86 units Vib Filter3 = 1.21 units Eng Vib = 4.9 units (LPT) AVM Alert OFF
T = - 192s -3min 12 sec 12:46:26 UTC 53 sec after T/O	Airspeed 204.5 knots Airplane climbing though 1,654 feet Fuel Cutoff - Run	ENG2 N1 at 97.63% ENG2 N2 at 104.38% ENG2 TRA at 113.73° Fuel Flow at 18,880 lbs./hr. EGT at 842°C	Vib Filter1 = 4.97 units Vib Filter2 = 4.98 units Vib Filter3 = 1.17 units Eng Vib = 4.99 units (LPT) AVM Alert OFF
T = -191s -3min 11 sec 12:46:27 UTC 54 sec after T/O	Airspeed 206 knots Airplane climbing though 1,628 feet Fuel Cutoff - Run	ENG2 N1 at 97.75% ENG2 N2 at 104.38% ENG2 TRA at 113.73° Fuel Flow at 19,040 lbs./hr. EGT at 844°C	Vib Filter1 = 4.99 units Vib Filter2 = 4.99 units Vib Filter3 = 1.16 units Eng Vib = 5 units (LPT) AVM Alert OFF
T = -63s 12:48:34 UTC 3 min 1 sec after T/O	Airspeed 251 knots Airplane climbing though 6,318 feet Fuel Cutoff - Run	ENG2 N1 at 72.13% ENG2 N2 at 99.63% ENG2 TRA at 113.55° Fuel Flow at 16,864 lbs./hr. EGT at 951°C	Vib Filter1 = 4.99 units Vib Filter2 = 4.99 units Vib Filter3 = 0.88 units Eng Vib = 5 units (LPT) AVM Alert ON

⁹ Even though the vibration filter values (ENG2 Filter(X) Vib) and engine vibration value (ENG2 Vib) are recorded within the same second, the filter values are recorded ahead of the engine vibration value by fractions of a second; therefore, the filter value and the engine vibration value may not always agree exactly.

¹⁰ lbs./hr. is pounds per hour

T = -62s 12:48:35 UTC 3 min 2 sec after T/O	Auto Throttle - DISENGAGED Airspeed 252 knots Airplane climbing through 6,343 feet Fuel Cutoff – Run	ENG2 N1 at 69.88% ENG2 N2 at 98.38% ENG2 TRA at 113.91° Fuel Flow at 15,936 lbs./hr. EGT at 934°C ENG2 Oil Pressure 64 psi	Vib Filter1 = 4.99 units Vib Filter2 = 4.99 units Vib Filter3 = 0.74 units Eng Vib = 5 units (LPT) AVM Alert ON
T = -59s 12:48:38 UTC 3 min 5 sec after T/O	MASTER CAUTION – ON (132 secs after max Eng Vib = 5 units) (Master Caution Triggered by Auto Throttle (AT) Disconnected) Airspeed 252 knots Airplane climbing through 6,403 feet Fuel Cutoff – Run	ENG2 N1 at 67% ENG2 N2 at 96.5% ENG2 TRA at 113.91° Fuel Flow at 13,696 lbs./hr. EGT at 895°C ENG2 Oil Pressure 62 psi	Vib Filter1 = 4.99 units Vib Filter2 = 4.99 units Vib Filter3 = 0.88 units Eng Vib = 5 units (LPT) AVM Alert ON
T = - 52s to -30s 12:48:45 UTC to 12:49:07 UTC		ENG2 Oil Pressure 60 to 57 psi	
T = - 28s to 24s 12:49:09 UTC to 12:49:13 UTC		ENG2 Oil Pressure 56 to 53 psi	
T = -23s 12:49:14 UTC 3 min 41 sec after T/O		ENG2 Oil Pressure 48 psi	
T = -22s 12:49:15 UTC 3 min 42 sec after T/O		ENG2 Oil Pressure 34 psi	
T = -21s 12:49:16: UTC 3 min 43 sec after T/O		ENG2 Oil Pressure 29 psi	
T = -20s 12:49:17 UTC 3 min 44 sec after T/O		ENG2 Oil Pressure 27 psi	
T = 0s 12:49:37 UTC 4 min 4 sec after T/O	ENG2 Fire - ON (59 secs after Master Caution) Master Caution - ON Airspeed 250 knots Airplane climbing through 7,949 feet Fuel Cutoff - Run	ENG2 N1 at 62.88% ENG2 N2 at 94% ENG2 TRA at 113.91° Fuel Flow at 10,240 lbs./hr. EGT at 788°C ENG2 Oil Pressure 27 psi	Vib Filter1 = 4.99 units Vib Filter2 = 4.99 units Vib Filter3 = 0.67 units Eng Vib = 5 units (LPT) AVM Alert ON
T = 14s 12:49:51 UTC 4 min 18 sec after T/O	ENG2 Fire - ON MASTER CAUTION -OFF (Master Caution on for 1 min 13 sec) Fuel Cutoff – Run Fire Extinguishing Switch - OFF	ENG2 N1 at 64.38% ENG2 N2 at 94.25% ENG2 TRA at 114.08° Fuel Flow at 10,208 lbs./hr. EGT at 772°C ENG2 Oil Pressure 27 psi	Vib Filter1 = 4.99 units Vib Filter2 = 4.99 units Vib Filter3 = 0.69 units Eng Vib = 5 units (LPT) AVM Alert ON
T = +36s 12:50:13 UTC 4 min 40 sec after T/O	ENG2 Fire - ON Master Caution – OFF	ENG2 N1 at 26.75% ENG2 N2 at 65.38% ENG2 TRA at 50.62°	Vib Filter1 = 3.37 units Vib Filter2 = 4.99 units Vib Filter3 = 0.68 units

	Fuel Cutoff –CUTOFF (Fuel Cutoff 36 sec after Fire Warning) Fire Extinguishing Switch – OFF	Fuel Flow at 1,248 lbs./hr. EGT at 412°C ENG2 Oil Pressure 6 psi	Eng Vib = 5 units (LPT) AVM Alert ON
T = +44s 12:50:21 UTC 4 min 48 sec after T/O	ENG2 Fire - ON Master Caution – OFF Fuel Cutoff – OFF Fire Extinguish Switch - PULLED (Last recorded fire switch OFF at 50:17) (1 st recorded fire switch Pulled at 50:21) (Therefore, fire handed pulled between 40 and 44 seconds after Fire Warning)	ENG2 N1 at 19.38% ENG2 N2 at 41.13% ENG2 TRA at 50.63° Fuel Flow at 0 lbs./hr. EGT at 233°C ENG2 Oil Pressure 3 psi	Vib Filter1 = 2.38 units Vib Filter2 = 4.99 units Vib Filter3 = 0.53 units Eng Vib = 5 units (LPT) AVM Alert ON
T = +49s 12:50:26 UTC 4 min 53 sec after T/O		ENG2 N1 at 18.63% ENG2 N2 at 34.5% ENG2 TRA at 50.62° Fuel Flow at 0 lbs./hr. EGT at 196°C ENG2 Oil Pressure 2 psi	Vib Filter1 = 0.33 units Vib Filter2 = 1.42 units Vib Filter3 = 0.03 units Eng Vib = 1.43 units (LPT) AVM Alert OFF
T = + 65s 12:50:42 UTC 5 min 9 sec after T/O		ENG2 N1 at 18.13% ENG2 N2 at 24.63% ENG2 TRA at 50.62° Fuel Flow at 0 lbs./hr. EGT at 141°C ENG2 Oil Pressure 0 psi	Vib Filter1 = 0 units Vib Filter2 = 0 units Vib Filter3 = 0 units Eng Vib = 0 units (LPT) AVM Alert OFF
T = +201s 12:52:58 UTC 7 min 25 sec after T/O	ENG2 Fire - OFF (3 min 21 seconds after Fire Warning) Master Caution – OFF Fuel Cutoff – OFF	ENG2 N1 at 21.38% ENG2 N2 at 20.38% ENG2 TRA at 50.62° Fuel Flow at 0 lbs./hr. EGT at 62°C ENG2 Oil Pressure 0 psi	Vib Filter1 = 0 units Vib Filter2 = 0 units Vib Filter3 = 0 units Eng Vib = 0 units (LPT) AVM Alert OFF
T = +291s 12:54:28 UTC 8 min 55 sec after T/O	Airplane Landing – WOW – GRND MODE		

9.0 HIGH ENGINE VIBRATION GUIDANCE

There are no specific procedures for engine high vibrations itself, but it is covered in part in the Boeing 767 Aircraft Flight Manual (AFM) Engine Fire, Severe Engine Damage, or Separation Emergency procedures (**FIGURE 12**) and in the Flight Crews Operations Manual (FCOM) Engine Failure or Shutdown procedure (**FIGURE 13**)¹¹; in both cases high vibration is dealt with as part of a broader engine failure event. Boeing Flight Crew Training Manual (FCTM) provides additional general guidance in the event of high engine vibrations such as reducing speed and altitude to reduce the vibration level, but it does not provide specific guidance relating vibration levels, such as what actions should be taken if for example 3, 4, or 5 vibration CUs.

¹¹ According to the FAA's Advisory Circular (AC) No. 25-1581-1, titled Airplane Flight Manual, the following definitions apply: Airplane Flight Manual (AFM) is an FAA-approved document that contains information (operating limitations, operating procedures, performance information, etc.) necessary to operate the airplane at the level of safety established by the airplane's certification basis and the Flight Crew Operating Manual (FCOM) is a document developed by a manufacturer that describes, in detail, the characteristics and operation of the airplane or its systems

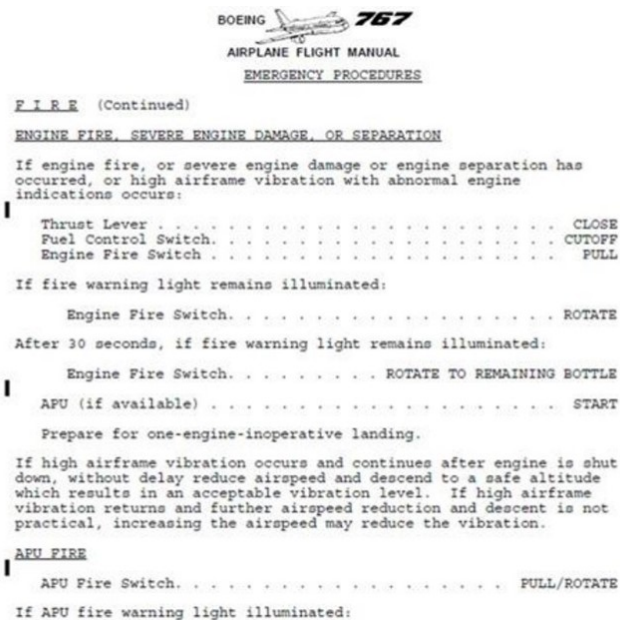


FIGURE 12: AFM ENGINE SEVERE DAMAGE CHECKLIST
FIGURE COURTESY OF MAS AIR - ©BOEING USED WITH PERMISSION

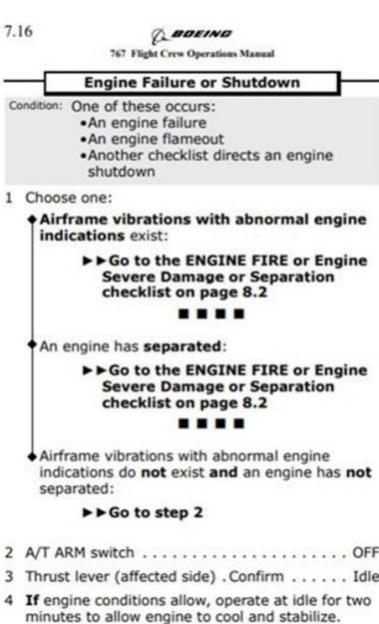


FIGURE 13: FCOM ENGINE FAILURE CHECKLIST
FIGURE COURTESY OF MAS AIR- ©BOEING USED WITH PERMISSION

10.0 METALLURGICAL EXAMINATION OF EXHAUST HARDWARE

The turbine exhaust sleeve and the exhaust plug were shipped to the Boeing Equipment Quality Analysis (EQA) laboratory in Seattle Washington and the Boeing Research and Technology (BR&T) group performed the analysis. The NTSB authorized Boeing to section from the turbine exhaust sleeve fractures of interest for closer examination. Metallurgical analysis of the removed fractures revealed that all the fractures were composed entirely of ductile separation fracture mode, consistent with a single event, and no other anomalies were observed contributing to the fractures. An Energy-Dispersive Spectrometry (EDS) analysis was conducted on the silver metal spatter on the turbine exhaust sleeve (See **PHOTO 12**) and determined that the silver-colored deposits to be consistent with 2000 series aluminum alloy.

Jean-Pierre Scarfo
Aerospace Engineer
Powerplant Lead

ATTACHMENTS

Attachment 1 *GE Mas Air Cargo ESN 695440 Failure Analysis Findings & Root Cause Summary*