



NATIONAL TRANSPORTATION SAFETY BOARD

Office of Aviation Safety

Washington, D.C. 20594

AIRWORTHINESS GROUP CHAIRMAN'S FACTUAL REPORT

February 21, 2018

A. ACCIDENT: WPR17FA128

Operator: Private
Location: Payette, Idaho
Date: June 17, 2017
Time: 1145 Mountain Daylight Time
Aircraft: Experimental Wilkie Prescott E (Vans) RV-3
Registration: N177TT

B. AIRWORTHINESS GROUP

Chairman: Clinton R. Crookshanks
 National Transportation Safety Board
 Denver, Colorado

Member: Jack Vanover
 National Transportation Safety Board
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Member: Robert Nance
 Federal Aviation Administration
 Boise, Idaho

Member: Dan Frandson
 Federal Aviation Administration
 Boise, Idaho

Member: Gus Funnell
 Van's Aircraft, Inc.
 Aurora, Oregon

C. SUMMARY

On June 17, 2017, about 1145 mountain daylight time, an experimental amateur-built Vans RV-3 airplane, N177TT, impacted terrain following the separation of the left wing while maneuvering near Payette, Idaho. The airline transport pilot sustained fatal injuries; the airplane was destroyed. The pilot was operating the airplane under the provisions of Title 14 Code of Federal Regulations (CFR) Part 91. The local personal flight departed Payette Municipal Airport (S75), Payette, Idaho about 1140. Visual meteorological conditions prevailed and no flight plan had been filed

D. DETAILS OF THE INVESTIGATION

1.0 Airplane

The Vans RV-3 is a single-seat, single engine, low-wing, experimental, amateur built, kit airplane sold by Van's Aircraft, Inc. (Figure 1¹). The airplane is of all metal construction, has a conventional tail and is equipped with fixed conventional landing gear. The airplane is 19 feet long, has a wingspan of 19 feet, 11 inches, and a tail height of 5 feet. The accident airplane was powered by a four-cylinder reciprocating engine driving a 2-blade constant speed tractor propeller. The accident airplane was modified from a standard RV-3 to have an open cockpit configuration.

The left and right wings are essentially identical and built separately before being fastened together at the airplane centerline, wing station (WS²) 0. The original RV-3 main (forward) wing spar was a built-up I-beam structure consisting of, from aft to forward at the airplane centerline, a 0.125-inch thick aluminum web, a 0.040-inch thick aluminum shim, and a stack up of seven 0.125-inch by 1.25-inches aluminum bars for both the upper and lower spar caps. The main spar web changes to a 0.040-inch thick aluminum channel at WS 14 and the 0.125-inch thick web extends to WS 18. The aft most upper and lower spar cap layers (layer 7) are continuous to the wing tip at WS 93.5. The other 6 upper and lower spar cap layers are continuous from WS 0 to the following points on the left and right wings; layer 1-WS 20, layer 2-WS 26.25, layer 3-WS 33, layer 4-WS 41.75, layer 5-WS 52.75, layer 6-WS 68.75. Steel wing fittings, about 12 inches long, are installed on the forward and aft sides of the upper and lower spar caps (4 total fittings) to attach the left and right wings together at the airplane centerline. Each fitting is attached to each wing spar cap with 5 bolts for a total of 10 bolts per fitting. The drawings and assembly instructions specify that the center 8 bolts in each fitting should be NAS1305-26 bolts and the outboard 2 bolts in each fitting should be NAS1304-26 bolts. Outboard of each of the fittings, the 0.125-inch web and the upper and lower spar caps are fastened with 10 additional bolts. The drawings and assembly instructions specify that these 10 bolts should be AN3-14A or AN3-15A bolts. Outboard of the bolt locations the web and spar caps are fastened together with rivets. The wings are attached to the fuselage with all or some of the above bolts depending on the design of the fuselage bulkheads. The main wing spar fits between forward and aft fuselage bulkheads with the bolts installed through the bulkheads, wing fittings, and upper and lower spar caps.

2.0 Flight and Debris Field

¹ All figures are presented in Appendix A to this report.

² Wing stations are measured in inches outboard of the airplane centerline.

Several witnesses were present for the flight and videos of the flight were provided to the investigation. The airplane departed runway 31 at Payette Municipal Airport and made a shallow right turn to head north while climbing. Seconds later the airplane made a sweeping left descending turn to align with runway 13. The airplane made a high speed, low altitude pass over the runway descending to less than 20 feet above ground level (AGL) before executing a sharp pull up to gain altitude. The airplane leveled off several hundred feet AGL and made a left turn to head in a northerly direction. Several seconds later the airplane initiated a descent and, while descending, the left wing separated from the airplane.

The airplane impacted flat farm land northeast of the airport. The main wreckage included the fuselage, engine, empennage, and right wing and was located about 3,025 feet northeast of the departure end of runway 31. The left wing was located about 3,200 feet east-northeast of the departure end of runway 31 and about 1,450 southeast of the main wreckage. The left flap was located about 2,800 feet east-northeast of the departure end of runway 31 and about 1,150 feet south-southeast of the main wreckage (Figure 2).

3.0 Wreckage Examination

The NTSB did not travel to the scene of the accident. Representatives from the FAA conducted the on-scene portion of the investigation. The wreckage was recovered and secured in the owner's hangar at the Payette Municipal Airport. The wreckage was examined by the group at the hangar August 8-9, 2017.

Main Wreckage

The main wreckage contained the entire airplane except for the left wing and left flap. There was no evidence of a post-crash fire. The wreckage was crushed and deformed consistent with impact. The vertical stabilizer, rudder, horizontal stabilizer, and elevators remained attached to the empennage. The rudder exhibited impact damage from the lower connection point to the bottom of the rudder. Rudder control continuity was established from the surface to the rudder pedals. The vertical stabilizer exhibited impact damage about 8 inches from the top. The right horizontal stabilizer exhibited impact damage to the outboard leading-edge. The right horizontal stabilizer and elevator tips were separated. The left horizontal stabilizer exhibited impact damage to the leading edge from about midspan outboard. The left horizontal stabilizer tip was separated. The trim tab was deflected downward about 35°.

Control continuity was established from the elevators to the control horn in the empennage and the elevator moved freely when actuated by hand. The aft elevator control tube was bent in multiple places and fractured at the forward end. The forward elevator control tube was fractured at the aft rod end about two feet aft of the control stick. All control tube separations had a dull, grainy appearance consistent with overstress separation. The access panel was removed below the control stick to access the control stick quadrant. A 5/16 ratcheting box-end wrench was found lodged in the structure near the aileron tube (Figure 3). There was no damage to the lower control stick or aileron tubes to suggest the wrench had jammed the flight controls.

The right wing was located with the main wreckage and had extensive crushing damage and fragmentation. The right aileron control tube was separated in multiple places, and the right aileron

remained attached to the wing. The aileron control tube fracture surfaces had a dull, grainy appearance consistent with overstress separation. The right flap remained attached to the wing. Right aileron control continuity was established from the fracture points to the aileron and the control stick.

Left Wing

The left wing separated from the airplane in the wing root area during the accident sequence. The left aileron remained attached to the wing and the left flap was separated. The left wing root rib and inboard portion of the wing were buckled and deformed upward consistent with wing separation in an upward direction. The left wing root rib measured 0.020-inch thick and there were no stiffeners or additional angles installed on it. There were no additional wing ribs installed in the wing walk area outboard of the root rib. The left aileron control tube was fractured at the wing separation point. Left aileron control continuity was established from the fracture point to the left aileron and control stick. The control tube fracture surfaces had a dull, grainy appearance consistent with overstress separation.

Main Wing Spar

The right main wing spar was mostly intact but damaged and deformed from WS 0 to the right wing tip. The left main wing spar was fractured in the left wing root area and damaged adjacent to the fracture point. The outboard portion of the left main wing spar was intact and installed in the left wing. The inboard portion of the left main wing spar remained attached to the right main wing spar at WS 0. All 4 wing fittings were intact and installed at the center portion of the main spar. Figure 4 shows an annotated photo of the center portion of the main spar as recovered, looking aft at the forward face of the spar. The bolts were numbered starting at 1 adjacent to WS 0 on both the left and right main spars.

The right upper spar cap at WS 0, from forward to aft, consisted of a 0.250-inch thick steel wing fitting, seven 0.125-inch thick aluminum bars, a 0.040-inch thick aluminum shim, a 0.125-inch thick aluminum web, a 0.040-inch thick aluminum aft fuselage bulkhead, and a 0.250-inch thick steel wing fitting. The 0.040-inch thick forward fuselage bulkhead only engaged holes 9 and 10 on the right upper spar cap. Bolts were installed in holes 1-10 and 12-15 with the heads of the bolts on the forward side of the spar cap. Washers were installed underneath the bolt heads in holes 3, 5-10, and 13-15. There was no bolt installed in hole 11. The bolts installed in holes 13-15 were painted green matching the spar cap, the remaining bolts were unpainted. Holes 1-4 had AN5-X(A)³ bolts installed based on the head markings. The aft side of the spar cap was inaccessible, so the grip length, drilling of the shank, or type of nut or washer could not be determined. Hole 5 had a NAS1304-27H bolt installed. The aft side of the spar cap was inaccessible, so the drilling of the shank and the type of nut or washer could not be determined. Holes 6-10 and hole 12 had AN3HX bolts installed with L, X, and SC marked on the head. Holes 13-15 had AN3-XA bolts installed with C, X, and S marked on the head. Self-locking nuts with no washers were installed on the bolts in holes 6-10 and 12-15. There was an L-angle stiffener installed between the upper and lower spar caps at hole 12. There was no evidence of adhesive between the spar cap layers.

The right lower spar cap at WS 0, from forward to aft, consisted of a 0.250-inch thick steel wing fitting, a 0.040-inch thick aluminum forward fuselage bulkhead, seven 0.125-inch thick aluminum

³ The X denotes the grip length of the bolt. The grip length of each bolt was not measured.

bars, a 0.040-inch thick aluminum shim, a 0.125-inch thick aluminum web, a 0.040-inch thick aluminum aft fuselage bulkhead, and a 0.250-inch thick steel wing fitting. Bolts were installed in holes 1-9 and 12-15. The heads of the bolts were on the forward side of the spar cap except for hole 4 which had the head on the aft side. Washers were installed underneath the bolt heads in holes 2, 3, 5-9, and 13-15. There were no bolts installed in holes 10 or 11 and there was no hole drilled in the forward fuselage bulkhead at the hole 10 location. The bolts installed in holes 13-15 were painted green matching the spar cap, the remaining bolts were unpainted. Holes 1-3 had AN5-X(A) bolts installed based on the head markings. The aft side of the spar cap was inaccessible, so the type of bolt in hole 4 and the grip length, drilling of the shank, and type of nut or washer in the other locations could not be determined. The bolt in hole 4 had a washer and self-locking nut installed. Hole 5 had a NAS1304-27H bolt installed. The aft side of the spar cap was inaccessible, so the drilling of the shank and type of nut or washer could not be determined. Holes 6-9 and hole 12 had AN3HX bolts installed with L, X, and SC marked on the head. Holes 13-15 had AN3-XA bolts installed with C, X, and S marked on the head. Self-locking nuts with no washers were installed on the bolts in holes 6-9 and 12-15. There was an L-angle stiffener installed between the upper and lower spar caps at hole 12. There was no evidence of adhesive between the spar cap layers. There was some evidence of corrosion between the right lower spar cap layers.

The left upper spar cap at WS 0, from forward to aft, consisted of a 0.250-inch thick steel wing fitting, seven 0.125-inch thick aluminum bars, a 0.125-inch thick aluminum web, a 0.040-inch thick aluminum aft fuselage bulkhead, and a 0.250-inch thick steel wing fitting. There was no 0.040-inch thick shim installed between spar cap layer 7 and the 0.125-inch thick aluminum web. The 0.040-inch thick forward fuselage bulkhead only engaged holes 9 and 10 on the left upper spar cap. Bolts were installed in holes 1-10 with the heads of the bolts on the forward side of the spar cap. Washers were installed underneath the bolt heads in holes 1-10. There were no bolts installed in holes 11-15 in the area where the left upper spar cap fractured and there was no damage to the holes. None of the missing bolts were recovered. Holes 1, 2, and 4 had AN5-X(A) bolts installed based on the head markings. The aft side of the spar cap was inaccessible, so the grip length, drilling of the shank, and type of nut or washer could not be determined. Hole 3 had a NAS1305-26H bolt installed and hole 5 had a NAS1304-27H bolt installed. The aft side of the spar cap was inaccessible, so the drilling of the shank and type of nut or washer could not be determined. Holes 6-10 had AN3HX bolts installed with L, X, and SC marked on the head. Self-locking nuts with no washers were installed on the bolts in holes 6-10. The L-angle stiffener normally installed between the upper and lower spar caps at hole 12 was separated and remained attached to the wing root rib. The upper hole in the stiffener was elongated. There was no evidence of adhesive between the spar cap layers. There was evidence of substantial corrosion between the left upper spar cap layers (Figure 5). The spar cap layer 1 was fractured about WS 17.25 through hole 15 and the outboard 3 inches was not recovered. Spar cap layers 2-7 were fractured about WS 16 through hole 14 and the fractures were matched with spar cap layers 2-7 on the left outboard wing. The upper spar cap layers were splayed open and exhibited S-bending in the area of fracture and the rivets through the spar cap were popped from about WS 17.5 to WS 22.5. The 0.125-inch thick aluminum web was intact to its production edge about WS 18. All the fractures exhibited a dull, grainy appearance consistent with overstress separation.

The left lower spar cap at WS 0, from forward to aft, consisted of a 0.250-inch thick steel wing fitting, a 0.040-inch thick aluminum forward fuselage bulkhead, seven 0.125-inch thick aluminum

bars, a 0.040-inch thick aluminum shim, a 0.125-inch thick aluminum web, a 0.040-inch thick aluminum aft fuselage bulkhead, and a 0.250-inch thick steel wing fitting. Bolts were installed in holes 1-9. The heads of the bolts were on the forward side of the spar cap except for hole 4 which had the head on the aft side. Washers were installed underneath the bolt heads in holes 1-3 and 5-9. There were no bolts installed in holes 10-15 in the area where the left lower spar cap fractured and there was no damage to the holes. None of the missing bolts were recovered. There was no hole drilled in the forward fuselage bulkhead at the hole 10 location. Holes 1-3 had AN5-X(A) bolts installed based on the head markings. The aft side of the spar cap was inaccessible, so the type of bolt in hole 4 and the grip length, drilling of the shank, and type of nut or washer in the other locations could not be determined. The bolt in hole 4 had a washer and self-locking nut installed. Hole 5 had a NAS1304-27H bolt installed. The aft side of the spar cap was inaccessible, so the drilling of the shank and type of nut or washer could not be determined. Holes 6-9 had AN3HX bolts installed with L, X, and SC marked on the head. Self-locking nuts with no washers were installed on the bolts in holes 6, 9, and 10. The L-angle stiffener normally installed between the upper and lower spar caps at hole 12 was separated and remained attached to the wing root rib. There was no evidence of adhesive between the spar cap layers. There was some evidence of corrosion between the left lower spar cap layers. The spar cap layer 1 was intact to the production edge at WS 20. Heads of three rivets at the outboard end were installed but the rivets were fractured. Spar cap layer 2 fractured about WS 10.5 through hole 9, layers 3-6 fractured about WS 13 through hole 11, and layer 7 fractured about WS 10.5 through hole 9. The fractures on layers 2-7 were matched with layers 2-7 on the left outboard wing. The left lower spar cap layers were splayed open and twisted and the rivets through the spar cap were popped from about WS 17.5 to WS 20. The 0.040-inch thick aluminum shim was intact to its production edge at WS 14 and the 0.125-inch thick aluminum web was intact to its production edge at WS 18. All the fractures exhibited a dull, grainy appearance consistent with overstress separation.

The bolts in holes 7, 8, and 9 were removed from the left lower spar cap and examined. Figure 6 is a photo of the bolts removed along with a new bolt. There were no nuts present on bolts 7 and 8 and the bolts were fractured through the cotter pin hole in the shank. The threads around the cotter pin hole were also stripped. The self-locking nut remained installed on bolt 9 with 1-2 threads visible. Removal of the nut showed the hole in the shank of the bolt was just visible beyond the material stack up and the nut was engaged in the area of the hole as installed. Examination of several other AN3HX bolts installed on the left and right spar caps showed a similar arrangement. The hole 9 bolt was deformed and measured 1.670 inches long which corresponds to a 15 length.

Aft Spar

The aft spar carry through was intact in the main wreckage and consisted of a 0.125-inch thick L-angle with an additional 0.125-inch thick bar riveted to the vertical flange. The right wing aft spar attach point consisted of a 0.125-inch thick lug with an additional 0.125-inch thick reinforcement plate added to the lug. The right side of the aft spar carry through remained attached to the right wing lug and the entire carry through was bent aft. An AN4-6A bolt with no washers and a low profile nut was in place in the right attach point. The bolt in the left side of the aft spar carry through was fractured with a portion of the shank remaining in the carry through. The head was fractured from the bolt and the nut pulled off the threads. The hole in the left side of the aft spar carry through was elongated in an outboard direction. The left wing aft spar attach point was fractured from the wing and not recovered.

Engine and Propeller

The airplane was equipped with a Lycoming O-235-N2C four-cylinder reciprocating engine, serial number L-23773-15. The engine exhibited impact damage. All four cylinders remained attached to the engine crankcase. The left magneto, starter, carburetor, oil cooler, and alternator separated from the engine and were found with the main wreckage. The oil filter was separated and not identified in the wreckage. The lower forward engine case had a hole about 4 inches in diameter. The engine case was cracked between the push rods on the No. 4 cylinder, from forward of the cylinder upward towards the case half. The oil pan displayed heavy impact damage.

The top spark plugs and rocker arm covers were removed. The rocker arms from each cylinder remained undamaged and exhibited no unusual wear signatures. The intake and exhaust exhibited crushing damage. The exhaust stacks were flattened and the exhaust tubes from the No. 2 and No. 4 cylinders remained attached to the cylinder. The intake tube from the No. 4 cylinder remained attached to the cylinder. The intake and exhaust pushrod tubes for the No. 2 and No. 4 cylinders exhibited crushing damage. The intake push rod and tube from the No. 1 cylinder was separated. The No. 1 exhaust pushrod and tube exhibited crushing damage. The engine crankshaft was not able to be manually rotated by hand due to the damage. The exposed portions of the crankshaft did not show any deformity.

The top and bottom spark plugs were removed. All the spark plugs were Champion REM-40E except 1-bottom and 2-top which were Auto Lite 40E. All eight spark plugs exhibited coloring consistent with normal operation. The right magneto had extensive impact damage and most of the body was separated. The right magneto accessory drive was rotated by hand. The left magneto was a Slick Magneto, Model No. 4371, SN 07050649. The distributor cap was removed and spark was observed between the contacts when the left magneto accessory drive was rotated by hand. The starter was heavily damaged. The gear teeth were retracted and did not appear to have any damage.

The oil cooler was separated from the engine and exhibited extensive crushing. The Marvel-Schebler Carburetor exhibited crushing around the air filter. The throttle body was separated and was not identified in the wreckage. The mixture control arm moved from stop to stop, but the control cable was separated. The fuel inlet screen was removed and no foreign debris was noted.

Portions of the wooden propeller remained sandwiched in the propeller hub, and the propeller hub remained attached to the engine. Both propeller blades were destroyed and separated.

4.0 Maintenance

The airplane, S/N 10508, was purchased by the accident pilot/owner in 1998 in an uncompleted state. A majority of the airplane was completed by up to three previous owners before it was acquired by the accident owner. Assembly of the airplane was completed in 2007 and the airplane was granted an airworthiness certificate in the Experimental category on May 2, 2007. Standard operating limitations for an experimental-amateur built airplane were issued in conjunction with the airworthiness certificate. The limitations included the requirement for an annual condition inspection and a prohibition from aerobatic flight.

The accident pilot/owner held an FAA mechanic certificate with airframe and powerplant ratings along with an inspection authorization (IA).

The records indicate that the owner moved from Hawaii to Idaho sometime between 2010 and August 2014. The airplane registration was updated in August 2014 to include the owner's Idaho address. It was reported that the airplane was disassembled and shipped in a container from Hawaii to Idaho. The airplane was reassembled in a hangar at Payette Municipal airport by the owner prior to the accident flight. The accident flight was the first flight after the airplane had been reassembled.

No airframe or engine logbooks were obtained for the accident airplane, so an accurate airplane or engine total time could not be established. The compliance with the conditional inspection requirements of the operating limitations could not be established.

5.0 Manufacturer Information

The original RV-3 wing spar described in Section 1.0 Airplane is referred to as a Type I spar and was supplied with kits between 1973 and 1983. The Type I spar was mathematically stress analyzed to design and ultimate load limits of 6.0 and 9.0 g's, respectively, at an aerobatic gross weight of 1,050 pounds (non-aerobatic gross weight is 1,100 pounds). As an option, the assembly instructions allowed for an epoxy adhesive to be used to bond the seven aluminum upper or lower spar cap bars together to form a single unit to facilitate drilling and riveting the spar caps to the spar web. It was later discovered that the adhesive provided some interbar shear and column strength, but the bonding could deteriorate, so it was not considered in the design as contributing to spar strength. In 1982, the company conducted a static load test on a Type I spar and found that it met the 9.0 g ultimate load criteria.

On March 16, 1981, the FAA issued GENOT (General Notice) TWA 1/40 SVCB, prohibiting aerobatics in the RV-3, due to some accidents involving wing failures. The action was permanent and could not be rescinded. Following this action, Van's Aircraft issued Change Notice 1 (CN-1) to RV-3 owners and builders in 1984. Briefly, CN-1 modified the wing by reinforcing the wing rear spar and carry through attach points, changing the rear spar attach point to a double shear connection, and strengthening the wing root ribs. The FAA and Canadian MOT (Ministry of Transport) reports on the RV-3 accidents suggested that these areas could have been the primary failure points. The airplane could be re-designated as an RV-3A at the completion of the modification in order to regain aerobatic operating authorization. Soon after the release of CN-1, the FAA issued another letter stating that RV-3 owners showing compliance with CN-1 could reapply for aerobatic operating limitations without re-designating the airplane. Thus, the RV-3A designation was adopted by some builders but does not necessarily mean there are any definite structural distinctions from an RV-3.

In 1984, the wing spar was redesigned and became referred to as the Type II spar. The Type II spar utilized five 0.1875-inch (3/16 inch) by 1.25-inches aluminum bars to make up the upper and lower spar caps. The spar caps were fastened together with bolts in the center section and rivets elsewhere similar to the Type I spar. Though the primary purpose was to simplify construction and

minimize the possibility of construction errors, a slight increase in calculated bending strength was achieved. Two additional ribs were added in the wing root area to increase the wing torsional stiffness, and the rear spar attach was strengthened.

In 1996, an RV-3 wing with a Type I spar was again static load tested with failure occurring below the 9.0 g ultimate load level. Examination revealed that the failure occurred due to compression buckling of the upper spar cap. A review of test data revealed that the 1982 test had been performed on a wing whose spar had been assembled with the optional epoxy bonding process. The 1996 test wing used a spar constructed without the epoxy adhesive. While the epoxy adhesive had not been calculated to add any spar strength, it appeared to have added compression buckling strength. As a result of further static load testing on both Type I and Type II wing spars, Van's Aircraft issued Change Notice 2 (CN-2-I for Type I spars and CN-2-II for Type II spars) specifying wing main spar modifications it deemed necessary for aerobatic strength. The change notices included a detailed history, explanation, and recommendations, and were sent to all known RV-3 and RV-3A owners and builders. Van's Aircraft maintained that both CN-1 and CN-2 (I or II) were necessary to achieve the aerobatic strength of the wing.

In November 1998, the FAA issued Special Airworthiness Information Bulletin (SAIB) ACE-99-10 to inform RV-3 owners of design, equipment, and operating limitations critical for safe aerobatic flight. The SAIB detailed the history of the issues and "highly recommended" adherence to the Vans specifications.

In 1999, the wing spar was redesigned again to include fully machined one-piece spar caps for the wings. Airplane kits with this configuration are designated as RV-3B.

Suggested airspeed limitations have changed slightly for the RV-3. The never exceed speed (V_{ne}) is 210 mph, the maneuvering speed (V_a) is 127 mph (down from 132 mph), and the stall speed in the clean configuration (V_s) is 54 mph. According to CN-1, because of the high ratio between V_{ne} and V_s , the RV-3 is more susceptible to pilot-induced overstress than most contemporary light airplanes.

6.0 Tests & Research

Van's Aircraft recommends in the assembly instructions and drawings that the bolts installed in the wing fittings should be NAS 1304 or NAS 1305 type bolts of the appropriate grip length. The remaining bolts in the upper and lower spar caps should be AN3-14A or AN3-15A bolts. All bolts should utilize self-locking nuts. The minimum tensile strength is 160 ksi for the NAS bolts and 125 ksi for the AN bolts per the specifications.

The AN3-14A bolts specified for installation in the spar caps are cad plated steel bolts with a diameter of 3/16-inch, thread pitch of 10-32, a grip length of 1.125 inches, and an overall length of 1.531 inches. The AN3-15A bolts are the same except the grip length is 1.250 inches and the overall length is 1.656 inches. These bolts do not have a hole in the head or shank.

Three of the AN3 bolts in the left lower spar cap were removed and examined in detail as documented earlier. The remaining AN3 bolts in the left and right upper and lower spar caps were

examined as installed and appeared the same as those removed. A box of new AN3H15 bolts was found in the pilot's hangar that had the same markings and appearance as those installed. The AN3H15 bolts were cad plated steel bolts with a diameter of 3/16-inch, thread pitch of 10-32, a grip length of 1.250 inches and an overall length of 1.656 inches. These bolts had a hole drilled in the head and in the shank. Drilled shank bolts such as these are designed to be used with castellated nuts. The castellated nut is tightened on the bolt so that the primary engaging threads are below the hole in the bolt shank and the slots align with the hole in the shank. A cotter pin should be inserted through the slots in the nut and the hole in the bolt shank to provide a locking mechanism for the nut.

The tension capabilities of a bolt are substantially different for a bolt with the thread engagement below the shank hole versus thread engagement at the shank hole due to the reduction in cross-sectional area at the hole location. For 10-32 threads, the minor thread diameter is 0.1519-inch which results in a cross-sectional area of $18.122 \times 10^{-3} \text{ in}^2$. The AN3 bolt specifications call for the hole in the shank to have a diameter of 0.070-inch which results in a minimum cross-sectional area of $7.878 \times 10^{-3} \text{ in}^2$ at the center of the shank hole. The failure tension load in the threaded area of the bolt can be calculated for the case where thread engagement occurs below the shank hole and for the case where thread engagement occurs through the center of the shank hole.

Stress equation:

$$\sigma = \frac{P}{A}$$

Calculation of failure tension load in full thread area:

$$125 \text{ ksi} = \frac{P}{18.122 \times 10^{-3} \text{ in}^2}$$

$$P = 2,265 \text{ lb}$$

Calculation of failure tension load in thread area with hole:

$$125 \text{ ksi} = \frac{P}{7.878 \times 10^{-3} \text{ in}^2}$$

$$P = 984 \text{ lb}$$