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CASE HISTORY—PEER-REVIEWED

Multiple-site Fatigue Cracking of Fuselage Structure Due to Improper Rivet Installation

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Abstract A large passenger airplane experienced an inflight rapid decompression and was forced to make an emergency landing. Post-accident inspection revealed that a section of the fuselage skin on the top portion of the airplane (crown) had fractured and flapped open during the flight. The National Transportation Safety Board (NTSB) Materials Laboratory examination of the fuselage structure determined that multiple-site damage (fatigue cracking) originated at the skin panel of a lap joint from areas of improperly drilled rivet holes. Fatigue crack growth analysis on the fracture face of the skin panel via quantitative fractography was performed by scanning electron microscopy revealed that cracking started approximately when the airplane entered service.

Keywords Fuselage lap joint · Aluminum alloy · Fasteners · Installation/assembly error · Quantitative fractography · Fatigue cracking · Multiple-site damage

Introduction

An airplane carrying 117 passengers experienced a rapid decompression after departing the airport [1]. The airplane made an emergency landing. Two passengers sustained minor injuries. Post-accident inspection of the airplane upon landing revealed that a section of fuselage skin about 60 in. long $\times$ 8 in. wide (152 cm $\times$ 20 cm) had fractured and flapped open on the crown portion of the fuselage. Figures 1 and 2 show the fractured skin panel on the top portion of the fuselage.

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Lap Joint Design

The exterior shell portion of the fuselage is made from many overlapping aluminum skin panels and where they overlap is called a lap joint. These lap joints are connected to stringers that run the length of the fuselage. Fracture of the fuselage skin occurred at the location of stringer number S-4L. Stringer number S-4L refers to the fourth stringer from the top side of the fuselage on the left side of the airplane. Figure 3 shows the cross section of the lap joint in stringer number S-4L. Adhesive sealant was applied at the factory between the three layers of skin panels. The adhesive sealant serves as a weatherproofing barrier and during flight keeps pressurized air inside of the passenger cabin. The lap joint was fastened by three rows of rivets (upper, middle, and lower). The fracture was through the lower skin of the lap joint and intersected the lower rivet row, as shown in Fig. 3.

Overall Approach and Equipment

The airplane was flown to an overhaul facility where the rupture area was removed from the fuselage and shipped to the NTSB Materials Laboratory for examination. Both halves of the fracture were examined with a Nikon SMZ1500 stereo microscope, which clearly revealed evidence thumbnail features typical of fatigue cracking that emanated from multiple adjoining holes in the lower rivet row. The length and depth of the fatigue cracks were

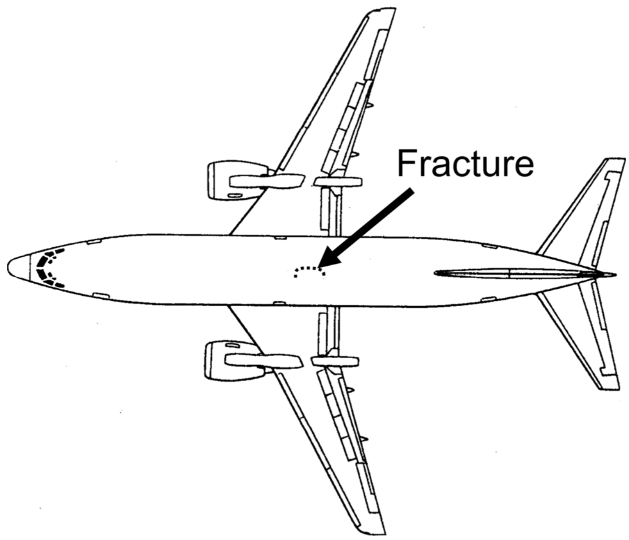

Fig. 1 Location of the fractured skin panel

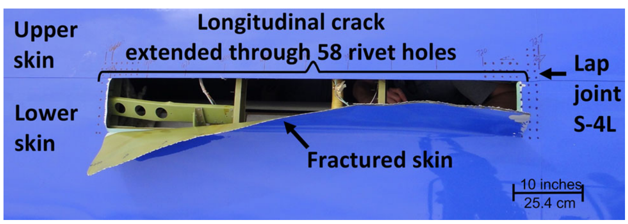

Fig. 2 Fractured skin panel as viewed from outside of the airplane

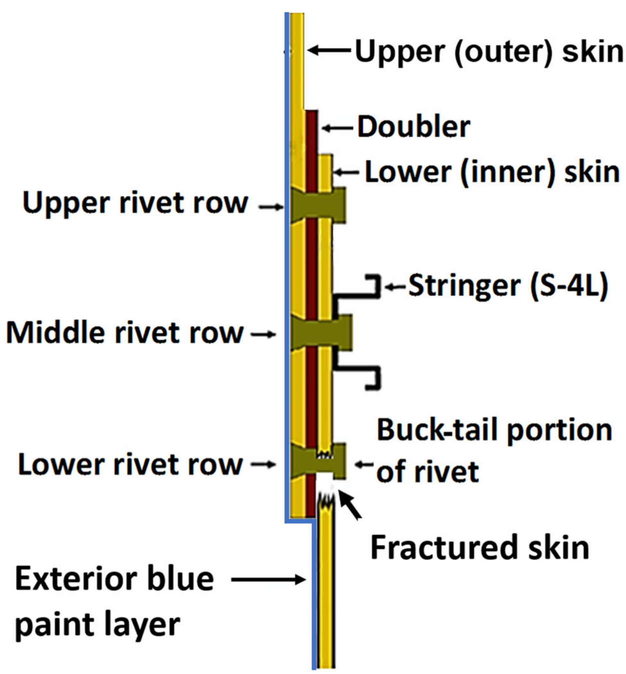

Fig. 3 Cross section diagram of the lap joint and fracture area (not to scale)

measured directly on the fracture face using a Keyence VHX-5000 digital microscope. The larger fatigue cracks were cut out of the fuselage, and they were examined utilizing a LEO 1455VP scanning electron microscope (SEM).

The head portion of each rivet was carefully drilled out from the lap joint and the shank portions with the buck-tail end were pressed out of the holes with a punch tool. The three skin layers were peeled apart (due to presence of adhesive sealants holding them together) so that the fracture face in each individual skin layer could be studied separately. The quality and size of the drilled rivet holes and their corresponding rivets were recorded with the digital microscope and scrutinized to determine whether the manufacturing process (drilling the holes or driving the rivets) was causal in the failure of the lap joint.

Metallurgical sections were made through several fuselage skin panels and rivets and the prepared sections were examined with a Reichert Leica MEF3A inverted microscope. The sections were etched with Keller's reagent. Microhardness testing of polished sections were assessed with a Matsuzawa model MHT2 microhardness tester using a Vickers indenter, 300 gram load for 10 seconds. The chemical composition of the rivets and core portion of the skin panels were determined by atomic emission spectroscopy analysis; and conductivity of the skin panels were measured with a Zetec MIZ-6 conductivity meter.

The submitted lap joints were inspected by eddy current method using a GE Inspection Technologies Hocking (formerly Krautkramer) Phase 2D instrument, prior to stripping paint from any area of a rivet, and before disassembly of the individual skin panels.

Laboratory Examination and Testing

Fracture of the Lower Skin

Stereo microscope examination of the lower skin fracture face (viewing the fracture face head on as shown in Fig. 4) revealed thumbnail fracture features typical of fatigue cracking emanating from 54 rivet holes. Fatigue cracking started on the forward and aft sides of each rivet hole. Figure 5 shows an example of one of the smaller fatigue cracks that did not intersect fatigue cracks from adjacent holes. Specifically, fatigue cracking started at the area where the rivet hole intersected the outer surface of the lower skin. The fatigue cracks in Fig. 5 emanated from diametrically opposite sides of the rivet hole, propagated forward and aft, and terminated in the areas indicated by a dashed line.

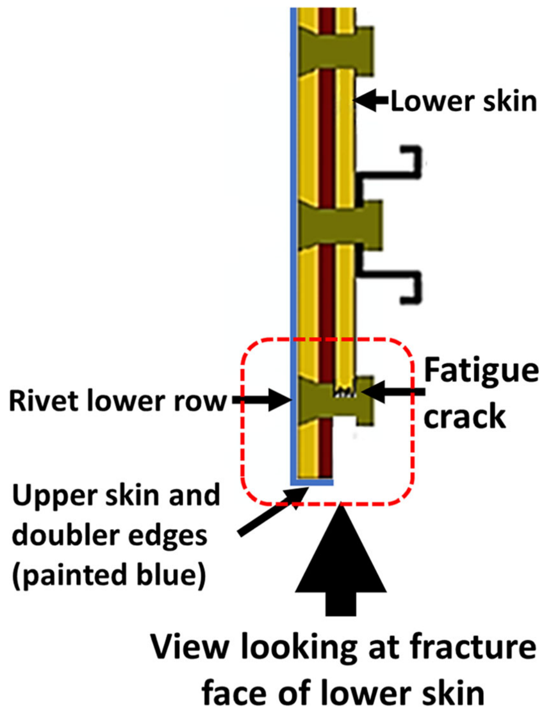

Fig. 4 Viewing the lower fracture face (line of sight)

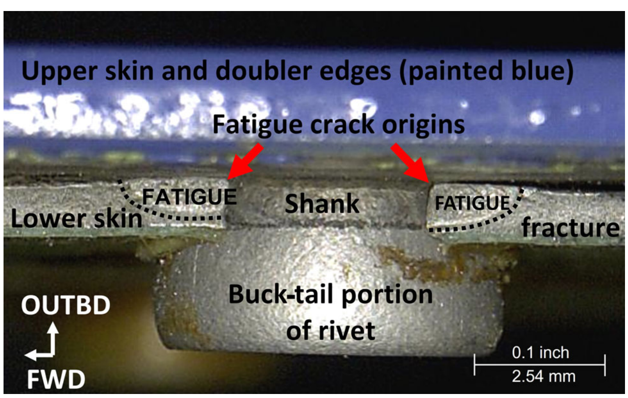

Fig. 5 Lower skin fracture face showing fatigue cracks (outlined by dashed lines) that emanated from the rivet hole

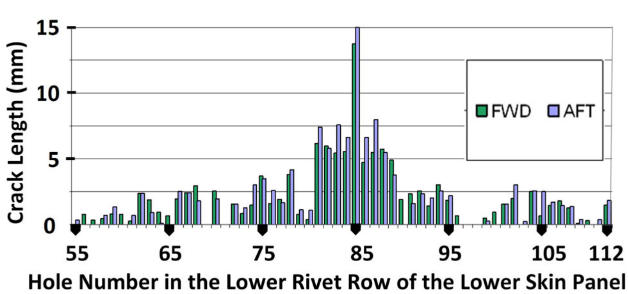

Fig. 6 Graphical representation of the rivet hole locations and length of respective fatigue cracks

Figure 6 shows a graphical presentation of the length of each fatigue crack. The rivet hole numbers were arbitrarily labeled on-site. The distance between the centers of each rivet hole was about 1-inch (2.5 cm). Typically, the longest fatigue crack represents the oldest fatigue crack and is the origin of fatigue cracking. The longest fatigue crack was located on the aft side of rivet hole "85" and it measured about 0.60 in. (15 mm). This observation gave us the first clue that fatigue cracking started on the aft side of rivet hole "85". As seen in Fig. 6, the length of a fatigue crack decreased as one moved away from the longest fatigue crack in rivet hole "85". The longer fatigue cracks penetrated through the wall of the lower skin, and they intersected the fatigue cracks at the adjoining rivet holes. In the aircraft industry, multiple fatigue cracks that emanate from a row of fastener holes is referred to as multiple site damage (MSD). The areas located outside of the fatigue regions exhibited micro-void coalescence features typical of overstress separation.

Quantitative Fractography

Fatigue cycles on the fuselage of an airplane are almost exclusively attributable to pressurization loads that normally occur once per flight. The internal cabin is pressurized and adjusted based on flight altitude and depressurized on landing. A flight cycle refers to a takeoff and landing event. Commercial airplane manufacturers and aviation forensic investigators utilize fatigue crack growth analysis by quantitative fractography to roughly determine the approximate number of pressure cycles that causes a crack in a fuselage to propagate to a specific size or to failure [2–5]. Typically, fatigue cracking is a slow process and can be monitored during the life cycle of the airplane. By estimating the number of cycles that led to a specific crack size or failure, the aircraft industry can set up inspection programs to detect a crack on other same model airplanes at corresponding areas before the damaged component can lead to injury or loss of life.

Fatigue striation counting was performed with the SEM on the longest fatigue crack (fatigue crack on the aft side of rivet hole "85") . Mating fractures involved in fuselage rupture often sustain mechanical damage because of the mating fractures contacting each other. The length of the longest fatigue crack was preserved and showed no evidence of mechanical damage. This permitted uninterrupted striation count of the entire length of the fatigue crack. The fatigue crack face was covered with heavy black deposits because of exposure to the environment and was cleaned with a chromic acid solution, a solution commonly used for

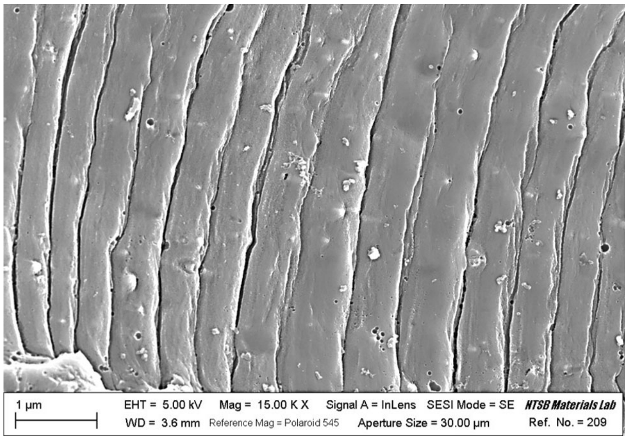

Fig. 7 Scanning electron micrograph of fatigue striations found on the aft side of rivet hole "85"

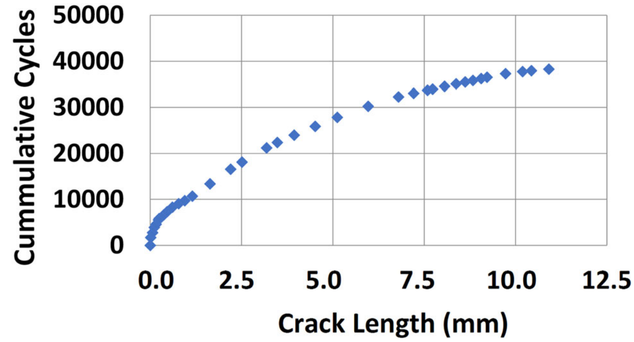

Fig. 8 Plot of cumulative cycles versus crack length for the fatigue crack that emanated from the aft side of rivet hole "85"

cleaning aluminum alloys [6]. This cleaning procedure exposed the fine fatigue crack features, see example in Fig. 7. Striation counting was performed at various intervals along length of the fatigue crack, in regions that contained continuous fatigue striations (having stable crack growth rate). A plot of accumulated flight cycles versus the crack length is shown in Fig. 8. The plot in Fig. 8 shows that the crack length increased with the number of cycles. The average crack growth rate at a specific distance from the origin was determined by adding the crack growth rate at a specific distance (point) to the crack growth rate at the previous point and then dividing those figures by two. The number of cycles was calculated at specific intervals of the fatigue crack path. The cycles for each interval were added to obtain the accumulated flight cycles. The total striations count on the fracture surface added up to 38,261 cycles [7]. At the time of the accident, the airplane logbook showed the airplane flew a total of 39,786 flights (cycles). The accumulated cycles obtained by the SEM quantitative fractography method was nearly as great as the accumulated service cycles on the airplane. When comparing these two pieces of information, the investigation concluded that the fatigue crack started approximately when the airplane entered service.

Fatigue crack growth follows three stages of growth. Stage I corresponds to crack nucleation; stage II corresponds to stable crack growth (follows Paris law and is used for fatigue life prediction methods in metallic material); and stage III corresponds to unstable crack growth that occurs at the end of the fatigue crack life. The number of cycles deduced from striation counting of the longest (and oldest) crack corresponded to the total crack propagation cycles, based on Paris-law regime, which coincided almost to the total flight cycles. The arbitrary, but commonly accepted, dividing line between high-cycle fatigue (HCF) and low-cycle fatigue (LCF) is about 10^4-10^5 cycles [8]. Another source indicated that failures in very low cycle fatigue (VLCF) are in the order of 10, say below 20 cycles; LCF applies to failures between 10^2 and 10^4 cycles; and HCF applies to failures greater than 10^5 cycles [9]. The accumulated cycles count obtained from SEM quantitative fractography work was consistent with LCF (closer to the LCF regime rather than the HCF regime). As indicated earlier in this paper, the origin of the fatigue cracks emanated from irregular drilled holes (origin was not free of damage). The irregular holes acted as stress raisers. The fast transition from crack nucleation to stable crack growth, as shown in the early part of the plotted curve in Fig. 8, most likely was due to stress raisers (irregular drilled rivet holes).

Examination of the Rivet Holes and Disassembled Rivets

Visual inspection of the holes (after drilling out the rivets) was performed with the three skin layers held together by adhesive sealant. As reference, Fig. 9a shows the outer face of a properly drilled hole. The hole is round—an ideal condition for driving a rivet. Figure 9b shows the side profile of the corresponding rivet that was pressed out of the round hole. The diameter along the entire length of the shank was uniform, testimony that the rivet was removed from a round hole.

In contrast, many of the rivet holes associated with fatigue cracking deviated from a round shape (they were referred to as irregular holes). Irregular holes result from improper drilling. Figure 10a shows the outer face of an irregular hole prior to peeling apart the skin panels. To further assess the condition of the irregular holes, the skin panels were peeled apart from each other (recall skin panels were held together by adhesive sealants), and each skin panel was examined separately as a single piece. Examination of the disassembled skin panels revealed many of the holes that contained fatigue cracks appeared like the outline of a figure eight, often referred as a double-

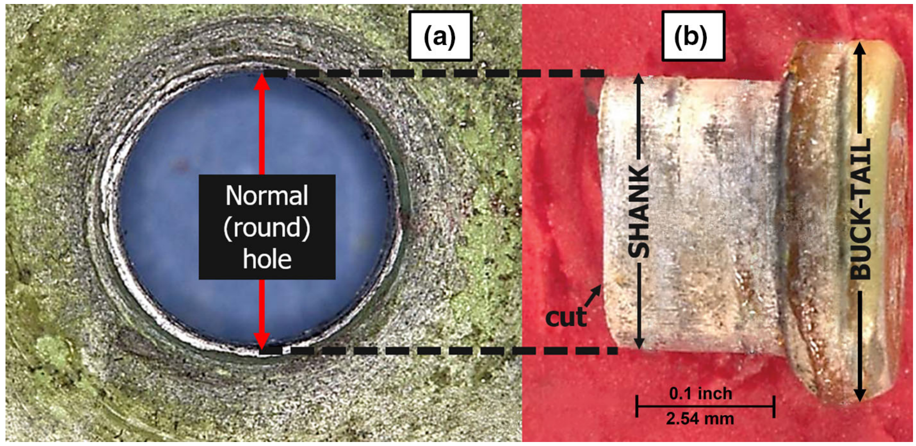

Fig. 9 (**a**) Exposed round hole prior to disassembly of panels and (**b**) side profile of removed corresponding rivet

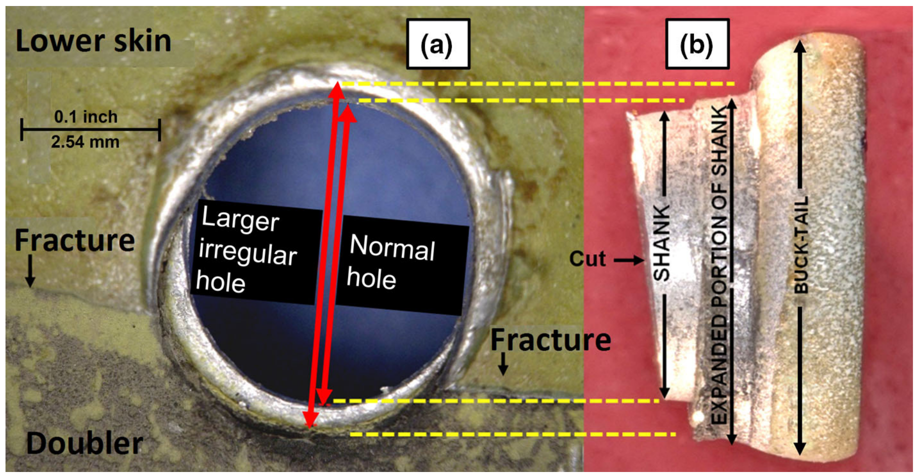

Fig. 10 (**a**) Exposed irregular hole prior to disassembly of panels and (**b**) side profile of removed corresponding rivet

drilled hole. Double drilled holes indicated that more than one drilling operation was performed at the same hole location. Double drilled holes are not permissible in the airplane manufacturer specification.

Rivets removed from irregular holes showed evidence that the diameter of the shank in the area adjacent to the bucked tail portion for most rivets was larger (expanded) compared to the diameter of the remaining portion of the shank, see Fig. 10b. The expanded portion of the shanks confirmed that the irregular holes in the lower skin were slightly larger compared to those in the doubler and upper skin.

Two worse-case examples of improperly driven rivets were found such that the buck-tail did not entirely cover the irregular hole in the lower skin, and they are shown in Figs. 11 and 12.

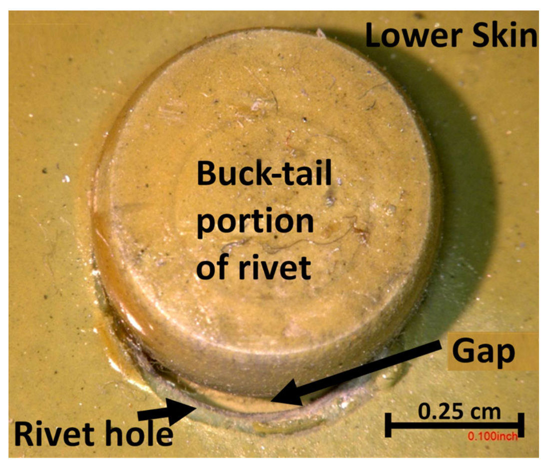

Fig. 11 Improperly installed rivet showing partially exposed hole (gap)

Material Properties of the Skin and Doubler

The skin panels and doubler were specified as 2024-T3 clad aluminum sheet. The material composition, hardness, electrical conductivity, and thickness of the upper skin, lower skin, and doubler were examined to determine whether they complied with manufacturer specifications.

In heat treatable aluminum alloys, the specified hardness range for many temper conditions overlap or are in proximity of each other, except for the hardness for as-quench condition. Conductivity and hardness can confirm a temper condition. So, conductivity measurements are crucial to verify the correct temper condition and is a good quality control method to determine deviation from standard manufacturing practice. In clad sheets, the major strength of the alloy is at the core. The clad material is a layer for corrosion resistance and must be removed prior to making conductivity measurements of the alloy in question. Conductivity measurements of clad material is permitted when correlation has been established between the conductivity readings and the acceptable range of heat-treat response with consideration of hardness measurements over the allowable range of cladding thickness. Conductivity values that fall outside of specified limits may indicate significant change to the mechanical strength values of the materials used in the lap joint. A loss of strength in the materials translate to a reduced life and early failure of the lap joint.

Table 1 shows the measured thickness, hardness, and electrical conductivity of the upper and lower skin panels, and doubler. Table 2 shows the chemical composition of samples removed from the upper and lower skin, and doubler. The material properties and thickness of the skin panels and doubler were consistent with those specified in the airplane manufacturer specifications. The etched sections showed grain structure that was typical for 2024 aluminum alloy in the T3 condition and no evidence of defects. No material anomalies were found in the samples that were measured, analyzed, or tested. The material of the skin panels and doubler were ruled out as a factor that led to the fuselage rupture.

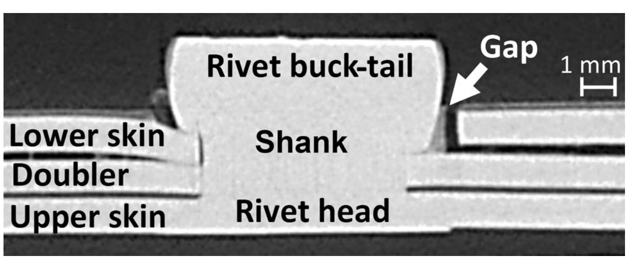

Fig. 12 X-ray computed tomography reconstructed image showing the cross section of an improperly installed rivet

Material Properties of the Rivets

The rivets were specified as 2017-T4 aluminum alloy, and the head portion of each rivet was specified as flush shear type (100-degree taper). Chemical analysis was performed on many rivet samples. Chemical analysis results from two typical rivet samples are shown in Table 3. Microhardness testing of the rivets produced an average hardness of 144 HV (77 HRB) each, consistent with driven 2017 aluminum rivets in the T4 condition. The size (diameter of the shank and head) of the disassembled rivets were measured and found to be within the airplane manufacturer specification [10]. Prior to drilling the rivet heads, none of the rivets showed evidence of fracture or cracking. No anomalies were found in the rivets that were tested. The material and size of the rivets were ruled out as a factor that led to the fuselage skin rupture.

Table 2 Composition of the core portion of the Alclad 2024 upper and lower skin panels (wt%)

Element	Min. Specified	Max. Specified	Upper skin Measured	Lower Skin Measured	Doubler Measured
Copper	3.8	4.9	4.54	4.25	4.32
Magnesium	1.2	1.8	1.53	1.44	1.45
Manganese	0.3	0.9	0.65	0.63	0.63
Iron	…	0.5	0.16	0.16	0.21
Silicon	…	0.5	0.1	0.1	0.15
Chromium	…	0.1	0	0	0
Zinc	…	0.25	0.06	0.06	0.05
Titanium	…	0.15	0.03	0.03	0.25
Others, each	0.05		0.002	0.002	0.002
Others, total	0.15		0.032	0.032	0.002
Aluminum	Balance		92.928	93.328	93.268

Table 1 Testing of Alclad 2024-T3 upper and lower skin panels

Skin panel	Thickness in. (mm)		Hardness (Rockwell HR15-T)		Electrical conductivity (% IACS)	
	Specified	Measured	Specified	Measured	Specified	Measured
Upper	0.037 − 0.042 (0.939 − 1.067)	0.039 (0.991)	81 − 88	84 − 85	28.5 − 35.0	33.3 − 33.7
Lower	Same as upper panel	Same	Same	Same	Same	Same
Doubler	0.032 − 0.040 (0.812 − 1.016)	0.037 (0.940)	Same	Same	Same	Same

IACS – International annealed copper standard

Table 3 Composition of several rivet samples (wt%)

Element	Rivet 2017-T4 Specified Min	Max	Sample rivet 60 Measured results	Sample rivet 109 Measured results
Silicon	0.2	0.8	0.37	0.33
Copper	3.5	4.5	3.88	3.80
Manganese	0.4	1.0	0.42	0.52
Magnesium	0.4	0.8	0.57	0.56
Iron	…	0.7	0.42	0.18
Chromium	…	0.1	< 0.01	0.01
Zinc	…	0.25	0.01	0.01
Titanium	…	0.15	0.01	0.01
Others, each	…	0.05	< 0.05	< 0.05
Others, total	…	0.15	< 0.15	< 0.15
Aluminum	Balance		Remaining	Remaining

QQ-A-430, Federal specification: aluminum alloy rod and wire; for rivets and cold heading, rev 2017

Eddy Current Inspection

Fatigue cracks on the lower (inner) skin were not visible from the exterior or interior of the airplane. When viewed from outside of the airplane, the fatigue cracks were covered by the upper (outer) skin panel and doubler. From inside the airplane, fatigue cracks that eventually penetrated the thickness of the lower skin were covered by insulation and wall panels. Inspection by a nondestructive method was needed to detect hidden fatigue cracks before they link together.

The lap joint assembly that was removed from the airplane was inspected at the NTSB Materials Laboratory by eddy current method, prior to any disassembly of the skin panels. The inspection was performed by the airplane manufacturer's nondestructive testing personnel using specific eddy current inspection methods developed by the airplane manufacturer to detect gaps and cracks in lap joints. The inspection method detected several cracks in areas slightly forward and aft of the longitudinal fracture.

The lap joints on the same model airplanes were inspected by specific eddy current inspection methods developed by the airplane manufacturer to detect gaps and cracks in lap joints in the crown portion of the airplane. There were no similar findings of multiple site damage in the lap joints of the same model airplanes as was found on the accident airplane; thus, the NTSB concluded that it was unlikely that there was a systemic quality assurance error at the airplane manufacturer facilities at the time of manufacture.

Manufacturing and Maintenance History

The internal portions of the skin panels in the fracture had the airplane manufacturer quality assurance stamp markings dated January through February 1996. This indicated the original skin panels had not been replaced after the airplane went into service. According to airline records, the airline had not repaired or replaced the crown skin or upper skin panel in the fracture area.

Fatigue Life of a Rivet Joint

Methods for manufacturing riveted joints have a great influence on the longevity of a mechanical joint, especially in structures that are subjected to cyclic stress. For example, reaming reduces the scatter of the hole diameters and increases hole surface smoothness. The sizing process reduces the roughness of the hole and strengthens it by introduction of compressive stress to internal layers of the material. The compressive stress hinders initiation of fatigue cracks on the hole surface [11]. When solid rivets are driven into a hole, the shank portion expands and fills the hole, creating a beneficial interference fit. By increasing the force applied to a driven rivet, interference increases, and fatigue life of the rivet joint is improved [12]. Complete filling of the rivet hole and even distribution of the contact force on mating surfaces is important in improving the fatigue resistance of a rivet joint [13]. Because many of the rivet holes were improperly drilled and the installed rivets did not completely fill the holes in the lower skin panel, the NTSB concluded that the fatigue life of the panel was significantly reduced.

Corrective Action/Lessons Learned

Manufacturing evolves around controlled and repeated processes. The accident serves as a reminder that a manufacturing error (deviation from a set standard), if not addressed appropriately, can lead to consequences many years after fabrication. Manufacturing errors can be eliminated/curtailed by training techniques and detected by quality control methods. To minimize the risk of future joint failures, the airplane manufacturer informed personnel involved with fabrication of riveted joints of this accident and provided them with updated training. The training emphasized awareness of acceptance and rejection criteria for fabricating drilled holes and riveted joints, extensively using examples of errors that were found on the lap joint from the accident airplane. As part of technology advancement, specialized eddy current inspection methods for detecting cracks in riveted joints were incorporated into scheduled maintenance and overhaul programs for the airplane model involved in the accident.

Conclusions

a. Fracture of the airplane fuselage was caused by fatigue cracking that initiated from multiple adjoining rivet holes in the lower (inner) skin portion of the lap joint at stringer S-4L.
b. Fatigue cracks in adjoining rivet holes eventually linked together forming one continuous crack, causing the lower skin to flap open that resulted in inflight rapid decompression of the airplane.
c. The fatigue cracks extended from improperly drilled rivet holes.
d. The hole quality in the skin panel at the area of fracture was not in accordance with the airplane manufacturer specifications or standard manufacturing practices.
e. Many of the rivet holes were improperly drilled and the installed rivets did not completely fill the holes in the lower skin panel, which significantly reduced the fatigue life of the panel.
f. Because the inner face of the skin panels in the area of fracture showed no evidence of quality assurance stamp marks that post-dated the manufacture dates of the airplane, and the airline showed no record of a repair in the fracture area, the evidence indicated that the crown skin panel in the area of fracture was replaced during manufacture.
g. Fatigue striation counting showed that the longest fatigue crack accumulated a total of 38,261 cycles, nearly as great as the accumulated service cycles on the airplane (39,781 cycles), which was supporting evidence that the fatigue cracking began approximately when the airplane entered service.

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