NATIONAL TRANSPORTATION SAFETY BOARD

Office of Aviation Safety
Washington, D.C. 20594

June 15, 2011

AIRWORTHINESS

Group Chairman's Factual Report

DCA11FA004

A. ACCIDENT: DCA11FA004

Operator:	American Airlines Flight 1640
Location:	Miami, Florida
Date:	October 26, 2010
Time:	2130 Eastern Daylight Time[1]
Aircraft:	Boeing 757-223, Registration Number: N626AA

B. AIRWORTHINESS GROUP

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C. SUMMARY

On October 26, 2010, about 9:30 pm eastern daylight time (EDT), American Airlines flight 1640, a B757-223, N626AA, experienced a rapid decompression while climbing through 32,000 feet. The flight crew executed an emergency descent and returned to Miami. The airplane landed

[1] All times are eastern daylight time (EDT) based on a 24-hour clock, unless otherwise noted. Actual time of accident is approximate.

without further incident, and no injuries were reported to the 6 crew and 154 passengers. A ground inspection of the airplane revealed a section of the fuselage crown skin, approximately 1-foot by 1.5-feet, had ruptured just above and aft of the forward, left passenger (L1) door. The airplane had accumulated about 63,010 hours and 22,450 cycles. The flight was operating under the provisions of 14 *Code of Federal Regulations* Part 121 on an instrument flight rules (IFR) flight plan and had departed from Miami International Airport (MIA), Miami, Florida, to fly to Boston Logan International Airport (BOS), Boston, Massachusetts.

D. DETAILS OF THE INVESTIGATION

1.0 Accident Aircraft Examination (N626AA)

Manufacturing Serial Number (MSN): 24584
Line Number (L/N): 304
Total Cycles (TC): 22,450 cycles
Total Time (TT): 63,010 hours
Manufacturing Date: August 14, 1990

The damaged section of fuselage crown skin was bounded by body station (BS[2]) 395 and BS 418 in the fore-aft direction and stringer (S[3]) 3L and S-4L in the circumferential direction. See Figure 1[4]. S-4 (L and R) are locations of longitudinal lap joints where the upper fuselage (crown) skin overlaps and is attached to the upper side fuselage skins with 3 rivet rows. The hole measured approximately 18 inches in the longitudinal direction by 7 inches in the circumferential direction. The aft 5 inches by 7 inches section of the flap separated from the aircraft and was not recovered. The forward 13 inches by 7 inches section of skin remained attached along the upper edge and was deformed upward. There were some abrasion marks on the fuselage skin above the hole that matched the shape of the lower, forward corner of the attached section. The fracture along the upper, aft, and lower edges appeared to match the edge of the chem-mill[5] pocket in the skin. There was no visible damage to the surrounding frames and stringers. There was some insulation missing from the area below the hole.

There was an exterior visual crack indication in the next stringer bay forward of the hole bounded by BS 377.7 and BS 395 in the fore-aft direction and S-3L and S-4L in the circumferential direction (Figure 2). The external crack indication was about 1-3/16 inches long and was centered at BS 386 about 3 inches above the lower edge of the skin panel, just above the upper lap joint rivet row. The red arrows in Figure 2 indicate the approximate ends of the visual crack indication. There was a dark gray stain from the crack indication aft to about the location of the hole. The crack was not visible from the interior of the aircraft.

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[2] Body Station numbers represent the number of inches measured along the length of the airplane from a set datum point at the forward end of the airplane.
[3] Stringers are numbered from stringer 1 at the top center of the fuselage sequentially down the left (L) and right (R) sides of the airplane as viewed from the tail of the airplane looking forward.
[4] All Figures are presented in Appendix A to this report.
[5] Chem-mill refers to areas of the skin that have been chemically milled. This is a process where select material is removed through the chemical action of strong acids or bases.

All of the longitudinal chem-mill edge details on the entire crown skin panel were checked for cracks using a draft procedure that Boeing developed. The eddy current sliding probe for chem-mill edge crack detection procedure included both high and low frequency eddy current inspections. Figure 3 shows an example area inspected by the procedure. The visible crack measured 1-1/2 inches long using high frequency eddy current and 1-3/4 inches long using low frequency eddy current. An additional 1 inch long subsurface crack was found using the sliding probe eddy current procedure centered about 2-3/4 inches forward of the center of the visible crack and in line with it. No other cracks were found.

A section of the skin panel from the lower left edge at the lap joint to 2 rivets above S-3L and from 2 rivets forward of BS 377.7 to the end of the panel at BS 439 was removed from the airplane and sent to the NTSB Materials Lab for further examination (Figure 4).

Materials Lab examination[6] of the hole area in the skin panel revealed multiple site fatigue crack initiation points along the longitudinal chem-mill step just above the S-4L lap joint. There was evidence of fatigue cracking propagating through the skin thickness along about 15 inches of the 18 inches long pocket. About 2.5 inches of the fatigue cracking was completely through the skin thickness. Where able to be measured, the skin thickness at the fatigue initiation sites ranged from 34.15 mils[7] to 34.97 mils. The lab also documented a visible depressed channel at the bottom of the chem-mill radius on the S-4L side of the pockets and a visible raised ridge at the bottom of the chem-mill radius on the S-3L side of the pockets. The measured skin thickness at the bottom of the channel in the pockets that were not fractured was consistently below 37 mils with a minimum value of 34.86 mils located at BS 387.5. The skin thickness at the peak of the ridge was also measured at several locations and was consistently above 43 mils. Materials characterization testing confirmed that the skin panel was manufactured from 2024-T3 aluminum alloy.

In addition, the NTSB Materials Lab performed a study of the fatigue region striation data[8]. The fatigue striation spacing was measured at a total of 12 locations in the hole area with 7 of the locations in the through crack area. Assuming a skin thickness and total crack length of 0.035 inch, the fatigue cycles were determined. The cycles to reach a crack length of 0.013 inch were also determined since Boeing reports that this represents the minimum detectable crack length. The interval for the crack to grow from 0.013 inch to 0.035 inch was then calculated. The average total cycles for the crack to grow through the skin thickness was 3,709 cycles and the average interval to grow from minimum detectable to through the skin thickness was 917 cycles.

2.0 Skin Panel Manufacturing

The P/N 141N3312-1, -4, or -7 skin panel spans from BS 297 to BS 439 and from S-4L to S-4R. The three different skin panels are the same in the pocket areas above the S-4L and S-4R lap joints. The -4 and -7 panels removed three chem-mill pockets relative to the -1 panel and added a

[6] See NTSB Materials Laboratory Factual Report No. 11-001 in the public docket for this accident for the details of the examination.
[7] 1 mil is equal to 0.001 inch.
[8] See NTSB Material Laboratory Study Report No. 11-001S in the public docket for this accident for the details of the study.

hole for an external antenna. The -4 skin panel has a painted finish and the -7 skin panel has a polished aluminum finish. The panels were manufactured from 0.063 inch thick Alclad 2024-T3 aluminum alloy sheet. The panels were stretch formed and then the interior surface was chemically milled into a waffle pattern as shown in Figures 5 and 6. The manufacturing drawing calls for chem-mill pockets with a thickness of 0.040 inch +/- 0.003 inch. The longitudinal full thickness areas are stringer pads and the full thickness circumferential areas are pads for shear tie attachment. This panel is unique on the airplane in that it is the only panel that has 0.040 inch thick pockets with single chem-mill steps from the stock gage to the minimum pocket gage. Chemical milling of panels is outlined in Boeing Chem-mill Process Specification BAC 5772. BAC 5772, section 11, allows up to a maximum of a 0.030-inch material removal in a single chem-mill process step.

The accident skin panel was likely manufactured some time during late 1989 or early 1990 based on the airplane build records. At the time of manufacture, the skin panels were stretch formed for contour before being masked, hand scribed, peeled, and placed on a rack. The rack was then dipped vertically in the chemical bath several times with measurements of select pocket thicknesses taken each time it was removed. Once the required amount of material was removed, the panel was rinsed and inspected. During the final inspection all pocket thicknesses were checked. The typical chem-mill rate achieved was about 1 mil per minute. There were no manufacturing records available for the accident skin panel and there was no requirement to keep them.

BAC 5772 has undergone 4 revisions between 1988 and 2010 but the recommended mill tank chemistry for Type II chemical milling has not changed during this time. The recommended chemistry was:
Sodium Hydroxide (NaOH) – 16-26 oz/gal
Dissolved Aluminum (Al) – 2.5-10.0 oz/gal
Sodium Sulfide (Na_2S) – 1.5-3.5 oz/gal
Triethanolamine (TEA) – 4.0-8.0 oz/gal

Temperature – 210°F to 220°F
Etch Rate – 0.0012-0.002 in/side/min (1.2-2 mils/side/min)

According to Boeing, the actual control ranges shall be determined at each location and will depend on experience, alloys being milled, sludge control, tank size, load size, and other variables to ensure that the requirements of BAC 5772, Section 11 are met. Boeing was able to find the tank chemistry data from January 2, 1990, to May 22, 1991, that included multiple samples per week. This data was examined and showed the following:

	NaOH (oz/gal)	Al (oz/gal)	Na_2S (oz.gal)	TEA (oz/gal)
Average	16.5	9.4	2.6	7.8
Minimum	10.7	4.9	0.7	6.0
Maximum	25.7	13.2	5.5	11.3

At the time of the accident skin panel manufacture, the manufacturing facility was a Boeing facility in Wichita, Kansas[9]. The facility had an aluminum recovery system with 2 crystallizers and centrifuges but no sock filters. The tank chemistry was sampled 3 times per week and standard practice was to dump half a tank and refill when smut build up became a problem. There were no regularly scheduled dumps or refills. The facility utilized a chromic acid anodize process and the parts were hand-scribed. In 1991, the facility underwent an extensive upgrade. The chem-mill tanks were upgraded to utilize an aluminum recovery system with sock filters and 1 of the 7 tanks was drained and cleaned each week. The new facility utilized a laser trace process instead of a hand scribe process for cutting the maskant. The new facility also eliminated roughness problems in the chem-mill areas of the panels. They installed a rotatable rack to turn the part during the milling process and utilized a phosphoric acid anodize process.

Two of the defects that can occur during chem-milling are channeling and ridging. Channeling is the formation of a groove or channel at the location of the fillet (Figure 7) while ridging is the formation of a ridge or raised area at the location of the fillet (Figure 8). It can be a common condition in the chem-mil process but doesn't always occur. Utilization of a single chem-mill step from maximum to minimum gage increases the potential for channeling. The condition can be detected by visual inspection alone. The possible causes are turbulence in the tank caused by out-gassing of the chemical reaction at the overhanging maskant, chemistry of the tank solution, part orientation, or over-scribing. Boeing has found that channeling can generally be blended out if the maskant is removed and the part is milled further creating additional chem-mill steps. The specification allows for channeling if the width to depth is greater than 6 and if the minimum gage thickness is above the minimum tolerance.

During fuselage build and lap joint installation, there is a stack up of tolerances that may affect how far the chem-mill step is from the edge of the opposite skin panel. The lower skin panel could overlap the upper skin panel chem-mill step with the installation tolerances at the extreme end of their ranges. The distance between the lap joint and the chem-mill step will affect the amount and severity of eccentricity in the lap joint under load.

3.0 Maintenance Records

The last A-check on the accident airplane was performed on October 15, 2010, and the last B-check was performed on October 5, 2010. The last Heavy C-check was completed on June 27, 2009. The only time the crown area was inspected was during the Heavy C-check. American Airlines work card 2211 – Fuselage Upper Lobe Interior BS 308-440 (Sec 41)-Inspection calls for removing the ceiling panels and insulation, cleaning the interior structure, and performing a general visual inspection of the interior of the skin panels, skin splices and joints, frames, stringers, and doublers. American Airlines work card 2250 – Upper Fuselage-Inspect calls for an external detailed inspection of the upper fuselage skin paying particular attention to the skin in areas of attachments to stringers, frames, and longitudinal splices. There were no non-routine work cards issued under work card 2211. There were 2 non-routine work cards issued under work card 2250 but they were for a dent in the skin at BS 1570 between S-15R and S-16R and a sheared rivet head at BS 1640 between S-15R and S-16R[10].

[9] The facility is currently owned and operated by Spirit Aero Systems.
[10] See Attachment 1 to this Factual Report for the pertinent maintenance records.

The 2010 Field Maintenance Reliability (FMR) was examined for any discrepancies in the area of the hole and nothing was found that involved the upper crown skin or provided an advance warning of the event.

The Boeing recommended maintenance inspections for the crown skin in the area of cracking were an external general visual inspection (GVI) every 6,000 cycles or 18 months and internal GVI every 24,000 cycles or 72 months.

4.0 Certification of Fuselage Skin

The Boeing 757 airplane fuselage was certificated under 14 *Code of Federal Regulations*, Part 25 at amendment level 45 with the exception of Federal Aviation Regulation (FAR) 25.365 which was at amendment level 54 for the aft cargo compartment. The forward crown skin panel was designed such that there were skin pad-ups installed for fail safety and the frames were shear tied directly to the fuselage skin. There are no specific regulations for the certification of the fuselage skin other than the pressure requirements in FAR 25.365. Based on this regulation, the accident skin panel was certified with an operating differential pressure of 8.6 psi ± 0.1 psi, a pressure due to malfunction in the pressure relief valve of 9.1 psi, an ultimate pressure of 18.2 psi, and ultimate flight loads with a pressure of 13.65 psi. The airplane is certified to withstand a hole of up to 6.2 ft^2 and still maintain its residual strength capabilities. The certification assumed the skin was manufactured from 2024-T3 Alclad with an ultimate tensile strength of 62 ksi, a minimum gage thickness at the pad up areas of 0.056 inch, and a minimum gage thickness in the pockets of 0.040 inch.

5.0 Fuselage Loads

The crown skin panel is subjected primarily to pressurization loads but also has some component of fuselage loads due to maneuver and gust. The pressurization of the airplane causes an outward force on the skin panels resulting in a hoop (or circumferential) tension stress in the skin. This hoop tension stress in the skin of a semi-monocoque structure is defined as:

$$\sigma_{hoop} = \frac{0.83PR}{t_{skin}}$$

Where P is the pressure applied, R is the radius of the fuselage and t is the skin thickness. This hoop stress for the accident airplane is about 12.8 ksi at an operating pressure differential of 8.6 psi and 13.2 ksi using the theoretical maximum values for a 9.1 psi pressure differential. All of this assumes a skin pocket thickness of 0.040 inch and a fuselage radius of 74 inches. The secondary load due to maneuver and gust is primarily in the longitudinal direction and will not add a significant hoop tension component to the stress in the skin. The pressurization of the fuselage will also cause a small amount of deflection at the lap joint due to its eccentricity. This is shown exaggerated in Figure 9. This deflection will induce a bending stress at the chem-mill step radius that is tensile in the upper skin chem-mill radius and compressive in the lower skin chem-mill radius. Assuming the nominal skin pocket thickness of 0.040 inch, a step change in thickness at the chem-mill radius, a nominal distance of the step from the lap joint, and the theoretical maximum pressurization stress, a Boeing analysis indicated that the peak stress in the chem-mill step radius, including both pressurization and induced bending stresses, was about 26 ksi (13.2 ksi hoop and 12.8 ksi bending).

5.1 Fatigue Loads

The Boeing 757 was designed with a designed service objective (DSO) of 50,000 flight cycles. The fatigue analysis performed by Boeing in accordance with FAR 25.571 indicated there would be no cracking at the chem-mill step assuming the skin and pockets were at nominal drawing thickness. The analysis showed that the lap joints were more critical in fatigue that the chem-mill steps. The 757 full scale fatigue test was run to more than 100,000 pressurization cycles with no crack indications on the crown skin at the S-4 chem-mill steps. The crown skin panel from the fatigue test article was re-inspected after the accident using the SB methods (spot probe mid frequency external eddy current, sliding probe external eddy current) along with external ultrasonic phased array and internal high frequency eddy current inspections with no findings. A fluorescent penetrant inspection of the panel also revealed no crack indications.

In order to quantify the effect of the channeling on fatigue life, the NTSB performed a fatigue analysis of a simplified lap joint and chem-mill step using AFGROW Fracture Mechanics and Fatigue Crack Growth Analysis software which utilizes linear elastic fracture mechanics principles. The problem was modeled as a flat plate with a width (W) of 15.5 inches and thickness (t) of 0.040 inch. The channel was modeled as a semi-elliptical crack with a total length (2C) of 0.5 inch centered in the flat plate. The crack growth rate model used was the NASGRO Equation, version 3, the material was set to 2024-T3 Al [Clad; plt & sht; L-T] and the failure criteria was based on the maximum stress intensity factor (K_{max}). A combined loading was applied to the flat plate with an axial stress of 13.2 ksi and a bending stress of 12.8 ksi obtained from the Boeing analysis. The axial stress was applied perpendicular to the long axis of the semi-elliptical crack and the bending stress was applied about the long axis of the semi-elliptical crack such that the stress would cause the crack to open. A constant amplitude applied stress with a stress ratio (R) of the minimum (0 ksi) to maximum (26 ksi) stress set to zero was applied to match the typical stress condition encountered by fuselage skin. The depth of the crack (a) was varied from 0.00 to 0.02 inch in 0.001 inch increments and the model was allowed to run until failure to obtain the fatigue life or cycles to failure. The percent change in the fatigue life between channel thickness values was calculated using:

$$\%\ change = \left(\frac{Cycles_i - Cycles_{i-1}}{Cycles_{i-1}}\right) \times 100$$

The total percent change in fatigue life from the life at the minimum drawing thickness of 0.037 inch was calculated using:

$$\%\ change = \left(\frac{Cycles_i - Cycles_{0.037}}{Cycles_{0.037}}\right) \times 100$$

The data obtained is presented in Table 1 below. A graph of the minimum thickness at the semi-elliptical crack location (channel) versus fatigue life for minimum thickness values between 0.020 inch and 0.038 inch is presented in Figure 10.

From the simplified analysis a fatigue life of 157,149 cycles was obtained for a panel with a channel with a minimum thickness of 0.038 inch and 81,240 cycles for a minimum thickness of 0.037 inch. For channels with a minimum thickness above this value, the life is essentially infinite; the model was run for several hundred thousand cycles with no failure. The data shows a significant decrease in the fatigue life for each 0.001 inch decrease in minimum thickness especially between 0.034 inch and 0.037 inch. This corresponds to the area on the graph where the curve steepens sharply.

t_{min}	a	Cycles	% change	Total % change
0.040	0.000	Infinite		
0.039	0.001	Infinite		
0.038	0.002	157149		
0.037	0.003	81240	-48.3%	
0.036	0.004	51849	-36.2%	-36.2%
0.035	0.005	36243	-30.1%	-55.4%
0.034	0.006	26650	-26.5%	-67.2%
0.033	0.007	20443	-23.3%	-74.8%
0.032	0.008	16050	-21.5%	-80.2%
0.031	0.009	13042	-18.7%	-83.9%
0.030	0.010	10748	-17.6%	-86.8%
0.029	0.011	9049	-15.8%	-88.9%
0.028	0.012	7845	-13.3%	-90.3%
0.027	0.013	6849	-12.7%	-91.6%
0.026	0.014	6144	-10.3%	-92.4%
0.025	0.015	5546	-9.7%	-93.2%
0.024	0.016	5049	-9.0%	-93.8%
0.023	0.017	4744	-6.0%	-94.2%
0.022	0.018	4445	-6.3%	-94.5%
0.021	0.019	4150	-6.6%	-94.9%
0.020	0.020	4043	-2.6%	-95.0%

Table 1 – Fatigue life data from AFGROW

6.0 Service Bulletin and AD

On November 22, 2010, Boeing released Service Bulletin (SB) 757-53-0097 instructing operators to inspect the crown skin panel from BS 297 to BS 439 at the S-4L and S-4R lap joints. The SB allows for an initial external detailed visual, external sliding probe eddy current (EC), or external spot probe mid-frequency eddy current (MFEC) inspection of the chem-mill step areas shown in Figures 5 and 6 followed by repetitive eddy current inspections. The SB is effective for all 757 models except the 757-200PF Package Freighter and the 757-300. The initial inspection should occur prior to 15,000 flight cycles and the repetitive inspections should be performed every 200 flight cycles if using MFEC and every 300 flight cycles if using EC. On January 6, 2011, Boeing released Revision 1 to the SB that clarified some of the figures but did not change the details of the inspection.

On January 10, 2011, the FAA issued AD 2011-01-15 requiring repetitive inspections of the crown fuselage skin of all 757 airplanes except the package freighter per the details presented in the original release of the Boeing SB. The AD became effective on January 25, 2011.

On March 28, 2011, Boeing provided an Alternate Method of Compliance (AMOC) Notice in SB 757-53-0097-01-AMOC-02 that allows the extension of intervals for the SPEC inspections in the Boeing SB to 620 flight cycles (from 300 cycles). This AMOC was approved by the FAA in a letter reference 120S-11-162 dated March 25, 2011. Striation data from the reported cracks was used to substantiate this change for the SPEC inspections.

7.0 Additional Airplanes

The NTSB was made aware of two additional airplanes with cracking similar to the accident airplane.

7.1 United Airlines 757-200 (N520UA)

L/N: 313
TC: 24,631 cycles
TT: 65,625 hours

On September 11, 2010, prior to the accident, United Airlines notified Boeing of a 10.75 inches long crack in the crown skin of their airplane between BS 420 and BS 439 above the S-4L lap joint common to the chem-mill step. A section of skin was removed from the airplane and sent to Boeing for examination. Boeing determined that there were multiple fatigue initiation sites in the skin at the bottom of a channel at the chem-mill step. The fatigue cracking then grew through the skin thickness. The total length of through thickness fatigue cracking was about 2.5 inches with the remainder of the crack being mixed fatigue and ductile overstress. At the area of crack initiation, there was evidence of a channel with a minimum thickness measured of about 32.6 mils. Several other areas of the submitted panel exhibited channeling with minimum thickness measurements about 33 to 35 mils. Material characterization testing confirmed that the skin panel was manufactured from 2024-T3 clad aluminum alloy.

7.2 American Airlines 757-200 (N618AA)

L/N: 260
TC: 23,665 cycles
TT: 65,700 hours

On December 5, 2010, American Airlines found three crack indications in the crown skin of their airplane between BS 400 and BS 439 above the S-4L lap joint common to the chem-mill step while performing the SB 757-53-0097 inspection. A section of skin containing the crack indications was removed from the airplane and sent to Boeing for examination. Boeing confirmed all 3 crack indications and found an additional crack location. The cracks were opened and examination of the fracture surfaces revealed multiple site fatigue initiation at the base of a channel adjacent to the chem-mill step. The crack depth was 0.010-0.015 inch and had not gone completely through the thickness. At the area of crack initiation, there was evidence of a channel with a minimum thickness measure of 35.6 mils. The rest of the channel measured about 36 mils. Materials characterization testing confirmed that the skin panel was manufactured from 2024-T3 clad aluminum alloy.

7.3 Other Boeing Airplanes

As mentioned earlier, this skin panel on the 757 airplane is the only panel that utilizes a single chem-mill step from stock gage to pocket gage and thus is uniquely susceptible to the cracking phenomenon described. The 737 airplane model does not have any single chem-mill step areas similar to those on the accident skin panel. The 767 airplane has some single chem-mill step areas on the fuselage skin panels but the milled step thickness is 0.007 inch to 0.014 inch as opposed to the 0.022 inch nominal step thickness on the accident skin panel. The 747 and 777 are being evaluated to determine if the skin panels have any single chem-mill step areas similar to the accident skin panel.

Submitted by:
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