NATIONAL TRANSPORTATION SAFETY BOARD

Office of Aviation Safety
Washington, D.C. 20594

March 14, 2022

Group Chairman's Factual Report

AIRWORTHINESS

DCA21FA085

A. INCIDENT

Location: Broomfield, CO
Date: February 20, 2021
Time: 1309 mountain standard time (MST)
2009 coordinated universal time (UTC)
Airplane: Boeing 777-200, N772UA, United Airlines flight 328

B. AIRWORTHINESS GROUP

Group Chairman	Clinton R. Crookshanks National Transportation Safety Board Denver, CO
Group Member	James Laubaugh Federal Aviation Administration Des Moines, Washington
Group Member	William (Bill) Williams The Boeing Company Seattle, Washington
Group Member	Michal Dudzik United Airlines Chicago, Illinois
Group Member	Eric Chesmar United Airlines San Francisco, California
Group Member	John McCartney International Brotherhood of Teamsters Denver, Colorado

C. DETAILS OF THE INVESTIGATION

NTSB personnel responded to the scene the evening of the incident and recovered most of the debris with the assistance of United Airlines personnel and local first responders. The debris was transported to a United Airlines hangar at Denver International Airport later that evening. The group examined the airplane and debris in the hangar February 21-26, 2021. The right engine was removed from the airplane and shipped to the United Airlines engine overhaul facility at San Francisco International Airport for examination by the Powerplants Group. The debris from the inlet, fan cowls, and thrust reversers was shipped to the Boeing Equipment Quality Analysis (EQA) Lab in Seattle, Washington for further examination and testing.

Members of the Airworthiness Group examined and reconstructed the inlet, fan cowls, and thrust reversers at the EQA lab March 11-18, 2021. The Airworthiness Group Chairman examined the fan case with the assistance of the Powerplants Group April 12-15, 2021.

D. FACTUAL INFORMATION

1.0 Airplane Overview

The Boeing 777-222 airplane is a long range, twin-engine, wide-body, transport category airplane (Figure 1) originally certified in 1995. The airplane is equipped with a conventional tail and retractable tricycle landing gear. The airplane is 209 feet, 1 inch long, 60 feet, 9 inches tall at the tail, has a fuselage diameter of 20 feet, 4 inches, has a wingspan of 199 feet, 11 inches, and has a horizontal stabilizer span of 70 feet, 7.5 inches. The airplane primary wing and fuselage structure is of all metal construction, primarily aluminum alloys. The horizontal stabilizer, vertical stabilizer, elevators, rudder, ailerons, flaperons, flaps, spoilers, floor beams, floor panels, and engine cowling are of composite construction along with many of the fairings, doors and panels. The composite structure consists of graphite epoxy, carbon fiber reinforced plastic (CFRP), fiberglass, or honeycomb sandwich. The incident airplane, SN 26930, was manufactured in September 1995 and delivered to United Airlines.

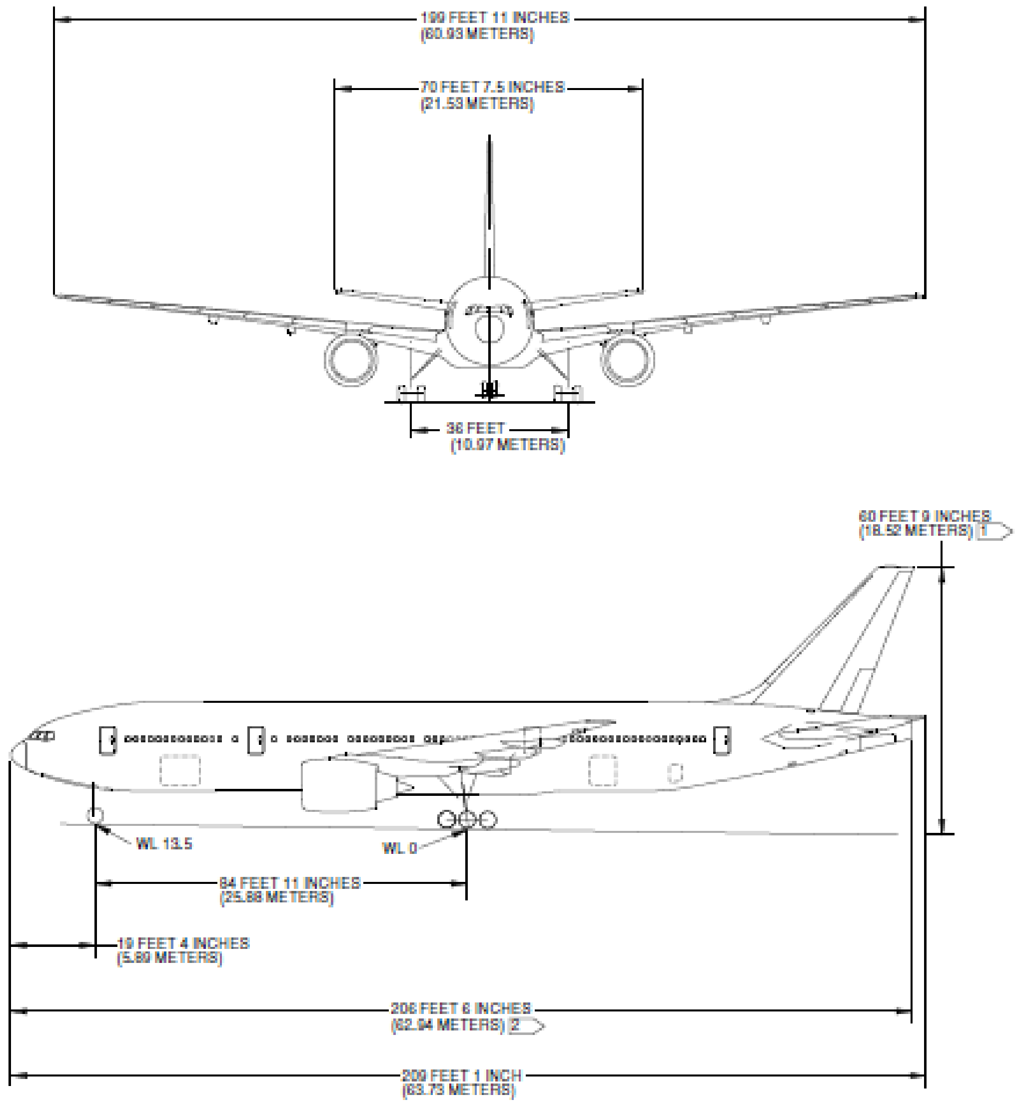

Figure 1 - Boeing 777-200 3-view drawing

The airplane is equipped with two Pratt & Whitney PW4000 series engines, one on each wing pylon (Figure 2). The engine is attached to the pylon, the inlet is attached to the forward end of the engine, the fan cowls are attached around the center portion of the engine and the thrust reversers are attached around the aft portion of the engine.

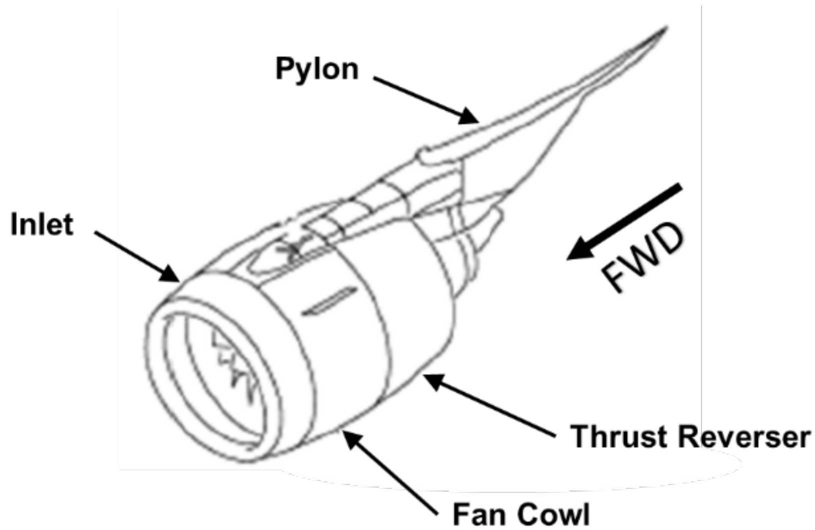

Figure 2 - Engine installation drawing for 777-200

1.1 Inlet

The inlet is a cantilevered structure attached to the forward flange of the engine fan case (A-flange) through the inlet attach ring with 52 bolts. The inlet consists of two concentric cylindrical structures joined together by forward and aft bulkheads (Figure 3). The hollow aluminum lip skin is attached to the forward bulkhead and provides an aerodynamic surface for the leading edge of the inlet and a passage for the engine anti-ice air. The inlet aft bulkhead consists of the aluminum inlet attach ring and aluminum outer ring chord with a CFRP honeycomb sandwich composite web. The inlet forward bulkhead consists of the aluminum inner and outer ring chords with a stiffened aluminum web. The inlet outer barrel is comprised of three CFRP honeycomb sandwich panels. A section of the outer barrel in the lower right quadrant is comprised of a titanium skin where the anti-ice exhaust duct is located. The inlet inner barrel is comprised of two CFRP honeycomb sandwich panels. The inner face sheet of the inner barrel is perforated for noise suppression and the outer face sheet is solid.

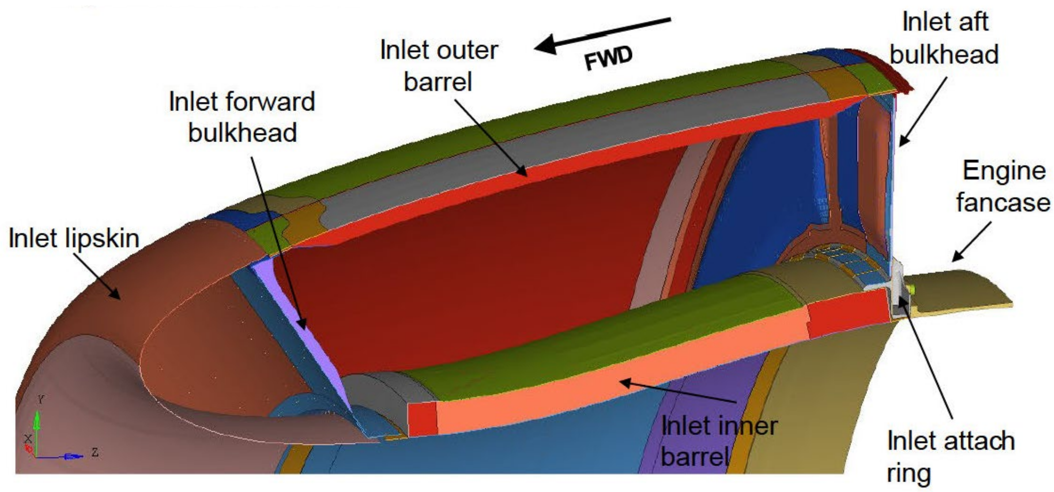

Figure 3 - Inlet cross-section drawing for 777-200

1.2 Fan Cowl

The fan cowl provides an aerodynamic enclosure around the engine fan cases and the doors open to allow maintenance access to the engine. The two fan cowls are semi-cylindrical doors that are each fastened to 4 hinges at the upper ends; 2 hinges on the fan cowl support beam, 1 floating hinge, and 1 hinge on the engine. The fan cowls are held together at the lower centerline through 4 latches (Figure 4). The fan cowls are CFRP honeycomb sandwich construction. The fan cowl support beam is a CFRP honeycomb sandwich panel attached at the forward end to the inlet attach ring and to the fan case at the aft end through aluminum fittings. The fan cowls interface with the inlet at the forward edge through a v-blade on the fan cowls that seats in a v-groove on the inlet aft bulkhead. The fan cowls interface with the thrust reversers at the aft edge through a sliding contact seal.

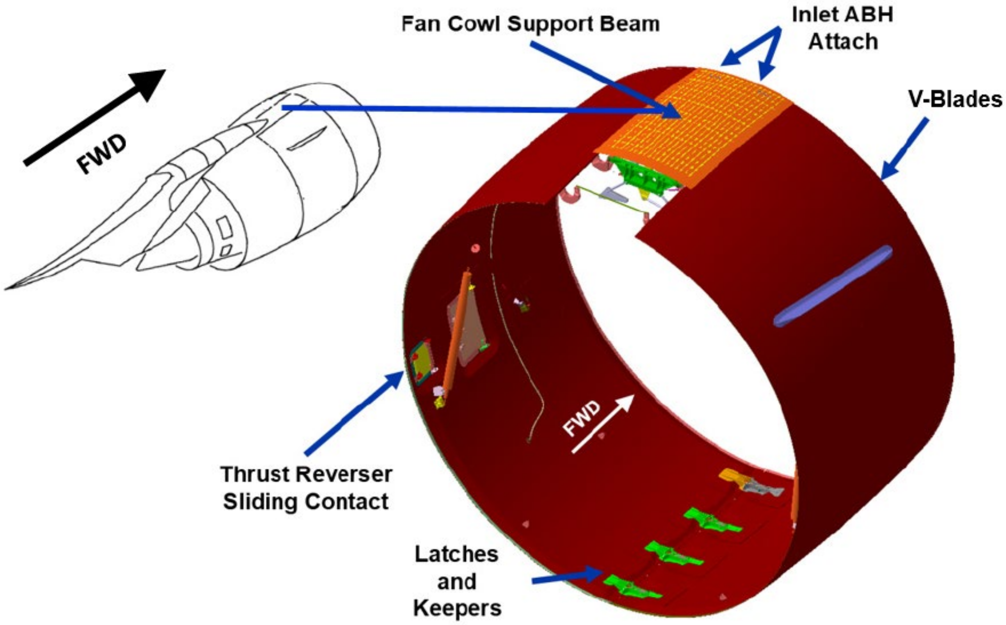

Figure 4 - Fan Cowl drawing for 777-200

1.3 Thrust Reversers

The thrust reversers (TRs) enclose the engine core to provide an aerodynamic enclosure around the engine, direct the fan exhaust, and actuate to provide reverse thrust during landing. The two semi-cylindrical TR halves are comprised of 3 main components, the translating sleeve, the fan duct cowl, and the aft cowl (Figure 5). The CFRP honeycomb sandwich inner wall of the fan duct cowl and the titanium aft cowl enclose the engine core and comprise the fire zone in the TR. The TRs are hinged at the upper end to the pylon and open to provide maintenance access. The main structural skeleton of the TR consists of the aluminum hinge beam at the upper end, the aluminum torque box at the forward end, the aluminum latch beam at the lower end, and the aluminum aft support ring and titanium aft cowl at the aft end. The CFRP honeycomb sandwich inner wall is connected to the TR at the upper and lower bifurcations. The CFRP honeycomb sandwich translating sleeve forms the outer surface of the TR and the outer wall of the fan duct cowl in the closed position. The translating sleeve slides aft along a mechanism attached to the torque box when actuated for reverse thrust. The mechanism also deploys the blocker doors to direct the fan exhaust. Once the translating sleeve moves aft, the fan exhaust is directed through the CFRP cascades installed between the torque box and aft support ring to direct the air forward for reverse thrust. The CFRP honeycomb sandwich fan duct cowl outer wall engages the fan case through the outer v-blade seal and the fan duct cowl inner wall engages the engine core through the inner v-blade seal when the TR is closed. The TR halves are held in the closed position with 2 load share latches, 2 hoop tie latches, and 2 split line latches on the aft cowl and 6 latch beam latches on the fan duct cowl. There are three latch beam access doors on the inboard TR to allow access to the latch beam and drain lines at the lower centerline of the TR.

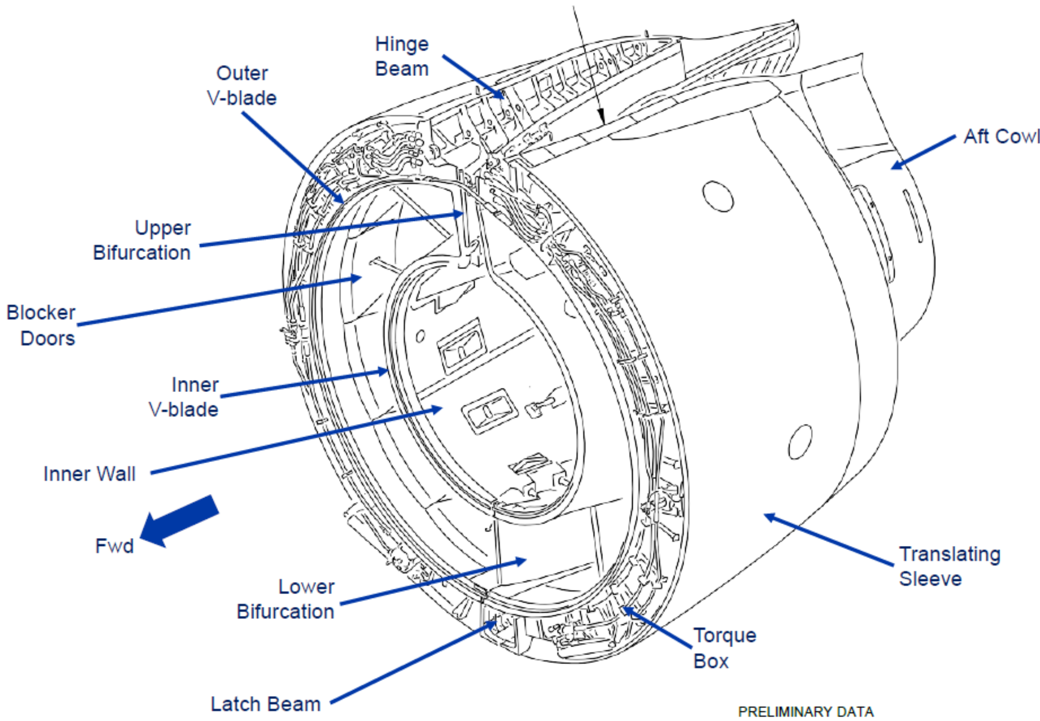

Figure 5 - Thrust Reverser drawing for 777-200

Rubber fire seals are installed in each TR half to help contain an undercowl fire within the interior of the fan duct inner wall and aft cowl. The fabric reinforced silicone rubber seals are installed along the upper and lower bifurcation walls and down the upper aft edge of the aft cowl. Kapton-faced thermal insulation blankets are installed on the upper and lower bifurcations and on the inside surface of the inner wall in the fire zone to protect the composite structure from radiant engine heat and fire.

2.0 Incident Site

Available data indicated the airplane was over Broomfield, Colorado at the time of the engine failure event. Sections of two fan blades separated during the event damaging the right engine, fan cowls, inlet, and thrust reversers. Multiple pieces of the inlet, fan cowls, and thrust reversers separated during the event and were recovered. The majority of the recovered debris was found scattered over about 40 acres bounded by East 14th Avenue on the north, East 13th Avenue and Broomfield Commons Drive on the south, Dover Street on the west, and the eastern edge of Broomfield Commons Park on the east. The 11.5 feet diameter inlet forward bulkhead and attached lip skin was found in the front yard of a residence located at 1372 Elmwood Street (Figure 6). The roof of a blue pickup truck parked in the driveway adjacent to the inlet lip skin was crushed. The fan cowl support beam with small portions of the fan cowls attached was found in the attic of a residence located at 13251 Sheridan Boulevard and had punctured the roof of the residence. Multiple items of debris were located in the open area of Broomfield Commons Park and in the neighborhood west of the park. The Nacelle Drain Access door was found in the yard of a residence located at 6104 Queen Court in Arvada, CO, about 9.8 miles south-southwest of the park. A portion of the right engine outboard thrust reverser lower outboard translating sleeve was recovered in a field about 100 yards north of East Quincy Avenue in Aurora, CO, about 30.4 miles southeast of the park, with significant fire damage.

Figure 6 - Inlet lip skin(left) and fan cowl support beam (right) as found

NTSB and United Airlines personnel recovered the debris around Broomfield Commons Park on the evening of the incident and transported everything to the United Airlines hangar at Denver International Airport. Additional smaller items of debris were recovered in the ensuing days as they were found by local residents. The section of translating sleeve was recovered on February 23 after being reported by an eyewitness.

3.0 Damage Assessment

The right engine failed, the inlet and fan cowls separated from the airplane, and the thrust reversers were damaged during the event. The engine failure and damage are expressly excluded from being considered substantial damage under 49 CFR 830.2.

The right engine pylon and lower wing skin were sooted adjacent to and aft of the engine. There was a hole in the right side of the wing-to-body and underwing fairings about 5 feet long by about 2 feet wide between fuselage stations 1030 and 1090. Fairings 192SR, 196AR, 196DR, 196FR, 196GR, 196LR, and 196JR were all damaged along with some of the underlying support structure. There was scuffing, scratching, paint transfer, and gouging of the lower wing skin adjacent to the hole. A section from the lower leading edge of the left fan cowl door was embedded between fairing 196FR and the fuselage. There was an area of impact damage about 20 inches long by 5 inches wide that penetrated the outer skin of fairing 196LR. There was an area of impact damage about 7 inches long by 4 inches wide that penetrated the outer skin of fairing 196JR. Multiple areas of scuffing, scratching, gouging, and paint/soot transfer were identified on the lower wing skin adjacent to both sides and aft of the right engine pylon from the wing root to about wing station (WS) 732. There was metal spray deposited on the lower wing skin outboard of the pylon between WS 547 and WS 732. The composite flap canoe fairings on the left and right sides of the right engine had impact damage. There was scuffing, cuts in the face sheet with exposed honeycomb, and black transfer marks on the lower surface of the right flaperon. There was a dent and a crack in the number 9 slat leading edge. There was a small puncture in the upper surface of slat 8. There were two punctures in the wing trailing edge panel 652HB. Several of the pylon access panels sustained punctures and gouging. There were several areas of damage or gouging noted to the right pylon support structure. The right horizontal stabilizer leading edge had multiple areas of scuffing and black transfer marks. There was some gouging of the upper leading edge near right horizontal stabilizer station 378 and a small dent in the outboard leading-edge panel. The damage to the fairings, doors, and slats is excluded from consideration under NTSB regulations and guidance. The remaining damage to the airplane would not adversely affect the structural strength, performance, or flight characteristics of the airplane and thus is not considered substantial damage under NTSB regulations.

4.0 Airplane Examination

The clock positions referenced are as viewed from aft of the right engine looking forward. The 12:00 position is the upper center line of the engine with the engine installed on the wing. A line between 12:00 and 6:00 is offset from vertical as installed due to the orientation of the engine and the wing dihedral angle. The engine rotates clockwise.

The details of the work performed by the Boeing EQA lab are documented in the Boeing EQA report in Appendix A.

4.1 Fan Case

The fan case was removed from the engine at the B-flange by removing the 163 bolts. The breakaway torque (loosening direction) was recorded for all fasteners (Figure 7). The fan case was installed in existing tooling for examination that allowed it to be rotated

B-FLANGE BREAKAWAY TORQUE (IN-lb)

1 - 75	31 - 10	61 - 15	91 - 20	121 - 75	151 - 20
2 - 120	32 - 25	62 - 35	92 - 40	122 - 15	152 - 10
3 - 85	33 - 20	63 - 20	93 - 25	123 - 30	153 - 20
4 - 85	34 - 10	64 - 25	94 - 10	124 - 25	154 - 15
5 - 70	35 - 5	65 - 40	95 - 25	125 - 25	155 - 25
6 - 110	36 - 25	66 - 50	96 - 30	126 - 15	156 - 50
7 - 90	37 - 15	67 - 25	97 - 30	127 - 10	157 - 90
8 - 40	38 - 15	68 - 35	98 - 30	128 - 15	158 - 100
9 - 20	39 - 15	69 - 55	99 - 15	129 - 25	159 - 75
10 - 15	40 - 15	70 - 25	100 - 15	130 - 15	160 - 70
11 - 40	41 - 25	71 - 45	101 - 5	131 - 25	161 - 50
12 - 25	42 - 30	72 - 10	102 - 10	132 - 20	162 - 40
13 - 45	43 - 30	73 - 20	103 - 15	133 - 15	163 - 20
14 - 40	44 - 50	74 - 15	104 - 20	134 - 15	
15 - 45	45 - 20	75 - 50	105 - 50	135 - 20	
16 - 30	46 - 40	76 - 20	106 - 15	136 - 15	
17 - 30	47 - 75	77 - 30	107 - 10	137 - 10	
18 - 50	48 - 75	78 - 30	108 - 10	138 - N/R	
19 - 20	49 - 25	79 - 60	109 - 40	139 - 25	
20 - 45	50 - 15	80 - 15	110 - 20	140 - 15	
21 - 40	51 - 10	81 - 20	111 - 30	141 - 15	
22 - 30	52 - 30	82 - 15	112 - 25	142 - 10	
23 - 50	53 - 10	83 - 15	113 - 25	143 - 20	
24 - 25	54 - 25	84 - 15	114 - 35	144 - 25	
25 - 25	55 - 25	85 - 10	115 - 35	145 - 10	
26 - 10	56 - 40	86 - 15	116 - 25	146 - 20	
27 - 15	57 - 30	87 - 10	117 - 25	147 - 20	
28 - 15	58 - 25	88 - 15	118 - 40	148 - 30	
29 - 45	59 - 20	89 - 25	119 - 50	149 - 35	
30 - 25	60 - 15	90 - 50	120 - 55	150 - 05	

Figure 7 - Breakaway torque values for fan case attach bolts

4.1.1 Fan Case Outer Surface

The outer two layers of the Kevlar® containment belt, inclusive of the environmental wrap, were torn with split and frayed fibers along the edges (Figure 8). The tear near the 11:30 position extended aft about 34 inches axially starting about 1 inch aft of the forward edge. There was a hole in the third layer of Kevlar® from the outside that measured about 5 inches axially and 3.5 inches circumferentially. All the remaining Kevlar® layers were penetrated by an embedded piece of fan blade that was centered about 15 inches aft of the A-flange near the 11:30 position. The hole through the layers measured 1 inch axially and 1.5 inches circumferentially. The Kevlar® containment belt layers were displaced outward about 3 inches from their nominal position with the blade fragment still in place.

The forward edge of the containment belt was displaced aft between 11:15 and 12:00 with a maximum displacement about 0.5 inch centered near 11:30 that cracked the sealant. The sealant bead on the aft edge of the containment belt was cracked and displaced forward from 9:00 to 1:30. The maximum displacement was about 1.75 inches between 11:15 and 11:45. The inner 3 layers of the containment belt remained in place while the rest were displaced forward. There was red sealant on the aft edge between 1:45 and 7:00 and between 8:00 and 9:45.

Figure 8 - Tear in outer layers of containment belt near 11:30 position

There was an area of multiple short scuff marks, each less than 1 inch in length, in the outer environmental layer with some frayed fibers between 6:15 and 8:15. There was a more concentrated area of scuffing and frayed fibers centered about 7:15 and 17 inches aft of A-flange that measured about 5 inches axially and 11 inches circumferentially. There was a linear witness mark/scuff on the outer environmental layer that extended from 7:30 and 13 inches aft of A-flange to 8:15 and 6.75 inches aft of A-flange. There was no obvious evidence of other contacts on the fan case containment belt.

There was a dark fan shaped discoloration of the outer environmental layer that began near the forward edge around 2:15 and expanded rearward. The outer layer in this area was removed for further analysis. The majority of the B-flange fastener holes were visibly elongated in the radial direction.

4.1.2 Fan Case Inner Surface

The interior surface of the fan case had significant damage (Figure 9[1]). There was an area of heavy gouging and scoring of the fan case aluminum isogrid material about 9 inches wide around the entire circumference in the fan blade plane of rotation except around where an embedded fan blade fragment was located. The

[1] See Appendix B for large format stitched photos of the fan case, inner barrel, outer barrel, and fan cowls.

gouging of the isogrid was heaviest from 4:30 to 10:00 where evidence of the underlying isogrid features were present. The area between 8:30 and 9:45 was gouged such that there was cracking at the isogrid rib-pocket interfaces. Almost all the abradable rub strip honeycomb and blue potting compound was missing around the entire circumference. There were small remnants of the blue potting compound along the leading edge of the rotation plane between 11:00 and 4:00. Three adjacent acoustic panels were mostly intact aft of the rotation plane from 6:00 to 7:30, 7:30 to 9:00 and 9:00 to 10:30, and the woven wire cloth exhibited evidence of multiple impacts. The inner face sheet was separated from the acoustic panel installed from 4:30 to 6:00 and the honeycomb was separated from 4:30 to 4:45. The honeycomb remained from 4:45 to 6:00. Portions of acoustic panel honeycomb between 4 inches and 15 inches forward of the B-flange remained from 10:30 to 4:30 with the free edges fractured and/or crushed. The inner face sheet and remainder of honeycomb was missing.

Figure 9 - Fan Case inner surface

There was a rectangular puncture of the aluminum isogrid that measured 13.5 inches long by 1 inch wide oriented about 35° from the plane of rotation centered near 11:00. Sections of aluminum isogrid were missing adjacent to the puncture in an area about 11 inches by 5 inches. The forward end of the puncture was located about 13 inches aft of A-flange and the aft end of the puncture was located about 21.5 inches aft of A-flange. There was a fracture of the aluminum isogrid from the forward end of the puncture that extended about 13 inches circumferentially clockwise. There was a fracture of the aluminum isogrid from the aft end of the puncture that extended about 45 inches in the clockwise direction that was oriented about -23° from the plane of rotation. A section of fan blade about 18 inches spanwise by 16 inches chordwise was lodged in the aluminum isogrid between 11:15 and 12:05 (Figure 10). The radial axis of the fan blade was oriented about -30° from the plane of rotation. The fan blade piece orientation was leading edge forward and the root section leading with respect to rotation direction. There was a fracture of the aluminum isogrid extending about 45 inches circumferentially clockwise from the leading edge of the fan blade. There was a fracture of the aluminum isogrid about 5 inches long between the fan blade and the puncture described earlier from 10:45 to 11:15. The fan case was cut axially in two places from the A-flange aft to the embedded blade in order to remove the blade fragment.

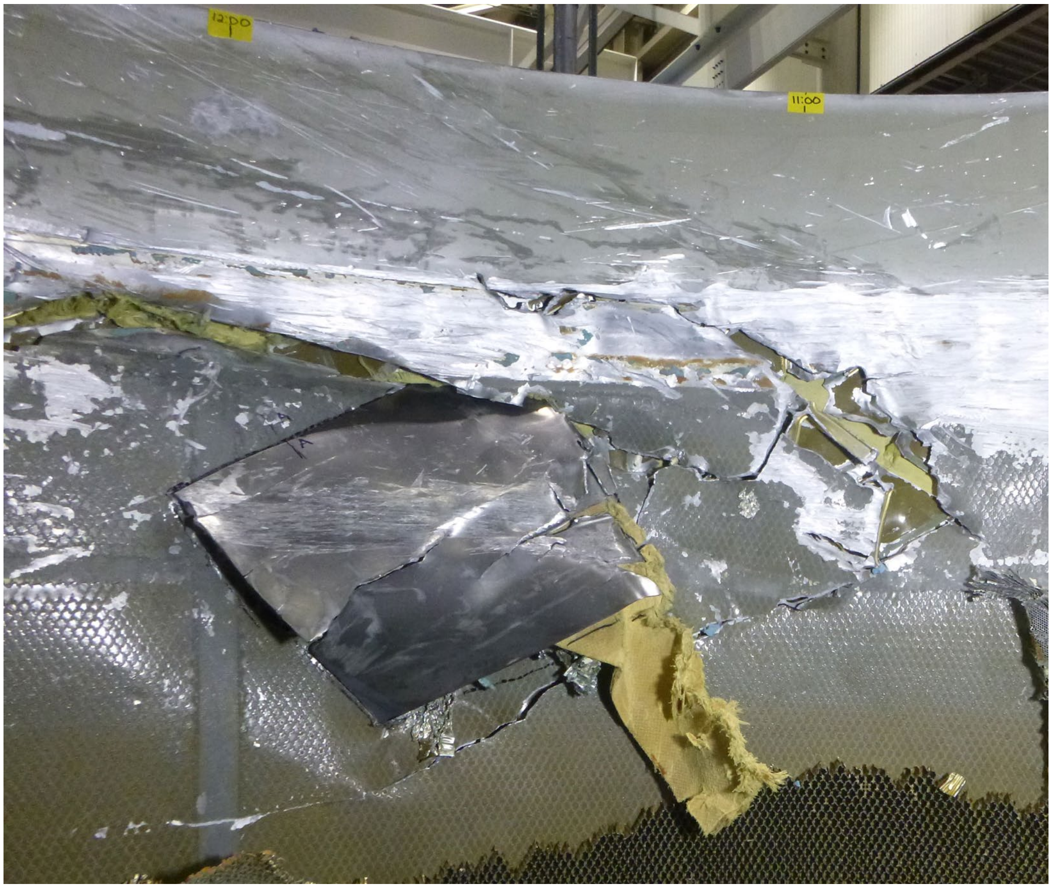

Figure 10 - Fan blade fragment lodged in fan case

The group identified multiple witness marks corresponding to fan blade fragment trajectories on the inner surface of the fan case. The group mapped the more significant witness marks that included 4 forward moving fragments, 1 circumferential moving fragment, and 2 aft moving fragments. See Appendix B for the fragment map annotated stitched photos. The fragment 1 witness marks consisted of dark scuffing, scratching, and gouging that measured about 7 inches wide and were oriented about 23° from the plane of rotation. The witness marks started between 10:45 and 11:15 at the forward edge of the rotation plane and crossed the A-flange between 11:30 and 12:30.

The fragment 2 witness marks consisted of dark scuffing, scratching, and gouging that measured about 5.5 inches wide and were oriented about 7° from the plane of rotation. The witness marks started between 10:45 and 12:30 at the forward edge of the rotation plane and crossed the A-flange between 2:00 and 3:30.

The fragment 3 witness mark was a linear gouge that started near 1:00 about 1 inch forward of the forward edge of the rotation plane. The gouge traversed circumferentially for about 12 inches before curving forward. The gouge crossed the A-flange near 4:00 and was oriented about 6° from the plane of rotation.

The fragment 4 witness mark was an intermittent linear gouge that started near 2:45 at the forward edge of the rotation plane. The gouging traversed circumferentially before curving forward and crossed the A-flange near 5:15 oriented about 7° from the plane of rotation. There were gouges and scratches between the fragment 3 and 4 witness marks that may or may not be associated with these same fragments.

The fragment 5 witness mark was an area of dark scuffing about 1.5-2 inches wide that traversed circumferentially between 4:15 and 6:15.

The fragment 6 witness mark was a set of aft moving dark scuffs that varied between 2 and 4 inches wide. The scuffs crossed the A-flange between 4:30 and 4:45 and entered the forward edge of the plane of the fan between 5:15 and 5:45.

The fragment 7 witness mark was a set of aft moving dark scuffs about 4 inches wide that crossed the A-flange between 2:30 and 2:45. They were present for about 6 inches aft of the A-flange before disappearing.

The group identified several other minor witness marks corresponding to pieces moving aft within the fan case. One witness mark was a set of 6 intermittent gouges each with twin parallel gouges that crossed the A-flange near 5:30 and traversed to near 7:45. They were oriented about -4° from the plane of rotation. Another witness mark was an area of parallel dark scuffs about 5.5 inches wide that began near 8:00 and traversed to about 10:30 that were oriented about -6° to the plane of rotation. Another witness mark was a linear scratch that crossed the A-flange near 9:30 and traversed to 11:30 oriented about -14° from the plane of rotation. The last witness mark was a linear scratch that crossed the A-flange near 9:00 and traversed to 11:30 oriented about -9° from the plane of rotation.

4.2 Inlet

The inlet separated from the right engine and was recovered in multiple pieces. The inlet attach ring remained attached to the engine A-flange after the event. The aluminum attach ring was mostly intact around the perimeter and all the fasteners were installed. The vertical leg of the attach ring was fractured where it attached to the fan cowl support beam. Most of the composite aft bulkhead web fractured from the attach ring at the outer edge of the vertical flange leaving small portions of the composite web attached. About 33 inches of the aft bulkhead web remained attached to the attach ring near the 3:00 position and about 30 inches remained attached near the 9:00 position where the aluminum splice brackets were installed. The inlet attach ring was removed from the A-flange and the breakaway torque for each fastener was recorded (Figure 11). Each fastener was numbered by location to preserve their installed position.

2/25/21
XAI7LA085
085

A/C 2372 # 2 ENGINE
N772UA
NOSE COWL ATTACH BOLTS BREAK
AWAY TORQUE INCH / POUNDS

	POS.	TORQUE	POS.	TORQUE	POS	TORQUE	
2:00 –	1	40	26	95	51	125	
NUMBERED	2	50	27	125	52	50	
CLOCKWISE	3	80	28	180			
FROM REAR	4	35	29	210			AMM 71-11-01-
	5	30	30	145			400-802-N00
	6	65	31	220			INITIAL TRQ
	7	55	32	150			32-38"#
	8	40	33	150			FINAL TRQ
	9	70	34	165			320-385"#
	10	70	35	120			
	11	105	36	115			
	12	110	37	105			
	13	40	38	30			
	14	60	39	20			
	15	80	40	35			
	16	145	41	130			
	17	130	42	135			
	18	145	43	95			
	19	105	44	90			
	20	105	45	110			
	21	80	46	55			
	22	105	47	65			
	23	100	48	90			
	24	120	49	100			
	25	155	50	95			

Figure 11 - Breakaway torque values for inlet attach bolts

The aft bulkhead web and outer ring chord were fractured into several pieces. There were 18 individual pieces of the aft bulkhead identified including the attach ring. The entire attach ring and outer chord were identified. Three small areas of the aft bulkhead web were not conclusively identified, one near 6:45, one near 8:15, and one near 11:30. There were several small pieces of bulkhead web recovered that could not be placed.

The inlet forward bulkhead and lip skin assembly was mostly intact. The titanium anti-ice exhaust panel remained attached to the forward bulkhead near the 7:00 position but most of the composite inner and outer barrel structure was separated around the perimeter. There was a fracture in the lip skin near 8:00 but the forward bulkhead was intact. Portions of the inner barrel remained attached around most of the perimeter. There were some blue paint transfer marks on the exterior portion of the lip skin near the 5:45 location. The aft inner edge of the lip skin and forward bulkhead inner chord was deformed outward between 1:30 and 2:00 and the lip skin was fractured between 1:30 and 2:45. The forward bulkhead web was fractured near 1:45 and the web was separated from the outer chord between 1:45 and 2:00. The aft inner edge of the lip skin and forward bulkhead inner chord was deformed outward between 3:15 and 3:30. The forward bulkhead inner chord and web were fractured between 3:30 and 3:45.

4.2.1 Inner Barrel

The inlet shipping container was modified to provide a framework for the inner barrel 3-dimensional reconstruction with the lip skin facing down. The inlet attach ring was fixed in place above the lip skin at the appropriate distance from the forward bulkhead. The recovered inner barrel pieces were identified, positioned between the forward and aft bulkheads, and taped in place. All the recovered inner barrel pieces, except for 4 small pieces, were able to be conclusively located on the reconstruction. About 65% of the composite inner barrel was recovered and placed on the reconstruction comprising 29 individual pieces. The largest piece of separated inner barrel spanned from about 11:15 to 1:15 and contained the P2/T2 sensor. Figure 12 shows a stitched photo of the reconstructed inlet inner barrel inner surface. A large format photo can be found in Appendix B.

Figure 12 - Inlet inner barrel reconstruction

The group identified 6 distinct witness marks corresponding to fan blade fragment paths on the inner barrel structure using the scuffing of the inner barrel skin, damage to the face sheet and core, transfer marks on the perforated face sheet, and damage to the lip skin and forward bulkhead inner and outer chords. Four of these fragment paths aligned with paths identified on the fan case inner surface (fragments 1, 2, 6, and 7). See Appendix B for the fragment map annotated stitched photos.

The largest witness mark, fragment 1, was an area of scuffing, damage, and transfer marks that were about 12-18 inches wide and entered the inner barrel at the A-flange near 12:00. The markings traveled forward and clockwise to about 4:00. There was an area of the perforated face sheet about 12 inches wide that was missing between 12:00 and 2:45 with little apparent damage to the underlying core material. There were dark transfer marks at the edges of the missing face sheet. The transfer marks stopped about 1.5 inches prior to the 3:00 split line and started again about 5 inches after the split line. The signatures disappeared near 4:00 where there was a hole in the reconstruction.

The fragment 2 witness mark entered the A-flange near 4:00 with scuffing and transfer marks and traveled slightly forward and clockwise. There were missing areas of inner barrel between 5:00 and 5:15 and between 5:45 and 7:00 in line with the fragment 2 path. The signature traveled a maximum of about 8 inches forward of the A-flange and appeared to start moving aft about 7:00 and exited the A-flange near 9:30.

The fragment 6 witness mark was an aft moving fragment with transfer marks that initiated about 12 inches forward of the A-flange near 4:00 and traveled aft and clockwise to exit the A-flange near 4:30. The signature was just dark colored transfer marks on the inner barrel perforated skin.

The fragment 7 witness mark was an aft moving fragment with transfer marks that initiated at the edge of the fractured perforated face sheet about 6 inches forward of the A-flange near 1:30 and traveled aft and clockwise to exit the A-flange near 2:30. The signature was just transfer marks on the inner barrel perforated skin.

The fragment 8 witness mark was a hole in the inner barrel about 4 inches by 2 inches between 1:15 and 1:30 just aft of the lip skin edge. The lip skin was fractured from the forward bulkhead and curled outward, and the forward bulkhead inner chord was deformed outward between 1:30 and 2:00. There was gouging and material transfer marks on the lip skin and chord.

The fragment 9 witness mark was a linear transfer mark that entered the A-flange near 1:15 that traveled forward and clockwise about 9 inches to the missing area of face sheet. The transfer mark appeared to line up with an area of fractured and deformed lip skin and forward bulkhead inner chord between 3:15 and 3:30. The lip skin was fractured from the forward bulkhead and curled outward, and the forward bulkhead inner chord was deformed outward. There was gouging and material transfer marks on the lip skin and chord.

4.2.2 Outer Barrel

Most of the composite outer barrel was recovered and conclusively identified. The pieces were placed on the floor in a 2-dimensional reconstruction. The panels were fractured axially across the width near 12:15, 1:15, 3:00, 5:30, 7:45, 9:00, and 11:15. There was an arc shaped mechanical cut through the panels between about 3:45 and 7 inches forward of the aft edge to about 5:30 and 7 inches aft of the forward edge. There was an area between 2:00 and 3:00 where several pieces were missing and unidentified in the recovered debris. A few pieces from this area could not be conclusively located. Both Powered Door Opening System (PDOS) switches were recovered separated from the outer barrel. The forward flange was fractured from the panels and remained attached to the inlet forward bulkhead between 12:30 and 3:00. There was an area of damage near the aft edge of the outer barrel panels between 12:15 and 1:45 that extended forward about 4 inches at its maximum. Figure 13 shows a stitched photo of the reconstructed inlet outer barrel inner surface. A large format photo of both the inner surface and outer surface can be found in Appendix B.

Figure 13 - Inlet outer barrel reconstruction, inner surface

Two metallic pieces of debris were embedded in the outer barrel panel near 6:00. The pieces fractured the inner face sheet and core but did not penetrate the outer face sheet. The pieces were documented and removed from the panel. Each piece was subjected to an X-ray Fluorescent (XRF) analysis to determine the gross material type. One piece had coloration and composition consistent with the aluminum fan case isogrid. The other piece had composition consistent with the fan blade titanium alloy with the presence of aluminum and vanadium.

The outer barrel fragments were inspected for fluid ingression using a handheld infrared camera. Two areas had evidence of fluid ingression into the honeycomb, one between 3:00 and 4:00 and one between 8:00 and 9:00.

4.3 Fan Cowl

A majority of the inboard and outboard fan cowl doors were recovered but had fractured into many pieces. The recovered fan cowl door fragments were placed on the floor in a 2-dimensional reconstruction. The items were first reconstructed with the inner surface up and then turned over with the outer surface up. Figure 14 shows a stitched photo of the reconstructed fan cowl inner surface. Large format photos of both the inner surface and outer surface can be found in Appendix B. There was an area on the inboard fan cowl door near the forward edge between 6:30 and 8:30 that was visibly more fragmented than elsewhere. A small section of the leading edge in this area was found embedded in the wing-to-body fairing. The eye bolts on the center two latch keepers on the inboard fan cowl door were fractured. The forward and aft latch keepers were mostly intact. The fan cowl latches on the outboard fan cowl door were mostly intact.

Figure 14 - Fan Cowl reconstruction, inner surface

The fan cowl fragments were inspected for fluid ingression using a handheld infrared camera. One area had evidence of fluid ingression into the honeycomb located on the inboard fan cowl door near 10:30 where the strake was installed.

The fan cowl support beam separated from the engine during the event. The forward attach point was fractured from the inlet attach ring and the three aft attach brackets installed on the fan case hinge beam were fractured. A portion of the inlet aft bulkhead and the fractured portion of the attach ring remained attached to the fan cowl support beam. Small portions of the right and left fan cowl doors remained attached to the forward two hinges on the left and right sides. The aft two fan cowl door hinges on the left and right sides remained on the airplane with small portions of the doors attached. The forward left (#1) hinge pin had visible crankshaft deformation and there was evidence of hinge over travel in both directions. The forward right (#1) hinge pin had very slight deformation and there was evidence of hinge over travel in both directions. The #2 left and right fan cowl hinges had scuffed paint and primer along the lower edges and the hinge pins were free to rotate and easily removed. The #3 and #4 left and right fan cowl hinges were free to rotate and easily removed.

4.4 Thrust Reversers

The left (inboard) and right (outboard) TRs remained installed after the event with the latches closed and latched. There was significant heat and fire damage to both TRs. The technicians reported that the latches felt like they had normal to tighter than normal tension when opening the thrust reverser halves. Detailed photos of the TRs can be found in the Boeing EQA Report in Appendix A.

The Nacelle Drain Access Door normally installed on the inboard TR separated during the event. The door had moderate fire damage. Samples of the sooting were obtained for analysis and a metallurgical analysis was performed on the fractured hinges with results presented in Appendix A.

4.4.1 Outboard TR

The outer V-blade was mostly intact and undamaged but disengaged from the fan case as installed. The V-blade was shifted aft of the associated V-groove between about 12:00 and 6:00 and displaced 1-inch radially outboard between about 12:00 and 3:00. There was a crack in the outer V-blade about 2 inches long near 12:00 that was removed for analysis.

The outboard TR translating sleeve outer skin was missing from about 1:00 to 6:00 from the leading-edge connection with the torque box to the aft edge of the translating sleeve inner cavity. The aftmost 26-inches of the translating sleeve outer skin remained. The upper portion of the outer skin panel between about 12:00 and 1:00 remained attached at the upper end, but the leading edge was detached from the torque box. The translating sleeve within the sleeve cavity was burned and charred with portions of the inner skin and honeycomb completely burned away and the composite material delaminated. The heaviest thermal damage was between about 3:00 and 6:00. All the cascades were missing between about 2:30 and 6:00.

The remaining cascades between about 12:00 and 2:30 were heavily sooted. The CFRP honeycomb sandwich translating sleeve inner skin in the area of the missing cascades was burned and delaminated. The inner skin outer face sheet was missing from about 3:00 to 6:00 in the area of the missing cascades. The translating sleeve inner skin had two burn throughs in the area aft of the cascade vanes between about 2:00 and 6:00. The two holes measured 7-inches circumferentially by 10-inches axially at about 2:30 and 58-inches circumferentially by 10-inches axially between about 3:00 and 5:30. The aft 26-inches of the translating sleeve was intact, sooted, and mostly undamaged except for 12 small impact holes located randomly around the sleeve. The latch beam in the lower bifurcation had significant thermal damage and about 24-inches was missing in the area adjacent to the under cowl drain masts (Figure 15). There was metal slag deposited on the outboard face of the latch beam, the lower surface of the translating sleeve, and on the aft side of the torque box around and forward of the missing area. The metal deposit was sampled and analyzed by the Boeing EQA lab with the results presented in Appendix A. The three thrust reverser hydraulic actuators were intact and appeared mostly undamaged aside from being sooted.

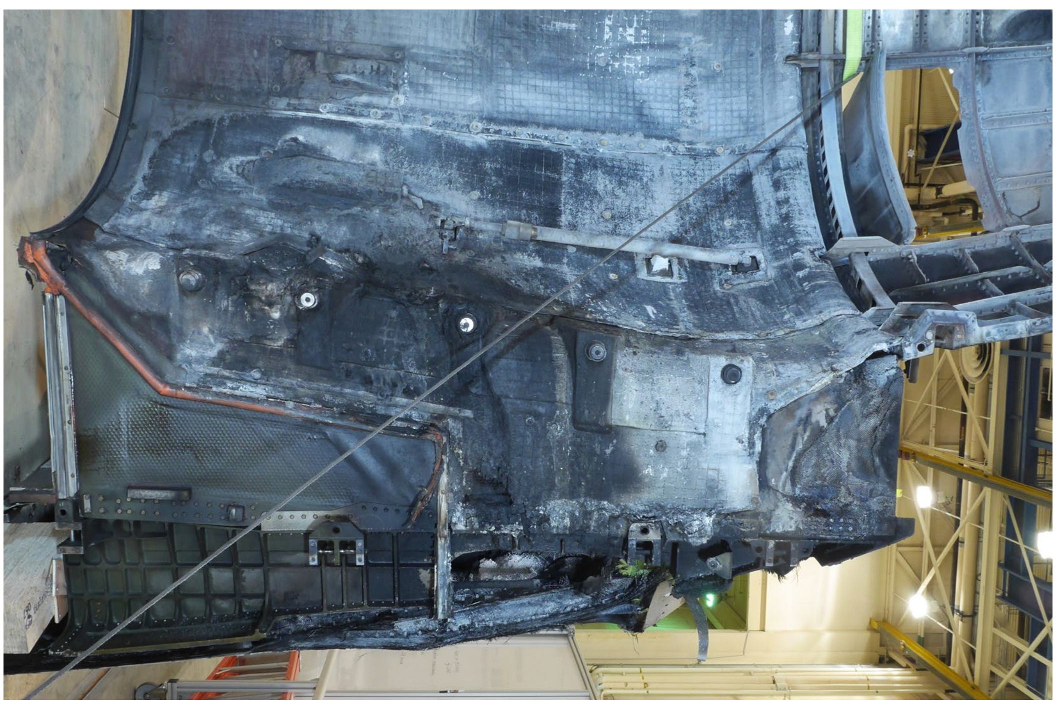

Figure 15 - Outboard thrust reverser lower bifurcation with blankets installed

A portion of the outboard TR translating sleeve outer skin about 52-inches wide circumferentially and 64-inches long axially was recovered separately. The panel was torn at the top edge but intact in the axial direction and corresponded to the lower section of missing translating sleeve outer skin. The panel contained the integrated lower latch beam cover and the lower TR actuator access port but was missing the access port cover assembly. The outer skin surface was sooted with the heaviest sooting being on the forward side. The outer surface had blistered and peeled paint along the forward and top edges. The latch beam cover outer skin was attached but peeled away, exposing the composite core material. The interior surface of the outer skin panel was heavily charred and delaminated. Where exposed, the honeycomb core was charred with sections missing. The outer skin panel lower attach fitting was fractured, and the fracture surface appeared rough and grainy. The access port cover attach flange was intact, with all 15 of the bolt threads and nuts retained. There were 14 bolt heads retained, with one missing.

The outboard aft cowl was intact and undamaged. The upper pressure relief door was in the closed position and the lower pressure relief door was in the open position after the event. There was a light gray colored soot trail aft of the lower relief door along its entire width that was sampled and analyzed.

The outboard TR translating sleeve fan duct side did not have any heat damage or sooting except in the areas of burn throughs. The perforated skin had numerous small tears and punctures. The blocker door struts were intact, still connected, and undamaged except for one that was slightly bowed.

The fan duct side of the inner wall was free of thermal damage and sooting except in the area of the lower bifurcation. The perforated skin had numerous small tears and punctures. The perforated skin had a circumferential crack about 6-inches aft of the leading edge between about 3:00 and 6:00. There were two triangular shaped areas of missing perforated skin aft of the crack centered about 4:00 and 5:00. The lower bifurcation perforated skin had a crack running axially along the bottom from the leading edge to approximately 16-inches forward of the aft edge. The perforated skin had a crack that extended vertically between the inner wall and outer wall in the center of the lower bifurcation that intersected the axial crack. The perforated skin was discolored at the bottom of the bifurcation where the vertical and axial cracks intersected.

The torque box aft bulkhead was exposed due to the missing translating sleeve outer skin panel. The aft bulkhead was sooted along its entire circumference from 12:00 to 6:00 with burn through holes in the lower section between about 3:00 and 6:00. The four separate burn through holes measured 11-inches circumferentially by 5-inches radially centered near 3:30, 28-inches circumferentially by 5-inches radially centered near 4:00, 10-inches circumferentially by 4 inches radially centered near 5:00, and 3-inches circumferentially by 2-inches radially centered near 5:30. The web around the burn throughs was pillowed aft. The forward bulkhead was intact and undamaged. The forward ends of the upper and middle TR hydraulic actuators and the pass-through holes in the forward bulkhead were sooted. The middle and lower actuator proximity flags were in the closed position.

The outboard TR inner wall with thermal blankets installed was sooted around the entire circumference with the heaviest signs of thermal damage being in the lower half. The inner wall thermal blankets were installed and mostly intact. The blanket face sheet was sooted between about 12:00 and 3:00 and was charred between about 3:00 and 6:00. The aft end of the lower bifurcation aft of the thermal blankets was charred and the composite was delaminated. The lower bifurcation fire seal was mostly intact. The aft vertical leg of the lower fire seal was charred, detached, and partially missing. The upper bifurcation fire seal was intact and installed.

The thermal blankets were removed to access the inner wall of the outboard TR. The inner wall was mostly undamaged and had some light sooting. There was an area of charred face sheet and thermal damage just aft of the vertical leg of the fire seal in the lower bifurcation. There was a circumferential crack in the inner wall skin about 6-inches aft of the leading edge between about 2:30 and 6:00.

4.4.2 Inboard TR

The outer V-blade was displaced radially inward through the thrust reverser/fan exit case/interface gap between about 6:00 and 9:00, disengaged radially outward between about 9:00 and 10:00, and engaged between about 10:00 and 12:00 as installed. The V-blade was cracked circumferentially between about 9:00 and 12:00 and between about 7:00 and 9:00. Portions of these fractures were removed for analysis.

The inboard TR translating sleeve was missing the outer skin from about 11:00 to 6:00 from the leading-edge connection with the torque box to the aft edge of the translating sleeve inner cavity. The aftmost 26-inches of the outer skin of the translating sleeve remained. The upper portion of the outer skin panel between 11:00 and 12:00 remained attached at the aft and upper edges, but the leading edge was detached, separated, and peeled back from the torque box. The translating sleeve within the sleeve cavity was burned and charred with the inner skin and honeycomb completely burned away and the composite material delaminated. The heaviest thermal damage was between about 6:00 and 8:00. Almost all the cascades were missing between 6:00 and 12:00. Some small sections of the cascades at the forward edges remained. The CFRP honeycomb sandwich inner skin in the area of the missing cascade vanes was burned and delaminated. The outer face sheet was missing except for a section about 2-feet by 2-feet near 9:00 under the middle TR hydraulic actuator. The translating sleeve inner skin had three burn throughs in the area aft of the cascade vanes between about 7:00 and 11:30. The three holes measured 43-inches circumferentially by 7-inches axially between about 7:00 and 8:30, 48-inches circumferentially by 13-inches axially between about 9:00 and 11:00, and 14-inches circumferentially by 22-inches axially near 11:30. The aft 26-inches of the translating sleeve was intact, sooted, and mostly undamaged except for 6 small impact holes located randomly around the sleeve. The latch beam in the lower bifurcation had significant thermal damage and about 24-inches was missing in the area adjacent to the under cowl drain masts (Figure 16). There was metal slag deposited on the outboard face of the latch beam, the lower surface of the translating sleeve, and on the aft side of the torque box around and forward of the missing area.

The metal deposit was sampled and analyzed by the Boeing EQA lab with the results presented in Appendix A. The three thrust reverser hydraulic actuators were intact and appeared undamaged aside from being sooted.

Figure 16 - Inboard thrust reverser lower bifurcation with blankets installed

The inboard aft cowl was intact and undamaged. The two pressure relief doors were in the open position after the event. There was a light gray colored soot trail aft of the lower relief door along its entire width that was sampled and analyzed.

The inboard translating sleeve fan duct side did not have any heat damage or sooting except for the areas of burn throughs. The perforated skin had numerous small tears and punctures. The blocker door struts were intact and still connected, but two struts were bowed, and one strut was bent.

The fan duct side of the inner wall was mostly free of thermal damage and sooting with two exceptions. The perforated skin had numerous small tears and punctures. There was an area of thermal damage and sooting on the inner wall that extended about 16-inches forward of the aft end and was centered near 7:00. The aft half of the lower bifurcation was charred and sooted and there was an area of burn through damage in the drain mast area that measured 6-inches radially and 24-inches axially. The lower bifurcation perforated skin had an axial crack at the top that extended 36-inches aft of the leading edge. The upper bifurcation perforated skin had a hole about 7-inches aft of the leading edge that measured 4-inches by 4-inches. There was an axial crack in the perforated skin between the hole and the leading edge.

The torque box aft bulkhead was exposed due to the missing translating sleeve outer skin panel. The aft bulkhead was sooted along its entire circumference from 6:00 to 12:00 with burn through holes in the lower section between about 6:00 and 7:30. The four separate burn through holes measured approximately 10-inches circumferentially by 3-inches radially centered near 6:30, 12-inches circumferentially by 4-inches radially centered near 7:00, 3-inches circumferentially by 2-inches radially centered near 7:30, and 4-inches circumferentially by 4-inches radially centered near 7:30. The web around the burn throughs was pillowed aft. There was an area of imminent burn through in the aft bulkhead between 9:30 and 10:30 with missing paint, no sooting, and aft pillowing of the web. The forward bulkhead was intact and undamaged. The forward ends of the upper and middle TR hydraulic actuators and the pass-through holes in the forward bulkhead were sooted. The middle actuator proximity flag and handle assembly was missing from the actuator housing. The lower actuator position flag was in the closed position.

The inboard TR inner wall with thermal blankets installed was sooted around the entire circumference with the heaviest signs of thermal damage being in the lower half. The inner wall thermal blankets were installed and mostly intact. The blanket face sheet was sooted between about 9:00 and 12:00 and charred between about 6:00 and 9:00. The thermal blankets installed near the aft end of the inner wall near 7:00 were burned through the face sheet material. The thermal blankets installed on the lower bifurcation were burned through exposing the composite material underneath, which was also charred and delaminated. The lower bifurcation had a burn-through hole consistent with the damage noted on the fan duct side. The aft end of the lower bifurcation aft of the thermal blankets was charred and the composite was delaminated. The lower bifurcation fire seal was mostly intact. The aft vertical leg of the lower fire seal and portions of the horizontal leg were charred. The upper bifurcation fire seal was intact and installed.

The thermal blankets were removed to access the inner wall of the inboard TR. The inner wall was mostly undamaged and had some light sooting. There was an area of charred face sheet and thermal damage near the aft edge of the inner wall near 7:00. The inner wall in the lower bifurcation area had thermal damage and charring along its length and was burned through as described earlier.

5.0 Systems Examination

An examination of certain relevant airplane systems was performed by the group.

5.1 Flight Recorders

The VOX RCDR, FLT RCDR DC, and FLT RCDR AC circuit breakers were pulled and locked out after the airplane landed. The Honeywell HFR5-V Cockpit Voice Recorder, S/N CVR-01238, and the Honeywell 980-4700-042 Flight Data Recorder, S/N 6543, were removed from the racks above the aft galley and shipped to the NTSB laboratory in Washington, DC, on United flight 1767 on February 21, 2021. The QAR disk was also removed and shipped to the NTSB lab along with the recorders.

5.2 Engine Fire Switch

The engine fire switches and engine fire bottle discharge lights are located on the engine fire panel (P8) in the flight deck (Figure 17). The engine fire panel has a fire switch for each engine and a discharge light for each fire bottle. The engine fire switches will give an indication of an engine fire, shut down the engine if activated, isolate the engine from the airplane systems if activated, and controls the engine fire extinguishing system. A solenoid locks the fire switch so it cannot be pulled accidently. When the sensors detect an engine fire, the fire warning light illuminates in the switch handle and the solenoid energizes to release the switch. When the fire switch is pulled, the push-pull switch contacts operate electrical circuits which shut down the engine and isolate it from the airplane systems. With the switch pulled, it can be rotated left or right to a mechanical stop at one of two discharge positions. The rotary switch contacts close and operate the fire extinguishing system.

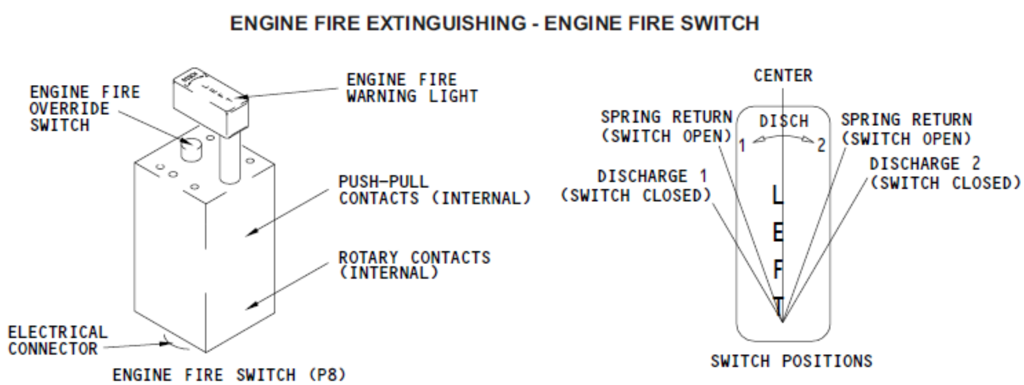

Figure 17 - Engine Fire Switch description and operation drawing

When the fire switch is pulled, the engine is isolated as the system closes the fuel spar valve, de-energizes the engine fuel metering unit (FMU) cutoff solenoid, closes the engine driven hydraulic pump supply shutoff valve, depressurizes the engine driven hydraulic pump valve, closes the pressure regulator and shutoff valve, removes power from the thrust reverser isolation valve, trips the generator field, and trips the backup generator field.

The right engine fire switch was observed in the flight deck P8 panel to be pulled and turned to the DISCH 1 position after the event (Figure 18). Both left and right engine fuel control switches were in the CUTOFF position. Both fire bottle discharge lights were observed to be illuminated when power was applied to the airplane.

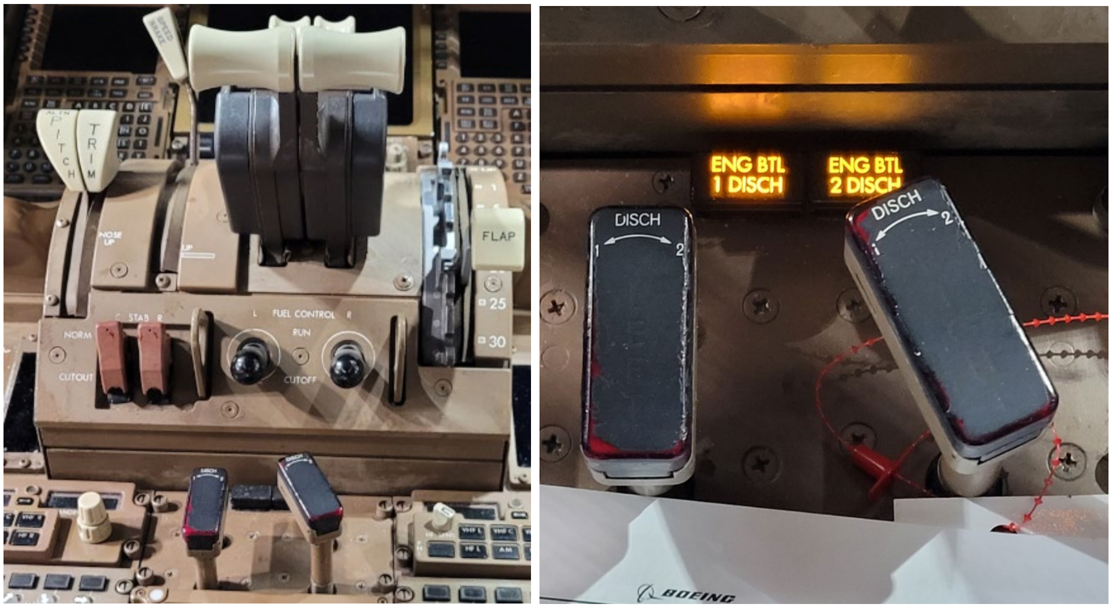

Figure 18 - Engine Fire Switches on incident airplane

5.3 Engine Driven Pump (EDP) Supply Shutoff Valve

The EDP supply shutoff valve stops hydraulic supply from the reservoir to the EDP when the engine fire switch is in the up position. The supply shutoff valve is a two-position valve operated by a 28v dc motor. A position indicator shows the position of the valve. The supply shutoff valve is located in the left side of each engine aft strut near the hydraulic reservoir (Figure 19).

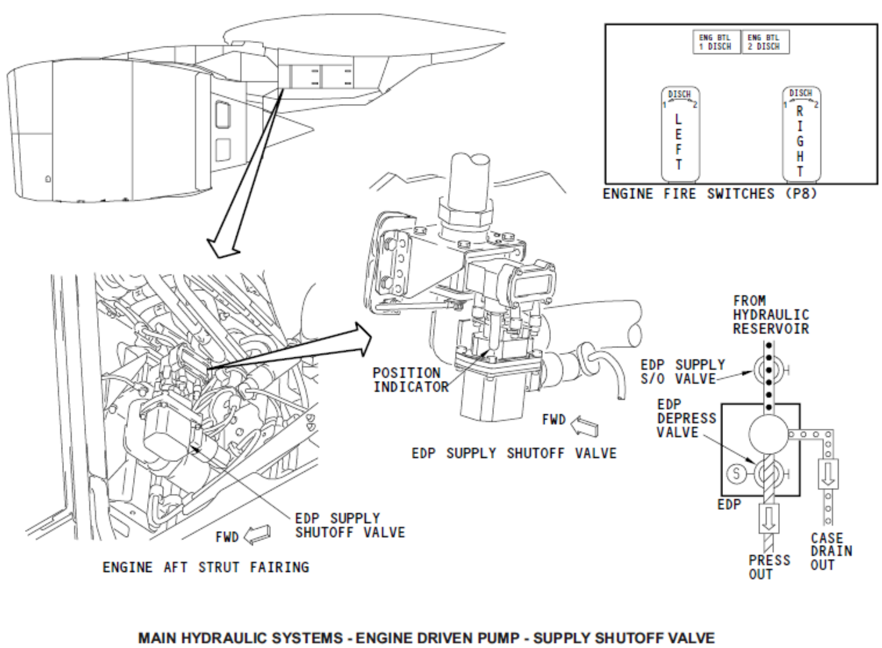

Figure 19 - Engine Driven Pump Supply Shutoff Valve description and operation drawing

The right EDP Supply Shutoff Valve on the event airplane was observed to be in the OPEN position with the right engine fire switch pulled (Figure 20).

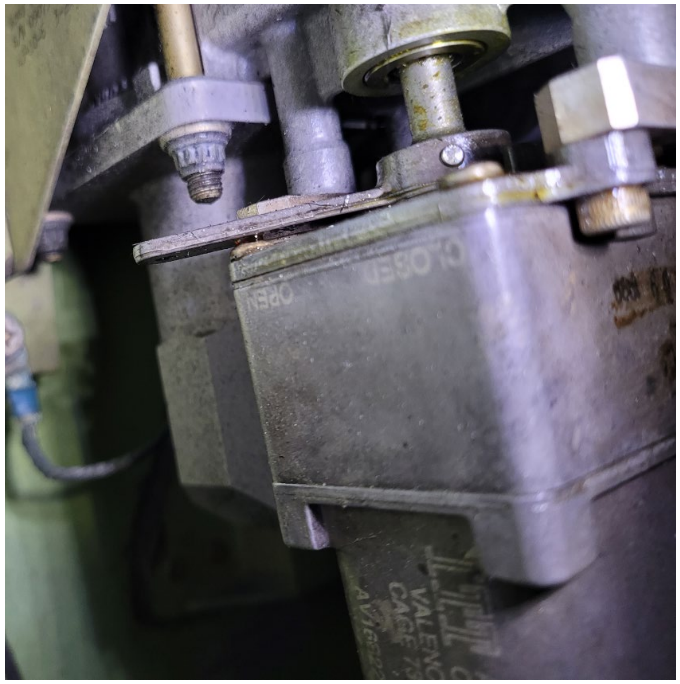

Figure 20 - Right Engine Driven Pump Supply Shutoff Valve on incident airplane

Connector and wiring checks were accomplished with the assistance of United Airlines mechanics as follows to verify continuity and functionality of the right fire switch and airplane wiring leading to the right EDP Supply Shutoff Valve with the right fire switch in the pulled position. Right fire switch and wire continuity was confirmed with no shorts or anomalies found.

From	To	Result	WDM Reference
Fire Switch S26201 Pin 18	Fire Switch S26201 Pin 36	Continuity	29-11-21
Fire Switch S26201 Pin 18	Fire Switch S26201 Pin 37	Open	29-11-21
Fire Switch S26201 Pin 36	Ground	Open	29-11-21
Connector DS26201 Pin 24	Ground	Continuity	26-20-11
Connector DS26201 Pin 36	Connector DV29202 Pin 3	Continuity	29-11-21
Connector DV29202 Pin 8	Ground	Continuity	29-11-21
Connector DV29202 Pin 3	Ground (with manipulation of harness	Open	29-11-21

The circuit breaker (C29602 R ENG-EDP SUPPLY VALVE) associated with this valve was found to be in the closed position on the P11 overhead panel in the flight deck (Figure 21).

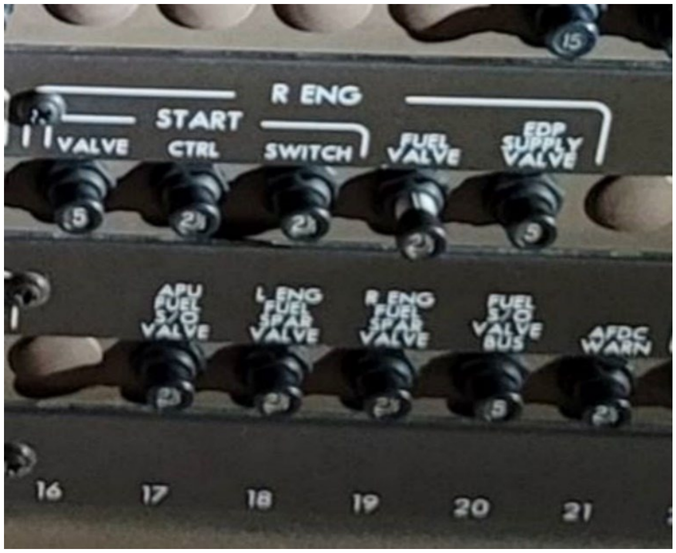

Figure 21 - Right EDP Supply Valve circuit breaker on incident airplane

With power applied to the airplane and circuit breaker C29602 R ENG-EDP SUPPLY VALVE in the closed position a voltage check of pin 3 on connector DV29202 showed a voltage ~28V and a voltage check of pin 2 on connector DV29202 showed a voltage ~0V. When circuit breaker C29602 was opened pin 3 on connector DV29202 showed a voltage ~0V.

The ATA 29 maintenance page accessible from the primary display system in the flight deck showed blank for the R S/O VLV indication with circuit breaker C29602 in both the closed and open positions (Figure 22). Maintenance message 29-10750 “Supply shutoff valve (EDP R) is not in commanded position” was active in the central maintenance computer existing faults (Figure 23). The valve, P/N S271W741-22, S/N T50052, was removed and retained for further examination and testing. See the Systems Group Chairman’s Factual Report for the details of the valve testing.

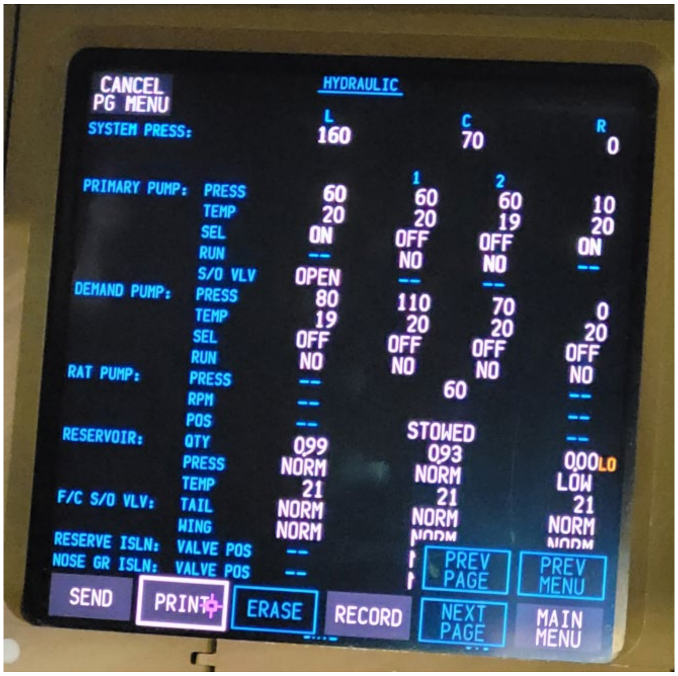

Figure 22 - ATA 29 Hydraulic System Maintenance page on incident airplane

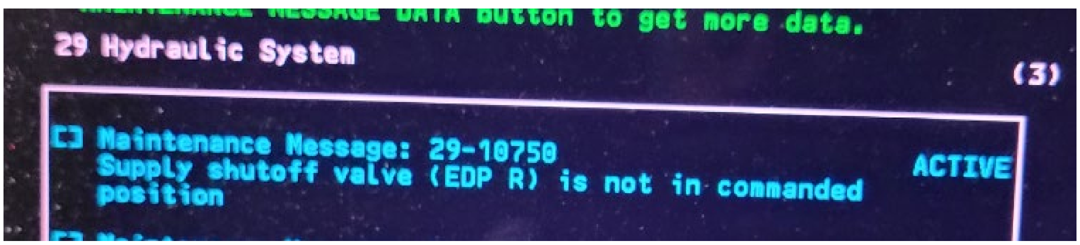

Figure 23 - Central Maintenance Computer message on incident airplane

5.4 Fuel Spar Valve

The fuel control switches control relays to open and close the fuel spar valves. The engine fire switch must be down to permit power to get through the relays to the spar valve. The fire switch also has a direct circuit to the spar valve. This closes the valve when the fire switch is pulled. The right fuel spar valve was found to be in the CLOSED position with the right engine fire switch pulled on the incident airplane (Figure 24).

Figure 24 - Right fuel spar valve on incident airplane

5.5 EDP Depressurization Solenoid Valve

The engine main gearbox turns the EDP when the engine turns. A solenoid valve in each EDP controls the pressurization and depressurization of the pump. When the depressurization solenoid valve is not energized, pump pressure output goes to the hydraulic system. When the pump is turned off, the depressurization solenoid valve gets electrical power. The solenoid valve blocks the pump output flow when the pump is turned off. This blockage removes the pressure from the EDP. The right EDP depressurization solenoid valve on the right engine was damaged by fire and the wiring leading to the electrical connector was found to be broken (Figure 25).

Figure 25 - Solenoid valve on engine driven pump on incident airplane

The circuit breaker (C29600 R D-PRESS VLV SPLY) for this valve was found to be tripped on the P210 panel in the EE bay (Figure 26).

Figure 26 - Depressurization solenoid valve circuit breaker on incident airplane

5.6 Fuel Metering Unit (FMU) Cutoff Solenoid

The fire switch permits FMU control power to go to the engine relay unit (ERU) when the switch is in its normal position (in). If the switch is pulled out, power connects directly to the FMU cutoff solenoid to stop metered fuel flow to the nozzles. The right engine FMU and associated wire harnesses were damaged by fire (Figure 27).

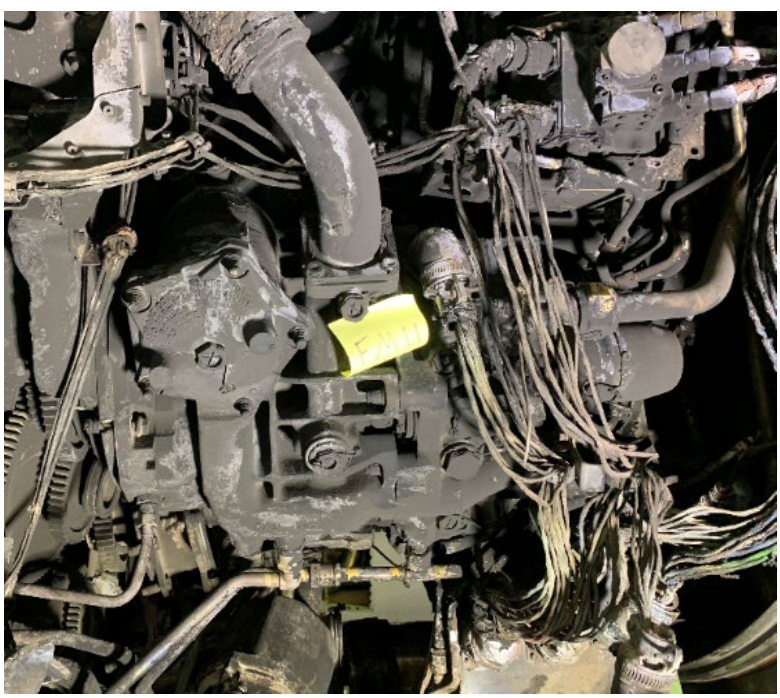

Figure 27 - Right engine fuel metering unit on incident airplane

The circuit breaker (C76600 R ENG FUEL VALVE) associated to this FMU function was found to be tripped on the P11 overhead panel in the flight deck.

5.7 Hydraulic Reservoir

The hydraulic reservoirs supply hydraulic fluid under pressure to the hydraulic pumps. The reservoirs also get return hydraulic fluid from the airplane systems that use hydraulic power. The left and right system reservoirs are the same. Each has a total volume of 12.6 gallons (47.8 liters) and normally contains 7.4 gallons (28 liters) of hydraulic fluid. The hydraulic system reservoirs are in the engine aft strut for each engine. The reservoirs are pressurized by the bleed air system for positive supply to the pumps. Each reservoir also has a standpipe. For the left and right reservoirs there are 2 gallons (7.6 liters) of fluid below the standpipe. The EDPs get a fluid supply from the standpipe. A port at the bottom of the reservoir supplies fluid to the AC electric motor pumps (ACMPs).

No fluid was observed in the upper and lower sight glasses of the right hydraulic system reservoir indicating the reservoir needs to be refilled. The reservoir quantity gage in the ground service bay also indicated a quantity of 0 when the right hydraulic system was selected (Figure 28). The ATA 29 maintenance page showed a right hydraulic reservoir quantity of 0.00 and an amber LO indication (Figure 22) when examined about 24 hours after the incident. The right hydraulic system continued to operate throughout the flight. A reservoir quantity about 90-95% was recorded at the time of the engine failure and a quantity about 65% was recorded at the time of the landing.

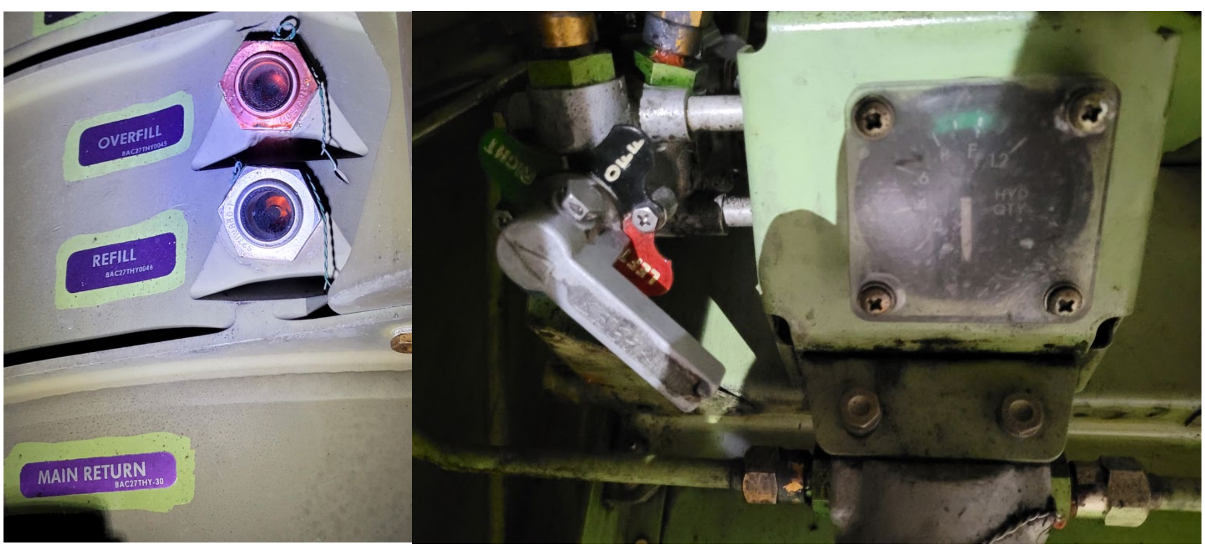

Figure 28 - Right hydraulic system reservoir (left) and quantity gage (right) ion incident airplane

6.0 Federal Aviation Administration (FAA) Information

The engine on the incident airplane is certified under Title 14 *Code of Federal Regulations* (14 CFR) Part 33 while the inlet, fan cowls, and thrust reversers are certified under 14 CFR Part 25.

After the event, the FAA issued Emergency Airworthiness Directive (AD) 2021-05-51 on February 23, 2021, requiring an inspection of the fan blades on all Pratt & Whitney PW4074, PW4074D, PW4077, PW4077D, PW4084D, PW4090, and PW4090-3 engines before further flight. The FAA followed up with AD 2021-05-51, effective on March 24, 2021, that clarified some items in the emergency AD but retained the same inspection requirements for the fan blades. These actions effectively grounded all PW4000-112 powered 777-200 and 777-300 airplanes.

The separation of the inlet and fan cowls from the airplane poses a risk of catastrophic outcome if the parts strike the empennage. The potential for inlet and/or fan cowl separation due to a fan blade failure event subsequently led the FAA to propose additional airworthiness directives to require corrective actions for the inlet and fan cowls before further flight

Under 14 CFR Section 25.901(c) "For each powerplant and auxiliary power unit installation, it must be established that no single failure or malfunction or probable combination of failures will jeopardize the safe operation of the airplane except that the failure of structural elements need not be considered if the probability of such failure is extremely remote." The FAA position is that the airplanes do not meet this requirement as currently designed due to the departure of the inlet and/or fan cowls resulting from a fan blade failure.

The shutoff valve for the hydraulic fluid supply to the engine driven hydraulic pump on the event engine should have closed when the engine fire handle was pulled, but the valve was found in the open position. While this did not significantly contribute to the fire in this case, it revealed that the airplane did not provide valve position indication required by FAA regulations. The lack of position indication to the flightcrew for the engine driven hydraulic pump shutoff valves poses a risk to the airplane in the event of a fire. Based on the assumption that all the flammable fluid sources to the engine are shut off by the fire handle, the flightcrew may delay landing in order to dump fuel to avoid an overweight landing or they may divert to a more convenient airfield unaware the flammable fluid sources to the fire zone may not be fully shut off. The FAA has proposed an airworthiness directive requiring a periodic check of valve function as an interim corrective action.

Under 14 CFR Section 25.1141 (f)(1) "For powerplant valve controls located in the flight deck there must be a means for the flightcrew to select each intended position or function of the valve; (2)(i) and to indicate to the flightcrew the selected position of the valve; (2)(ii) and when the valve has not responded as intended to the selected position or function." The FAA position is that the lack of indication to the flightcrew of the engine driven hydraulic pump shutoff valve position does not meet these requirements.

The fire that penetrated the TR inner wall and in the translating sleeves poses a risk to damaging the airplane from parts departing or the spread of the fire to other areas. The fire in the engine core compartment spread to the TR translating sleeves after failure of the firewall in the lower bifurcation area. One possible cause of the firewall failure in the lower bifurcation of the inboard TR was that debris from the fan blade failure struck the firewall puncturing it allowing pressurized air from the fan duct to intensify the fire. Another possible cause was that the firewall burned through due to the intensity of the undercowl fire. The FAA has proposed an airworthiness directive requiring the installation of debris shields in the lower bifurcation of the TRs as an interim corrective action to address the unsafe condition which would address either possible cause for the fire in the TRs.

Under 14 CFR Section 25.1191(a) "the combustion, turbine, and tailpipe sections of turbine engines, must be isolated from the rest of the airplane by firewalls, shrouds, or equivalent means." 14 CFR Section 25.1191(b) states that "Each firewall and shroud must be (1) Fireproof; (2) Constructed so that no hazardous quantity of air, fluid, or flame can pass from the compartment to other parts of the airplane; (3) Constructed so that each opening is sealed with close fitting fireproof grommets, bushings, or firewall fittings; and (4) Protected against corrosion." The portion of the TR structure that forms the engine core compartment and the lower bifurcation functions as a firewall and is required to be fireproof as listed above. The FAA defines fireproof in 14 CFR 1.1 "With respect to materials and parts used to confine fire in a designated fire zone, means the capacity to withstand at least as well as steel in dimensions appropriate for the purpose for which they are used, the heat produced when there is a severe fire of extended duration in that zone." The FAA accepted method of compliance contained in Advisory Circular 20-135 states "the material or part will perform this function under conditions likely to occur in such zones and will withstand a 2000°F flame (±150°F) for 15 minutes minimum." The FAA position is that the airplane as designed does not meet this requirement if the lower bifurcation firewall failed due to the fire. If the firewall failure occurred due to impact from fan blade debris, the FAA position is that the airplane design does not meet the requirements of 14 CFR Section 25.901(c) previously stated.

In August 2021, Boeing petitioned the FAA for a time-limited, partial exemption from Title 14 CFR Sections 25.901(c), 25.361(b) Special Condition 25-ANM-78 Item No.6, 29.903(c), and Part 25 Appendix K, K25.1.1 for fan blade failure. The exemption requested was for 5 years so that Boeing could incorporate design changes to the inlet, fan cowls, and TRs sequentially as needed, with the interim changes not yet bringing the design into compliance with those regulations. Approval of the exemption would allow the PW4000 powered 777 airplanes to resume flying while the various modifications necessary to fully comply with the regulations are developed and incorporated.

The FAA Administrator said in September 2021 the agency would require that the cowling and other structure around the engines on PW4000 powered 777 airplanes be strengthened to prevent similar incidents. In testimony to a U.S. House committee, the administrator said that Boeing and the FAA were working together to ensure "the structure around the engine, the cowling and the inlet area, does not damage the aircraft structure" due to a fan blade out event.

In January 2022, the FAA published two Notices of Proposed Rulemaking (NPRMs) stating their intention to release one AD requiring that all engine inlets on PW4000 powered 777 airplanes be modified before further flight to withstand fan blade failure event loads. The second AD would require the installation of debris shields on the lower bifurcation area of the TRs, require an inspection of the fan cowl doors for moisture ingression, and require a functional check of the left and right hydraulic pump shutoff valves before further flight.

The FAA granted the time-limited, partial exemption to The Boeing Company from 14 CFR 25.901(c) and 25.903(c), Appendix K25.1.1 to part 25, and item 6 of Special Conditions 25-ANM-78 as they pertain to non-compliance with the "no single failure" provision of § 25.901(c) and affected structural changes for failure of an engine fan blade, to allow approval of the incremental design changes for the propulsion installation on Model 777-200 and 777-300 airplanes equipped with PW4000-112 engines on March 4, 2022.

7.0 Other Fan Blade Out (FBO) Events

The group researched the results of a certification FBO test and examined several other FBO events that have occurred on the PW4000 powered 777-200 airplanes.

7.1 Certification FBO Test

In 1994 Pratt & Whitney performed a FBO test for certification of the PW4000 engine. The certification test was designed to demonstrate containment and safe shutdown of an engine after the intentional fracture of a fan blade at redline speed. The main focus of the test was on meeting applicable 14 CFR Part 33 engine certification requirements including the response of the engine during an FBO event. Data from the test was also used in meeting 14 CFR Part 25 airplane certification requirements including the response of engine mounts and TRs during an FBO event. Typically, an inlet and TRs are installed for the test, but fan cowls are not installed so that containment case behavior can be captured. The inlet installed for the test does not need to be a production part, but it must be representative of a production part. The engine mounts must be equivalent to production mounts. For the certification test, the inlet installed was of a different design than the production inlet and included an aluminum aft bulkhead, instead of the CFRP aft bulkhead, and a fiberglass and aluminum outer barrel, instead of the CFRP outer barrel installed on production units. During the test the intended blade was released near the blade root and a portion of the following blade also fractured and separated. The containment case successfully retained the blade fragments. Within about two revolutions after blade release, fan blade segments traveled forward of the fan case, sliced through and separated about half of the inlet inner barrel, and one blade fragment exited the inlet inner and outer barrels (Figure 29). After design changes and analysis to show the observed failure effects relevant to 14 CFR Part 33 compliance were addressed to the satisfaction of the FAA Engine Certification Office, the engine was approved without repeating the fan blade out test. However, at least some of the other failure effects observed in that test that were relevant to the airplane compliance with 14 CFR Part 25 do not appear to have been brought to the attention of the FAA Seattle Aircraft Certification Office. At the time, there was no anticipated transmittal of impact displacements from the Kevlar® wrapped containment case to the inlet while the loss of the inner barrel load path from a single fan blade fragment was anticipated.

Figure 29 - Certification FBO test engine and inlet damage

7.2 UAL 2 FBO Event

On April 7, 2010, United Airlines flight 2, a PW4084 equipped Boeing 777-222 airplane, N210UA, suffered a fan blade failure on the left engine during the takeoff roll from Honolulu, HI. The takeoff was aborted, and the airplane returned to the gate. Examination of the airplane showed that a portion of one fan blade separated during the event. The containment case successfully retained the blade fragments, but some fragments traveled forward of the fan case. The inlet inner barrel had damage around about half the circumference and was penetrated in 3 locations. The inlet outer barrel was penetrated in two places in the lower outboard quadrant (Figure 30). Several fan exit guide vanes also separated during the event, damaging the thrust reversers. The inlet and fan cowls did not separate from the airplane during the event. The NTSB was notified of the event and gathered data but did not perform an investigation due to the lack of substantial damage received by the airplane.

Figure 30 - UAL 2 inlet outer barrel punctures

7.3 UAL 1175 FBO Event (DCA18IA092[2])

On February 13, 2018, United Airlines flight 1175, a PW4077 equipped Boeing 777-222 airplane, N773UA, suffered a fan blade failure on the right engine over the Pacific Ocean enroute to Honolulu, HI. The airplane descended and landed safely. Examination of the airplane showed that portions of two fan blades separated during the event damaging the inlet, fan cowls, and TRs. The fan cowls, most of the fan cowl support beam, inlet lip skin, inlet forward bulkhead, half of the inlet aft bulkhead, inlet outer barrel and a majority of the inlet inner barrel separated during the event and were not recovered. The containment case successfully retained the blade fragments though the outer Kevlar® environmental wrap was torn and deflected outward about 2.5 inches and some blade fragments traveled forward of the fan case. The inlet attach ring remained attached, the inner barrel remained attached between 9:00 and 12:00, and the inlet aft bulkhead remained attached between 9:00 and 3:00 (Figure

[2] The full NTSB report and public docket can be found on www.ntsb.gov under this accident number.

31). Examination of the fan case showed three distinct blade trajectory paths that spiraled forward across the A-flange. There were also two small punctures in the right fuselage below the window belt.

Figure 31 - UAL 1175 left engine, fan cowl, and inlet damage

7.4 JAL 904 FBO Event

On December 4, 2020, Japan Airlines flight 904, a PW4074 equipped Boeing 777-289 airplane, JA8978, suffered a fan blade failure on the left engine during climb out from Naha, Okinawa, Japan. The airplane returned to the airport and landed safely. The incident is being investigated by the Japan Transport Safety Board (JTSB) and the NTSB is participating in the investigation as the accredited representative for the state of design and manufacture under ICAO Annex 13. Examination of the airplane showed that portions of two fan blades separated during the event damaging the inlet, fan cowls, and TRs. Portions of the left fan cowl separated during the event but most of the right fan cowl, the fan cowl support beam, and inlet remained attached. The containment case successfully retained the blade fragments, but some fragments traveled forward of the fan case. The inlet was mostly intact after the event (Figure 32). There was evidence of 5 distinct blade path trajectories that damaged the inlet inner barrel and there was significant fracturing of the inner barrel. Two of these fragments penetrated the inner barrel but did not penetrate the outer barrel. The inlet aft bulkhead was fractured from 9:45 to 11:45 but remained attached. The damaged inlet, fan cowls, and TRs were shipped to the Boeing EQA lab for further examination.

Figure 32 - JAL 904 left engine and inlet damage

7.5 Boeing Failure Analysis

Following the UAL 1175 FBO event Boeing began work on a dynamic simulation model to examine the failure scenarios of the inlet and fan cowl structures in comparison to the data collected during the certification FBO tests. The analysis utilized finite element modeling and progressive failure analysis with the physical evidence collected from the known FBO events at the time. The analysis expanded the damage simulation beyond the typical 20-30 milliseconds following a FBO to capture the progressive failure effects up to 2 seconds following the FBO event. The loading conditions and simulations were broken down into 4 phases. Phase 1, the impact phase lasts for about the first 0.02 seconds (0.0 to 0.02 sec) and includes the impact loads from the blade hitting the fan case and the resulting displacement wave traveling around the case. Blade fragments spiraling forward into the inlet occurs during this phase. Phase 2, the Surge phase occurs between 0.02-0.20 sec and is when the engine experiences a stall with the compressed gasses expanding forward and aft causing a pressure wave. Phase 3, the rundown phase occurs between 0.02-2.0 sec and includes the fan imbalance loads as it decelerates. Phase 4, the windmilling phase occurs from 2.0 sec to landing and includes the much lower imbalance loads generated from a windmilling engine.

The UAL 1175 event analysis indicated the separation of the inlet and fan cowl would occur within the first 2 seconds following the fan blade separation if the fan blades damage the inlet sufficiently. During the UAL 1175 FBO event, three fan blade fragments traveled forward of the A-flange and penetrated both the inner and outer face sheets of the inner barrel, and two of those fragments penetrated the outer barrel. The post-event simulation studies indicated that a combination of the aft bulkhead damage and the inner barrel damage consistent with the observed aircraft incident damage would cause portions of the inlet to separate. During the UAL 2 FBO event, three fragments penetrated the inner barrel and two of these penetrated the outer barrel. The analysis indicated that the amount of inner barrel damage would not cause the inlet to separate.

During a FBO event, the released fan blade has a significant amount of centrifugal and circumferential energy. The fan case and Kevlar® containment belt were designed to absorb the energy and prevent the fan blade fragments from exiting radially through the fan case. The fan case and Kevlar® containment belt will deflect outward as they absorb the energy, but it was not anticipated that there would be significant impact deflections at A-flange. Pratt & Whitney estimated the radial displacement at the A-flange would be about 0.1 inch in certification documents. Working with Pratt & Whitney, Boeing examined the displacement wave deflections in their dynamic simulation model. The simulation studies identified a critical threshold for the displacement of the inlet aft bulkhead caused by the displacement wave and failure of the CFRP bulkhead. The certification test simulation studies with the aluminum aft bulkhead predicted a bulkhead displacement of 0.47 inch with localized yielding, but without a failure of the inlet structure or the inlet to fan case interface. The UAL 1175 incident analysis predicted a displacement of 0.55 inch and delamination of the installed CFRP bulkhead face sheets, which exceeded the face sheet laminate rupture strain in compression leading to the failure of the inlet aft bulkhead. The aluminum structure in the certification inlet had the ability to yield and absorb the same amount of energy and redistribute the FBO loads between the fan case and the inlet without causing failure of the inlet aft bulkhead. The UAL 2 event analysis predicted a displacement of 0.46 inch and compression buckling of the CFRP aft bulkhead but no failure.

Boeing records indicated that evidence of moisture ingression had been found on multiple other 777 fan cowls, and although varied in extent and location, was on some occasions reported in the area of the latches on the lower fan cowl panels and the area of the hinge attachments on the upper fan cowl panels. Such moisture ingression would degrade the strength of the cowls. Although the UAL 1175 event fan cowls were not recovered to verify the presence of moisture ingression, it is possible that this type of degradation existed and contributed to the fan cowl separation.

The analysis indicated that large fan cowl deflections can be induced following the departure of portions of the inlet. The forward portion of the fan cowl would no longer be capable of remaining engaged after portions of the inlet and the aft bulkhead departed the airplane. As a result, the fan cowl would no longer maintain its shape when subjected to the engine imbalance loads and buffeting from air loads. Large panel deflections could induce internal stresses in the fan cowls that exceed those observed during certification and exceed the capability of the fan cowl panels and latches.

Simulation work by Boeing for the JAL 904 FBO event and the current incident event is ongoing.

8.0 Boeing Design and Development Work

Following the UAL 1175 FBO event and the dynamic simulation work, Boeing began development of modifications to the inlet and fan cowls that would prevent the loss of both during an FBO event. The necessity of these modifications was further reinforced after the JAL 904 FBO event and the current UAL event when the airplanes were grounded. The documented damage from the two most recent events changed the focus of the modifications somewhat due to the observed damage patterns.

Pending the partial exemption request, Boeing intends to provide instructions for modification of the inlets as a first and most critical step. The improved inlet will incorporate several changes. The fasteners used to attach the inlet to the engine at the A-flange will be replaced with increased strength fasteners. The fasteners used to attach the aft edge of the outer barrel panels to the aft bulkhead will be changed from countersunk head fasteners to pan head fasteners and additional fasteners will be installed on the panels at the upper quadrant of the inlet to prevent pull through failures along the interface. Metallic doublers will be added to the aft side of the aft bulkhead web to add strength and prevent the aft bulkhead failure from the FBO displacement wave. Metallic ballistic shielding panels will be added between the inner barrel and outer barrel to prevent fan blade fragments from penetrating both barrels of the inlet. In conjunction with the modification, an inspection and repair of the outer barrel panels for moisture ingression will be recommended.

Boeing intends to provide modification instructions for the TRs on the affected airplanes. Metallic debris shields will be installed on the inner surface of the fan duct inner wall behind the thermal blankets in the lower bifurcation area on both the inboard and outboard TRs. The debris shields will also add some level of fire protection in this area.

An inspection and repair of the fan cowls for moisture ingression will also be recommended in another service action for the affected airplanes. Ongoing work is examining possible redesign or modification of the fan cowls to prevent failure and separation during an FBO event.

E. LIST OF ATTACHMENTS

Appendix A – Boeing EQA Report AS13328
Appendix B – Stitched Photos of Inlet, Fan Cowls, and Fan Case

Submitted by:

Clinton R. Crookshanks
Aerospace Engineer (Structures)