



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

November 12, 2018

AIRWORTHINESS GROUP CHAIRMAN'S FACTUAL REPORT

NTSB No: DCA16FA217

A. INCIDENT

Location: Pascagoula, Mississippi

Date: August 27, 2016

Time: 0922 central daylight time

Aircraft: Southwest Airlines, Boeing 737-7H4, Registration N766SW, Flight No. 3472

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TABLE OF ACRONYMS

3D	3 DIMENSIONAL	IGV	INLET GUIDE VANE
AC	ADVISORY CIRCULAR	In	INDIUM
AD	AIRWORTHINESS DIRECTIVE	IR	INSPECTION REPORT
AGB	ACCESSORY GEARBOX	JAA	JOINT AVIATION AUTHORITIES
Al	ALUMINUM	JAR	JOINT AVIATION REQUIREMENTS
ALF	AFT LOOKING FORWARD	LCF	LOW-CYCLE FATIGUE
AMM	AIRCRAFT MAINTENANCE MANUAL	LE	LEADING EDGE
AOW	ALL OPERATORS WIRE	LLP	LIFE-LIMITED PART
APB	AUXILIARY POWER BREAKER	LPC	LOW PRESSURE COMPRESSOR
APU	AUXILIARY POWER UNIT	LPT	LOW PRESSURE TURBINE
ARINC	AERONAUTICAL RADIO INCORPORATED	m/s	METERS PER SECOND
ASM	AMERICAN SOCIETY OF METALS	MCD	MAGNETIC CHIP DETECTOR
ASTM	AMERICAN SOCIETY FOR TESTING AND MATERIALS	MG	MAGNESIUM
ATP	ACCEPTANCE TEST PROCEDURE	mm	MILLIMETERS
AU	AIRCRAFT UNIT (SCALAR)	MPa	MEGAPASCALS
BPCU	BUS POWER CONTROL UNIT	msec	MILLISECONDS
BS	BODY STATION	MW	MOMENT-WEIGHT
BSI	BORESCOPE INSPECTION	N1	FAN/LOW ROTOR SPEED IN PERCENT rpm
CD	CONSIGNE DE NAVIGABILITE	N2	HIGH ROTOR SPEED IN PERCENT rpm
CDU	CONTROL DISPLAY UNIT	NA	NOT APPLICABLE
CESM	COMMERCIAL ENGINE SERVICE MEMORANDUM	NDI	NON-DESTRUCTIVE INSPECTION
CFM	CFM INTERNATIONAL	Ni	NICKEL
CIS	CYCLES-IN-SERVICE	NPRM	NOTICE OF PROPOSED RULEMAKING
CMM	COMPONENT MAINTENANCE MANUAL	NTSB	NATIONAL TRANSPORTATION SAFETY BOARD
CP	CONSUMABLE PRODUCT	NVM	NON-VOLATILE MEMORY
CSLSV	CYCLES SINCE LAST SHOP VISIT	OEM	ORIGINAL EQUIPMENT MANUFACTURER
CSN	CYCLES SINCE NEW	OGV	OUTLET GUIDE VANE
CTAI	COWL THERMAL ANTI-ICE	PAD	PROPOSAL TO ISSUE AN AIRWORTHINESS DIRECTIVE
CU	COCKPIT UNIT (SCALAR)	PDP	POWER DISTRIBUTION PANEL
Cu	COPPER	PS	PRESSURE SIDE
DGAC	DIRECTION GÉNÉRALE DE L'AVIATION	PSI	POUNDS PER SQUARE INCH
EAD	EMERGENCY AIRWORTHINESS DIRECTIVE	PTI	PROPULSION TECHNOLOGIES INTERNATIONAL
EASA	EUROPEAN AVIATION SAFETY AGENCY	RII	REQUIRED INSPECTION ITEM
ECI	EDDY CURRENT INSPECTION	rpm	REVOLUTION PER MINUTE
ECU	ELECTRONIC CONTROL UNIT	RS	RESIDUAL STRESS
EDS	ENERGY DISPERSIVE X-RAY SPECTROSCOPY	RSS	REPAIR SOURCE SUBSTITUTIONAL

EQA	EQUIPMENT QUALITY ANALYSIS	SB	SERVICE BULLETIN
ESM	ENGINE SHOP MANUAL	SC	SPECIAL CONDITION
ESN	ENGINE SERIAL NUMBER	SEM	SCANNING ELECTRON MICROSCOPE
EVM	ENGINE VIBRATION MONITORING	SNECMA	SOCIÉTÉ NATIONALE D'ETUDE ET DE CONSTRUCTION DE MOTEURS D'AVIATION
FAA	FEDERAL AVIATION ADMINISTRATION	SPM	STANDARD PRACTICES MANUAL
FBO	FAN-BLADE-OUT	SRM	STRUCTURAL REPAIR MANUAL
FC	FLIGHT CYCLES	SS	SUCTION SIDE
FDR	FLIGHT DATA RECORDER	STA	STATION
FEM	FINITE ELEMENT MODEL	SWA	SOUTHWEST AIRLINES
FL	FLIGHT LEVEL	SWAPA	SOUTHWEST AIRLINES PILOT'S ASSOCIATION
FLA	FORWARD LOOKING AFT	TAI	THERMAL ANTI-ICE
FMC	FLIGHT MANAGEMENT COMPUTER	TC	TYPE CERTIFICATE
FPI	FLUORESCENT PENETRANT INSPECTION	TCDS	TYPE CERTIFICATE DATA SHEET
fps	FEET PER SECOND	TE	TRAILING EDGE
GCB	GENERATOR CONTROL BREAKER	TGB	TRANSFER GEARBOX
GCU	GENERATOR CONTROL UNIT	Ti	TITANIUM
GE	GENERAL ELECTRIC	TR	THRUST REVERSER
GEOWS	GE ON-WING SUPPORT	TRF	TURBINE REAR FRAME
GMT	GREENWICH MEAN TIME	TSLSV	TIME SINCE LAST SHOP VISIT
GP	GROUND POWER	TSN	TIME SINCE NEW
HMU	HYDROMECHANICAL UNIT	UI	ULTRASONIC INSPECTION
HPC	HIGH PRESSURE COMPRESSOR	UTAS	UNITED TECHNOLOGIES AEROSPACE SYSTEMS
HPSOV	HIGH PRESSURE SHUTOFF VALVE	V	VOLT
HPT	HIGH PRESSURE TURBINE	V	VANADIUM
HV	HARDNESS VALUE	WO	WORK ORDER
IDG	INTEGRATED DRIVE GENERATOR	WOW	WEIGHT-ON-WHEELS

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C. SUMMARY

On August 27, 2016, about 0922 central daylight time, a Boeing 737-7H4 airplane, registration number N766SW, powered by two CFM International (CFM) CFM56-7B24 turbofan engines, operated by Southwest Airlines (SWA) as flight number 3472, experienced a left engine (No. 1) fan blade and inlet separation and cabin depressurization event while passing through flight level (FL) 310. The oxygen masks were deployed, pilots declared an emergency, an in-flight shutdown of the left engine was performed, and the airplane was diverted to Pensacola International Airport (PNS) where an uneventful single engine landing was performed. The incident flight was conducted under provisions of 14 Code of Federal Regulations (CFR) Part 121 instrument flight rules (IFR) flight from New Orleans, Louisiana (MSY) to Orlando, Florida (MCO) and was the first flight of the day for the airplane and crew. There were 99 passengers, 1 lap child, and 5 crewmembers on board the incident flight; no injuries were reported.

The Airworthiness Group convened at the PNS on August 28, 2016, to perform a detailed examination of the event airplane and engine, and the group completed its work on September 1, 2016. Examination of the airplane revealed: (1) a 16-inch x 5-inch through-puncture on the left side of the fuselage just below passenger windows 11 and 12 with no corresponding damage on the interior liner, (2) the left winglet front spar was partially severed, (3) numerous impacts along the left side of the fuselage, left wing, and left horizontal stabilizer, and (4) the left engine inlet was severely damaged and almost completely missing. The left winglet front spar along with a piece of the left wing leading edge (LE) No. 1 slat that exhibited impact damage were sent to National Transportation Safety Board (NTSB) Materials Laboratory in Washington DC for analysis. What remained of the inlet was sent to Boeing in Renton, Washington and then to United Technologies Aerospace Systems (UTAS), the inlet manufacturer, in Chula Vista, California, for further evaluation and debris trajectories exit assessment.

Examination of the No. 1 engine revealed: (1) a single fan blade had fractured below the platform and the entire airfoil and shank exited the engine and inlet and were not recovered, (2) the dovetail of the fractured blade remained secure within the fan disk, (3) the remainder of the fan blades remained in place and exhibited varying degrees of hard impact airfoil damage, (4) the fan case exhibited numerous impact marks and gouges, distortion, and loss of the all abrasion resistant seal material but remained intact¹ and exhibited no through-holes penetrations or uncontainments, and (5) no signs of an undercowl fire were observed. After the on-site examination of the engine was complete, all the fan blades and their associated hardware were sent to the NTSB Laboratory for initial examination with some hardware subsequently sent to Safran and other testing facilities for additional specialized testing. The fan case was sent to CFM in France for further evaluation and a debris trajectory exit assessment.

The flight data recorder (FDR) was removed from the airplane and shipped to the NTSB's Vehicle Recorders Division in Washington DC for readout. Readout of the FDR revealed that it had stopped recording data during the diversion to PNS due to an electrical power loss. Based on this information, the Airworthiness Group conducted a series of on-scene airplane electrical checks to determine potential causes for power interruption to the FDR. The various on-scene test scenarios were unable to duplicate the observed in-flight electrical anomaly that caused the FDR to lose power. Subsequent off-wing component tests were also unable to positively identify a specific part or component that could have contributed to the electrical power loss. Prior to the aircraft returning to service, the components removed for additional testing were replaced, and the on-scene test scenarios were accomplished again on the airplane. The additional tests were also unable to duplicate the observed in-flight electrical anomaly that

¹ For the purposes of this report, the word intact is used to indicate that the part/component/assembly is complete (not lacking any part that properly belongs to it) but may be damaged or distorted.

caused the FDR to lose power. The exact cause of the electrical power interruption to the FDR was indeterminate. SWA performed a scheduled FDR download and parameter validation test on February 28, 2017 with no faults found, consistent with a properly functioning FDR.

Metallurgical examination of the event fan blade set found that the fractured fan blade, identified as fan blade No. 23, had fractured due to a fatigue crack that initiated on the convex side of the blade root dovetail centered about 2.1-inches from the front face and near the outboard edge where the copper-nickel-indium (Cu-Ni-In) coating (outboard is toward the blade tip and inboard is toward the blade root). The fractured blade was originally manufactured as part number (PN) 340-001-022-0 and was later modified to PN 340-001-027-0. The Cu-Ni-In coating on the dovetail was missing and spalled in several locations on the convex side; most notably in an area adjacent to the fracture origin. The coating thickness was measured optically adjacent to the fracture origin and at several locations along the length of the dovetail. Variations in coating thickness were found; however, on average, the coating thickness was within the required specification but at the upper limit. Optically looking at the fracture surface revealed several distinct crack arrest lines. Using a scanning electron microscope (SEM), striations were consistent with low cycle fatigue (LCF) growth. The NTSB and CFM conducted striation density estimates along the crack propagation path and estimated that about 15,000 flight cycles had accumulated between crack initiation and fracture and that the striation count between arrest lines matched well with the fan blade dovetail relubrication intervals for the event engine. On average SWA relubricated the fan blade dovetails about every 1,300 cycles between the time the fan blade set had been installed on the event engine in 2007 to when the event occurred. At relubrication, the fan blade coating is inspected and if deemed serviceable a lubricant is applied to the dovetail and shims to reduce friction between the surfaces before the fan blade hardware is reinstalled.

Additional examination of fan blade No. 23 confirmed the material composition and microstructure to be consistent with the specified titanium alloy (Ti-6Al-4V) and no surface anomalies or and no material defects were found. The inner surface of fan blade No. 23 shim appeared as a mirror image of the fan blade No. 23 dovetail. At locations where the coating was missing on the fan blade dovetail, no corresponding impression (wear) mark was observed on the shim. At locations where the coating was intact on the fan dovetail, corresponding impression marks consistent with dovetail coating spalling was present. This is consistent with the dovetail coating spall being present when the shim was reinstalled.

Six additional fan blades from the event fan blade set were found to have dovetail surface cracks; the cracks were not detected visually with the Cu-Ni-In coating still in place but were detected using alternate inspection methods. The dovetail cracks were detected using an ultrasonic inspection technique while the coating still in place or after the coating was removed using either traditional fluorescent penetrant or eddy current non-destructive inspection (NDI) techniques. In each case, the dovetails exhibited a primary crack and a network of microcracks located approximately 2-inches aft of the forward face. The cracked dovetails were laboratory fractured and examination of the fracture surfaces confirmed the primary cracks had features consistent with LCF. On six of the seven cracked fan blades (this includes the fractured blade No. 23), spalled coating was observed just below the crack initiation site. In each case the amount of spalling would have allowed the fan blades to continue-in-service but with a reduced inspection interval; 50 flight cycles or 75 flight hours whichever comes first according to the engine manual.

The Cu-Ni-In coating thickness on sixteen of the event fan blade dovetails plus a non-event exemplar fan blade was scanned using a 3-dimensional (3D) laser coordinate scanner before and after the coating was stripped to determine the overall coating thickness. Nearly all the event fan blade dovetails

had localized areas of coating thickness values that exceeded specification on the concave side and five event blades exceeded the specification on the convex side of the blade dovetail as well. Variability in coating thickness was also observed in nearly all the event blades.

Residual stress (RS) measurements were conducted on the dovetails of a mix of event fan blades with and without a primary crack as well as on non-event exemplar fan blades for comparison. In general, the RS were less compressive than desired on all the event fan blades when compared to a reference profile and this was especially true near the crack initiation site. The RS for fan blade No. 23 dovetail was almost zero near the surface and had consistently the least-compressive RS values versus depth when compared to the other event and exemplar fan blades. As part of the fan blade repair, shot-peening of the entire fan blade, including the dovetail, root, and airfoil, is used to achieve a compressive RS level and it effectively serves to provide improved fatigue strength of the material and relieve surface tensile stresses that can aid in fatigue crack initiation. Shot-peening is a cold work process in which compressive stresses are induced into the surface of the part by shot impingement at specified velocity, angle, and duration.

Metallurgical examination of the left winglet front spar revealed elemental peaks of titanium (Ti), which is consistent with the fan blade material and not an element of the material of the winglet front spar. The same analysis was performed on the left wing LE No. 1 slat and no elemental Ti peaks were noted.

The event fan and booster assembly configuration was such that, based on the fan disk configuration, a fan blade spacer and shim were required. The shim limits the radial movement of the fan blade in the fan disk dovetail slot and reduces possible damage to the forward face of the fan blades by the fan disk retention lugs. All the fan blades were originally manufactured as part number (PN) 340-001-022-0 and were subsequently modified to PN 340-001-027-0. Regardless of the fan blade PN, CFM classified all the fan blade as non-life-limited parts even though all the fan blades are serialized. There is no Chapter 5 life-limit, mandatory inspection interval, or tracking requirement. The entire event fan blade set remained together as a set throughout its entire service life including overhaul and repair; thus, the entire fan blade set was tracked from first engine installation until the event. At the time of the fan blade failure, the entire fan blade set had accumulated 62,851 hours and 38,152 cycles since new and 30,237 hours and 17,956 cycles since overhaul and repair. Since the event engine was installed on the event airplane in October 2007, the fan blades were inspected and relubricated on average every 1,274 cycles and, based on the entire relube history of the fan blade, this inspection interval seemed to be consistent throughout the life of this fan blade set. The last dovetail relubrication before the event occurred 1,396 cycles prior. According to the Boeing 737-600/700/800/900 aircraft maintenance manual, fan blades are to be removed and inspected. If the fan blades are serviceable, the dovetail is relubricated before reinstallation. According to CFM and Boeing guidance, the relubrication of the fan blade dovetail occurs every 3,000 cycles or 5,000 hours whichever comes first. Fan blade lubrication intervals may be adjusted by each operator based on its own aircraft maintenance program. In the case of SWA, the fan blade dovetail relubrication was performed more frequently to match other maintenance tasks.

The event fan blade set was repaired in 2001 and again in 2007; both times at a GKN Aerospace/Chem-Tronics facility. The Airworthiness Group convened at the GKN facility in El Cajon, California to review the 2001 and 2007 repair records and to observe the Cu-Ni-In coating application process with special emphasis on those operations that may affect or account for the low RS observed during the metallurgical examination. The Cu-Ni-In coating repair was originally in the CFM56-7B engine shop manual and the 2001 GKN repair was performed using the engine shop manual instructions that applicable at that time. Subsequently, the Cu-Ni-In coating became a source approved repair that required a technical licensee agreement and repair source substantiation (RSS) prior to performing the repair. This means that the repair source (an approved Part 145 repair/overhaul shop) must prove that

they can perform the repair in accordance with the repair specifications; GKN received their RRS in June 2003. Review of the Cu-Ni-In coating process and procedures did not reveal any substantive deviations from the specified procedures.

As it relates to fan-blade-out (FBO) containment requirements for the Boeing 737 airplane with a CFM56-7B engine installed, CFM is responsible for adhering to the applicable Federal Aviation Regulation (FAR) Part 33 AIRWORTHINESS STANDARDS for aircraft engines while Boeing is responsible for the applicable FAR Part 25 TRANSPORT CATEGORY AIRWORTHINESS STANDARDS. Boeing created the design specifications for the inlet and Rohr Industries, Inc. designed, manufactured, and delivered the inlet to Boeing. Subsequently, Rohr Industries, Inc. was acquired and is currently referred to as UTAS.

To meet the applicable FAR Part 33 requirements, and to obtain data that would be used to meet FAR Part 25 certification requirements, CFM performed a total of eight development FBO rig tests and an engine FBO containment certification test, followed by a second engine FBO certification test specifically to demonstrate the fan blade axial retention capability. Rig tests are specialized hardware specific tests to validate the capability and performance of a specific of the test hardware prior to an actual engine FBO certification test. Each test had a specific set of objectives and based on those objectives various configurations of the fan blade, fan case, and inlet were used. The purpose of these FBO tests were to: (1) understand the behavior of the fan blade after separation (kinematics), (2) test the fan case containment capability (radial containment - kinetic energy of each impact), (3) define the loads from the ensuing imbalance, (4) calculate the speed and weight of ejected fragments (forward containment), if any, and (5) demonstrate the proposed production hardware configuration. This included gathering data on the fan blade fragments relative size, weight, speed, and forward exit trajectory (the helical angle of travel from the center of rotation as the fragment departs the fan case A1 flange and crosses over into the inlet). Based on the FBO rig tests prior to the engine FBO containment certification test, areas of improvement were identified. CFM redesigned the fan case to strengthen it to prevent through-case penetrations and changed the fan blade axial retention feature while Boeing instructed UTAS to design a containment shield to stop an FBO fragment from penetrating the outer barrel of the inlet and to modify the inlet attach flange. This first Boeing production representative inlet used during one of the FBO rig tests did not have a containment doubler, as it was not known at that time whether inlet containment was needed.

The CFM56-7B engine FBO containment certification test was conducted in April 1996 with a newly modified fan case, fan blade, and redesigned representative inlet configuration with a containment shield (doubler). The results of the engine FBO containment certification test showed that the engine met the FAR Part 33 requirements in that the fan case sustained the initial FBO impact with no penetrations, the engine did not catch fire, did not fail its mounting hardware, and was able to be shutdown. However, along with the initial fan blade release, additional fan blades failed their axial retention feature and were released penetrating through the fan case and inlet. To address the fan blade axial retention feature deficiency, CFM conducted additional FBO rig tests and an engine FBO certification test specifically to demonstrate and validate the new fan blade axial retention design. CFM and Boeing jointly correlated the engine FBO containment certification test results, the transient forcing function that defined the various FBO related events and loads, such as initial fan impact loads, surge loads, rundown loads, torque loads, imbalance loads, and windmilling loads for Boeing and UTAS to use in their analysis of the inlet and its capability. Based upon CFM input that the damage to the inlet and containment shield was due to the subsequent release of additional fan blades, Boeing and UTAS concluded that despite the amount of inlet damage, the inlet remained attached to the engine, so it withstood the initial FBO impact, the surge pressure loads, and the rundown loads. Boeing and UTAS agreed a redesign was not necessary and additional testing was not required.

The Boeing inlet design philosophy was, and remains currently, that the failure of an engine fan blade and subsequent damage should not cause separation of the inlet or separation of components from the inlet cowl that would be a hazard to the aircraft. This post FBO phase is commonly referred to as the 'Fly-Home' phase which accounts for the effect of the engine failure engine has on the airframe, systems, and occupants on board and the ability for a continued safe flight and landing. To meet the 'Fly-Home' intent, Boeing developed inlet design requirements and final damage tolerances based in part based on CFM FBO testing and in part by analysis and provided them to UTAS for them to assess the capability of the inlet. Based on the information provided by CFM and Boeing, UTAS concluded that there was positive margin for the expected post-FBO inlet damage and windmilling that the inlet would be expected to remain attached to successfully complete the 'Fly Home' requirement, which includes a successful safe landing. Simply put, based on this analysis, Boeing and UTAS determined that the failure of a fan blade and subsequent damage, based on the results of the FAR Part 33 engine FBO containment certification test, would not cause a hazard to the aircraft.

CFM and Boeing have implemented corrective actions to improve and enhance the current overhaul and on-wing fan blade inspection requirements based on this event. Previously during overhaul, the fan blades were inspected for cracks using a fluorescent penetrant inspection (FPI) technique. In November 2016, CFM added a fan blade dovetail eddy current inspection (ECI) procedure in the engine shop manual at overhaul. The concave and convex sides of the fan blade dovetail are scanned after the Cu-Ni-In coating has been removed to detect discontinuities such as cracks at the surface or near-surface of the part at a higher sensitivity level than the previous FPI method. Laboratory analysis of the event fan blade set revealed that several additional fan blades were cracked; however, the cracks were not visibly apparent through the coating. Based on this finding, the visual inspection of the fan blade dovetail for cracks during relubrication was deemed insufficient. CFM developed an ultrasonic inspection (UI) technique that could be performed with the coating still on the dovetail; hence it could be utilized during the fan blade field inspection and dovetail relubrication. The UI is a manual scan on both the concave and convex sides of the dovetail with a transducer placed at the bottom of the blade root to minimize coating interference and allows inspection of coated blades, which is not possible with the ECI technique.

Based on the SWA flight 3472 fan blade separation and inlet departure incident and before the SWA flight 1380 fan blade separation and inlet departure accident that occurred on April 17, 2018 in Philadelphia, CFM issued two different service bulletins (SB) related to on-wing ultrasonic fan blade dovetail inspections; SBs CFM56-7B 72-1019 (March 24, 2017 basic, revision 1 issued June 24, 2017) and 72-1024 (July 28, 2017). The SBs required to have a one-time ultrasonic dovetail inspection; the inspection requirements vary amongst the SBs and are based in part by the fan blade PN and the number of flight cycles the fan blade or engine on which it was installed has accumulated. On March 26, 2018, European Aviation Safety Agency (EASA) issued Airworthiness Directive (AD) 2018-0071 requiring a one-time ultrasonic inspection of dovetail as recommended in SBs 72-1019 and 72-1024. In August 2017, the Federal Aviation Administration (FAA) issued a Notice of Proposed Rulemaking (NPRM) to propose a mandatory inspection of fan blades in accordance with the SBs but had not issued an AD mandating any inspection prior to April 20, 2018. The fan blade set installed on the engine involved in the SWA flight 1380 fan blade separation and inlet departure accident that occurred on April 17, 2018 was not in any inspection population identified by SB 72-1019, SB 72-1024, or the FAA NPRM issued in August 2017.

Three days after the SWA 1380 accident, April 20, 2018, CFM issued SB CFM56-7B 72-1033 recommending an initial and repetitive ultrasonic fan blade dovetail inspection for all PNs and prioritized the inspection of fan blades that had accumulated high flight cycles or those installed on high flight cycle engines. That same day both the FAA and EASA issued Emergency ADs 2018-09-51 and 2018-0093-E respectively mandating that all fan blades with greater than 30,000 flight cycles (if known) or installed on

engines with greater than 30,000 flight cycles (if the fan blade flight cycles are unknown) have the dovetail ultrasonically inspected within 20 days. According to CFM, there were approximately 640 engines worldwide that met or exceed 30,000 cycles since new threshold and as of May 10, 2018, all 640 engines had been inspected. CFM issued additional revisions to SB 72-1033 to accelerate the inspection of all fan blades and to reduce the UI on-wing inspection interval and the FAA and EASA followed with ADs. As of this report the most current ADs are FAA 2018-18-01 and EASA AD 2018-0211 mandating the revisions to SB 72-1033.

In total, as of this report, there are an estimated 356,000 CFM56-7B fan blades and 14,659 CFM56-7B engines delivered. According to CFM, as of August 22, 2018, all CFM56-7B fan blades targeted by the various ADs have been inspected, either using the on-wing UI or at shop visit using ECI; or cleared through paperwork evidence that the fan blades had less than 20,000 cycles; essentially all fan blades have complied with SB 72-1033, FAA AD 2018-10-11, and EASA AD 2018-019. According to SWA, as of May 18, 2018, all their CFM56-7B fan blades with greater than 3,000 cycles since new had been inspected using either UI or ECI.

D. DETAILS OF THE INVESTIGATION

1.0 ENGINE AND AIRPLANE INFORMATION

1.1 ENGINE DESCRIPTION

The CFM56-7B24 is a high bypass, dual-rotor, axial flow turbofan engine. A single-stage high pressure turbine (HPT) drives the nine-stage high pressure compressor (HPC). The integrated fan and low pressure compressor (LPC, sometimes referred to as the booster) are driven by a four-stage low pressure turbine (LPT) (**FIGURE 1**). The annular designed combustion chamber increases the HPC discharge air velocity to drive the high and low pressure turbines. An accessory drive system provides drive requirements for engine mounted aircraft accessories. The engine consists of three major assemblies: fan, core engine, and LPT. It also includes the following components which are removed or installed at engine level: two modules (accessory gearbox (AGB) and transfer gearbox (TGB)), spinner front cone, LPT shaft plug and coupling nut, aft rotating air/oil separator, and oil inlet cover.

According to the engine's FAA Type Certificate Data Sheet (TCDS) No. E00056EN, Revision 10, dated August 9, 2016, the CFM56-7B24 has a maximum takeoff thrust rating of 24,200 pounds (10,765 dekanewton (daN)) flat-rated² to 86°F (30°C) and a maximum continuous thrust rating of 22,800 pounds (10,142 daN) flat-rated to 77°F (25°C).

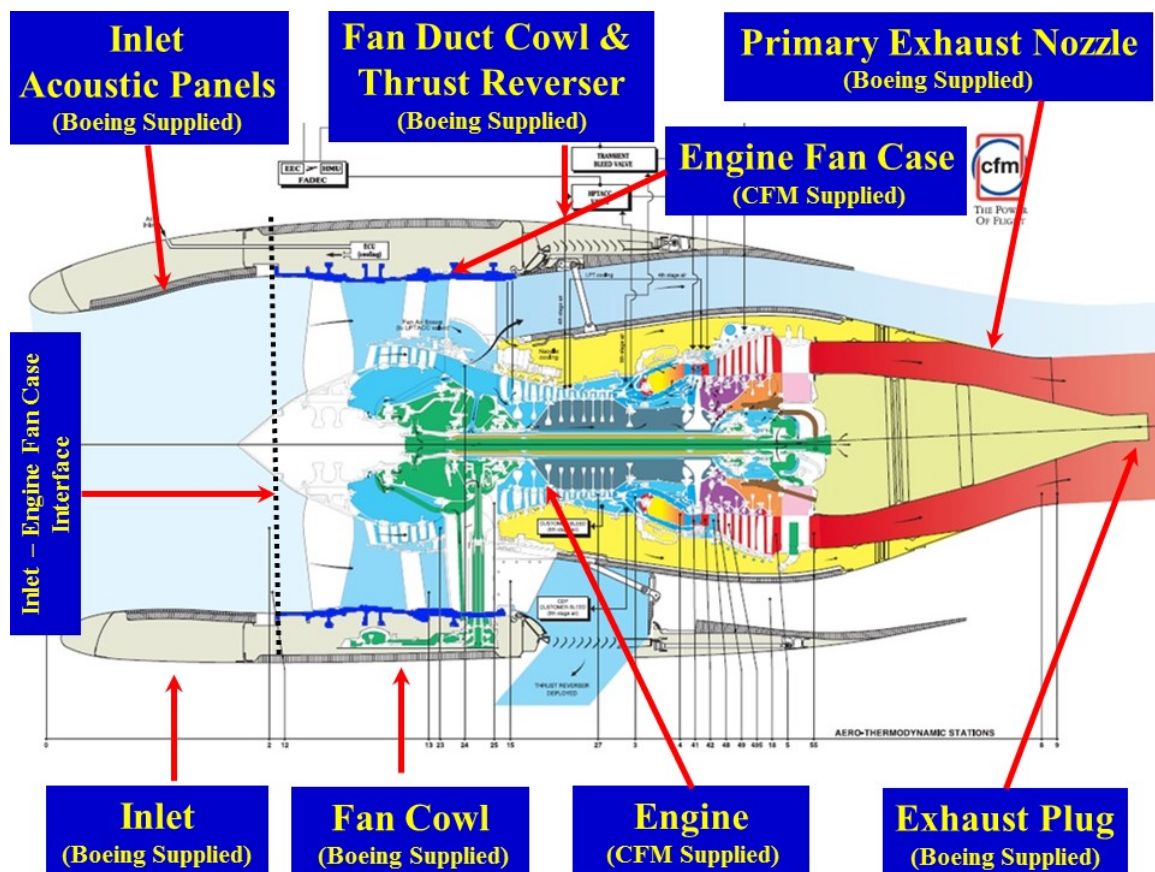


FIGURE 1: ENGINE AND NACELLE CROSS-SECTION SCHEMATIC

FIGURE COURTESY OF CFM

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

The CFM56 engine is a dual certificated engine – certificated by the United States (USA) FAA and French Direction Générale de L'Aviation Civile (DGAC). EASA took over responsibility for certification of the CFM engines in 2004. The FAA issued the type certificate (TC) for the CFM56-7B24 on December 17, 1996 and the certificate basis was 14 CFR Part 33 effective February 1, 1965 with Amendments 33-1 through 33-15.

Since the CFM56 engine is a dual certificated engine, the certification basis was a Joint Aviation Requirements (JAR) JAR-E Change 8 (04 May 1990) required by the DGAC (French Direction Générale de l'Aviation Civile) were also complied with. The DGAC originally certificated the engine model under DGAC certificate de Type Moteur M21 superseded by the EASA TC number EASA.E.004 since 2006 and the FAA validated the product under TC No. E00056EN. According to TCDS No. E00056EN, engines that are produced by General Electric (GE) are identified by an even number engine serial number (ESN) prefix (i.e., “874”, “876”, “890”, “892”, or “654”) and engines that are produced by Safran Aircraft Engines are identified by odd number prefix; therefore, the event ESN 874112 was produced by GE.³

CFM International is a partnership between GE in the USA and Safran Aircraft Engines (formerly Snecma of France). CFM is not an acronym; however, the company (CFM International) and product line (CFM56) receive their names by a combination of the two parent companies' commercial engine designations: GE's CF6 and Snecma's M56. Snecma changed its name to Safran Aircraft Engines, a subsidiary of the Safran Group, in May 2016. The division of labor is such that Safran Aircraft Engines is responsible for the fan and LPT modules while GE is responsible for the remainder of the engine – HPC, combustion, and HPT. For brevity, Safran Aircraft Engines will be referred to as Safran for the remainder of this report.

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

The fan blades were numbered sequentially in the counterclockwise direction forward looking aft (FLA) in accordance with blade numbering instructions provided in the Boeing 737-600/700/800/900 Aircraft Maintenance Manual (AMM) where the No. 1 blade is located immediately above an offset hole on the fan disk bolt flange.

³ TCDS No. E00055EN applies to CFM56-7B engine models produced by GE in the USA under license from CFM International whereas TCDS No. E00056EN applies to CFM56-7B engine models produced by Safran Aircraft Engines in France under license from CFM International. The most current version of TCDS No. E00056EN is version 6 dated June 14, 2006.

1.2 EVENT ENGINE HISTORY

The left engine installed on the incident airplane N766SW, was a CFM56-7B24 turbofan engine, ESN 874112 (**PHOTO 1**). According to SWA, at the time of the fan blade separation, ESN 874112 had accumulated 60,790 hours time since new (TSN); 36,329 cycles since new (CSN); 30,237 hours time since last shop visit (TSLSV), and 17,956 cycles since last shop visit (CSLSV). The last shop visit for the engine occurred in August 2007 at the GE Celma shop located in Petrópolis, Brazil. After completion of the engine repair, it was shipped back to SWA where it was installed on the accident airplane in engine position No. 1 (left) on October 17, 2007 where it remained until the event.



PHOTO 1: LEFT ENGINE DATA PLATE

1.3 FAN ROTOR ASSEMBLY CONFIGURATION DESCRIPTION

The fan and booster assembly is comprised of the spinner cone, fan disk, fan blades, booster rotor, booster vane assemblies, and small associated hardware. The front and aft spinner cones are made of an aluminum (Al) alloy. The aft flange of the aft spinner cone is bolted to the fan disk and is part of the fan blade retention system. Twenty-four (24) Ti alloy fan blades are installed into the fan disk and are individually retained. To limit fan blade radial movement, a spacer is placed under each blade root⁴; the spacer is retained axially by the retention ring (See **FIGURE 31**). To limit fan blade axial movement, the blade dovetail contacts the spacer forward rabbet and the aft face of the retaining ring. The fan disk is bolted to the booster by 24 bolts. See **FIGURES 2** and **14** for additional fan configuration details.

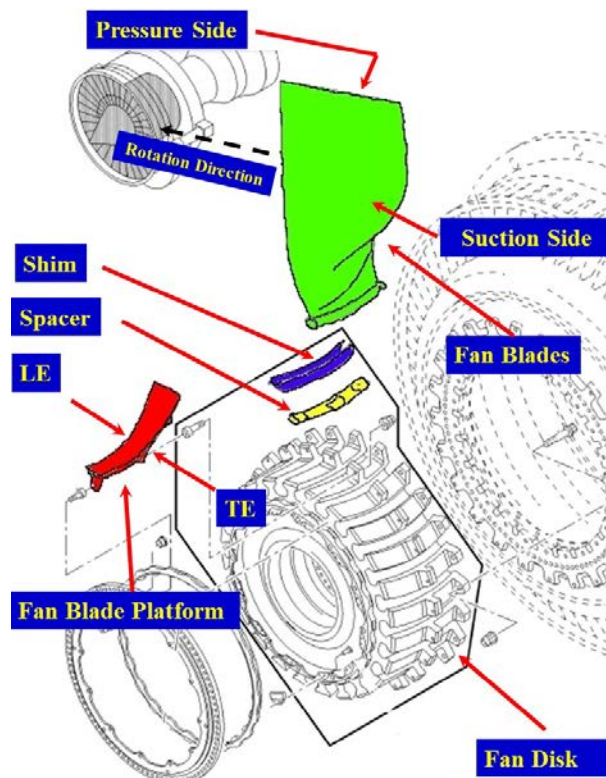


FIGURE 2: FAN BLADE AND ASSOCIATED HARDWARE ASSEMBLY SCHEMATIC

FIGURE COURTESY OF CFM

⁴ Although the root of the fan blade is technically comprised on the airfoil shank and dovetail, the airfoil shank is commonly referred to as simply the root. For purposes of this report, the airfoil shank and root can be used interchangeable and the dovetail will be used to reference that part of the fan blade root that slides into fan blade disk. See **FIGURES 18-20** and footnote 20 for additional clarification.

1.3.1 Fan Disk

The fan disk on the event engine was part number (PN) 340-000-420-0, serial number (SN) DD763459-E (**PHOTO 2**). This fan disk configuration requires both a spacer and shim; this configuration was introduced by CFM56-7B⁵ service bulletin (SB) 72-0485 titled *Engine – Fan and Booster Assembly (72-21-00) – Introduction of a New Fan Disk and New Spacers and New Shims*. The current version of SB 72-0485 is Revision 2 dated April 23, 2008, and the original was released on August 13, 2004. Prior to the release of the SB, there had been reports of premature flaking (spalling) on the Cu-Ni-In coating on the fan blade dovetail and wear on the pressure faces of the fan disk determined to be caused by high local crush stress concentration. To address these in-service findings, SB 72-0485 was issued to introduce a new fan rotor configuration that included a new fan disk PN 340-000-420-0 (slot geometry was changed to accommodate a shim), a shim PN 340-252-306-0 between the fan blade and fan disk to improve the lubrication duration and thus reduce the stress levels, and new spacer PN 340-001-261-0 (spacer height was changed to accommodate the shim). The fan disk is made of wrought Ti alloy; Ti-6Al-4V (often shortened to Ti-6-4).⁶



PHOTO 2: EVENT FAN DISK PN AND SN

PHOTO COURTESY OF CFM

1.3.2 Fan Blade

CFM did not classify and certify the fan blade as a life limited part (LLP); therefore, there is no regulatory Chapter 5 life-limit requirement. However, all the fan blades are serialized and according to SWA, starting in 2015, each fan blade time and cycle were tracked within their fleet.

The following describes how the fan blade SN is decoded; for illustration purposes the fractured blade No. 23, SN BA824478-6, is used for reference (**PHOTO 3**). The first digit 'B' represents the fan blade forging facility; in this case the 'B' represents Safran Aircraft Engines. The next 7 digits 'A824478' is the serial number followed by a dash ('-'), and then followed a 'Checksum' identifier, in this case '6'. To improve engine parts configuration control and traceability, a 'Checksum' algorithm was introduced to the serial number marking in the mid-1990s. This algorithm adds one or two alphanumeric characters to the serial number marking, separated by a dash ('-'). According to CFM, the dash and Checksum identifier are **not** part of the serial number, are not to be recorded by customers in their serial number tracking system, and the checksum suffix should not appear on EASA/FAA documents. **ATTACHMENT 1** provides the list of fan blades SNs from the last shop visit at GE Celma.

⁵ The CFM56-7B will be dropped from the SB number for brevity for the remainder of the report.

⁶ The nominal composition of Ti-6Al-4V is Ti with 6% Al and 4% V by percentage weight.

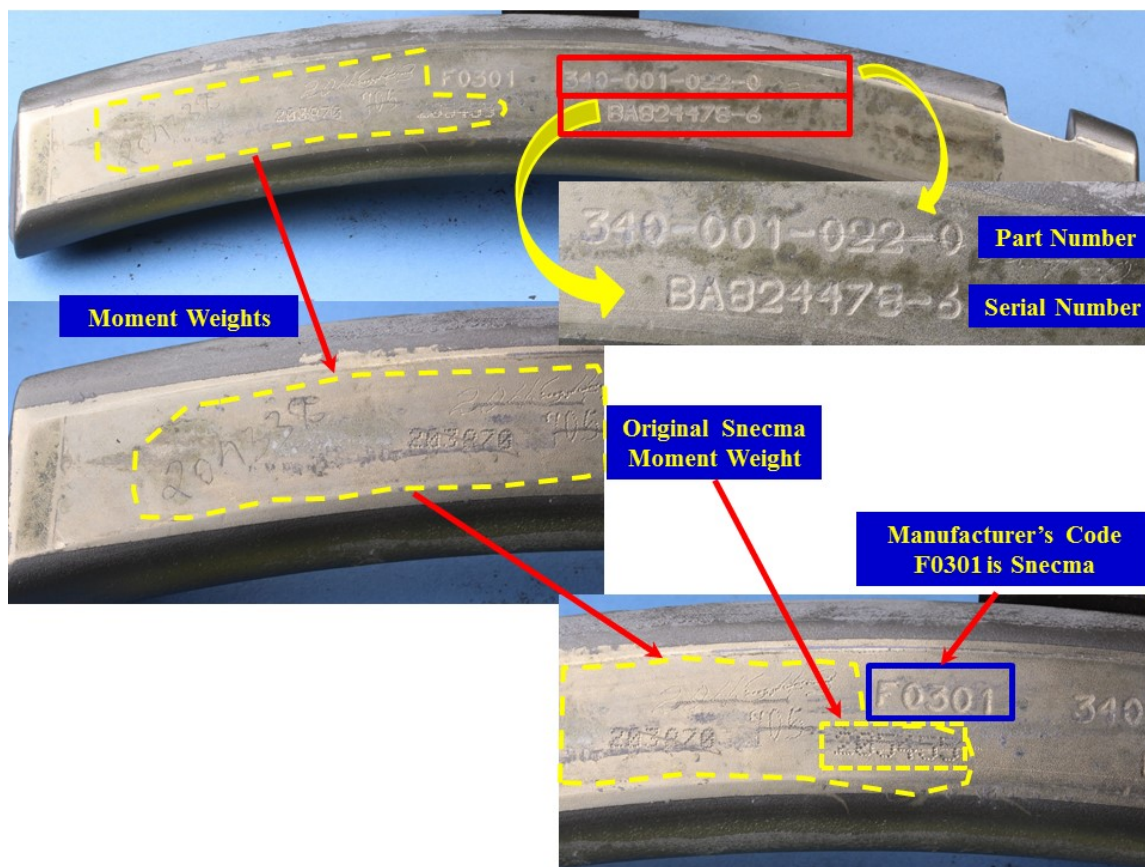


PHOTO 3: FAN BLADE MARKING IDENTIFICATION

All the fan blades were originally PN 340-001-022-0 and the -022-0 was marked out (~~-022-0~~) and replaced with a -027-0 making the new PN 340-001-027-0 after SB incorporation. Fan blade PN -027-0 was introduced by CFM SB 72-0253 titled *Engine – Fan and Booster (72-21-00) Introduction of a New Fan Blade and Rework of the Fan Blade Pressure Face Undercut*. The current version of SB 72-0253 is revision 6 dated November 6, 2002, and the original was released on July 11, 2000. The SB was issued to reduce the local crush stress level during operation on the fan blade and fan disk pressure faces that caused premature flaking of the Cu-Ni-In coating on the fan blade dovetail resulting in loss of lubrication; this is done by a geometrical modification of the fan blade root undercut. SB 72-0253 introduced both a new production fan blade PN 340-001-026-0 and an approved field modification fan blade PN 340-001-027-0. Because material is removed, the moment-weight (MW)(gram-inches) of the fan blade changes; therefore, the older MW was marked out and a new MW was marked on the blade root. Each fan blade appears to have a total of five different MW marking on the bottom of the blade root; this is consistent with 4 separate shop visits for inspections and repair.

The fan blades are made of Ti-6-4 and a Cu-Ni-In coating and varnish are applied to the dovetail (**FIGURE 3**). The Cu-Ni-In coating (purple for illustration purposes) provides a better fit with the fan disk (plastic behavior of the coating under load) and wear protection while the dry film lubricant varnish (green - Molydag®) provides low frictional resistance. Before the shims are installed over the fan blade dovetail, a molybdenum disulfide lubricant⁷ (Molykote™) is applied for additional low frictional resistance used for tight tolerance mating parts. The nominal weight of the PN -027-0 blade configuration is approximately 10.83 pounds (4,913 grams). Although the dovetail from the failed No.

⁷ Red if sprayed or black if applied by brush – only one is required.

23 fan blade was not weighed before it was sectioned for the metallurgical examination, the estimated total weight of the dovetail pieces was about 0.7-0.9 pounds (318-408 grams).

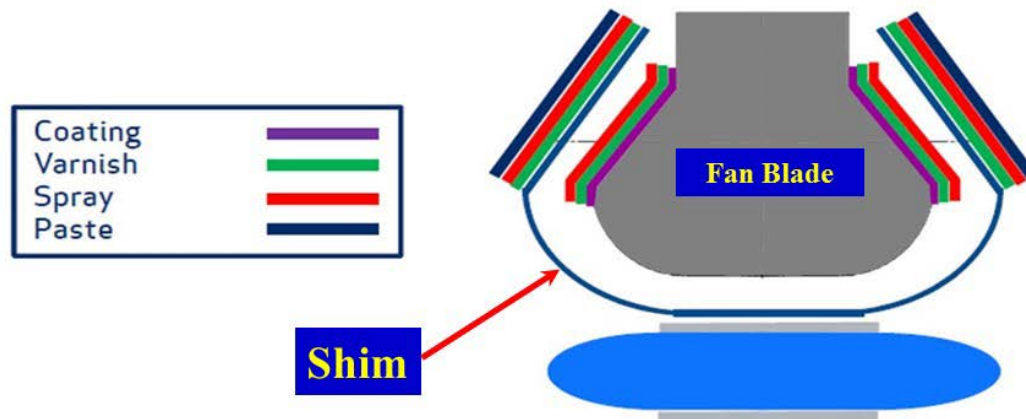


FIGURE 3: COATINGS ON FAN BLADE DOVETAIL CONTACT SURFACES AND SHIM

FIGURE COURTESY OF CFM

1.3.3 Fan Blade Shim

The event fan rotor had two different authorized configurations of fan blade shims: PNs 340-252-306-0 and 340-252-307-0. See **FIGURE 4** for shim configuration differences. The shim is used to avoid fretting of the fan disk pressure faces, and it also improves lubrication durability which reduces stress concentrations. SB 72-0481 titled *Engine – Fan and Booster Assembly (72-21-00) – In Field Introduction of Fan Blade Shims with New Axial Stop* introduced a modified shim design to improve reliability and to reduce possible damage to the forward face of the fan blades by the shim retention lugs and to compensate for the material removed from the fan disk slots when SB 72-0485 was introduced. In-service reports indicated the shim edges were too flexible allowing the shim to move forward. To address this condition, a new shim configuration, PN 340-252-307-0, was introduced by CFM SB 72-0649 titled *Engine – Fan and Booster Assembly (72-21-00) – Introduction of New Fan Blade*. The current version of SB 72-0649 is Revision 1 dated January 18, 2011, and the original was released on February 8, 2008. The new shim deleted the interference fit of the shim chamfer on the fan blade root, deleted the interference

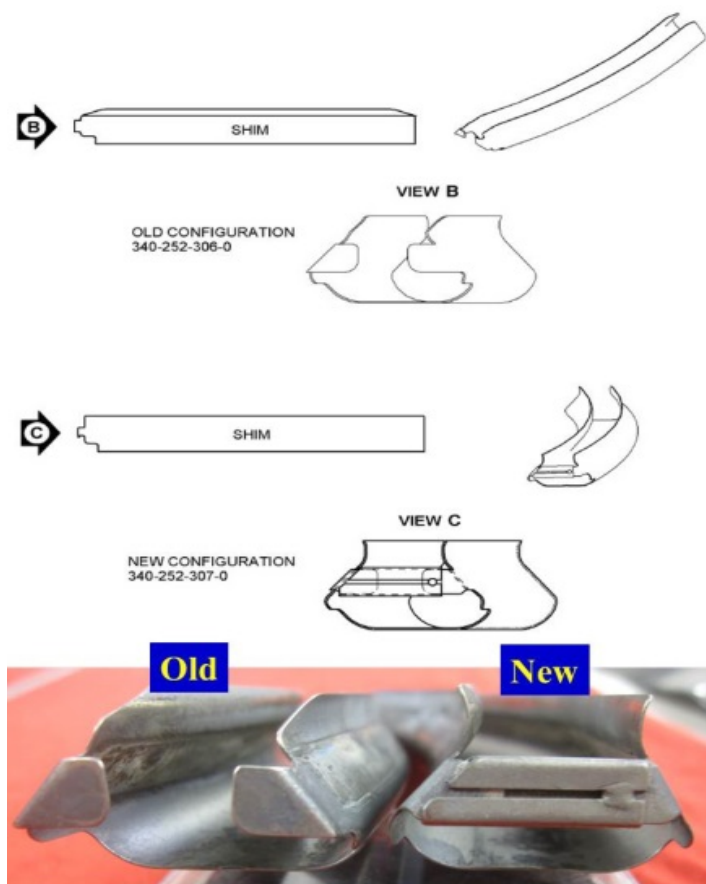


FIGURE 4: FAN BLADE SHIM CONFIGURATION

FIGURE COURTESY OF CFM

fit on the fan blade platform by limiting the axial shim displacement, and limited the distortion on opening of the lugs by introduction of a welded retaining plate. Both the older and newer styles of the shim are made of alloy 718⁸ and both configurations are allowed on the fan assembly. Examination of the shims on the incident fan rotor revealed that all the shims were the older style configuration PN 340-252-306-0 except for shims on fan blades Nos. 22-24. Like the fan blades dovetail lubrication prior to installation, molybdenum disulfide lubricant is also applied to the shim; the lubricant can be sprayed-on (red) or a brushed-on paste (black) (See **FIGURE 3** for dovetail coating details).

1.3.4 Fan Blade Platform

All the fan blade platforms were PN 340-001-816-0; with a mix of new production and reworked by CFM SB 72-0586 titled *Engine – Fan and Booster Assembly (72-21-00) – Introduction of a New Fan Blade Platform with a New Elastomer Seal*. The current version of SB 72-0586 is Revision 2 dated August 31, 2015 and the original was released on November 24, 2006. The SB was released to improve the convex seal reliability as there were reports of elastomer seal distress during fan blade dovetail relubrication.⁹ **PHOTO 4** provides the fan blade platform nomenclature. The platforms are intended to provide a smooth aerodynamic flow path between the fan blades.



**PHOTO 4: FAN BLADE
PLATFORM ORIENTATION**

1.3.5 Fan Blade Spacer

All the fan blade spacers were PN 340-001-261-0 which was introduced along with the new fan disk and shims by SB 72-0485. The spacers are made of Ti-6-4 and are intended to maintain the fan blades radially during engine stops or very low rotational speed and ensure axial load transmission from the fan blade to the forward axial retentions system in case of the FBO event or bird ingestion. According to the useable-on codes the spacer PN -261-0 can be installed on fan disk PN -420-0. The event fan disk was PN -420-0.

⁸ Alloy 718 is wrought precipitation hardenable Ni alloy.

⁹ For the remainder of this report, relube(s) will be used to replace relubricate-relubricated-relubricating-relubrication

1.4 FAN CASE DESCRIPTION

The fan frame assembly supplies front handling mounts and is the main forward support for the installation of the engine to the airplane. It consists of three major assemblies: the fan frame assembly, fan inlet case assembly (sometimes referred to as the fan containment case and will be referred to as the fan case for simplicity), and the fan outlet guide vane (OGV) assembly; only the fan case assembly will be discussed in this section.

The fan case is made of Al alloy and has flanges and ribs on its outer surface to provide additional strength and stiffness to the case during engine operation. Inside the fan case is an abradable (blue) shroud located in-line with the fan blade rotational plane that provides proper tip clearance (**FIGURE 5**). The fan case provides for fan blade radial containment following an FBO event, the fan case transmits FBO loads to both the fan frame and inlet.

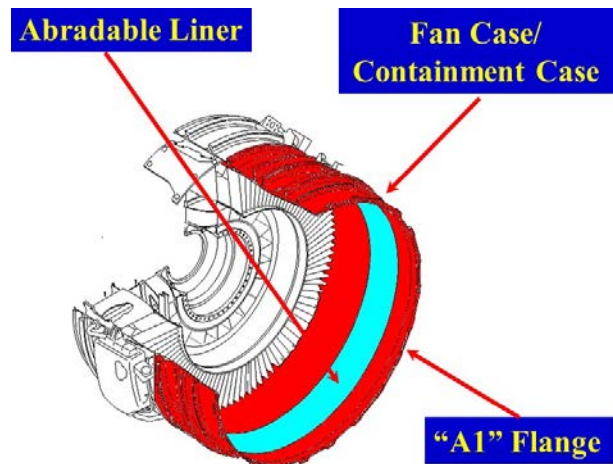


FIGURE 5: FAN CASE/CONTAINMENT CASE

FIGURE COURTESY OF BOEING

The fan case provides for the primary FBO radial containment protection; the Boeing 737-700 inlet, which is attached to the fan case A1 flange (engine forward flange), also provides additional fan blade protection. The fan case A1 flange-to-inlet attach ring interface is considered critical for the FBO capability of the inlet and is where engine loads are reacted with the inlet.

1.5 ENGINE NACELLE DESCRIPTION

The engine inlet¹⁰, fan cowl, fan duct cowl and thrust reverser (TR) (subsequently referred to in the rest of the report as simply the TR), the primary exhaust nozzle, and exhaust plug, comprise the boundaries of the engine nacelle and consist of fixed and hinged components (**FIGURES 1** and **6** and **PHOTOS 5-7**). The fixed components include the inlet, primary exhaust nozzle, and exhaust plug; while the hinged components include the fan cowl and TR. The inlet is bolted to the engine fan case A1 flange, the primary exhaust nozzle to the outer portion of the turbine rear frame (TRF), and the exhaust plug (two-piece design) is bolted to the forward centerbody which is attached to the inner portion of the TRF. The fan cowl and the TR are each in two halves, hinged at either side of the pylon, and joined by latch hooks on the bottom split line. The fan cowl incorporates three hinges and three latch assemblies while the TR incorporates four hinges and six latch assemblies, including a lower radial restraint and four axial locators. All the nacelle hardware, including the TR, is provided by Boeing, with detailed design and build responsibility accomplished by several different suppliers.¹¹

¹⁰ The inlet is also referred to as the inlet cowl or inlet cowl assembly in various Boeing and UTAS documents; for brevity, only the term “inlet” will be used throughout this document.

¹¹ The inlet and fan cowl are designed and manufactured by UTAS. The TR design is owned by Boeing and manufactured by Spirit AeroSystem. The exhaust is a Boeing design and was manufactured by Doncasters.

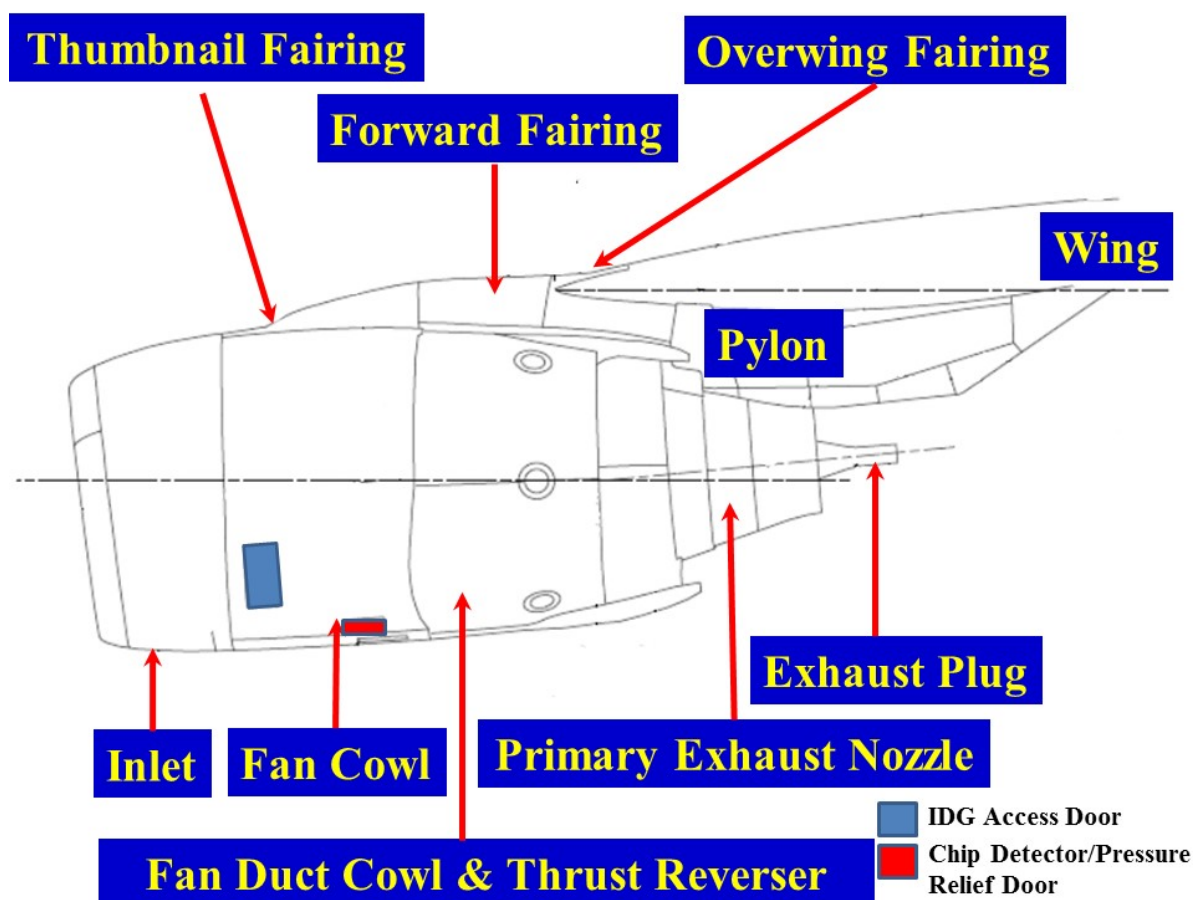


FIGURE 6: BOEING 737 NACELLE NOMENCLATURE AND LOCATION - ALL BOEING SUPPLIED

FIGURE COURTESY OF BOEING



PHOTO 5: LEFT SIDE OF EXEMPLAR ENGINE NACELLE



PHOTO 6: RIGHT SIDE OF EXEMPLAR ENGINE NACELLE



PHOTO 7: FRONTAL VIEW OF EXEMPLAR INLET

1.5.1 Inlet Description – Inlet PN 314-2100-2

The inlet is an aerodynamic fairing assembly that is mounted to the engine at the A1 flange and directs smooth uninterrupted airflow into the fan and core sections. The inlet also provides acoustic attenuation for engine noise and acts as an external aerodynamic pressure surface that directs smooth airflow over the fan cowl and the rest of the nacelle. There are two hoist points on each side of the inlet to permit removal of the inlet from the engine with a hoist and sling. The entire inlet weighs approximately 356 pounds and measures approximately 42-inches-axial-length, 80-inches-tall, and 62-inches inside diameter.

The inlet is comprised of the inlet lip, inner barrel, forward and aft bulkheads, outer barrel skin panels, and attach ring (FIGURES 7-8). The combination of the inlet lip and forward bulkhead is sometimes referred to as the D-duct assembly. The inner and outer barrels are connected to bulkheads at their forward and aft ends. The inner and outer aft ends of the lip assembly (referred to as the inlet lip) are attached to the forward bulkhead. The inlet lip forms an aerodynamic surface to reduce airplane drag. The aft end of the inlet structure, the attach ring, forms the aft closeout of the inner barrel and inner chord of the aft bulkhead and is attached to the engine flange with 24 bolts. The aft bulkhead segments are fastened together by rivets and are secured to the attach ring by rivets while the inner barrel is fastened to the attach ring by a series of Hi-Lok™ fasteners. The inlet lip skin and the forward bulkhead form a D-duct for hot engine bleed air to flow to and inhibit ice formation on the inlet lip. The inlet cowl thermal anti-icing (CTAI) duct supplies engine bleed air to the swirl nozzle inside the D-duct. Bleed air is let out through the thermal anti-ice (TAI) exhaust duct at the bottom of the inlet immediately aft of the inlet lip. There is an engine control unit (ECU) cooling scoop and a T12 access/pressure relief door as well as a panel to permit access to the TAI exhaust duct for maintenance.

The inlet lip and outer barrel skin, as well as the attach ring, are made of Al while the forward and aft bulkheads are made of Ti-6-4 and the inner barrel is an acoustic honeycomb structure primarily made from Al. The inlet lip is made of 2219 Al skin, the outer barrel is made of 2024 Al skin and frame panels – an upper and lower panel, and the attach ring is made of forged 7075 Al with 24 holes for attachment bolts. Each of the 24 attachment bolt assemblies is comprised of a nut, bolt, washer and crushable spacer; the attachment bolt assembly is provided by Boeing. The inner barrel upper and lower panels are mechanically spliced longitudinally in two places 180° apart (at about the 3:00 and 9:00 o'clock positions) using Ti splice plates (the splice plates are sometimes referred to as splice straps).

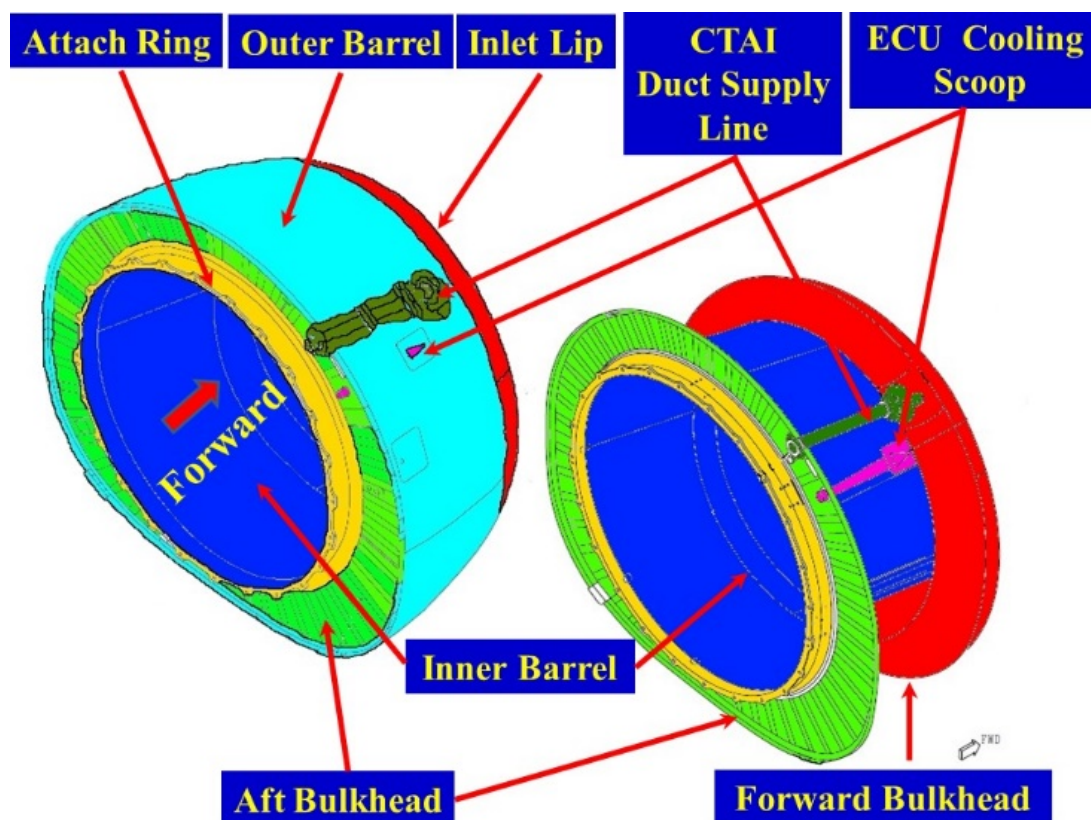


FIGURE 7: INLET NOMENCLATURE

FIGURE COURTESY OF BOEING

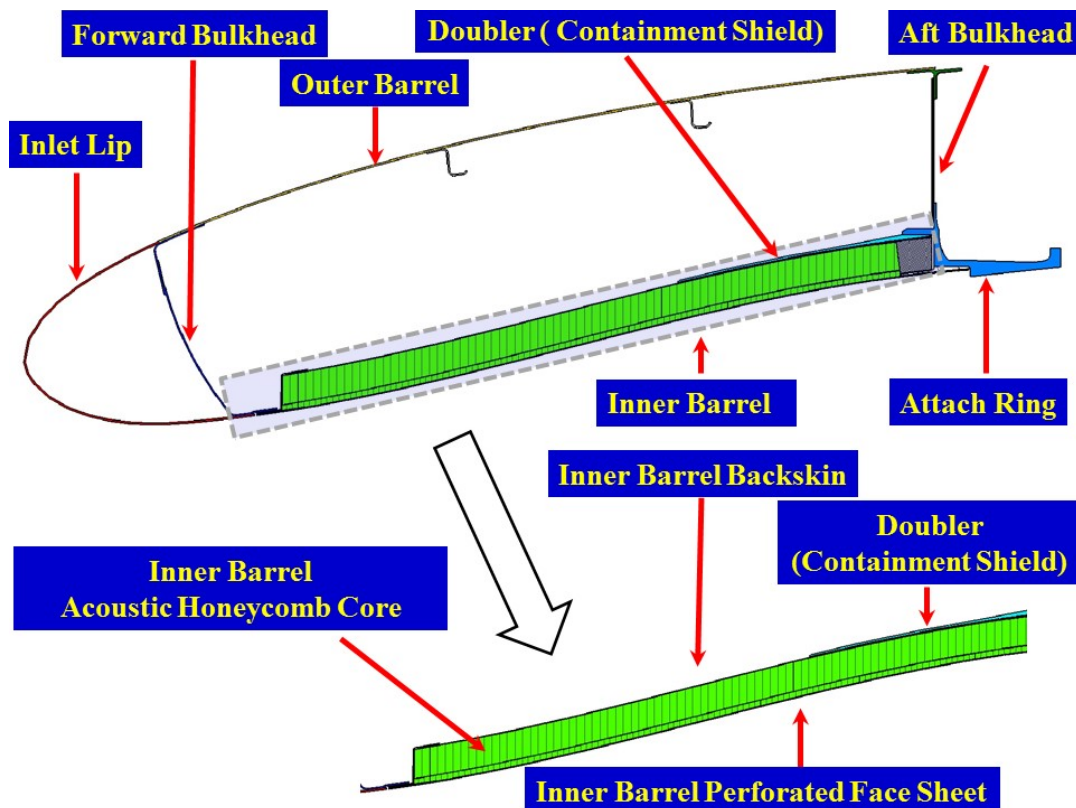


FIGURE 8: CROSS-SECTION OF INLET

FIGURE COURTESY OF UTAS

The inner barrel is made from two acoustic honeycomb sandwich panels consisting of an outer 2024 Al face sheet, Al honeycomb split core with a non-metallic septum, and an inner 2024 Al perforated face sheet for sound suppression. On the backside of the inner barrel outer face sheet, a 2024 Al axial doubler (sometimes referred to as a plate or containment shield) is bonded over about the aft 11-inches of the panel. This doubler has a tapered thickness from the aft edge to the forward edge. The Al face sheets, core and containment shield are co-cure bonded together in a single autoclave bond operation. The inner barrel is fabricated in two sections secured together by two horizontal Ti splice plates 180° apart located at the 3:00 and 9:00 o'clock positions (FIGURE 9). The inner barrel is fastened to the attach ring by a series of Hi-Lok™ fasteners that go through the containment shield, outer face sheet, core, perforated face sheet and attach ring.

Aerodynamic and inertia loads on the outer barrel are transmitted to the inner barrel through the inlet lip and the forward bulkhead and directly through the aft bulkhead where they are transmitted to the attach

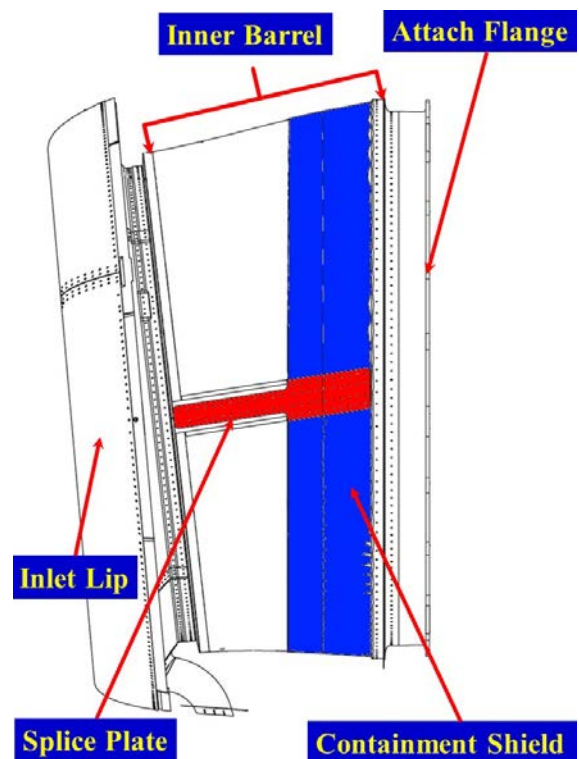


FIGURE 9: INLET NOMENCLATURE

FIGURE COURTESY OF BOEING

ring and finally to the engine fan case. In addition, axial force and aft loads from the inlet movement are reacted at the inlet aft bulkhead into the fan cowls.

Boeing created the design specifications for the inlet and Rohr Industries, Inc. designed, manufactured, and delivered the inlet to Boeing. Subsequently, Rohr Industries, Inc. was acquired by Goodrich Aerospace and became Goodrich Aerostructures respectively before being acquired by United Technologies Corporation where the name changed to UTAS. Though the company name at the time of the design and certification was Rohr Industries, Inc., for the remainder of this report, the inlet manufacturer will be referred to as UTAS.

1.5.2 Fan Cowl Description – Fan Cowl PNs 314-2200-3 (left) and 314-2200-11 (right)

The fan cowls are aerodynamic fairings that enclose the engine fan case and extends from the aft end of the inlet to the forward end of the thrust reverser. The fan cowl is constructed in two halves (left and right) with each half hinged at three locations at the top to the pylon fan cowl support beam by a bolt to a clevis assembly (**FIGURE 10**). At the bottom, along the 6:30 o'clock axial line, the two halves are joined together by three latch assemblies; each latch assembly is comprised of a latch hook (left cowl) and a latch keeper (right cowl); the latch keeper is sometimes referred to as the clevis. The fan cowl hinges are cast Ti, the fan cowl skin and frame construction are 2024 Al and the latch housings are Al forgings. A radial support fitting at the bottom of the fan cowl engages with a radial support bracket from the engine fan case so to help the fan cowl to maintain its shape. Pressure loads encountered throughout the flight and ground loads are carried by the skin primarily by hoop tension and compression. The hoop loads are transferred to the fan cowl support beam at the hinge interfaces. Axial fore and aft loads are reacted at four axial locators at the inlet aft bulkhead. In addition, radial contact loads are reacted at the inlet aft bulkhead, aft interface with the thrust reverser, and radial support fitting.

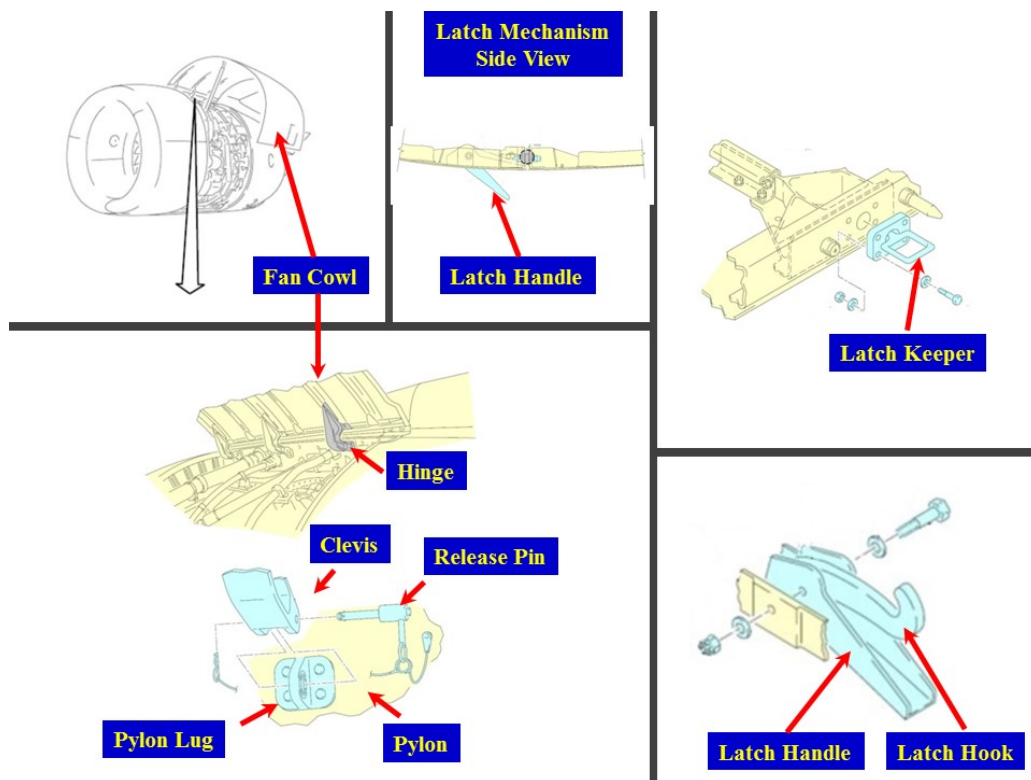


FIGURE 10: FAN COWL HARDWARE AND NOMENCLATURE

FIGURE COURTESY OF BOEING

1.6 EVENT INLET MAINTENANCE HISTORY

SWA provided maintenance records for all the discrepancies and repairs performed on the inlet assembly since its installation on the accident engine back in 2007. Only two discrepancies were noted; a lightning strike in February 2008 and missing sealant on the inlet lip in December 2015. The February 2008 non-routine discrepancy work cards for the lightning strike indicate that the inlet was struck twice; 3-inches and 19-inches aft on the inlet lip on the left side of the nose cowl. In both cases, the damaged area was repaired in accordance with the Boeing structural repair manual (SRM) 54-10-01.

2.0 ON-SCENE EXAMINATION

The Airworthiness Group, comprised of members from CFM, Boeing, SWA, FAA, Southwest Airlines Pilot's Association (SWAPA), and the NTSB, convened at the PNS on August 28, 2016, to perform a detailed examination of the event airplane and engine, and the group completed its on-scene work on September 1, 2016.

2.1 LEFT ENGINE INLET AND FAN COWL HARDWARE EXAMINATION

On the event engine, a 360° section of the inlet inner barrel forward of the containment doubler separated from the rest of the inlet (**FIGURE 11** and **PHOTOS 8-10**). This corresponds to a location on the inlet where a design thickness step change occurs at the forward end of the containment shield (doubler), approximately 17-inches forward of the A1 flange. The only other part of the inner barrel that remained were the two bolted inner barrel splice plates, located at the 3:00 and 9:00 o'clock positions, where the two halves of the inner barrel mate (See **FIGURE 9** and **PHOTOS 8-9**). The blue dashed line in **FIGURE 11** provides the approximate location of the fracture plane of the inlet inner barrel at the containment forward end.

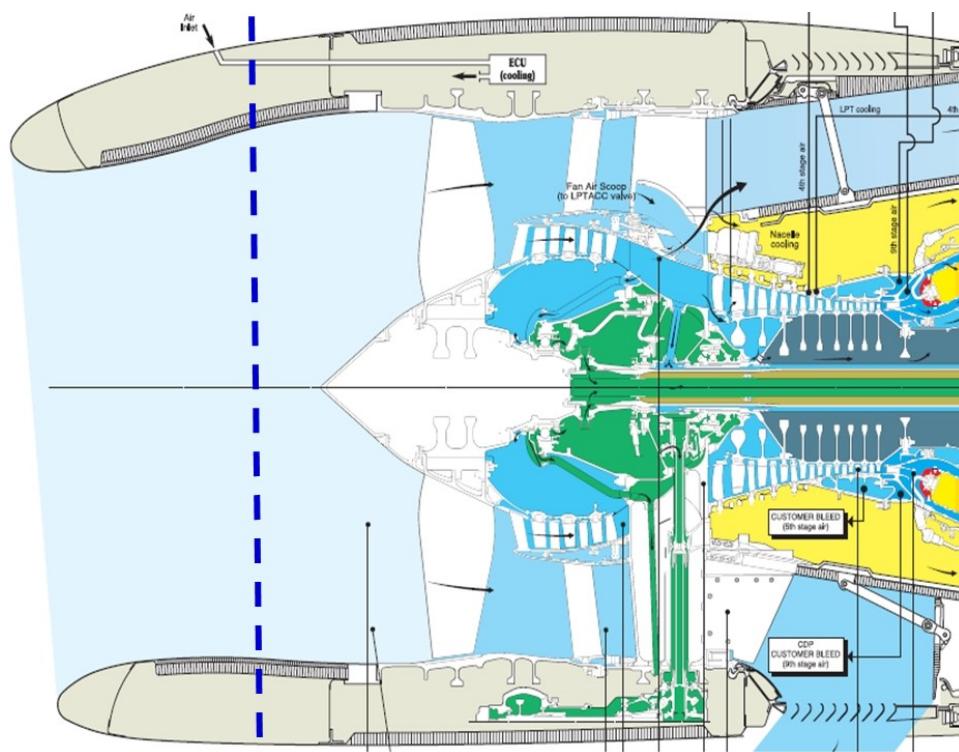


FIGURE 11: DASHED LINE APPROXIMATE FRACTURE LOCATION

FIGURE COURTESY OF CFM

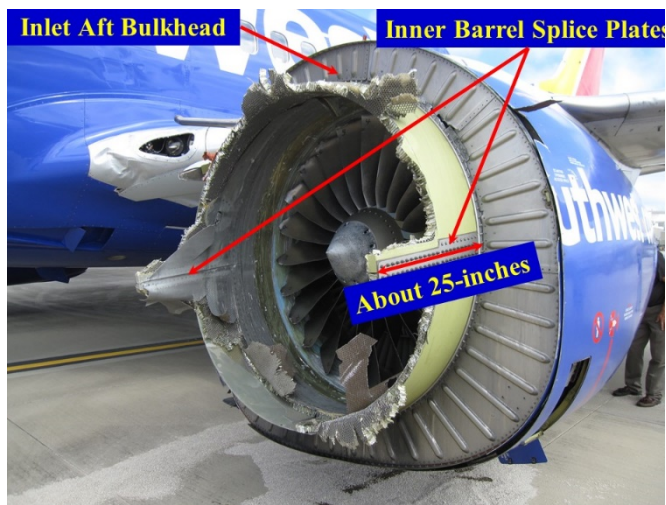


PHOTO 8: LEFT ENGINE INLET – OUTBOARD

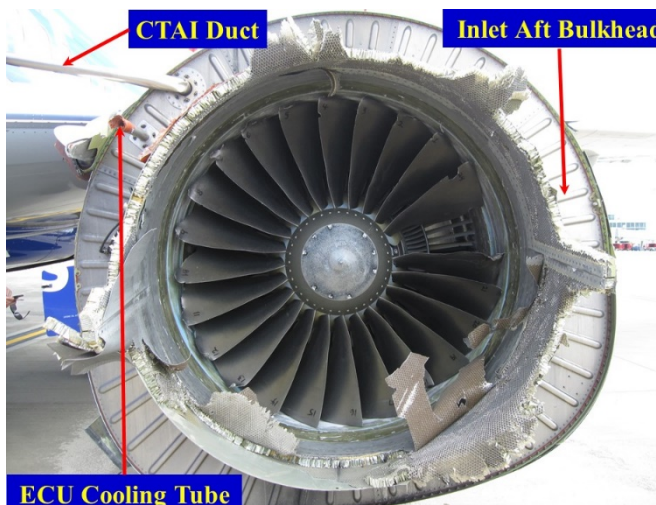


PHOTO 9: LEFT ENGINE INLET – FRONTAL

The inner barrel honeycomb core and inner skin that remained was flared and torn outward. The inner barrel splice plate at the 9:00 o'clock position exhibited impact damage and outward (engine outboard, aircraft inboard) deformation while the splice plate at the 3:00 o'clock position appeared visually undistorted. The inlet CTAI duct and the ECU cooling tube that both pass through the inlet aft bulkhead located between the 1:00-2:00 o'clock positions were fractured and the CTAI duct was bent outwards. The entire outer barrel was missing forward of the inlet aft bulkhead; the inlet lip and forward bulkhead (D-duct) was also missing. The T12 temperature probe was still attached to the inlet inner barrel but was flattened (**PHOTO 11**). All the rivets that attach the still-present aft bulkhead to the inner barrel flange were either missing or sheared (**PHOTO 12**).

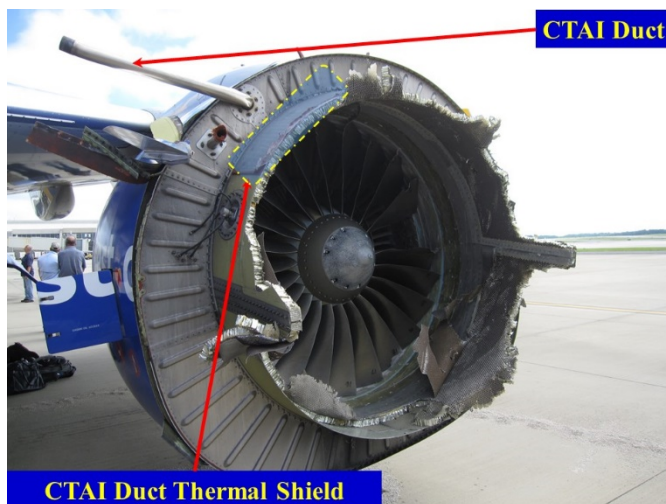


PHOTO 10: LEFT ENGINE INLET – INBOARD



PHOTO 11: LEFT ENGINE DAMAGED T12 PROBE

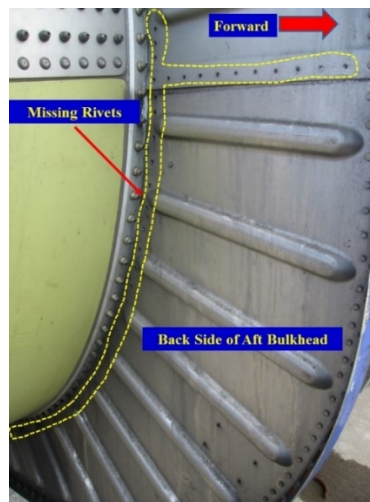


PHOTO 12: LEFT ENGINE INLET AFT BULKHEAD MISSING RIVETS - INBOARD SIDE

With the fan blades removed, the damage to the fan case and the inlet was examined in greater detail for possible fan blade debris paths and exit locations. Blue tape was used to follow potential debris paths based on the observed damage orientation and direction in the fan case and inlet (**PHOTOS 13-15**). More precise trajectory mapping was conducted by CFM and UTAS, after the fan case and inlet case were removed from the engine. See Section 10.0 POST EVENT FAN CASE AND INLET DOCUMENTATION & ASSESSMENT for details.

After FBO release, fan blade airfoil/tip fragments, if not influenced by other factors, typically wants to travel in a forward spiral/helical pattern around the fan case and inlet; the forward movement is due to several factors such as any slope/conical nature of the fan case and inlet surfaces, the angle of attack of the blade, and aerodynamic loading in the forward direction (**PHOTO 13**).

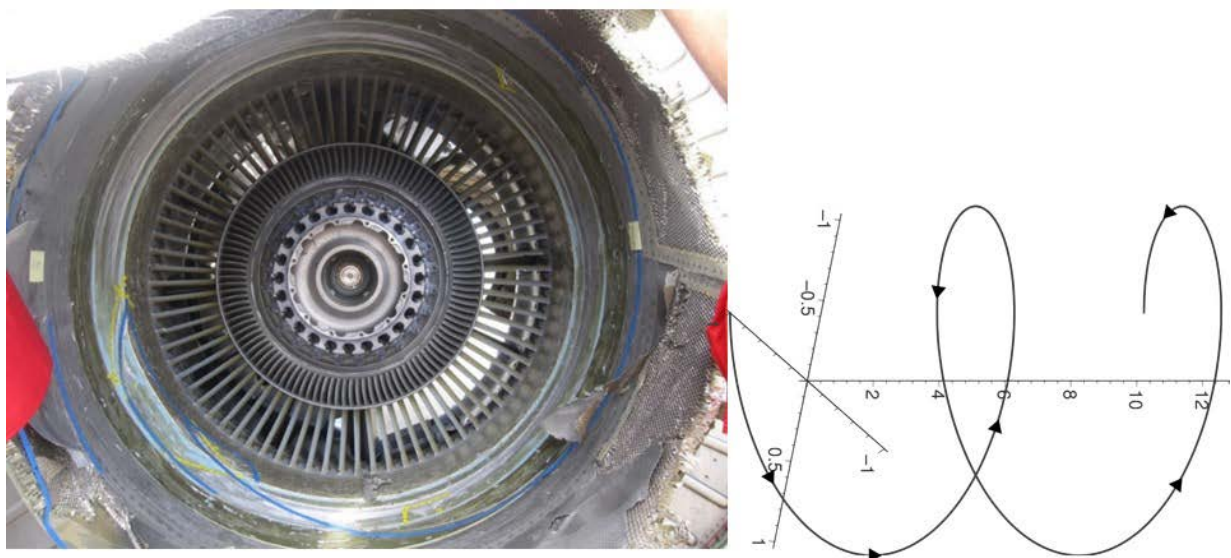


PHOTO 13: LEFT ENGINE FAN CASE AND INLET OVERALL VIEW OF DEBRIS PATH¹²

¹² The figure of the helix is by an unknown author and is licensed under CC BY-SA-NC (<https://creativecommons.org/licenses/by-nc-sa/3.0/>).



PHOTO 14: LEFT ENGINE INLET DAMAGE AND DEBRIS PATH AT THE 3:00 O'CLOCK POSITION



PHOTO 15: LEFT ENGINE INLET DAMAGE AND DEBRIS PATH AT THE 9:00 O'CLOCK POSITION

The inlet is attached to the fan case by 24 nuts and bolts with crushable spacers. There are two different configurations of this installation, a preferred and an alternate (**FIGURE 12**); the event engine had the preferred installation (**PHOTO 16**). All the bolts, nuts, crushable spacers, and washers were accounted for, in place, and some of the crushable spacers were compressed. A gap was observed between the rear flange of the inlet attach ring and the fan case A1 flange at several locations around the circumference; gaps appeared between bolts locations and the gaps were observed at the 3:00, 5:00, and between the 7:00-9:00 o'clock locations. The torque was not measured on any of the inlet attach bolts.

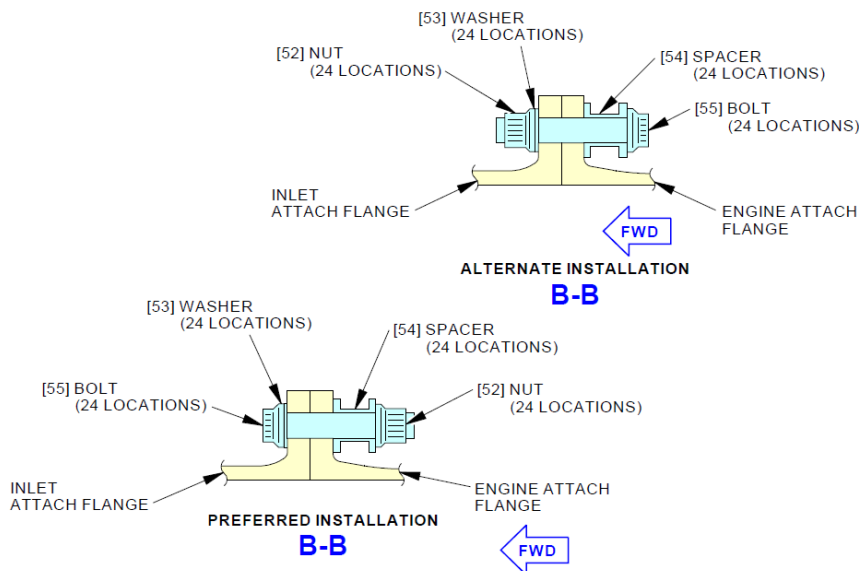


FIGURE 12: INLET-TO-FAN CASE ATTACHMENT CONFIGURATION

FIGURE COURTESY OF BOEING

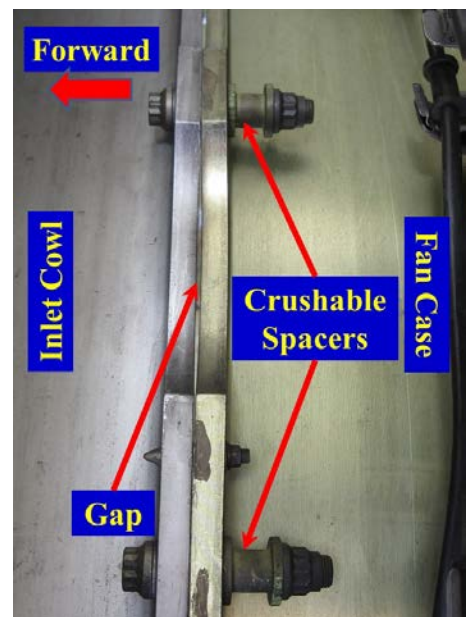


PHOTO 16: INLET-TO-FAN CASE ATTACHMENT CONFIGURATION AND GAP BETWEEN ASSEMBLIES

The forward most fan cowl latch was latched and secure; the aft most two latches were not latched and found hanging. All the latch and latch keepers were intact.

On the left fan cowl are access panels for the integrated drive generator (IDG) and the engine magnetic chip detectors (MCD). The IDG access panel was fractured and not recovered; the only part that remained with the fan cowl was the hinge assembly. The MCD access door was found open; the locking mechanism and access cutout in the cowl appeared to be undamaged. The forward edge of the left fan cowl was peeled outboard at about the 10:00 o'clock position; the peeled section measured 6-inches-circumferential x 4-inches-axial.

On the right fan cowl: (1) the forward edge was peeled outboard at about the 3:00 o'clock position (28-inches-circumferential x 4-inches-axial), (2) the forward edge was missing at about the 4:00 o'clock position (15-inches-circumferential x 4-inches-axial piece), (3) a flap located at about the 2:30 o'clock position (10-inches-circumferential x 4-inches-axial) was also peeled outboard, and (4) the outboard skin exhibited longitudinal scuffing at the 3:00 o'clock position.

On both fan cowl, the LE rub strip was heavily worn and loose. The fan cowls were opened and examination of the radial restraint shear pin (located on the right fan cowl near the bottom where the latch keepers are located), the latch, and latch keepers revealed that all appeared undamaged. The fan cowl axial locator was found dislodged from its normal 4:00 o'clock position but was found when the fan cowls were opened.

The TR uses six latch and latch keeper assemblies to secure the two halves; all six were latched (**FIGURE 13**). The left TR aft hinge bolt bush was found loose and displaced. The right TR inner wall trailing edge (TE) edge was found scored at the 4:00 o'clock position coincident with similar scoring noted on the primarily exhaust nozzle outer surface. The TR was manually translated OPEN and CLOSED with no binding.

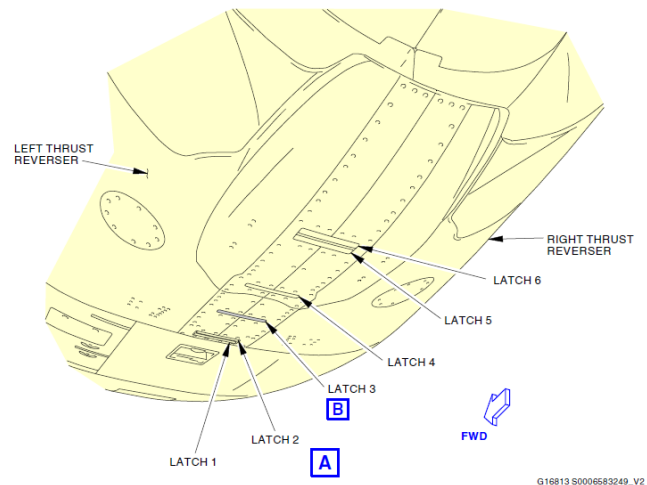


FIGURE 13: THRUST REVERSER LATCH LOCATIONS

FIGURE COURTESY OF BOEING

The eight bolts (four forward and four aft) that secure the forward and rear engine mount hanger fitting to the pylon were in place, secure, and tight. No removal torques were taken on any of the engine mount bolts when the engine was removed from the pylon. After the engine was removed from the airplane, the bolts were examined and they all appeared to be normal, each had a washer and the outer aft rear bolt also had a grip length adjustment washer; this washer is used for ensuring proper bolt length. Examination of the forward engine mount hardware revealed that: (1) the forward fan case fitting was still attached to the fan case and secured, (2) the two forward mount links appeared undamaged and were secured to both the fan case fitting and the forward engine mount hanger, and (3) the forward engine mount hanger and its two alignment pins were intact and undamaged. The two thrust links appeared intact,

undamaged, and secured to both the forward and rear engine mount hanger fittings. Examination of the rear engine mount hardware revealed that: (1) the two rear mount links appeared undamaged and were secured to the TRF and the rear engine mount hanger and (2) the shear pin hole appeared undistorted. The shear pin and the shear pin hole (located in the rear engine mount hanger) were measured at two locations - inboard and outboard. The diameter of the shear pin at both the inboard and outboard locations measured 1.649-inches while the diameter of shear pin hole at both the inboard and outboard locations measured 1.651-inches. The left engine pylon appeared to be undamaged.

2.2 LEFT ENGINE EXAMINATION

FIGURE 14 provides the nomenclature and location of fan and booster parts discussed in this section. Note that the fan blade (in green) shown in **FIGURE 14** is just of the airfoil; the dovetail resides in the fan disk and is not depicted.

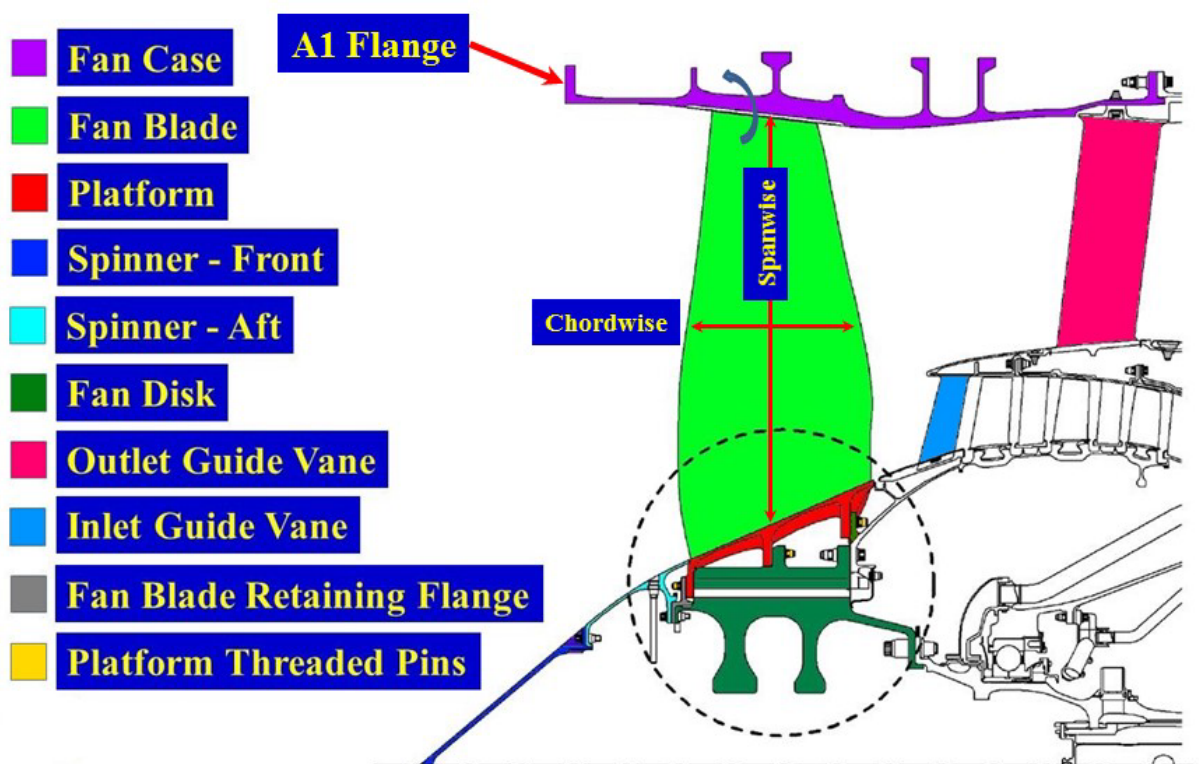


FIGURE 14: FAN BLADE AND ASSOCIATED HARDWARE SCHEMATIC – (FAN ROTATES INTO THE PAGE)

FIGURE COURTESY OF BOEING

The front and rear spinner were intact and appeared undamaged. The front spinner was missing most of the black paint and its white swirl pattern. All six front-to-rear spinner retaining bolts were present and a torque check was performed in the tightening direction using the lowest allowable installation torque (torque specification between 110-120 inch-pounds); none of the bolts rotated under the lowest allowable torque. With the front spinner removed, all the rear spinner-to-fan disk attachment bolts along with rear spinner counterweights were observed. All the hardware was in place and intact. A torque check of the rear spinner-to-fan disk attachment bolts was performed in the tightening direction using the lowest allowable installation torque (torque specification between 143-158 inch-pounds); none of the bolts rotated under the lowest allowable torque.

All the fan blades were present and remained installed in the fan disk except for one; identified as fan blade No. 23 (**PHOTO 17**). The fan blade had separated at the root with the dovetail remaining installed in the fan disk (**PHOTO 18**). All the remaining fan blades exhibited a combination of hard body impact damage, tears, missing material, and airfoil curling or distortion. Adjacent to fan blade No. 23 location in the counterrotating direction, platform No. 22 was missing except for the forward lug. The platform No. 22 middle and aft retaining pins were fractured (**PHOTO 18**) but the forward bolt remained.



PHOTO 17: FAN BLADE FRACTURE – NO. 23

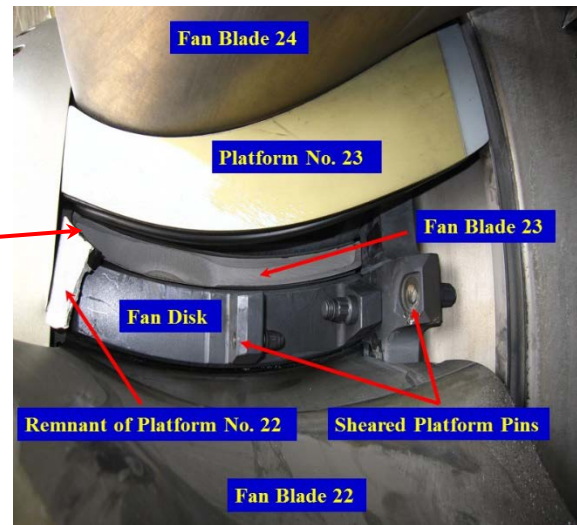


PHOTO 18: NO. 23 ROOT STILL INSTALLED

The fan rotor was rotated by hand; slight resistance was felt which appeared to be coming from the booster. The fan blades tips were not dragging against the fan case when rotated. Examination of the fan case revealed no pass-through hole punctures. The fan blade abradable material (originally light blue in color) had rubbed away exposing the fan case parent material beneath. Circumferential scoring and rubbing were noted along almost the entire circumference of the fan case blade running path and several hard impact marks were noted along and aft of the fan blade running path. At the 3:00-4:00 o'clock position, there were two hard impacts in the blade path identified as impact ❶ and ❷ that were about 2.5-inches from center-to-center of their respective impacts. Impact mark ❶ was a relatively straight impact mark with a slight curvature that measured about 11-inches-lengthwise x 0.75-inches at its widest x 0.30-inches-deep. Impact mark ❷ was a crescent-shaped impact mark that measured about 9.0-inches-lengthwise (deep trench length was about 7.5-inches) x 2.5-inches at its widest (this is the gouging width) and the deepest trenching was about 0.5-inches-wide x 0.30-inches-deep; scuffing (no appreciable depth) was noted about 1.0-inch-wide (**PHOTO 19**). At about the 5:00 o'clock position, there were two hard impacts identified as impacts ❸ and ❹; impact mark ❸ was within the blade running path and impact mark ❹ was aft of the blade running path. Impact mark ❸ was relatively straight and was about 60° relative to the stationary blade tip orientation. Scuffing of the fan case in this area measured about 10-inches-circumferential with the actual gouge measuring about 4.5-inches-lengthwise x 1.5-inches at its widest x 0.2-inches-deep. Blade rub was also noted from the 5:00-7:00 o'clock position circumferentially from impact mark ❸. Impact mark ❹ was located entirely aft of the fan case abradable, was oriented in the normal blade tip orientation, and measured about 8.0-inches-lengthwise x 0.5-inches at its widest x 0.25-inches-deep (**PHOTO 20** – red outline is consistent in shape of the fan blade root). At about the 7:00 o'clock position there was single impact mark, identified as impact mark ❺, consistent with the blade running position that measured about 5-inches-lengthwise x 2-inches-wide x 0.2-inches-deep (**PHOTO 21**). At about the 10:30 o'clock position there was a single impact mark, identified as impact mark ❻, located

at the forward edge of the fan blade running position that measured about 2.5-inches-lengthwise x 0.5-inches at its widest x 0.2-inches-deep (**PHOTO 22**). Looking at the external surface of the fan case, two distress and distorted areas were located just ahead of the oil tank at about the 3:00-4:00 o'clock position coincided with inner impacts marks **①** and **②**. These two areas measured about 2.5-inches from center-to-center and were labeled **X** and **Y** (**PHOTO 23**). One area, **X**, exhibited a slight local tearing in the case and this corresponded with the fan case inner impact mark **①**; the fan case B3 flange was distorted and pushed aft. The other area **Y**, had a visible tear and opening of the outer skin of the fan case; however, no through hole or through tear was observed. A third slight tear was also noted at the 5:00 o'clock position that coincides with inner impact mark **④**; this was labeled **Z** (**PHOTO 24**).



PHOTO 19: FAN CASE IMPACTS ① AND ②

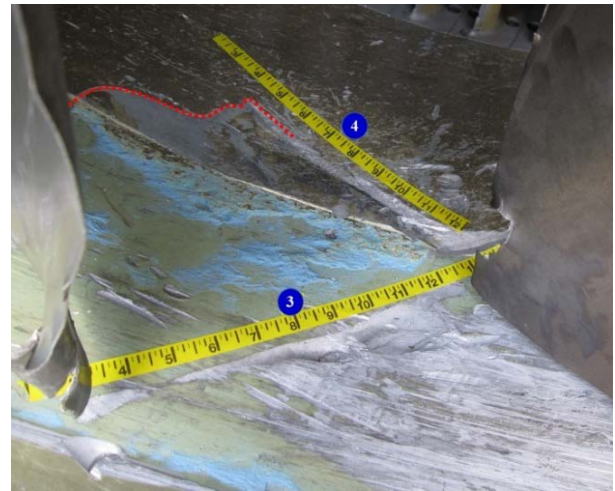


PHOTO 20: FAN CASE IMPACTS ③ AND ④

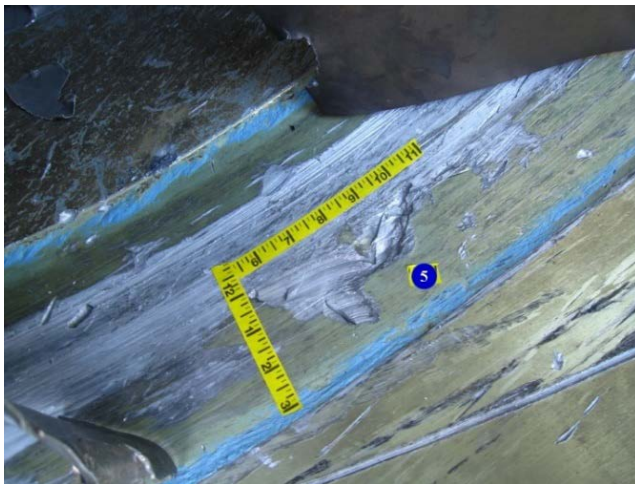


PHOTO 21: LEFT ENGINE FAN CASE IMPACT ⑤

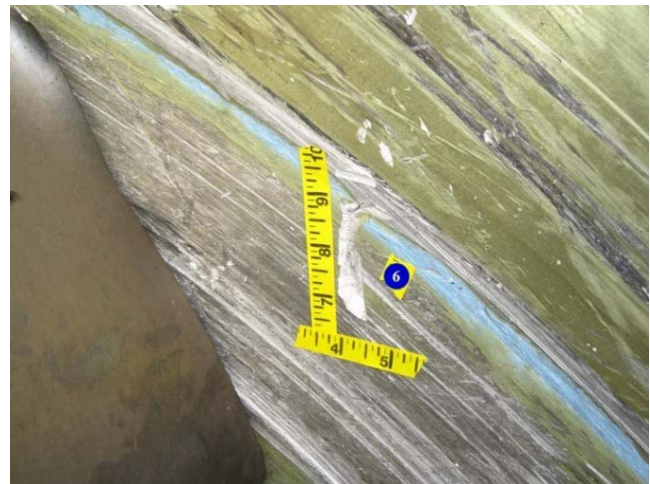


PHOTO 22: LEFT ENGINE FAN CASE IMPACT ⑥



PHOTO 23: TEARS IN THE OUTER SKIN OF THE LEFT ENGINE FAN CASE JUST FORWARD OF THE OIL TANK



PHOTO 24: TEAR IN THE OUTER SKIN OF THE LEFT ENGINE FAN CASE AT THE 5:00 O'CLOCK POSITION

All the fan OGVs were present and full length; several of the fan OGVs exhibited LE impact damage and gouges (**PHOTO 25**). Near the 4:00 o'clock position, in the general area of the fan case impacts ① and ②, several fan OGVs had pulled out slightly from their shroud but were still engaged and the fan OGV seals were fractured with most missing. The booster inlet guide vanes (IGVs) appeared in good condition with no appreciable impact damage. Looking through the booster IGVs, the stage 2 booster blades were visible, and several blades exhibited minor blade tip damage and curling.

Looking through the TRF, all the LPT stage 4 blades were present, intact, and one exhibited minor damage. The primary exhaust nozzle exhibited 360° circumferential buckling (**PHOTO 26**), was torn almost continuous from the 3:00-9:00 o'clock position and was torn intermittently from the 9:00-3:00 o'clock position coinciding with each buckling location. A section of aft edge of the outer exhaust nozzle measuring 13-inches-circumferentially was bent radially outward at around 4:00 o'clock position.



PHOTO 25: LEFT ENGINE FAN OGV DAMAGE



PHOTO 26: LEFT ENGINE LPT STAGE 4 BLADES INTACT AND PRIMARY EXHAUST NOZZLE BUCKLED AND DISTORTED

All three MCDs were removed and examined for debris. A light coating of metallic particles was noted on all the MCDs; according to the Boeing 737-600/700/800/900 AMM they were considered serviceable.

With the fan cowls and TR opened, the outside of the engine was visible. There was no evidence of an undercowl fire or fuel leaks; there was evidence of an oil leak. The CTAI duct located at about the 1:00 o'clock position was fractured just forward of the anti-ice valve (**PHOTO 27**). The ECU remained attached to the engine and all four integral mounting supports were intact. The ECU was secured to the fan case by four support brackets; all the brackets were present and secured except for the upper forward support bracket which was loosely attached to the fan case. The engine oil tank filler cap was found dislodged from the tank but was still secured by its lanyard. The AGB was fractured near the IDG installation pad (**PHOTO 28**). Oil was found leaking from the AGB at the fracture location. An AGB drain line was found disconnected from its b-nut; however, the b-nut and the tube threads did not appear to have any damage. The starter air duct, located at about the 9:30-10:00 o'clock position, was fractured.

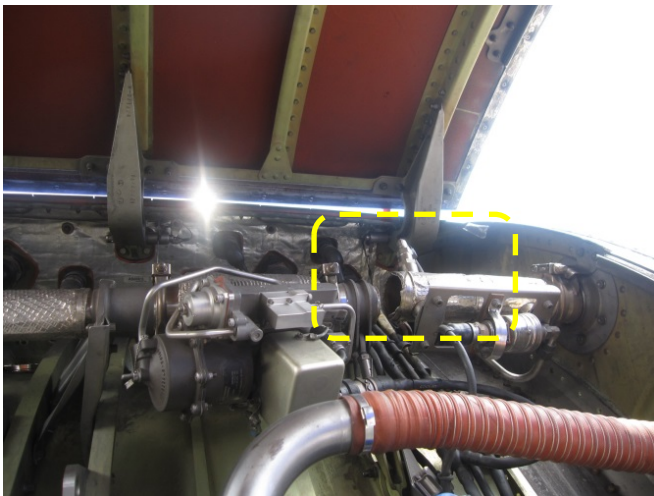


PHOTO 27: CTAI DUCT SEPARATION

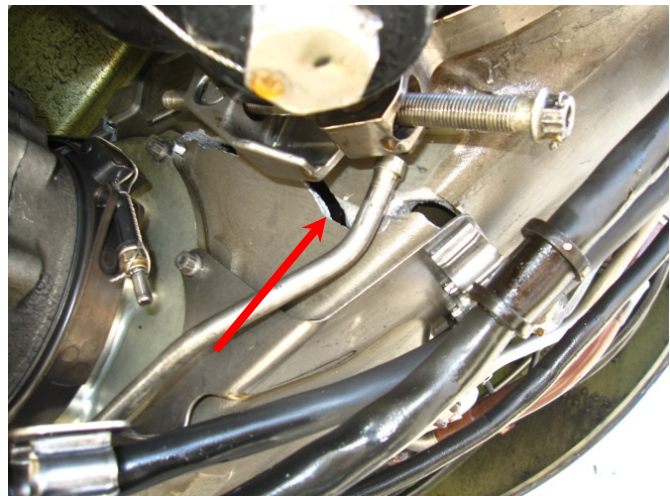


PHOTO 28: LEFT ENGINE AGB FRACTURE

After the general engine inspection was completed, the fan blade hardware was removed, and a more detailed on-scene examination of their condition was performed. SWA provided the last shop visit records that listed the fan blade PNs, SNs, and installation position from the last shop visit (**ATTACHMENT 1**) and this list was compared with actual information stamped on the fan blade root. The fan blade information provided from that last shop visit matched the PNs and SNs stamped on the blade roots and were in the as-specified position. All the fan blade dovetails were examined with a 10x magnification sight-glass and no other fan blades had any observed indications of a crack besides fan blade No. 23; the failed blade. **TABLE 1** provides the on-scene condition documentation of each fan blade and their associated hardware. Two small loose pieces of fan blade hardware were recovered; one piece was found wedged in the fan OGVs at about the 10:00 o'clock position and another wedged in the inlet at about the 2:00 o'clock position.

TABLE 1: FAN BLADE HARDWARE CONDITION

Coding: **BLUE** is a new configuration shim, **RED** is the failed blade at the root
 LE – Leading Edge, TE – Trailing Edge, PS – Pressure Side, SS – Suction Side

Position	Fan Blade Part Markings	Serial Number (Photo No.1)	Condition	Picture
1	PN 340-001-022-0 PN 340-001-027-0 F0301 - Snecma manufacturer's identification number 5 different MWs	BA786555-H (PHOTO 29)	Airfoil: LE and TE exhibited hard impact damage, tears, and missing material. LE tip curled in the direction opposite rotation. Blade tip intact; scuffing on airfoil pressure side. Blade Root: Loss of coating on PS towards the TE. Shim: Still installed and intact, no missing material. PN 340-252-306-0 (old configuration). Spacer: No damage. PN 340-001-261-0 (F0301, F0732, and 07W25). Platform: Intact, seals in good condition, was reworked PN 340-001-814-0 PN 340-001-816-0 (F0301, FA9L6, and DL272004-5).	 PHOTO 29: FAN BLADE 1
2	PN 340-001-022-0 PN 340-001-027-0 F0301 5 different MWs	BA725664-P (PHOTO 30)	Airfoil: LE tip curled in direction of rotation and torn. LE hard impact damage and edge missing material. TE dented; no missing material. Scuffing on airfoil PS. Blade Root: Loss of coating on PS towards the TE. Loss of coating on SS towards LE. Shim: Present and intact. No missing material. PN 340-252-306-0 (old configuration). Spacer: No damage. PN 340-001-261-0 (F0301, F0732, and 07W28). Platform: Intact but cracked on LE, seals in good condition, was reworked PN 340-001-814-0 PN 340-001-816-0 (F0301, FA9L6, and DL240321-8).	 PHOTO 30: FAN BLADE 2
3	PN 340-001-022-0 PN 340-001-027-0 F0301 5 different MWs	BA725666-M (PHOTO 31)	Airfoil: LE tip curled in direction opposite of rotation. LE and TE hard impact damage and tears. Missing material on TE tip. Scuffing on airfoil pressure side. Blade Root: Loss of coating on PS towards the TE. Loss of coating on SS towards LE. More coating loss on SS than PS. Shim: Present and intact. No missing material. PN 340-252-306-0 (old configuration). Spacer: No damage. PN 340-001-261-0 (F0301, F0732, and 07W28). Platform: Intact, LE seal missing material, no rework original PN 340-001-816-0 (F0301, FA9L6, and DL272113-K).	 PHOTO 31: FAN BLADE 3

TABLE 1: FAN BLADE HARDWARE CONDITION

Coding: **BLUE** is a new configuration shim, **RED** is the failed blade at the root
 LE – Leading Edge, TE – Trailing Edge, PS – Pressure Side, SS – Suction Side

Position	Fan Blade Part Markings	Serial Number (Photo No.1)	Condition	Picture
4	PN 340-001-022-0 PN 340-001-027-0 F0301 5 different MWs	BA788424-5 (PHOTO 32)	Airfoil: Tip curl in the direction opposite rotation; LE and TE hard body impact and missing material. TE tip missing. Minor scuffing on airfoil pressure side. Blade Root: Loss of coating on SS towards LE. Shim: Present and intact. No missing material. PN 340-252-306-0 (old configuration). Spacer: No damage. PN 340-001-261-0 (F0301, F0732, and 07W28). Platform: Intact, LE seal missing material, no rework original PN 340-001-816-0 (F0301, FA9L6, and DL272123-4).	 PHOTO 32: FAN BLADE 4
5	PN 340-001-022-0 PN 340-001-027-0 F0301 5 different MWs	BA725665-N (PHOTO 33)	Airfoil: Tip curl in direction opposite rotation and missing tip material near LE. LE and TE tears. Scuffing on airfoil PS. Blade Root: No visual coating loss. Shim: Present and intact. No missing material. PN 340-252-306-0 (old configuration). Spacer: No damage. PN 340-001-261-0 (F0301, F0732, and 07W25). Platform: LE missing material. TE deformed, seal missing same area as LE missing material. Was reworked PN 340-001-814-0 PN 340-001-816-0 (F0301, FA9L6, DL227301-N and 611307).	 PHOTO 33: FAN BLADE 5
6	PN 340-001-022-0 PN 340-001-027-0 F0301 5 different MWs	BA725668-K (PHOTO 34)	Airfoil: Tip curled in direction opposite of rotation. Entire tip is missing and jagged. LE and TE hard impact damage and missing material. TE dented and torn. Scuffing on airfoil PS. Blade Root: No visual coating loss. Shim: Present and intact. No missing material. PN 340-252-306-0 (old configuration). Spacer: No damage. PN 340-001-261-0 (F0301, F0732, and 07W25). Platform: LE and TE missing material. Seal missing in same areas as missing material. No rework original PN 340-001-816-0 (F0301, FA9L6, and DL272113-K).	 PHOTO 34: FAN BLADE 6

TABLE 1: FAN BLADE HARDWARE CONDITION

Coding: **BLUE** is a new configuration shim, **RED** is the failed blade at the root
 LE – Leading Edge, TE – Trailing Edge, PS – Pressure Side, SS – Suction Side

Position	Fan Blade Part Markings	Serial Number (Photo No.1)	Condition	Picture
7	PN 340-001-022-0 PN 340-001-027-0 F0301 5 different MWs	BA824504-V (PHOTO 35)	Airfoil: Tip curled in direction opposite of rotation. Entire tip is missing and jagged. LE and TE hard impact damage and missing material. Minor scuffing on airfoil PS. Blade Root: Loss of coating on PS towards the TE. Shim: Suction side LE is torn but still attached. Rest of the shim is present. No missing material. PN 340-252-306-0 (old configuration). Spacer: No damage. PN 340-001-261-0 (F0301, F0732, and 07W25). Platform: LE missing material. TE cracked. LE seal missing material at two spots. No rework original PN 340-001-816-0 (F0301, FA9L6, and DL272113-K).	 PHOTO 35: FAN BLADE 7
8	PN 340-001-022-0 PN 340-001-027-0 F0301 5 different MWs	BA786510-R (PHOTO 36)	Airfoil: Tip curled in direction opposite of rotation. Entire tip is missing. LE and TE hard impact damage and missing material. TE edge torn. Blade Root: Loss of coating on PS towards the TE. Shim: SS LE is torn but still attached. Rest of the shim is present. No missing material. PN 340-252-306-0 (old configuration). Spacer: No damage. PN 340-001-261-0 (F0301, F0732, and 07W25). Platform: Intact but LE. Seal missing material. No rework original PN 340-001-816-0 (F0301, FA9L6, and DL272113-K).	 PHOTO 36: FAN BLADE 8
9	PN 340-001-022-0 PN 340-001-027-0 F0301 5 different MWs	BA824502-Y BA824502-R (the R was X-out and replaced with the Y) (PHOTO 37)	Airfoil: Tip curled in the direction opposite rotation. Entire tip missing and jagged. LE near tip has what appears to be acoustic material wedged into the tear. TE hard impact damage, torn, and missing material. Blade Root: Loss of coating on PS towards the TE. Shim: Present and intact. No missing material. PN 340-252-306-0 (old configuration). Spacer: No damage. PN 340-001-261-0 (F0301, F0732, and 07W28). Platform: LE aft is cracked and LE seal is missing material. No rework original PN 340-001-816-0 (F0301, FA9L6, and DL272113-K).	 PHOTO 37: FAN BLADE 9

TABLE 1: FAN BLADE HARDWARE CONDITION

Coding: **BLUE** is a new configuration shim, **RED** is the failed blade at the root
 LE – Leading Edge, TE – Trailing Edge, PS – Pressure Side, SS – Suction Side

Position	Fan Blade Part Markings	Serial Number (Photo No.1)	Condition	Picture
10	PN 340-001-022-0 PN 340-001-027-0 F0301 5 different MWs	BA824435-B (PHOTO 38) Last shop visit records had a -8 instead of -B. This is thought to be a clerical error. See description of the dash number (checksum suffix)	Airfoil: Tip curled in the direction opposite rotation. Large portion of tip missing, more so on the TE. LE and TE hard impact damage and torn in several spots. Minor scuffing on airfoil PS. Blade Root: Loss of coating on PS towards the TE. Shim: Present and intact. No missing material. PN 340-252-306-0 (old configuration) Spacer: No damage. PN 340-001-261-0 (F0301, F0732, and 07W25). Platform: LE and TE aft missing material. Seal missing material LE aft. Was reworked PN 340-001-814-0 PN 340-001-816-0 (F0301, FA9L6, and DL240321-8).	 PHOTO 38: FAN BLADE 10
11	PN 340-001-022-0 PN 340-001-027-0 F0301 5 different MWs	BA770956-G (PHOTO 39)	Airfoil: Tip curled in the direction opposite rotation. Almost entire tip missing and jagged. TE hard impact damage, missing material, and torn. Blade Root: Loss of coating on SS towards the LE. Shim: Present and intact. PN 340-252-306-0 (old configuration). Spacer: No damage. PN 340-001-261-0 (F0301, F0732, and 07W28). Platform: Crack LE aft. Slight deformation on TE aft. Seal missing on LE forward. No rework original PN 340-001-816-0 (F0301, FA9L6, and DL272113-K).	 PHOTO 39: FAN BLADE 11
12	PN 340-001-022-0 PN 340-001-027-0 F0301 5 different MWs	BA746470-U (PHOTO 40)	Airfoil: Tip curled direction opposite rotation, LE tip torn, tip TE missing, TE hard impact damage and torn, and minor scuffing on airfoil pressure side. Blade Root: No visual coating loss Shim: Present and intact. PN 340-252-306-0 (old configuration) Spacer: No damage, PN 340-001-261-0 (F0301, F0732, and 07W28). Platform: Missing material leading and TE aft, seal missing material LE forward, and no rework original PN 340-001-816-0 (F0301, FA9L6, and DL272113-K).	 PHOTO 40: FAN BLADE 12

TABLE 1: FAN BLADE HARDWARE CONDITION

Coding: **BLUE** is a new configuration shim, **RED** is the failed blade at the root
 LE – Leading Edge, TE – Trailing Edge, PS – Pressure Side, SS – Suction Side

Position	Fan Blade Part Markings	Serial Number (Photo No.1)	Condition	Picture
13	PN 340-001-022-0 PN 340-001-027-0 F0301 5 different MWs	BA786481-6 (PHOTO 41)	Airfoil: Tip curled in direction opposite rotation. TE tip missing. LE tip curled and torn. TE hard impact damage, torn and missing material. Scuffing on airfoil PS. Blade Root: Loss of coating on PS towards the TE. Shim: Present and intact. PN 340-252-306-0 (old configuration). Spacer: No damage. PN 340-001-261-0 (F0301, F0732, and 07W25). Platform: Cracked on LE and TE aft. Seal missing material LE forward. No rework original PN 340-001-816-0 (F0301, FA9L6, and DL272113-K).	 PHOTO 41: FAN BLADE 13
14	PN 340-001-022-0 PN 340-001-027-0 F0301 5 different MWs	BA783812-6 (PHOTO 42)	Airfoil: Minor tip curl in the direction opposite rotation. LE impact and missing material. TE tip torn. Minor scuffing of airfoil PS. Blade Root: Loss of coating on PS towards the TE. Shim: Present and intact. PN 340-252-306-0 (old configuration). Spacer: No damage. PN 340-001-261-0 (F0301, F0732, and 07W28). Platform: Cracked TE aft. Seals in place and good condition. Was reworked PN 340-001-814-0 PN 340-001-816-0 (F0301, FA9L6, DL209709-11 and 5123006).	 PHOTO 42: FAN BLADE 14
15	PN 340-001-022-0 PN 340-001-027-0 F0301 5 different MWs	BA786482-5 (PHOTO 43)	Airfoil: Minor LE tip bent. TE tip bent in the direction opposite rotation. TE hard impact damage, torn, and missing material. Blade Root: Significant loss of coating on PS towards the TE. Shim: Damage on LE PS. Cracked but in intact. PN 340-252-306-0 (old configuration). Spacer: No damage. PN 340-001-261-0 (F0301, F0732, and 07W27). Platform: Intact and missing seal material LE forward. No rework original PN 340-001-816-0 (F0301, FA9L6, and DL272123-4).	 PHOTO 43: FAN BLADE 15

TABLE 1: FAN BLADE HARDWARE CONDITION

Coding: **BLUE** is a new configuration shim, **RED** is the failed blade at the root
 LE – Leading Edge, TE – Trailing Edge, PS – Pressure Side, SS – Suction Side

Position	Fan Blade Part Markings	Serial Number (Photo No.1)	Condition	Picture
16	PN 340-001-022-0 PN 340-001-027-0 F0301 5 different MWs	BA783814-4 (PHOTO 44)	Airfoil: Slight bending of the LE tip in the direction opposite rotation. TE dented and torn. Blade Root: No visual coating loss Shim: Present and intact. PN 340-252-306-0 (old configuration) Spacer: No damage. PN 340-001-261-0 (F0301, F0732, and 07W28). Platform: Platform intact and undamaged. Missing seal LE edge forward. No rework original PN 340-001-816-0 (F0301, FA9L6, and DL272113-K).	 PHOTO 44: FAN BLADE 16
17	PN 340-001-022-0 PN 340-001-027-0 F0301 5 different MWs	BA783810-8 (PHOTO 45)	Airfoil: Slight tip rub and rolled material. LE tip impact damage and missing material. LE hard impact damage and missing material. TE torn near tip. Blade Root: Significant loss of coating on PS towards the TE. Loss of coating on SS towards the LE. Shim: Present and intact. PN 340-252-306-0 (old configuration). Spacer: No damage. PN 340-001-261-0 (F0301, F0732, and 07W25). Platform: Intact and no missing seal material. Was reworked PN 340-001-814-0 PN 340-001-816-0 (F0301, FA9L6, and DL227283-H).	 PHOTO 45: FAN BLADE 17
18	PN 340-001-022-0 PN 340-001-027-0 F0301 5 different MWs	BA783811-7 (PHOTO 46)	Airfoil: LE tip missing chunk (4 x 2-inches) and airfoil curled in direction opposite rotation. TE hard impact damage and missing material. Slight scuffing and imprint consistent with acoustic paneling geometry on airfoil PS. Blade Root: Loss of coating on PS towards the TE. Shim: Present and intact. PN 340-252-306-0 (old configuration). Spacer: No damage. PN 340-001-261-0 (F0301, F0732, and 07W25). Platform: Intact and missing seal LE forward. No rework original PN 340-001-816-0 (F0301, FA9L6, and DL272110-K).	 PHOTO 46: FAN BLADE 18

TABLE 1: FAN BLADE HARDWARE CONDITION

Coding: **BLUE** is a new configuration shim, **RED** is the failed blade at the root
 LE – Leading Edge, TE – Trailing Edge, PS – Pressure Side, SS – Suction Side

Position	Fan Blade Part Markings	Serial Number (Photo No.1)	Condition	Picture
19	PN 340-001-022-0 PN 340-001-027-0 F0301 5 different MWs	BA767285-L (PHOTO 47)	Airfoil: Minor tip curl LE in the direction opposite rotation. Scuffing and imprint consistent with acoustic panel geometry on airfoil PS. Blade Root: Loss of coating on SS towards the LE. Shim: Present and intact. PN 340-252-306-0 (old configuration). Spacer: No damage. PN 340-001-261-0 (F0301, F0732, and 07W27). Platform: Intact and missing seal material on LE forward. No rework original PN 340-001-816-0 (F0301, FA9L6, and DL272121-6).	 PHOTO 47: FAN BLADE 19
20	PN 340-001-022-0 PN 340-001-027-0 F0301 5 different MWs	BA725669-J (PHOTO 48)	Airfoil: LE tear near tip. LE tip corner missing. Scuffing on airfoil PS. Blade Root: Loss of coating on PS towards the TE. Shim: Present and intact. PN 340-252-306-0 (old configuration). Spacer: No damage. PN 340-001-261-0 (F0301, F0732, and 07W28). Platform: About 80% LE missing material and entire LE seal missing. No rework original PN 340-001-816-0 (F0301, FA9L6, and DL272113-K).	 PHOTO 48: FAN BLADE 20
21	PN 340-001-022-0 PN 340-001-027-0 F0301 5 different MWs	BA824505-U (PHOTO 49)	Airfoil: Tip curled in the direction opposite rotation. LE tip missing material. Tip rub along entire surface. TE distorted. Scuffing on airfoil PS. Blade Root: No visual coating loss. Shim: Heavily damaged. Small amount of material missing on PS and large portion torn. SS 80-90% is missing. PN 340-252-306-0 (old configuration). Spacer: No damage. PN 340-001-261-0 (entire PN no legible but the 261 can be seen, all other marking not legible). Platform: Entire LE missing material and seal. TE missing 60% material and seal. Aft pin location lug is cracked. No rework original PN 340-001-816-0 (F0301, FA9L6, and DL272113-K).	 PHOTO 49: FAN BLADE 21

TABLE 1: FAN BLADE HARDWARE CONDITION

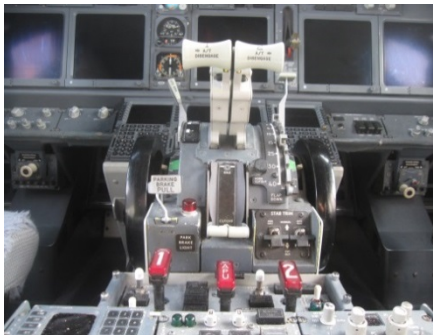
Coding: **BLUE** is a new configuration shim, **RED** is the failed blade at the root
 LE – Leading Edge, TE – Trailing Edge, PS – Pressure Side, SS – Suction Side

Position	Fan Blade Part Markings	Serial Number (Photo No.1)	Condition	Picture
22	PN 340-001-022-0 PN 340-001-027-0 F0301 5 different MWs	BA767290-A (PHOTO 50)	Airfoil: Tip curled in the direction of rotation and missing material. LE curled in the direction opposite rotation. LE missing material and torn. LE, distorted. Radial imprint marks on SS of airfoil. TE impact damage, missing material and torn. Significant scuffing on airfoil pressure side. Blade Root: Significant loss of coating on PS towards the TE and minor loss of coating on SS towards the TE. Shim: LE and TE missing material. PN 340-252-307-0 (new configuration). Spacer: No damage. PN 340-001-261-0 (F0301, F0732, and 07W25). Platform: Only piece that remained was the forward most pin lug. Unable to determine PN configuration.	 PHOTO 50: FAN BLADE 22
23	PN 340-001-022-0 PN 340-001-027-0 F0301 5 different MWs	BA824478-6 (PHOTO 51)	Airfoil: Airfoil missing, all that remains is the blade root. Blade Root: With shim removed at lab confirmed loss of coating on the SS towards the LE. Shim: Present and intact. PN 340-252-307-0 (new configuration). Spacer: No damage. PN 340-001-261-0 (F0301, F0732, and 07W27). Platform: Present and in good condition. Was reworked PN 340-001-814-0 PN 340-001-816-0 (F0301, FA9L6, and DL227283-H).	 PHOTO 51: FAN BLADE 23
24	PN 340-001-022-0 PN 340-001-027-0 F0301 5 different MWs	BA824482-U (PHOTO 52)	Airfoil: Tip curled in the direction opposite rotation. LE hard impact damage and missing material (1.5- x 3-inch). TE impact damage, missing material (1- x 2-inches) and torn. Scuffing on pressure side of airfoil. Blade Root: Loss of coating on PS towards the TE and loss of coating on SS towards the LE. Shim: Present and intact. PN 340-252-307-0 (new configuration). Spacer: No damage. PN 340-001-261-0 (F0301, F0732, and 07W27). Platform: Present, undamaged and in good condition. Was reworked PN 340-001-814-0 PN 340-001-816-0 (F0301, FA9L6, and DL227254-3).	 PHOTO 52: FAN BLADE 24

2.3 COCKPIT AND CABIN DOCUMENTATION

Examination of the cockpit revealed that both thrust levers (throttles) were against the aft stops, both fuel control cutoff levers were in the down and CUTOFF position (**PHOTO 53**), and the left engine fire handle was pulled up and straight (not turned). The passenger oxygen activation toggle switch was still gated and safety wired, consistent with the passenger oxygen system not being activated manually (**PHOTO 54**).

The cabin (**PHOTO 55**), lavatory, and flight attendant oxygen masks had all deployed. Each row has eight masks and four oxygen generators; each aisle-row therefore has four masks and two oxygen generators. A total of seven oxygen generators were not actuated. Two cabin masks and their associated hose were found disconnected from their coupling.



**PHOTO 53: THROTTLES,
CUTOFF LEVERS, AND FIRE
HANDLES**



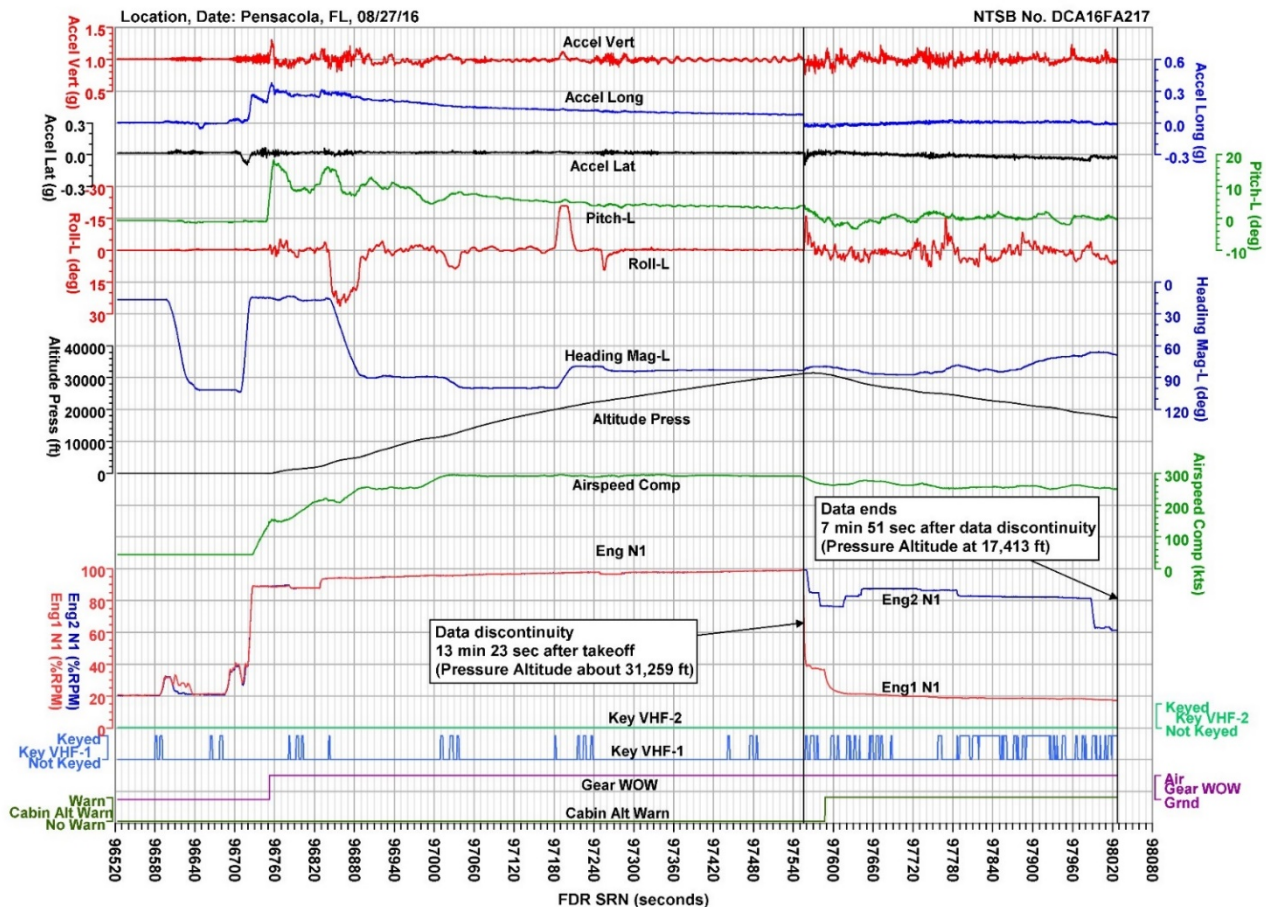
**PHOTO 54: PASSENGER
OXYGEN SWITCH**



**PHOTO 55: CABIN OXYGEN
MASKS DEPLOYED**

2.4 AIRPLANE ELECTRICAL CHECK

The FDR was shipped to the NTSB's Vehicle Recorders Division in Washington DC for readout (See the FLIGHT DATA RECORDER SPECIALISTS FACTUAL REPORT for details). A preliminary review of the FDR data showed a momentary power interruption at 13 minutes 23 seconds after takeoff and about 7 minutes 51 seconds later (after the momentary power interruption) the FDR ceased to record data (**PLOT 1**).



Revised: 28 March 2018

Southwest Airlines, Boeing B737-700, N766SW

National Transportation Safety Board

PLOT 1: FDR PLOT SHOWING FAN BLADE EVENT, POWER INTERRUPTION, AND RECORDER STOPPAGE

According to Boeing, the FDR is normally powered from the Transfer Bus No. 1, which is primarily powered by the No. 1 engine generator. The FDR starts recording if electrical power is available on Transfer Bus No. 1 and either engine start lever is in the commanded-ON position or the airplane is in the air. Upon loss of the No. 1 engine and its generator, electrical power to the Transfer Bus No. 1 is automatically switched to either the auxiliary power unit (APU) or the No. 2 engine generator. At the time of the fan blade failure, the APU was not in operation. A combination of the loss of the No. 1 engine generator, the loss of electrical power to the Transfer Bus No. 1, and transfer of electrical power to the Transfer Bus No. 1 from the No. 2 engine generator; resulted in the FDR momentary loss of data discontinuity. According to the pilot statements, the APU was eventually started but APU power was not able to be connected to the airplane. The attempt to connect APU power to Transfer Bus No.1 would result in the Transfer Bus No. 1 dropping off-line from the No. 2 engine generator to then receive power from the APU. The APU was unable to connect to the Transfer Bus No. 1 resulting in the FDR recording ceasing.

To identify and isolate those components that may have caused or contributed to the power loss to the FDR, a series of electrical checks were performed. First, a check of the status of the battery was performed. The battery switch in the cockpit overhead panel was selected from OFF to the ON position and the digital voltage readout was 20 volts, which according to the SWA mechanic performing the test, would be insufficient to start the APU.

Following the battery check, a test was performed to see if the airplane would accept ground power (GP) and APU power and if the standby power (battery) was operating properly. The GP cart was connected to the airplane, its generator was turned ON, and the GP voltage and frequency were checked on the cockpit overhead panel and were found to be satisfactory according to the SWA mechanic performing the tests. After everything was stabilized for a few minutes, the SWA mechanic selected the GP switch on the cockpit overhead panel to the ON position. Then the SWA mechanic started the APU and it started normally utilizing GP.¹³ During the entire test, the GP cart was still physically connected to the airplane. After the APU start, GP was deselected (by moving GP switch to the OFF position) and the APU generator was manually turned ON independently to each electrical bus (Transfer Buses Nos. 1 and 2), then to both buses simultaneously. This was done by selecting the APU generator switches on the electrical panel in the cockpit as follows:

- 1) Left bus ON - right bus OFF
- 2) Left bus OFF - right bus OFF - Electrical System Powered by Battery (standby power)
- 3) Left bus OFF - right bus ON
- 4) Left bus OFF - right bus OFF - Electrical System Powered by Battery (standby power)
- 5) Left bus ON - right bus ON
- 6) Left bus OFF - right bus OFF - Electrical System Powered by Battery (standby power)

In all configurations, standby power functioned as intended and the APU generator powered each bus independently or both buses simultaneously as intended. No anomalies or faults were found.

The bus power control unit (BPCU), box G15, located in the electrical and electronics compartment just aft of the nose landing gear wheel well, was found to have a fault light illuminated (**PHOTO 56**). Box G15 was reset by holding the BPCU test button for three seconds; after which the fault light extinguished. The standby power and APU generator test were conducted again (same procedures listed above) and again in all configurations. Standby power functioned as intended and the APU generator powered each bus independently or both buses simultaneously as intended. No anomalies or faults were found and the BPCU fault light did not re-illuminate at any time during the tests.



PHOTO 56: BUS POWER CONTROL UNIT WITH FAULT LIGHT ILLUMINATED

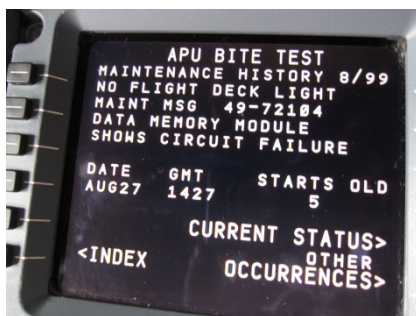
Three generator control units (GCU), (APU GCU, GCU-1, and GCU-2 are all located in the electrical and electronics compartment) were checked for fault lights. On the APU GCU box, the GCB/APB (Generator Control Breaker/Auxiliary Power Breaker) FAULT light was illuminated. On the GCU-1, the IDG FAULT light was illuminated. On GCU-2 box, no FAULT light was illuminated. The faults were cleared on the GCU-1 and the APU and the standby power test was performed again. The

¹³ No engine power (engine generator power) was available during this test.

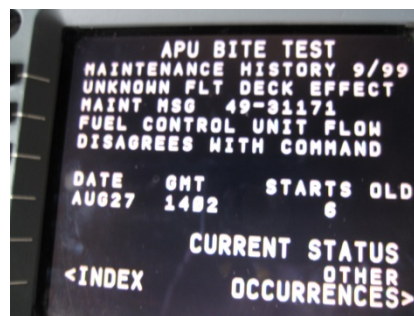
only difference from the previous tests was that after the APU was started, the APU generator available light illuminated, but this time the GP cart was disconnected from the airplane. In all configurations, standby power functioned as intended and the APU generator powered each bus independently or both buses simultaneously as intended. No anomalies or faults were found. Since the various test scenarios were not able to duplicate the observed in-flight electrical anomaly that caused the FDR to lose power, the APU GCU, BPGU, and the APB were removed for further evaluation (See Section 12.0 ADDITIONAL ELECTRICAL HARDWARE TESTING for details).

Review of the FDR data showed multiple APU faults; therefore, the APU bite test was performed using the first officer's Flight Management Computer (FMC) control display unit (CDU). A series of faults were noted and of interest were the following faults from August 27 (event date):

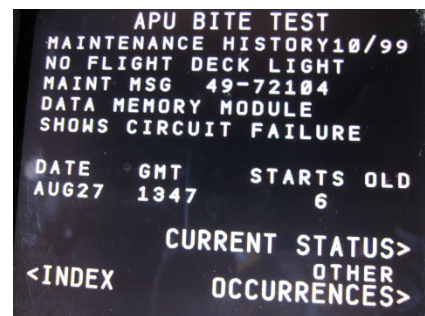
- 1) 1427 Greenwich Mean Time (GMT) maintenance message was 49-72104 '*data memory module shows circuit failure*'. This was approximately the time the APU was started during the incident flight (**PHOTO 57**).
- 2) 1402 GMT maintenance message was 49-31171 '*fuel control unit flow disagrees with command*' (**PHOTO 58**).
- 3) 1347 GMT maintenance message was 49-72104 '*data memory module shows circuit failure*' (**PHOTO 59**).



**PHOTO 57: APU FAULT –
MEMORY MODULE CIRCUIT
FAILURE 1427 GMT**



**PHOTO 58: APU FAULT – FUEL
FLOW DISAGREE FAILURE
1402 GMT**



**PHOTO 59: APU FAULT –
MEMORY MODULE CIRCUIT
FAILURE 1347 GMT**

2.5 AIRFRAME EXAMINATION

FIGURES 15-16, and **TABLE 2** provides the location and structural damage description for the event airplane.

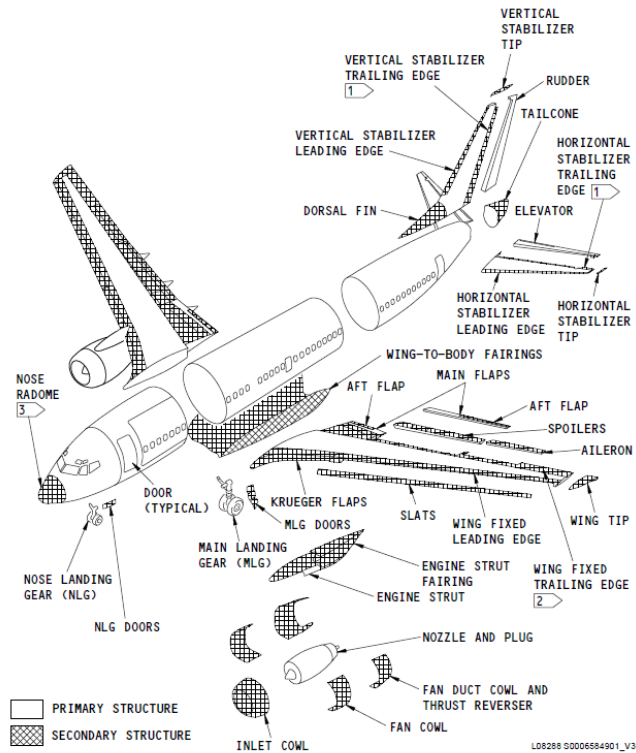


FIGURE 15: AIRPLANE STRUCTURAL LOCATION AND NOMENCLATURE

FIGURE COURTESY OF BOEING

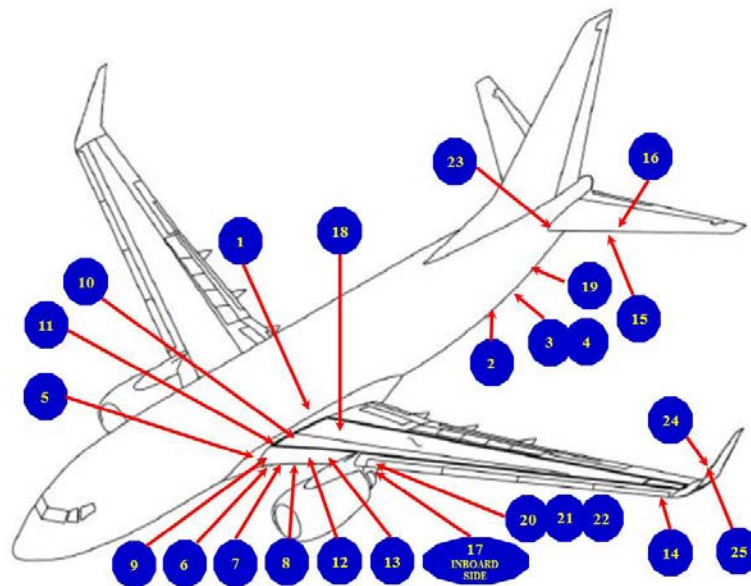


FIGURE 16: AIRPLANE STRUCTURAL DAMAGE LOCATIONS

FIGURE COURTESY OF BOEING

TABLE 2: AIRPLANE STRUCTURAL DAMAGE

Item Number	Location	Part Number (PHOTO NO.)	Condition	Photo
1	Section 43 between body station (BS) 500D and BS 520. Between stringers 13L and 16L. Between windows 11 and 12 at seat row 8.	143A3150-63 (S4L-S14L), 143A3210-19 (S14L-S24L) (PHOTO 60)	Skin is dented inboard between stringers 13L and 16L.	 PHOTO 60: FUSELAGE DAMAGE
		(PHOTO 61)	The yellow tape is the location of the fuselage skin puncture.	 PHOTO 61: FUSELAGE DAMAGE
1A	Section 43 between BS 500D and BS 520. Between stringers 13L and 14L.	143A3150-63 (PHOTO 62)	Skin puncture between stringers 13L and 14L. The puncture is about 16-inches-long x 5-inches-wide. There was damage to the air ducting behind the skin. No damage was found on the interior liner. No foreign material was found in the cabin area. Picture is without the air duct.	 PHOTO 62: FUSELAGE DAMAGE
1A	Section 43 between BS 500D and BS 520. Between stringers 13L and 14L.	(PHOTO 63)	Damage to the air duct at skin puncture location from picture 62.	 PHOTO 63: FUSELAGE DAMAGE

TABLE 2: AIRPLANE STRUCTURAL DAMAGE

Item Number	Location	Part Number (PHOTO NO.)	Condition	Photo
1B	Section 43 BS 520 between stringers 13L and 14L.	143A1200-145 (PHOTO 64)	The frame web connection to the shear tie is cracked the full length between stringers 13L and 14L. The shear tie is bent. There is a second crack in the frame web at the near the end of the failsafe chord. The visible portion of this crack is about 0.25-inches, but it extends an unknown length underneath the failsafe chord.	 PHOTO 64: FUSELAGE DAMAGE
1C	Section 43 BS 500D to 520. Stringers 13L to 16L.	143A3150-63, 143A3210-19 (PHOTO 65)	Stringers 13L to 16L are bent inboard. Stringer 15L is bent more, about 0.1-inch inboard displacement.	 PHOTO 65: FUSELAGE DAMAGE
1C	Section 43 BS 500D to 520. Stringer 15L.	(PHOTO 66)	See above description.	 PHOTO 66: FUSELAGE DAMAGE
1C	Section 43 BS 520. Stringer 14L.	143A1215 (PHOTO 67)	The stringer clip is buckled.	 PHOTO 67: FUSELAGE DAMAGE

TABLE 2: AIRPLANE STRUCTURAL DAMAGE

Item Number	Location	Part Number (PHOTO No.)	Condition	Photo
1C	Section 43 BS 520. Stringer 15L.	143A1215 (PHOTO 68)	The stringer clip is buckled.	 PHOTO 68: FUSELAGE DAMAGE
1C	Section 43 BS 520. Stringer 16L.	143A1215 (PHOTO 69)	The stringer clip is buckled.	 PHOTO 69: FUSELAGE DAMAGE
1C	Section 43 BS. 500D Stringer 14L	143A1225 (PHOTO 70)	The stringer clip is buckled.	 PHOTO 70: FUSELAGE DAMAGE
2	Section 47 BS 890-895. Between Stringers 24L and 25L.	(PHOTO 71)	5-inches-long skin gouge. Depth is about 0.020-inches at its deepest point.	 PHOTO 71: FUSELAGE DAMAGE

TABLE 2: AIRPLANE STRUCTURAL DAMAGE				
Item Number	Location	Part Number (PHOTO NO.)	Condition	Photo
3	Section 47 BS 948, above and below Stringer 23L.	147A3110-3 (S14L-S23L), 147A3400-4 (S23L-S23R) (PHOTO 72)	Multiple areas of minor surface gouges.	 PHOTO 72: FUSELAGE DAMAGE
4	Section 47 Fuselage Skin, BS 906-967. Stringer 25	147A3400-4 (PHOTO 73)	A long narrow region of minor surface gouging was evident.	 PHOTO 73: FUSELAGE DAMAGE
19	Section 47, BS 990.	417T2093-10 (PHOTO 74)	Drain mast impact damage. There was a puncture about 3-inches-long x 1.5-inches-wide. A small piece of metal was found inside the drain mast.	 PHOTO 74: DRAIN MAST DAMAGE
19		(PHOTO 75)	See above description.	 PHOTO 75: DRAIN MAST DAMAGE

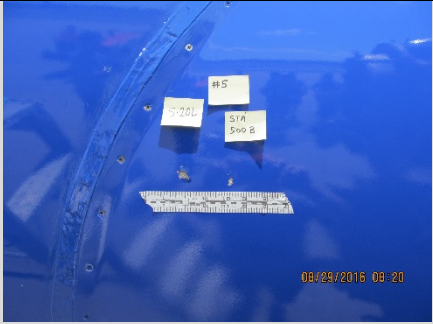
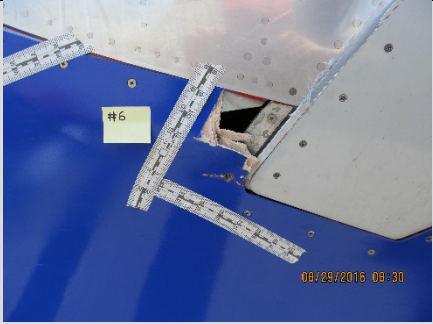
TABLE 2: AIRPLANE STRUCTURAL DAMAGE				
Item Number	Location	Part Number (PHOTO NO.)	Condition	Photo
19		Unknown (PHOTO 76)	A 1.75-inches x 1-inch piece of metal was found inside the drain mast. The thickness was about 0.049-inches. The piece was not consistent with fan blade, fan blade platform, fuselage, or drain mast material.	 PHOTO 76: DEBRIS IN DRAIN MAST
19		(PHOTO 77)	See above description.	 PHOTO 77: DEBRIS IN DRAIN MAST
5	Wing-to-Body Fairing, BS 500B, stringer 20L. Access panel 191AL.	149A7151-1 (PHOTO 78)	Gouges in the wing to body fairing panel.	 PHOTO 78: WING-TO-BODY FAIRING DAMAGE
6	Wing-to-Body Fairing, BS 540. Access panel 191FL.	149A7155-1 (PHOTO 79)	About 2-inches x 2-inches corner of the wing to body fairing was missing.	 PHOTO 79: WING-TO-BODY FAIRING DAMAGE

TABLE 2: AIRPLANE STRUCTURAL DAMAGE

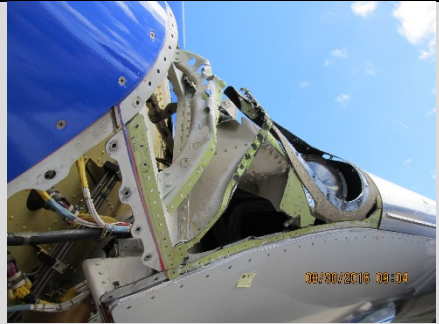
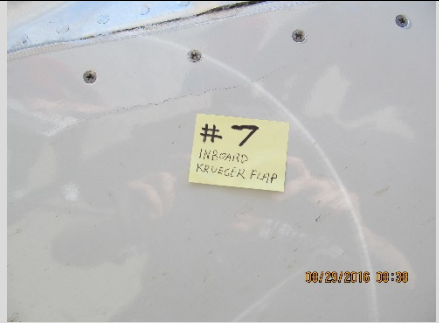
Item Number	Location	Part Number (PHOTO NO.)	Condition	Photo
9	Side of Body to Krueger station 125.75	(PHOTO 80)	Impact damage to the strakelet assembly. The LE skin and support ribs are dented. The light lens was shattered.	 PHOTO 80: STRAKELET DAMAGE
9		(PHOTO 81) (After removal for the damage skin)	The strakelet substructure was also damaged. No damage was found to wing front spar or fuselage structure behind the strakelet assembly.	 PHOTO 81: STRAKELET DAMAGE
9	Krueger station 125.75	116A3211-5 (PHOTO 82)	A 6-inch crack was found in the outboard clevis leg of the strakelet seal rib.	 PHOTO 82: STRAKELET DAMAGE
7	Inboard Krueger flap	114A1221-3 (PHOTO 83)	Approximately 5-inches-long crack on the inboard Krueger flap. Crack is about 10-inches from the inboard edge and 1-inch aft of the forward edge.	 PHOTO 83: KRUEGER FLAP DAMAGE

TABLE 2: AIRPLANE STRUCTURAL DAMAGE				
Item Number	Location	Part Number (PHOTO NO.)	Condition	Photo
8	Outboard Krueger flap	114A1111-3 (PHOTO 84)	Paint transfer and minor surface gouges 14-inches from the inboard edge	 PHOTO 84: WING DAMAGE
10	Krueger station 125.75. This panel is above and aft of the strakelet assembly.	116A1210-3 (PHOTO 85)	Screws have pulled through the panel.	 PHOTO 85: WING DAMAGE
11	Krueger station 125.75	116A4410-7 (PHOTO 86)	The rivets attaching the LE skins were sheared.	 PHOTO 86: WING DAMAGE
10 and 11		(PHOTO 87)	See above items 10 and 11 descriptions.	 PHOTO 87: WING DAMAGE

TABLE 2: AIRPLANE STRUCTURAL DAMAGE				
Item Number	Location	Part Number (PHOTO NO.)	Condition	Photo
12	LE skin inboard of the engine.	116A4410-7 (PHOTO 88)	Multiple minor surface scratches on the LE skin inboard of the engine.	 PHOTO 88: WING DAMAGE
12	Krueger station 141	116A4410-7 (PHOTO 89)	See above description.	 PHOTO 89: WING DAMAGE
13	Krueger station 191	116A4410-7 (PHOTO 90)	Additional surface scratches on the LE skin inboard of the engine. Gouges are from 9-inches to 24-inches from the aft edge and located 50-inches from the inboard edge.	 PHOTO 90: WING DAMAGE
13	Krueger station 191	116A4410-7 (PHOTO 91)	See above description.	 PHOTO 91: WING DAMAGE

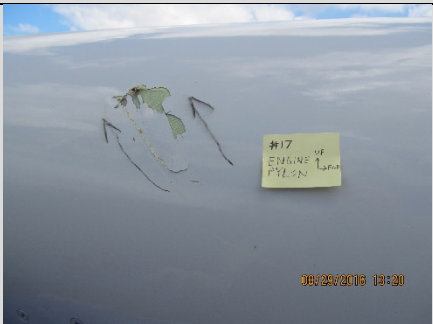
TABLE 2: AIRPLANE STRUCTURAL DAMAGE				
Item Number	Location	Part Number (PHOTO No.)	Condition	Photo
18	Wing Station (WS) 221. The forward/aft black line is WS 204 (Rib No. 5).	112A3102 (PHOTO 92)	Gouges on the upper wing skin inboard of the engine just aft of the front spar chord.	 PHOTO 92: WING DAMAGE
18		(PHOTO 93)	See above description.	 PHOTO 93: WING DAMAGE
17	Engine pylon fairing on the inboard side. Engine pylon fairing panel 431BR.	313A2121 (PHOTO 94)	Surface gouges existed on the engine pylon fairing.	 PHOTO 94: WING DAMAGE
20, 21, and 22	Just outboard of the engine pylon.	(PHOTO 95)	Photo to the right shows the general location of items 20, 21, and 22.	 PHOTO 95: WING DAMAGE

TABLE 2: AIRPLANE STRUCTURAL DAMAGE

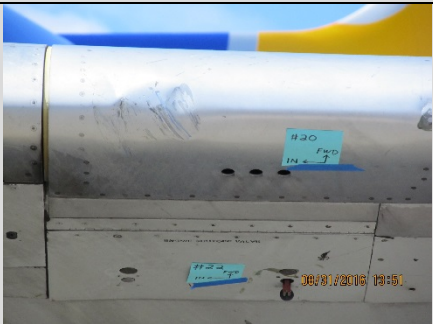
Item Number	Location	Part Number (PHOTO NO.)	Condition	Photo
20	Slat No. 4	114A8041-3 (PHOTO 96)	A small dent was found near the inboard edge of slat No. 4. Additional minor scraping damage was found along its length. See photo above.	 PHOTO 96: WING DAMAGE
21	Lower wing skin No. 3 just outboard of the engine pylon approximately at WS 240.	112A4103-5 (PHOTO 97)	Minor surface gouging on the lower wing skin.	 PHOTO 97: WING DAMAGE
22	Engine shutoff valve access door panel 621CB.	(PHOTO 98)	Minor gouging was evident. The left latch was found in the open position.	 PHOTO 98: WING DAMAGE
14	Slat No. 1 about 8-inches inboard from the outboard edge of the part (Slat station 511)	114A8011-1 (PHOTO 99)	10-inches-long x 5-inches-wide dent. The dent is approximately 0.5-inches-deep.	 PHOTO 99: WING DAMAGE

TABLE 2: AIRPLANE STRUCTURAL DAMAGE

Item Number	Location	Part Number (PHOTO NO.)	Condition	Photo
14		(PHOTO 100)	See above description.	 PHOTO 100: WING DAMAGE
14		116A5411-1 (PHOTO 101)	10-inches-long x 5-inches-long dent. The dent was about 0.5-inches-deep.	 PHOTO 101: WING DAMAGE
14		114A8211-1 (PHOTO 102)	Upon removal of the LE skin, the 114A8211-1 nose beam was found dented/buckled.	 PHOTO 102: WING DAMAGE
25	About 6-inch outboard of the inboard edge of the outboard LE skin.	737-0350 (LE Skin), 737-0221 (Front Spar) (PHOTO 103)	The winglet LE skin was punctured, and the winglet front spar chord was partially severed. The puncture was about 5-inches-long at the LE and it about 6-inches-deep. The front spar chord was severed about 2-inches-deep at on the inboard edge and 1.5-inches on the outboard edge. The slot cut into the spar chord was about 0.375-inches-wide.	 PHOTO 103: WINGLET DAMAGE

TABLE 2: AIRPLANE STRUCTURAL DAMAGE				
Item Number	Location	Part Number (PHOTO NO.)	Condition	Photo
25		(PHOTO 104)	See above description.	 PHOTO 104: WINGLET DAMAGE
25		(PHOTO 105)	See above description.	 PHOTO 105: WINGLET DAMAGE
25		(PHOTO 106)	See above description.	 PHOTO 106: WINGLET DAMAGE
25		(PHOTO 107)	See above description. The slot cut into the spar chord was about 0.375-inches-wide.	 PHOTO 107: WINGLET DAMAGE

TABLE 2: AIRPLANE STRUCTURAL DAMAGE				
Item Number	Location	Part Number (PHOTO NO.)	Condition	Photo
25		(PHOTO 108)	See above description. View of outboard side.	 08/20/2016 11:41 PHOTO 108: WINGLET DAMAGE
25		(PHOTO 109)	See above description. View of the winglet front spar damage.	 08/20/2016 11:45 PHOTO 109: WINGLET DAMAGE
25		(PHOTO 110)	See above description. View of the outboard flange of the winglet front spar chord.	 08/20/2016 11:44 PHOTO 110: WINGLET DAMAGE
24	Winglet outboard LE skin	737-0350 (PHOTO 111)	There was a dent 3-inch-wide x 0.2-inches-deep located about 16.5-inches outboard from the inboard edge of the part.	 08/20/2016 11:50 PHOTO 111: WINGLET DAMAGE

TABLE 2: AIRPLANE STRUCTURAL DAMAGE

Item Number	Location	Part Number (PHOTO NO.)	Condition	Photo
15	Horizontal stabilizer LE. LE station 133.15 to outboard tip.	186A1210 (PHOTO 112)	Dent damage to the horizontal stabilizer from LE station (STA) 149 to 180. The LE skin torn and the LE ribs were bent.	 08/29/2016 11:04 PHOTO 112: HORIZONTAL STABILIZER DAMAGE
15 and 23		(PHOTO 113)	The LE from LE STA 133.15 to tip was removed and is shown in the picture to the right. Scrapping damage was noted along much of its length.	 08/30/2016 11:04 PHOTO 113: HORIZONTAL STABILIZER DAMAGE
23	Horizontal stabilizer LE skin just outboard of LE STA 133.15	(PHOTO 114)	Minor denting and surface scoring noted. The surface was imprinted with a honeycomb pattern characteristic of the engine inlet acoustic liner.	 08/30/2016 10:29 PHOTO 114: HORIZONTAL STABILIZER DAMAGE

TABLE 2: AIRPLANE STRUCTURAL DAMAGE

Item Number	Location	Part Number (PHOTO NO.)	Condition	Photo
15		(PHOTO 115)	The location labeled as location 10 at the deepest portion of the dent is LE STA 161.04. The dent was about 35-inches-long and 1.4-inches-deep. The LE skin had a honeycomb pattern imprinted on it and had red material transfer. Portions of the engine inlet material were found embedded in the tear of the skin.	 PHOTO 115: HORIZONTAL STABILIZER DAMAGE
15		(PHOTO 116)	Inlet material found embedded in the horizontal stabilizer LE skin tear.	 PHOTO 116: DEBRIS FOUND IN HORIZONTAL STABILIZER
15		(PHOTO 117)	Inlet material found embedded in the horizontal stabilizer LE skin tear.	 PHOTO 117: DEBRIS FOUND IN HORIZONTAL STABILIZER

TABLE 2: AIRPLANE STRUCTURAL DAMAGE				
Item Number	Location	Part Number (PHOTO NO.)	Condition	Photo
15	Horizontal Stabilizer LE STA 151.75 to 170.34 (Locations 9 to 11)	(PHOTO 118)	The horizontal stabilizer LE ribs at LE STA 151.75, 170.34, and 179.64 were buckled.	 PHOTO 118: HORIZONTAL STABILIZER DAMAGE
16	LE STA 155 just aft of the front spar.	182A3002 (PHOTO 119)	5-inches-long x 3-inches-wide x 0.095-inches-deep dent with minor gouge damage in the horizontal stabilizer upper skin.	 PHOTO 119: HORIZONTAL STABILIZER DAMAGE

3.0 METALLURGICAL EXAMINATION

All the event fan blades and their associated hardware (shims, platforms, and spacers) were shipped to the NTSB Materials Laboratory in Washington DC, along with pieces of the left wing LE and left wing winglet, for examination and evaluation. After an initial examination conducted at the NTSB Materials Laboratory, most of the blades were shipped to the GE Engineering Materials Systems Laboratory in Evendale, Ohio for further examination and inspection. Several selected event fan blades were also shipped to: (1) the Safran Failure Analysis Laboratory Moissy-Cramayel, France for additional examination and evaluation including RS analysis, (2) Lambda Technologies Group in Cincinnati, Ohio for RS analysis, and (3) GE Global Research Center in Niskayuna, New York for grain mapping orientation. In addition to the event fan blades, exemplar and in-service fan blades with inspection findings were also sent to the GE and Safran laboratories in Evendale and Moissy-Cramayel respectively for evaluation. The following sections are a synopsis of the NTSB Materials Laboratory report that includes information from various other laboratory examinations; for complete details see *NTSB Materials Laboratory Factual Report No. 17-043*.

3.1 FRACTOGRAPHIC AND METALLOGRAPHIC EXAMINATION OF EVENT BLADES

3.1.1 Fractured Fan Blade No. 23 Hardware

Initial optical examination of fan blade No. 23 in the as-received condition revealed that it fractured in the dovetail of the blade root. The fracture surface exhibited a relatively smooth area with multiple curved crack arrest lines emanating from the convex side of the blade through almost the entire thickness of the dovetail (0.695-inches-wide); these features are consistent with fatigue crack initiation and propagation. Crack arrest lines generally represent changes in stress state, environment or time interval associated with fatigue crack growth. Beyond the smooth fatigue region was a smaller region with a lighter gray appearance with a prominent shear lip; features consistent with ductile overstress fracture. The fracture origin was centered about 2.1-inches from the dovetail front face on the convex (suction) side of the blade. Ratchet marks were observed in the fracture origin; ratchet marks are consistent with multiple fatigue origins (**PHOTO 120**). The highest fan blade operational stress is located in the dovetail LE convex side area.

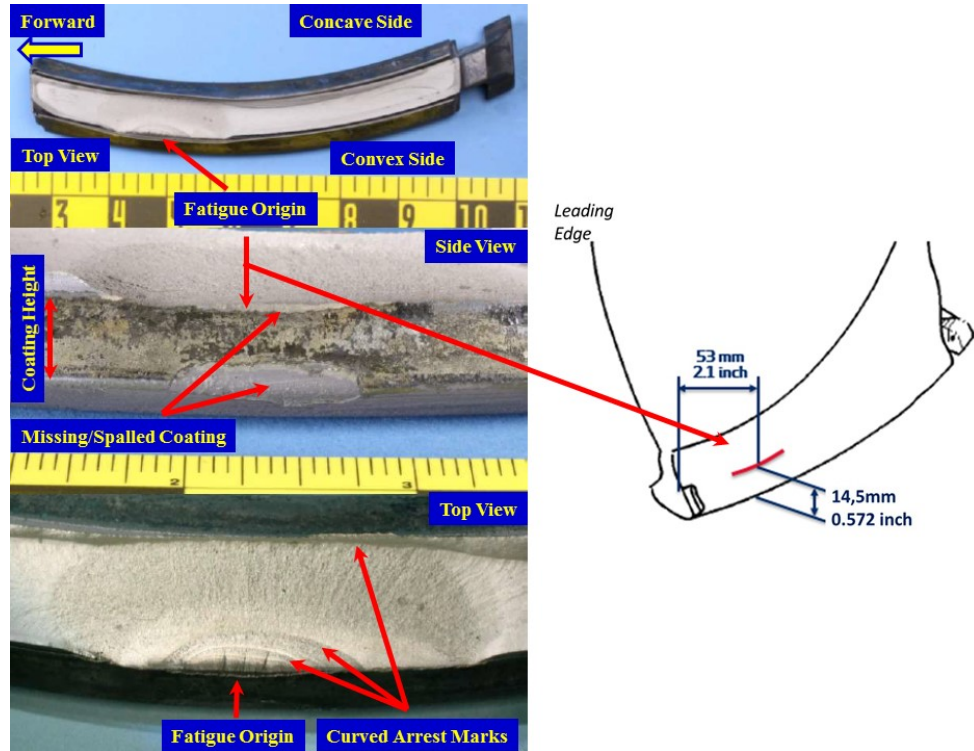


PHOTO 120: FRACTURE SURFACE OF FAN BLADE NO. 23

With the shim removed, the fracture origin was confirmed to be in the dovetail near the outboard edge of the Cu-Ni-In coating (See **FIGURE 3** for location of coating; inboard is toward the root and outboard is toward the blade airfoil). The Cu-Ni-In coating was missing and spalled in several locations on the convex side; most notably in an area adjacent to the fracture and a larger area directly inboard from the fracture origin (See center picture in **PHOTO 120**). The larger spalled area measured 0.87-inches-long and extended up to 35% of the width of the contact face. Within this larger spalled area were gray linear features consistent with brush strokes when the lubricant was applied; these features were also observed on the smaller inboard spalled area. The coating thickness at the outboard edge of the coating adjacent to the spalled area was measured optically using a microscope, the thickness was about 0.007-inch thick; the coating thickness requirement specifies the coating should be 0.0047-0.0071-inches (0.12-0.18 millimeters (mm))

The shim was sectioned lengthwise down the centerline to facilitate examination of the interior contact surface. The interior of the shim visually appeared like a mirror image of the blade dovetail. The areas where the coating was present on the blade dovetail had corresponding contact marks on the shim interior surfaces. The areas where the coating was missing and spalled on the blade dovetail had no corresponding contact marks or wear (unworn) on the shim interior surface. This is indicative of the blade dovetail not contacting the shim in those unworn locations (**PHOTO 121**).

After the initial optical examination, the fracture surface was inserted into a SEM. The fracture surface and coating were analyzed using energy dispersive X-ray spectroscopy (EDS). The EDS spectra obtained on the fracture surface had elemental peaks consistent with Ti-6-4 plus peaks of carbon and oxygen. The coating had a spectra consistent with the Cu-Ni-In coating plus peaks of molybdenum, sulfur, Al, and oxygen. A transverse cross-section was cut from the aft end of the blade dovetail away from the fatigue region to examine the microstructure of that blade without destroying the fatigue features. The cross-section was polished then etched with 15/15 reagent (15 milliliters hydrofluoric acid and 15 milliliters nitric acid in 70 milliliters water) revealing that the microstructure had an even mix of alpha grains interspersed with areas of alpha plus beta; consistent with the expected microstructure for the specified material. No surface anomalies, such as distorted grains or alpha case, were observed and no defect at the fracture origin area was found.

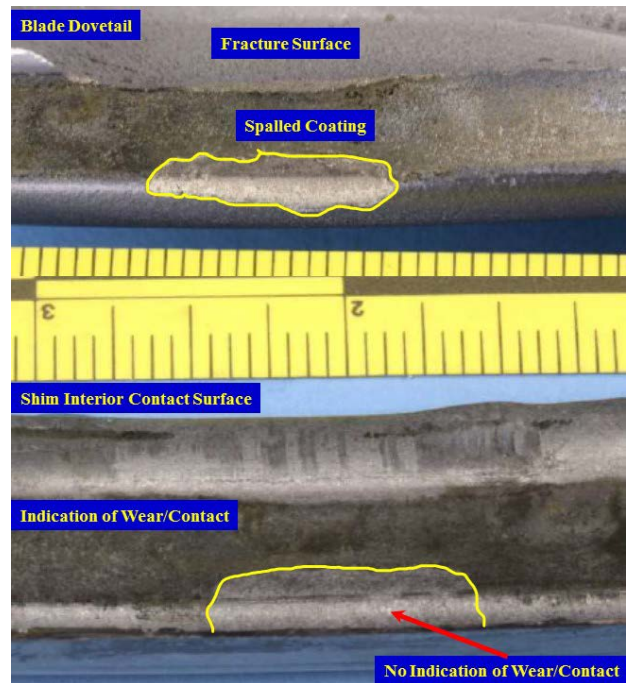


PHOTO 121: BLADE NO. 23 DOVETAIL AND SHIM INTERIOR SURFACE COMPARISON

Four distinct crack arrest lines were noted on the fracture surface; a possible fifth arrest line between lines three and four was present particularly near the aft side of the fatigue region, but the boundary was not as distinct around the entire crack front as the other arrest lines (**PHOTO 122**). Crack striations (observed using the SEM) were consistent with low-cycle fatigue (LCF) crack growth were observed along the fracture surface and the striation spacing generally became larger as the depth of the crack increased and as the crack advanced toward the arrest line. At crack arrest lines, the striation spacing showed an abrupt change. The distance between striations immediately past each arrest line was noticeably shorter than the spacing leading up to the arrest line. This is consistent with a temporary reduction in crack growth rate as the crack grew after each arrest line had been formed.

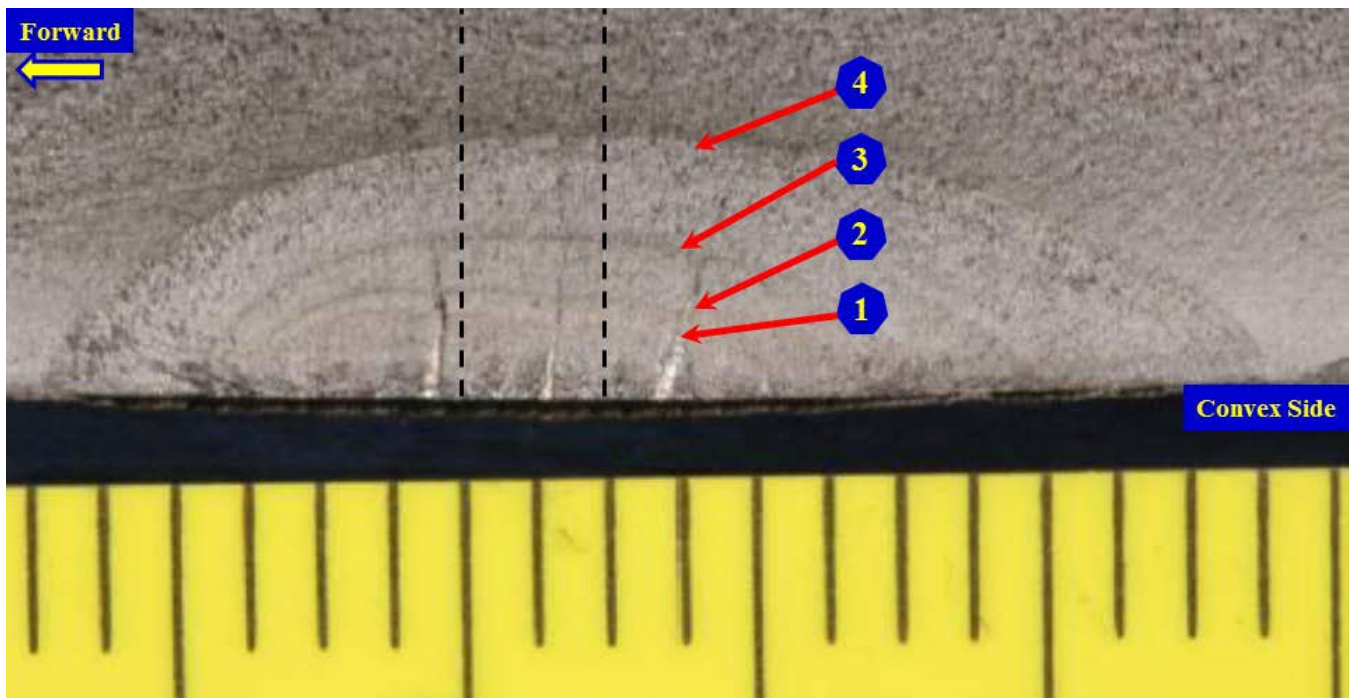


PHOTO 122: CRACK ARREST LINES

The NTSB and CFM conducted independent striation density estimates, a technique to estimate the approximate number of stress cycles from crack initiation to failure, and both came up with similar results. The purpose of striation density measurements was to estimate the number of flight cycles that elapsed between crack initiation and fracture. One striation correlates with one flight cycle where a flight cycle is assumed to include one takeoff and landing. The total estimate for crack growth from a size of 0.0005-inches to failure for the No. 23 fan blade was about 15,000 cycles. As mentioned previously, arrest lines can represent changes in stress state, and according to CFM, lubrication/relubes of the dovetails reduces (changes) the overall stress state of the blade root. Therefore, estimates for striation count between arrest lines 1 and 2, 2 and 3, and 3 and 4 was conducted and were 1,020 cycles, 1,709 cycles, and 2,438 cycles respectively. Since the engine was last overhauled in October 2007, the fan blades had been relubed 13 times and the cycles between relubes was on average every 1,274 cycles (See Section 6.2 OPERATION HISTORY - INSTALLATION, REMOVAL, AND RELUBE for details).

3.1.2 Fan Blade No. 24 Hardware

Since fan blade No. 24 had similar spalled and missing Cu-Ni-In coating (inboard on the convex side contact towards the forward front face) as the fractured fan blade No. 23 and was manufactured from the same manufacturing lot (See Section 6.1 MANUFACTURING HISTORY for details), a transverse cut through the blade dovetail and a polish and etch of the cross-section was performed like what was performed on the No. 23 fan blade dovetail but this cross-section was prepared approximately 2-inches from the LE. Optical examination of the dovetail cross-section revealed a primary crack and several secondary cracks emanating from under the Cu-Ni-In coating near the outboard end of the contact face on the convex side of the blade (**PHOTO 123**); this corresponded to the fracture origin area for fan blade No. 23. Examination of the microstructure revealed it to be similar to fan blade No. 23; consistent with the material specification and no anomalies noted. The primary crack was lab fractured revealing fatigue features with multiple origins located at the convex side of the dovetail. The crack striations were consistent with LCF crack growth and the striation spacing generally became larger as the depth of the crack increased. A striation density measurement was performed for the primary crack using the SEM, and the total estimated number of striations was about 16,000 cycles from a crack depth of 0.0005-inch to the final depth at the time of the event. Six crack arrest lines were observed in the fatigue region, and the estimated number of striations between crack arrest lines corresponded to the number of cycles between relubes. The Cu-Ni-In coating had disbonded/peeled from the contact surface at the outboard end of the convex side where the primary and secondary cracks were located but remained intact covering the cracks from view during the optical examination of the blade prior to sectioning.

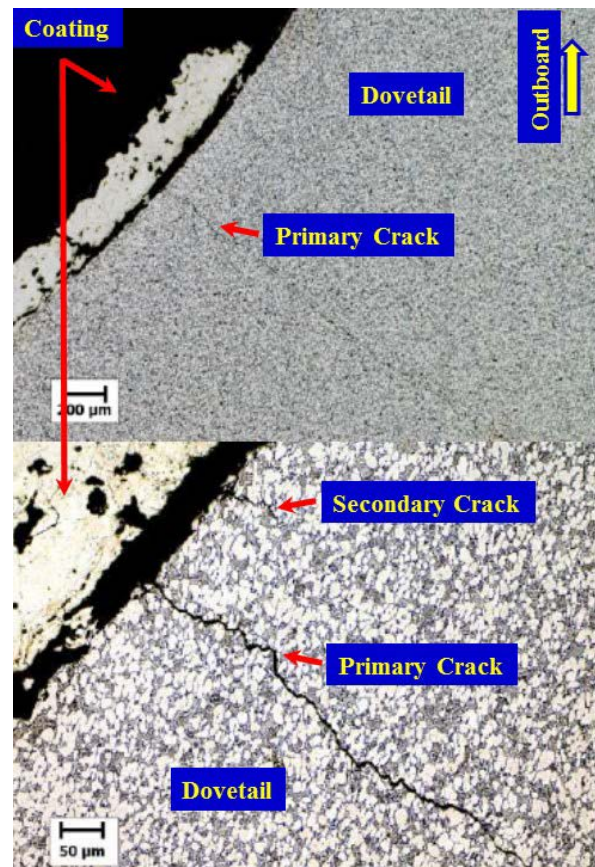


PHOTO 123: MAGNIFIED VIEWS OF BLADE NO. 24 CRACKS AND COATING DISBONDED

The No. 24 blade shim was sectioned lengthwise down the centerline exposing the interior contact surface. Just like with fan blade No. 23 shim, the interior contact surface of shim No. 24 appeared as a mirror image of the blade dovetail showing contact marks in areas where the coating was intact and an absence of contact marks in areas where the coating had spalled and was missing (See bottom of **PHOTO 121** for reference). Along with the interior of the shim showing a mirror image of the dovetail coating, the exterior of the surface also showed a contact pattern matching the spalled and missing coating.

Except for one area of spalling and missing coating found on the blade dovetail, the coating remained attached and appeared consistent with a properly sprayed coating with no indications of excessive porosity, unmelted particles, or entrained oxides observed in the cross-section. Vickers hardness measurements using a 300-gram load were taken on both the convex and concave sides of the blade dovetail where the coating appeared intact and undamaged; the hardness requirement for the coating

when it was last applied on the fan blade dovetail was specified as a minimum of 120HV0.3.¹⁴ On the contact surfaces, the hardness varied from 140HV0.3-180HV0.3; consistent with the requirements at the last coating application.

The convex and concave contact surfaces of the blade dovetail were scanned using a laser coordinate scanner before and after the coating was removed. The difference between the measurements taken before and after coating removal represented the coating thickness. Variations in coating thickness were noted on both the convex and concave sides and the variations were radially outboard to inboard as well as longitudinally from front to back. In some locations, such as the forward end on both the concave and convex sides, the coating was slightly thicker than specified; 0.009-inches and 0.008-inches respectively versus the specified maximum of 0.007-inch thickness.

3.1.3 Examination of Remaining Event Fan Blades Plus Exemplar Fan Blades

Because cracking was found beneath the Cu-Ni-In coating on fan blades Nos. 23 and 24, it was decided that all the event blade dovetails would be inspected after the coating was removed by chemical stripping using a variety of NDI techniques that included visual, FPI, ECI, or UI before any further destructive evaluation (other than removal of the coating by chemical stripping) was performed on the fan blade set. Along with the event fan blades, 2 additional non-event exemplar fan blades were included in the NDI; one of which was manufactured in the same lot as fan blade No. 23. Not every fan blade was subjected to all three NDI techniques.

All the other fan blade dovetails were examined optically, and no other cracks were detected. Since no cracks were detected during the initial optical examination of the event blade set and the crack in fan blade No. 24 dovetail was only found after sectioning and mounting in a SEM, it was thought that the Cu-Ni-In coating might mask any surface indications and cracks from being detected using just surface or near-surface inspection techniques such as FPI and ECI while the coating was still in place. All the fan blades dovetails were inspected using UI either before and/or after the coating was removed. Following the UI, the dovetails were inspected using a combination of FPI, ECI, or SEM with the Cu-Ni-In coating removed to identify or confirm cracks (See Table 2 in *NTSB Materials Laboratory Factual Report No. 17-043* report for a complete list of NDI inspection results). A total of seven fan blade dovetails were found to have cracks either by fracture (No. 23) or detected by traditional NDI methods (UI, ECI, or FPI); Nos. 1, 2, 11, 17, 19, and 24. Neither of the two exemplar blades were found to have cracks. In each case, the cracked dovetails exhibited a primary crack and a network of microcracks located at the surface; at some locations the microcracks appeared to link together to form a longer crack feature. The cracks and microcracks were all located in the grit blast region where the coating had been stripped from the surface and approximately 2-inches aft of the forward face; no cracks were found in the shot-peened¹⁵ surface outboard of the grit blast zone. Additionally, the crack surfaces were relatively flat with

¹⁴ The Vickers hardness test uses a calibrated machine to force a square-based pyramidal diamond indenter into the material being tested. The impression made is then measured from corner-to-corner on a diagonal, and the hardness calculated using this measurement. Hardness requirements read as follows: 120HV0.3 $\Rightarrow$ 120 is the hardness value, HV indicates hardness was determined using a Vickers test and 0.3 is the applied test load in kilogram force (or 300-gram load).

¹⁵ Shot-peening is a cold work process in which compressive stresses are induced into the surface of the part by impingement shot (typically metallic) at specified conditions (velocity, angle, and duration). Shot-peening increases the fatigue strength of the material and relieves surface tensile stresses that can aid in cracking. Because quantitative measure of the coverage can be difficult, coverage is typically estimated based on the intensity of the peening over a specified duration. Peening intensity is typically expressed as the arc height of an Almen test strip at or more than full saturation (100%) coverage. Arc height is the measured curvature of the Almen test strip that had been shot-peened only on one side and is a gauge for the effectiveness of the shot-peening operation. Almen test strips come in three designations ('N', 'A', and 'C'), each is a varying thickness depending on the density (arc height) specified.

curving crack arrest lines and boundaries; features consistent with fatigue. Fan blades 1, 2, 11, 17, 19, and 24 were lab fractured and an SEM examination of the cracks showed striations consistent with LCF in Ti-6-4; relatively fine striations were observed near the origin that generally decreased in density with increasing crack depth.

Several representative blades from the event set and one non-event exemplar blade that showed no crack indications by traditional NDI methods were sectioned to facilitate an SEM examination in the area approximately 2-inches forward of the front face (this is where the failed blade No. 23 crack origin was located). After sectioning through the blade dovetail, the convex surface of the dovetail was etched and examined using a SEM. Microcracks were observed in all the examined event blades in the grit blast area where the coating had been removed; no microcracks were observed in the shot-peened surface outboard of the grit blast area nor were any observed in the exemplar non-event fan blade dovetail.

The Cu-Ni-In coating on many of the blades showed signs of sliding and deformation in the radial direction; however, several blades exhibited missing and spalled coating on either the concave or convex side. The coating loss on the convex side (See Table 1 in *NTSB Materials Laboratory Factual Report No. 17-043* for location and size) was typically near the forward end of the blade dovetail and on the concave side was typically near the aft end of the blade dovetail. Spalled coating was observed on six of the seven total cracked event fan blades just below the crack initiation site (this includes blades No. 23 and 24). According to CFM, the coating loss locations were consistent with their field experience and consistent with the operational loading of the fan blade. The coating loss on the convex side was significantly greater than what was observed on the concave side; again, consistent with CFM field experience. Comparing the observed missing and spalled Cu-Ni-In coating on the event blades with the inspection limits specified in the Boeing AMM 737-600/700/800/900 AMM section 72-21-02 titled *Fan Blade – Inspection/Check* in effect at the time of the event showed that all fan blades with missing and spalled coating on the convex side of the dovetail in the contact area would be allowed to continue-in-service but a reduced inspection interval; the limit of 50 flight cycles or 75 flight hours whichever comes first (See Section 4.0 AMM FAN BLADE DOVETAIL & SHIM INSPECTION, AND RELUBE REQUIREMENTS and **FIGURES 18-20** for details of the Cu-Ni-In coating inspection in the contact area <3>).

Sixteen event fan blades plus a non-event exemplar fan blade were scanned using a laser coordinate scanner before and after the coating was removed to determine the Cu-Ni-In coating thickness (See Tables 6 and 7 in *NTSB Materials Laboratory Factual Report No. 17-043* for complete coating thickness results).¹⁶ Five of the blades had coating thickness values that exceeded the specification on the convex side of the blade, including exemplar fan blade A. Nearly all the blades had coating thickness values that exceeded specification on the concave side. Most of the blades from the event engine had lower thickness values near the forward and/or aft ends of the contact face.

For most blade dovetails that exhibited missing and spalling Cu-Ni-In coating on the convex side (See Table 1 in *NTSB Materials Laboratory Factual Report No. 17-043*), their corresponding shims were sectioned lengthwise similar to what was performed on the Nos. 23 and 24 shims. One shim, with the older configuration, exhibited that same mirroring that was noted on shim No. 23, but the area corresponding to the missing coating also appeared shiny and worn consistent with prior contact with the coating. In several shims, the spalled dovetail coating had adhered to the interior of the shim.

¹⁶ Same scanning process as what was performed on the No. 24 blade; No. 24 is included in the sixteen event blades measured.

3.2 DOVETAIL RESIDUAL STRESS PROFILES

RS is defined as the stress that remains in the solid material that is not being subjected to external forces. RS can be caused or altered by processing operations where the material is distorted, such as shot-peening, machining, heating during coating application etc., as well as by operating environment, such as operating at high stress levels. As part of the fan blade repair, shot-peening of the entire fan blade, including the dovetail, root, and airfoil, is used to achieve a compressive RS level and it effectively serves to provide improved fatigue strength of the material and relieve surface tensile stresses that can aid in fatigue crack initiation. RS measurements were conducted on nine of the event blades, those with and without primary cracks, as well as two non-event exemplar blades. Measurements were taken at multiple areas along the blade dovetail convex and concave contact surface within and outside of the grit blast area (**PHOTO 124**) and RS profiles were created; compressive stress vs depth curves.

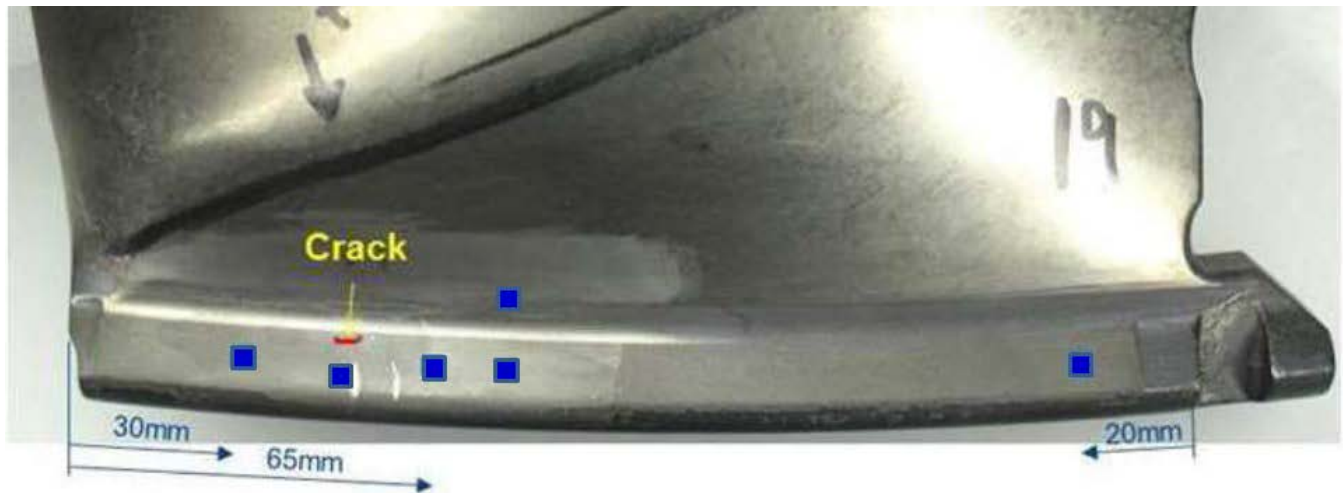


PHOTO 124: RS MEASUREMENT LOCATIONS ON BLADE NO. 19 – TYPICAL OF OTHER BLADE LOCATIONS

CFM provided an expected RS stress profile to use as a baseline when comparing the RS profiles from the event blades and non-event exemplar blades. The typical RS profile for Ti-6-4 shot-peened with steel shot is expected to have a small compressive stress at the surface, then the RSs continue to increase (negative number represent compression; lower value represent more compressive stress and higher values represent less compressive stress) until it reaches a maximum absolute value at a certain depth. After that, the compressive stresses start to gradually decrease and dissipate until the RS becomes tensile (See **FIGURE 17**). According to CFM, the maximum compressive stress absolute value in the blade dovetail with proper shot-peening is expected to be greater than 500 megaPascals (MPa).

Lambda Technologies Group and CFM conducted RS measurements using X-ray diffraction at the surface and at varying depths starting at the LE and translating toward the TE. X-ray diffraction provides quantitative results and measures strain instead of stress. Stress calculations were performed based on strain values and their orientation to the surface of interest in combination with the elastic properties of the material. **TABLE 4** in *NTSB Materials Laboratory Factual Report No. 17-043* lists the blades tested, locations where the RS was measured, what facility conducted the measurement, and selected RS values.

In general, the compressive stresses were less compressive than expected on all the event fan blade dovetails when compared to a reference profile (**FIGURE 17**).¹⁷ The light blue curves represent a typical RS profile in shape and in stress level. Abnormal RS profiles were consistently found in the tested event blades on the convex side of the dovetail in the grit blast contact face in the areas corresponding to the crack initiation sites. Adjacent shot peened areas (not grit blasted) and some other area of grit blasted contact faces had maximum compressive RS absolute values more than 500 MPa (See TABLE 4 in *NTSB Materials Laboratory Factual Report No. 17-043*).

The top chart in **FIGURE 17** shows that for the event fan blade No. 23, the surface RS value was almost zero and had a consistently reduced compressive stress value verses depth and did not intersect (reach the expected RS value) the reference value RS until almost at a depth of 100 micrometers (about 0.004-inch). Similar abnormal RS profiles were observed for other blades with primary cracks and in other event blades in grit blasted area away from the crack locations (lower chart in **FIGURE 17**). The RS profiles were also created for the three non-event exemplar fan blades and their profiles were comparable to the reference profile.

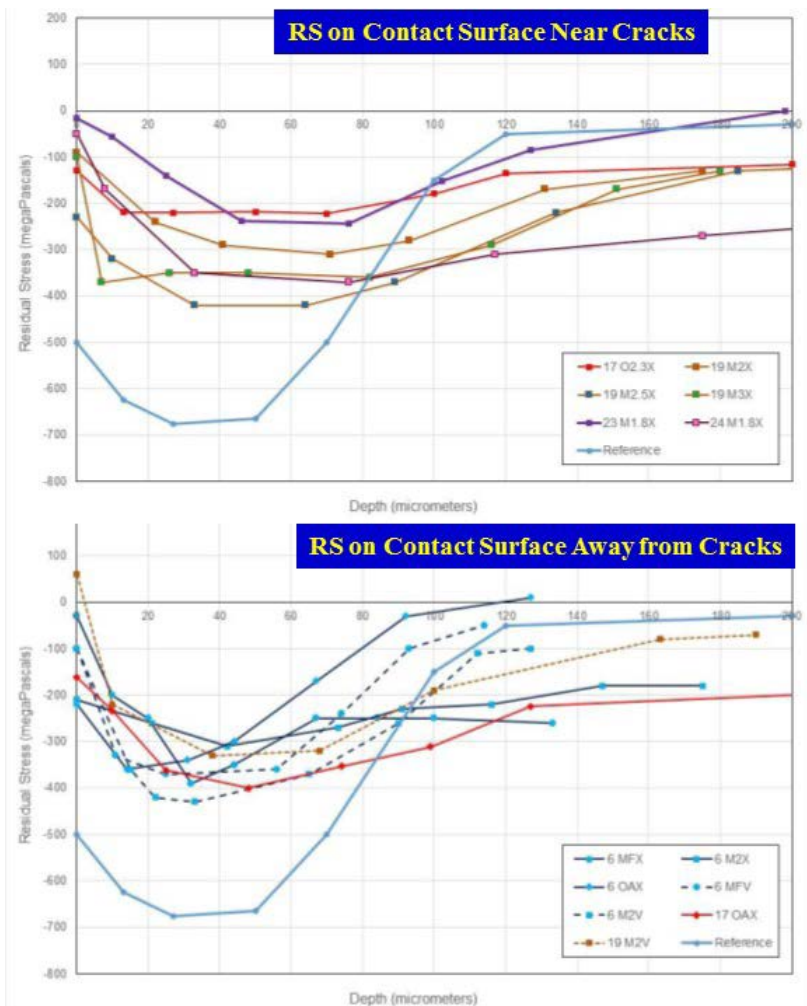


FIGURE 17: RESIDUAL STRESS PROFILES

3.3 ADDITIONAL AIRPLANE HARDWARE EXAMINATION

A piece of the left wing LE No. 1 slat that exhibited an impact dent (See **PHOTOS 99-101**) and sections of the left winglet above and below the slice (See **PHOTOS 103-111**) were removed from the airplane and sent to the NTSB Materials Laboratory to determine if liberated fan blade fragments had impacted the airplane in those areas. The items were not cleaned prior to the examinations. See *NTSB Materials Laboratory Factual Report, No. 16-107* for complete details.

The LE slat piece exhibited sliding contact marks with an associated inward deformation. The sliding contact marks were examined using SEM and EDS. The typical spectra within and outside of

¹⁷ Each RS measurement has a unique identifier - first the number of the blade, radial position (M=middle of contact face, O=outboard edge of contact face, S=shoot-peened only area outside of grit blast area), longitudinal position (A=near aft face, F=near forward, and if number distance from forward face in inches), last letter denotes which side of the blade (V=concave, X=convex). RS was performed on a total of 9 event engine fan blades.

the sliding contact area had elements of Al, Copper (Cu), Magnesium (Mg), and Zinc (Zn) consistent with 7000-series Al alloy, the specified material. No elemental peaks of Ti consistent with fan blade material were observed.

The surfaces of winglet slices (See [PHOTO 111](#)) were torn, deformed outboard and aft, and portions exhibited sliding contact and smearing. EDS spectrum for the sliding contact areas revealed elements consistent 7000-series Al alloy and Ti, amongst other elements. Outside the sliding contact area, no elemental peaks of Ti were observed.

3.4 SHIM EVALUATION CONDUCTED BY CFM

NTSB and CFM evaluated 10 of the shims from the event blade set; all the shims installed on cracked fan blades were evaluated except for the No. 1 shim. The shim thickness was evaluated at three locations (LE, middle, TE) on the concave, convex, and bottom face for a total of nine measurements per shim. These measurements were compared with the production specification; all the surfaces of the shim should be of a uniform thickness, $0.4\text{mm} \pm 0.1\text{mm}$ ($0.0157\text{-inches} \pm 0.004\text{-inches}$) after forming. NTSB and CFM found that there was no significant preexisting thickness variation identified in the shims. Although shims Nos. 23 and 24 showed imprints of the spalled and missing fan blade dovetail coating, this was not universally the case on some of the other shims that were evaluated.

3.5 ADDITIONAL GKN FAN BLADE EVALUATION

GKN, the repair shop that overhauled the event blade set (See Section 5.0 ESM FAN BLADE DOVETAIL INSPECTION AND REPAIR REQUIREMENTS and Section 6.0 ESN 874112 EVENT FAN BLADE SET HISTORY for details) provided a list of fan blade SNs for blades that had been recoated around the time the incident fan blade set was repaired; 48 fan blades before and 48 fan blades after the event blade set had been recoated. As of October 26, 2018, 84 blades from this pool of 96 fan blades have been inspected; 83 passed the inspection (3 ECI, 76 UI and 4 ECI+UI) and 1 had a UI indication that was later confirmed to be a crack.

4.0 AMM FAN BLADE DOVETAIL & SHIM INSPECTION, AND RELUBE REQUIREMENTS

4.1 FAN BLADE INSPECTION REQUIREMENTS AT THE TIME OF THE EVENT

According to the CFM56-7B Commercial Engine Service Memorandum (CESM) No. 005, titled *Stage 1 Fan Blade and Disk Dovetail – On-Wing Periodic Maintenance Action* Revision 3 dated September 9, 2005,¹⁸ CFM recommended a maximum on-wing fan blade inspection and lube/relube interval of 1,500 to 3,000 flight cycles or 5,000 hours whichever comes first but stated that the task interval or threshold may be adjusted by each operator based on its own aircraft maintenance program. The current version of CESM No. 005 at the time of the incident was Revision 3 dated September 9, 2005.¹⁹ In all three revisions of CESM No. 5, the upper cycle interval for inspection and relube was 3,000 flight cycles. See SB 72-1033, revision 2 in See Section 13.6.1 SERVICE BULLETINS for update to the fan blade respective inspection and relube cycle.

¹⁸ Original release of the CESM was on May 2, 2000, with Revision 3 being the most current at the time of the event.

¹⁹ Revision 3 recommended a maximum on-wing fan blade inspection and lube/relube interval of 3,000 cycles or 5,000 hours whichever comes first but stated that the task interval or threshold may be adjusted by each operator based on its own aircraft maintenance program.

A review of the Boeing Maintenance Planning Document (MPD) applicable at the time of the accident for the 737-600/-700/-800/-900/-900ER had the relube maintenance task threshold (first time the task was to be performed) of 3,000 flight cycles or 5,000 flight hours whichever comes first and a repetitive interval of 3,000 flight cycles or 5,000 flight hours whichever comes first; this task was applicable to all airplane and engine models covered by the MPD. The relube maintenance task was added to the Boeing MPD on February 15, 2001. SWA had previously performed the fan blade cleaning, inspection, and relube every 2,600 hours; however, in July 2016, a month prior to the event, SWA revised their requirement to match the CFM recommendation of every 3,000 cycles. According to the maintenance records, the last fan blade relube had occurred in November 2015 under the 2,600 hour interval (See [TABLE 4](#) and [FIGURE 23](#) for details).

CESM No. 005 calls out the fan blade cleaning, inspection, and relube tasks in accordance with the Boeing 737-600/700/800/900 AMM TASK 72-21-02 titled *Fan Blade – Inspection/Check*. AMM TASK 72-21-02 provides the inspection requirements for the fan disk, retaining flange, retaining ring, spacer, platform, shim, and fan blade. For the remainder of this section, the focus will be on the inspection of fan blade dovetail, shim, and the dovetail relube procedures only.

According to AMM SUBTASK 72-21-02-200-001 titled *Fan Blade Examination*, no cracks or tears are permitted. A NOTE is provided that defines the pressure faces and it reads as follows (See [FIGURES 18-20](#)²⁰):

NOTE: The fan blade dovetail pressure faces are defined as the contact area and the non-contact area. The contact area is the critical area where the fan disk dovetail slot pressure faces mate the fan blade dovetail pressure faces. Carefully examine the contact area for damage.

After the blade is removed from the fan disk, the quality of the Molykote™ lubricant is assessed. Missing lubricant in the **contact** and **non-contact** areas are permissible, since before the fan blades are reinstalled into the fan disk they are relubed ([FIGURE 18](#)).

The next inspection of the dovetail relates to the protective Cu-Ni-In coating. Although not specifically stated, to visually inspect the protective coating, the Molykote™ must first be removed; typically, a cloth is used to remove it. The Cu-Ni-In coating on the dovetail is visually inspected for any missing and/or chipped coating in the contact area on both the concave and convex sides of the blade and if damage is observed, a transparent graduated tool (SPL-2330 (inch) or SPL-2331 (millimeter)) is used

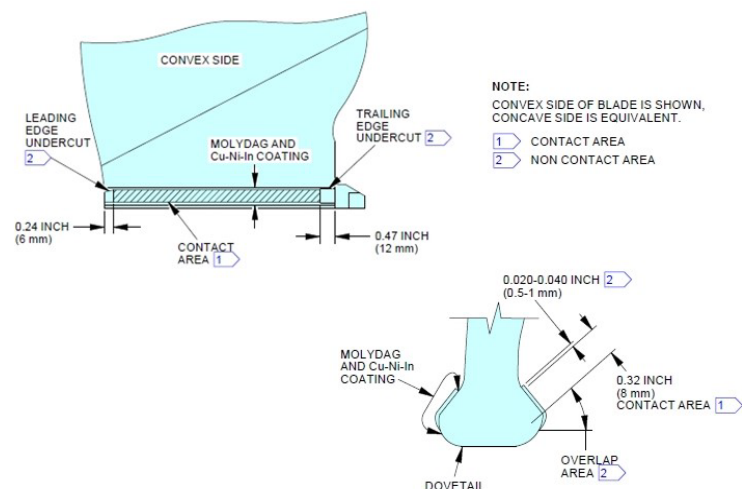


FIGURE 18: FAN BLADE DOVETAIL CONTACT AND NON-CONTACT AREAS AND MOLYKOTE™ APPLICATION AREA
FIGURE COURTESY OF CFM

²⁰ The transverse line across the fan blade shown in [FIGURES 18-20](#) define the blade root area. Anything below that line is considered the blade root area for inspection purposes.

to assess the extent of the coating damage. In the **non-contact** area, defined as the pressure area where the LE and TE undercuts are located, coating loss is permissible provided there is no base material wear.

The **contact area** on both the convex and concave sides of the dovetail are divided into three areas/zones for inspection purposes: “Area X”, “Area Y”, and “more than Area Y”; “Area X” and “Area Y” overlap each other. In the **contact area**, coating loss is permissible in certain areas and may be allowed to continue in service but at a reduced operational interval from the general interval called out in CESM No. 005 (3,000 cycles or 5,000 hours whichever comes first). The following are the requirements for continued service:

<1> In “Area X” only, missing/chipped coating with no base material wear is permissible with no flight restriction (**FIGURE 19**). As part of the “Area X” inspection, the **NOTE** states that repeat extensions are permitted

<2> In “Area Y” (but not in the overlapping area X) missing/chipped coating with no base material wear is permissible with a continue-in-service restriction of 1,500 flight cycles or 2,500 flight hours whichever comes first. As part of the “Area Y” inspection, the **NOTE** states that repeat extensions are permitted (**FIGURE 20**).

<3> In “More than Area Y” (this is the area of the contact surface not identified by “Area X” or “Area Y”), missing/chipped coating with no base material wear is permissible with a continue-in-service restriction of 50 flight cycles or 75 flight hours whichever comes first. Unlike “Area Y”, there is **no** repeat extension permitted in this location.

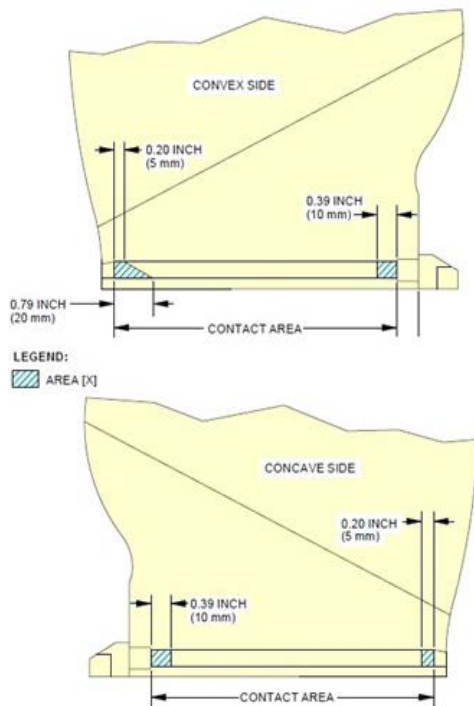


FIGURE 19: “AREA X” LOCATIONS

FIGURE COURTESY OF CFM

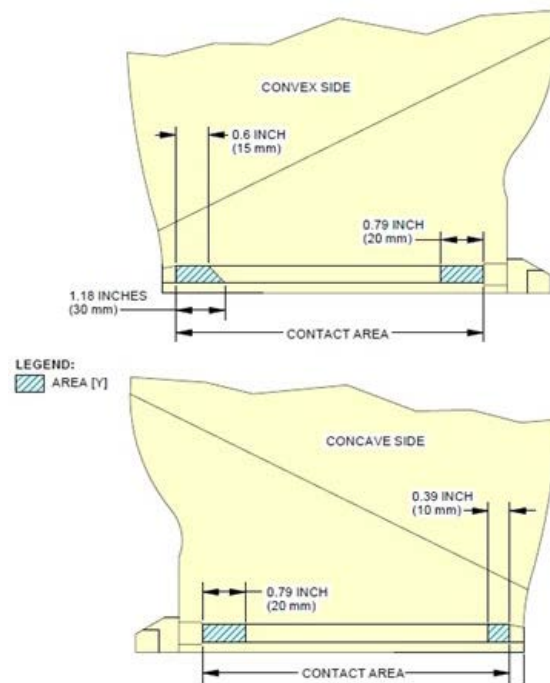


FIGURE 20: “AREA Y” LOCATIONS

FIGURE COURTESY OF CFM

For the event blade No. 23, dimensional measurements of the chipped/missing Cu-Ni-In coating located on convex (suction) side of the dovetail showed that the coating damage was centered about 2-inches aft of the LE spanning approximately 1.3-2.7-inches from the LE of the dovetail. The loss of coating on the event blade No. 23 spanned across the “Area Y” and “More than Area Y” areas. The

No. 23 blade crack initiation site was determined to be on the same side as the coating damage and measured about 2.1-inches from the LE near the outboard edge of the contact area within the “More than Area Y” area. The growth of the fatigue crack (shear lip starts at about 1.489-inches from LE) was also in the “More than Area Y” area.

Since the fan blade failure event, CFM provided Boeing revised visual inspection criteria to clarify and simplify the allowable missing or chipped coating limits on the dovetail contact area in AMM TASK 72-21-00-640-801 and introduced additional dovetail crack inspection techniques (See Section 13.4 NEW INSPECTION FAN BLADE OVERHAUL REQUIREMENTS AND TECHNIQUES for details on the additional dovetail cracks inspection).

4.2 SHIM INSPECTION REQUIREMENTS AT THE TIME OF THE EVENT

The shims are inspected in accordance with AMM SUBTASK 72-21-02-220-056 titled *Shim* for missing coating (varnish) on the pressure face (reconditioning of the coating if missing is permissible per engine shop manual (ESM) TASK 72-21-22-300 Repair 001 – titled *Shim - Repair 001 -Replacement of the Lubricant Coating*). General hardware conditions such as cracks, tears, punctures, and wear not permitted. Scratches, nicks, dents, and distortion within certain allowable limits (**FIGURE 21**).

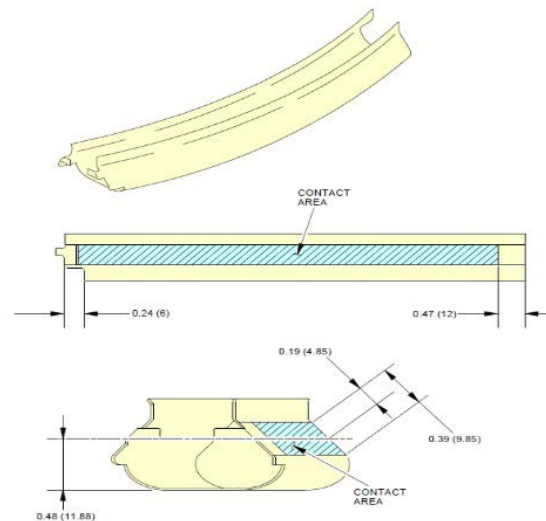


FIGURE 21: SHIM CU-NI-IN CONTACT AREAS
FIGURE COURTESY OF CFM

4.3 DOVETAIL RELUBE REQUIREMENTS AT THE TIME OF THE EVENT

After the fan blade hardware is deemed acceptable they are lubricated before installation. AMM TASK 72-21-00-640-801 titled *Fan Rotor Blades and Fan Disk Lubrication* provided instructions for the application of the lubricant; the lube is applied to the fan blade spacer; threaded pins on the fan disk, booster spool and retaining flange; fan disk dovetail slots pressure faces, aft face of the fan blade shank, the dovetail pressure faces, and the shims inner and outer pressure faces. Prior to the application of the lube, the surfaces are cleaned of all dirt and unwanted material with one of three specified solvents. TASK 72-21-00-640-801 references Boeing standard practices manual (SPM) TASK 70-10-05-600-801 titled *Application of Molybdenum Disulfide And Other Lubricants Dispersed In An Inorganic Binder And Non-Flammable Solvent - Maintenance Practices* calls-out the lube specification. Two types of lube are authorized; a spray application Molykote™ D321 R or D00641 Dow Corning 321 lubricant [consumable product (CP) 5062] or a brush-on application D50019 Molykote™ G-n Plus lubricant [CP2444] or D50308 lubricant [CP 2171]. For the spray application, coating thickness shall be approximately 0.0003-0.0005-inches (0.007-0.013mm) and for the brush-on the coating thickness shall be approximately 0.004-0.008-inches (0.1-0.2mm).

SWA created a TASK card, Work Instruction No. 772-21-20 titled *CFM56-7 Fan Blade, Disk and Disk Dovetail Slot Cleaning, Inspection and Relubrication* which provides the list of references,

materials, equipment, and instructions for performing the relube task. According to the SWA task card, the fan blade inspection and relube is a Required Inspection Item (RII)²¹ that requires the sign off the mechanic performing the task and as well as that of an inspector who ensures that the task was performed in accordance with the maintenance instructions. The lube that is specified on the SWA TASK card is the brushed-on solid film (paste) Molykote™ G-n Plus lubricant [CP2444] only.

5.0 ESM FAN BLADE DOVETAIL INSPECTION AND REPAIR REQUIREMENTS

5.1 FAN BLADE DOVETAIL INSPECTION AND REPAIRS

There are no authorized on-wing repairs for the fan blade dovetail; all fan blade dovetail repairs procedures are either in the ESM or in stand-alone repair documents. When the fan blades are removed from the engine for overhaul, either an on-wing removal or during an engine shop visit, the fan blades are inspected in accordance with the CFM56-7B ESM TASK 72-21-01-200-001 titled *Fan Rotor Blade – Inspection*. The fan blades are subjected to a visual, dimensional, and an ultra-high sensitivity level 4²² FPI per SPM TASK 70-32-14-230-008 inspection. Based on the condition of the fan blades, the ESM calls-out various fan blade repairs; however, this section will only focus on those tasks that directly impact the fan blade dovetail and shim, especially the application of the Cu-Ni-In coating and shot-peening.

Since the Cu-Ni-In coating is removed during fan blade overhaul, no “incoming” or “as-received” inspection criteria for the coating is provided; instead the dovetail is inspected for fretting on the dovetail pressure faces; nicks, dents, and scratches on the dovetail pressure faces; and rub or wear on the shim contact area (SUBTASK 72-21-01-220-055-0 titled *Blade Root Area*). GKN was asked if they track or trend the amount and location of spalled Cu-Ni-In coating on the fan blades they receive; they indicated that coating condition is not tracked nor recorded when the fan blades are received and processed. However, they did anecdotally indicate receiving fan blades from various operators with spalled Cu-Ni-In coating but could not confirm the exact location nor extent of the spalling but that it was not that uncommon to see the Cu-Ni-In coating spalled or missing.

It should be noted that cracks in the dovetail are not specifically called out; however, the ESM addresses this in SUBTASK 72-21-01-220-051-0 titled *Blade Critical Area* and by stating in the introduction section that:

These instructions do not purport to cover all details or variations in equipment nor to provide for every possible contingency to be met in connection with installation, operation or maintenance. Should further information be desired or particular problems arise which are not covered sufficiently for purchaser's purpose, contact CFM International Product Support.

and

²¹ According to FAA Advisory Circular (AC) 120-16 titled *Air Carrier Maintenance Programs* Section 7, FAR Sections 121.369(b) and 135.427(b) require you to designate certain tasks as RII. RII are defined as

²² FPI techniques are classified based on the strength or detectability of the indication that is produced for several very small and tight fatigue cracks. The five sensitivity levels are: Level ½ - Ultra Low Sensitivity, Level 1 - Low Sensitivity, Level 2 - Medium Sensitivity, Level 3 - High Sensitivity, Level 4 - Ultra-High Sensitivity.

..., first inspect the part to determine what defect is present, and then refer to the defect line within the specific part or area paragraph. When the defect is located, check the Maximum Serviceable Limits line and Maximum Repairable line.

According to CFM, and corroborated by SWA and GKN, this is interpreted as any anomaly or damage that is not specifically covered in the ESM that makes the part serviceable or repairable is automatically considered **unserviceable**; therefore, detected cracks in the blade dovetail are neither serviceable nor repairable.

ESM TASK 72-21-01-300-002 titled *Fan Rotor Blade - Repair 002 - Reconditioning of Cu-Ni-In Thermal Spray Coating and on The Dry Film Lubricant Varnish on The Fan Rotor Blade Root*, provides the instructions for the application of the Cu-Ni-In coating and varnish. The general process steps for Repair 002 is as follows:

- Shot-peen blade root
- Abrasive blast - prepare blade root for application of the Cu-Ni-In coating
- Plasma spray Cu-Ni-In coating
 - Checking thickness
 - Tensile test
 - Metallographic examination
 - Hardness Test
- Prepare blade for dry film varnish
- Apply varnish
 - Check polymerization
 - Check thickness

If the Cu-Ni-In coating or dry film varnish are deemed unacceptable, the repair provides procedures for redoing either task by removing the unacceptable coating and preparing the surface for coating reapplication. There are two acceptable procedures for coating removal: chemical strip or abrasive blast.²³

Portions of this repair are proprietary and requires a technical licensee agreement and Repair Source Substitutional (RSS) prior to use; therefore, these procedures are not included in the ESM, instead TASK 72-21-01-300-002 references CFM repair REP 15-93387²⁴ titled *Reconditioning of the Cu.Ni.In Thermal Spray Coating and of the Dry Film Lubricant Varnish on the Fan Rotor Blade Root* for the repair instructions. As part of REP 15-93387, a shot-peening operation was specified but was not proprietary and did not require an RSS at the time that GKN performed the repairs; that has now been changed and the shot-peening operation does require an RSS (See Section 13.4 NEW FAN BLADE INSPECTION AND REPAIR OVERHAUL REQUIREMENTS AND TECHNIQUES for details). The details of the shot-peening operation at the time GKN performed the repair were provided in ESM TASK 72-21-01-300-004 titled *Stage 1 Fan Rotor Blade - Repair 004 - Shot-Peening of The Fan Rotor Blade*; CFM also issued a standalone repair document REP 15-93389 *Shot-Peening of the Fan Rotor Blade* that has the same instructions as what is provided in ESM Repair 004. For details of the revised shot-peening procedure

²³ Because abrasive blast can remove more parent material than the chemical strip, doing it twice – once to remove the unacceptable coating and then again to roughen the surface for the coating reapplication – could potentially adversely affect the benefit from the shot-peek process. Therefore, if the abrasive blast is used to remove the unacceptable coating, a shot-peening operation must occur before further processing.

²⁴ There are currently 10 revisions to REP 15-93387.

see Section 13.4 NEW FAN BLADE INSPECTION AND REPAIR OVERHAUL REQUIREMENTS AND TECHNIQUES for details.

The repair instructions for the reconditioning of the dovetail coating for CFM56-7B fan blades were issued in March 1999 at which time the repair did not require RSS. Therefore, all the technical data for reconditioning the Cu-Ni-In coating was once included in ESM TASK 72-21-01-300-002. Revisions 001 and 002, issued November 1999 and March 2001 respectively, introduced several process changes to REP 15-93387; however, the repair still did not require RSS. Between the time Revision 002 (March 2001) and Revision 003 (February 2003) were issued, CFM changed the repair status from “no substantiation required” to “substantiation required”. In doing so, all the technical data to perform the reconditioning of the dovetail coating that had been previously in the ESM was removed and replaced in part with a reference to Repair REP 15-93387 and a substantiation requirement (**FIGURE 22**).

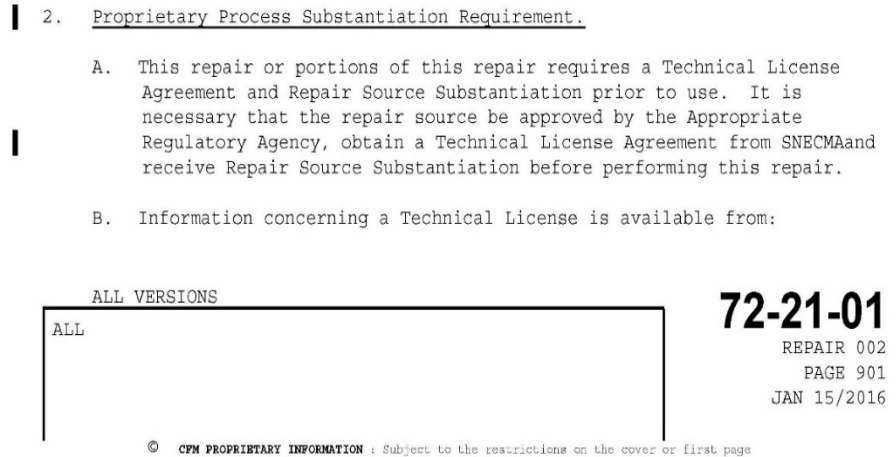


FIGURE 22: FAN BLADE DOVETAIL CU-NI-IN COATING AND VARNISH REPAIR – 002

FIGURE COURTESY OF CFM

REP 15-93387-009 stated that:

This repair contains critical processes of a high technology nature which could affect engine reliability and/or performance if improperly accomplished. Accordingly, CFM requires substantiation of these processes. Substantiation of repair processes or specific portions thereof, may be obtained by the repair source submitting to SNECMA documentation in accordance with the Substantiation Requirements for the specific repair. These requirements are available upon request from SNECMA

This means that the repair source (an approved Part 145 repair/overhaul shop) must prove that they can perform the repair in accordance with the repair specifications. In general, the approval/substantiation process starts with the repair shop applying to the original equipment manufacturer (OEM), in this case CFM, for source approval to perform a specific repair. The OEM then provides the repair shop the technical documentation to perform that repair and the substantiation program package that outlines what is needed to gain approval. The repair shop then submits the substantiation package, which can consist of paperwork, test specimens, actual repaired hardware, etc., to the OEM for review and evaluation. If the repair shop meets the requirements outlined in the substantiation program, the repair shop is granted repair source approval for that specific repair for which they applied, and they are placed on the OEMs Approved Repair Source List for that specific repair for which it received source approval. Once the process has been approved, it is locked and any deviations or alternates to the process or procedures require a process change approval from the OEM.

The CFM industrial substantiation process for the reconditioning of the Cu-Ni-In coating was and still is a two-step process; first step is process substantiation that is comprised of test specimens (coupons) than must be prepared and coated in accordance with the parameters specified in the repair document. The second step is part substantiation that is performed after the results of the first step are acceptable and require that an actual fan blade be prepared and coated using the same parameters as the test specimens. Once the first two steps have been successfully met, a provisional industrial substantiation sheet is provided to the repair source so that a re reproducibility check can be performed on the first set of blades to ensure that the repair conforms to the repair document. Once this reproducibility check is found acceptable, a final industrial substantiation sheet is provided to the repair source.

The event fan blade set was repaired in 2001 and again in 2007; both times at a GKN facility. GKN received their source approval in June 2003 in accordance with REP 15-93387 Revision 003. REP 15-93387 Revisions 003 – 009 (February 2003, May 2005, March 2008, June 2009, July 2009, November 2011, and again in November 2011, respectively) are part of the substantiation requirement, Revision 001 and 002 were issued when the Cu-Ni-In coating repair was not a source approved repair and the instructions were included in the ESM. According to CFM, a re-substantiation or validation of the repair process is not required when subsequent revisions are issued.

5.2 SHIM INSPECTION AND REPAIR

When the fan blades are sent to the repair/overhaul vendor (in the case of the SWA flight 3472 event fan blade set, the repair/overhaul vendor was GKN – See Section 6.3 REPAIR HISTORY OF THE EVENT FAN BLADE SET for details) the shims do not typically accompany the fan blades; this was confirmed by GKN, SWA, and CFM. Therefore, in these cases the assessment of the shims is performed either in the field by the aircraft mechanic in accordance with AMM SUBTASK 72-21-02-220-056 titled *Shim Examination* or at the engine maintenance shop in accordance with the ESM TASK 72-21-22-200-01 titled *Fan Blade Shim – Inspection/Check*. The AMM and ESM inspection criteria are essentially the same as far as defects, such as cracks, nicks, tears, wear etc.; the only differences are that the ESM serviceable limits for the scratch depth is less (in other words more restrictive), the AMM has axial retentions distortion permitted where the ESM has no serviceable limit (therefore, distortion is not allowed), and the AMM allows wear if the shim thickness is not lower than 0.014-inches (0.36mm) whereas the ESM does not allow wear.

5.3 FAN BLADE HARDWARE RELUBE

After the fan blade and associated hardware are deemed acceptable, the fan blade rear shank, the fan blade platform mating faces, the fan disk and booster spool locating pins, spacers and shims are lubricated using ESM TASK 70-63-15 titled *Application of Molybdenum Disulfide And Other Lubricants Dispersed In An Inorganic Binder And Non-Flammable Solvent* before installation. This is similar to the procedures outlined in AMM TASK 72-21-00-640-801. This task is not performed at the fan blade overhaul shop (GKN) but rather at the engine repair shop; GE Celma disassembled, repaired, and reassembled the engine during both event fan blade set shop visits (See Section 6.2 OPERATIONAL HISTORY - INSTALLATION, REMOVAL, AND RELUBE for details).

6.0 ESN 874112 EVENT FAN BLADE SET HISTORY

In this section, the SN for each blade will be truncated for simplicity as follows: (1) the “BA” will be dropped since all the event set fan blades started with “BA” and (2) ‘ - ’ (dash) number will not be included since technically it is not part of the SN. For example, fan blade No. 23, SN BA824478-6 will

be identified solely as SN 824478 and all the other fan blades from the event set will be identified accordingly.

6.1 MANUFACTURING HISTORY

All the fan blades from the event set were manufactured around the same time from 7 different material lots of round diameter Ti Imperial Metal Industries (IMI) 318²⁵ billets produced by Timet UK Ltd., Swansea, United Kingdom and subjected to UI before they were saw cut into long bar pieces and sent to Snecma²⁶, Gennevilliers, France for forging and machining. After the fan blade is forged and final machined, a blue etch inspection²⁷ and an FPI were performed to detect material anomalies. After the inspections were completed, the dovetail was shot-peened and the Cu-Ni-In coating applied. Metallurgical examination of the event fan blade set found that 7 fan blades, including the separated blade (No. 23) were found cracked in the dovetail; blades Nos. 1, 2, 11, 17, 19, 23, and 24 and they were produced from different material lots (**TABLE 3**). No material or manufacturing anomalies were noted during the laboratory examination; therefore, extensive documentation of the manufacturing records was not warranted. The entire event fan blade set was manufactured and introduced into service as PN 340-001-022-0.

TABLE 3: FAN BLADE MANUFACTURING MATERIAL LOT NUMBER		
Blade Number	Blade SN	Material Lot Number
1	786555	DL047812
2	725664	DL047734
11	770956	DL047806
17	783810	DL047806
19	767285	DL047810
23	824478	DL047831
24	824482	DL047831

6.2 OPERATIONAL HISTORY - INSTALLATION, REMOVAL, AND RELUBE

As previously mentioned, CFM56-7B fan blades are serialized but are not life-limited so fan blade time and cycles were not required to be tracked; however, according to SWA they started to track fan blade time and cycles starting in 2015. Since the event fan blade set stayed together as a set, the TSN and CSN for each fan blade was calculated based on the engine times and cycles for the time they were installed on that engine. **FIGURE 23** gives a graphical representation of the event fan blade operational history.

TABLE 4 provides the event fan blade set ON/OFF and fan disk slot position record for the life of the event fan blades. All the fan blades were originally delivered to SWA as individual blades, PN 340-001-022-0, and were not part of an engine build nor a spare fan and booster assembly. The fan blade

²⁵ The British IMI designation is similar to the United States American Society for Testing and Materials (ASTM) or American Society for Metals (ASM) standards.

²⁶ For the manufacturing sections, Snecma will be used instead of the current name of the company – Safran Aircraft Engines – as Snecma was the name of the company (and is on all the paperwork).

²⁷ Blue etch anodize inspection technique is one in which a bluish conversion coating applied to Ti surfaces. This provides a high visual contrast distinction for certain surface anomalies such as cracks, material segregation, heat-treating imperfections, and abnormalities caused by machining.

set was composed of new fan blades fabricated from a variety of material lot numbers and those blades were originally installed into fan disk PN 340-000-410-0, SN BA599558. The fan disk configuration PN -410 **did not** require a shim (See Section 1.3.2 FAN BLADE for details). Fan disk SN BA599558 was installed on ESN 874177 and on March 28, 1998 the engine was installed new on airplane N710SW (0 TSN and 0 CSN) in the right position (No. 2) where it remained until June 2001. In May 2001, the first fan blade relube occurred with no fan blade hardware anomalies noted; the fan blade set had accumulated 11,242 hours TSN and 7,452 CSN at that time.²⁸ On June 21, 2001, ESN 874177 was removed from N710SW due to booster spool damage and the engine was sent to the GE Celma engine shop in Petrópolis, Brazil for repair. At the time the fan blade set was removed from ESN 874177, all the fan blades had accumulated 11,719 hours TSN and 7,754 CSN. The fan blade set was sent to GKN Aerospace/Chem-Tronics, Inc.²⁹ facility in Tulsa, Oklahoma for repair and reworked (See Section 6.3.1 GKN 2001 REPAIR OF THE EVENT FAN BLADE SET for details). Review of the GKN work orders (WO) for each of the 24 individual fan blades revealed that they were all processed and repaired alike.

While the event fan blade set was at GKN, SB 72-0253 was incorporated; the last three digits of the fan blade PN changed from -022-0 to -027-0 and a new static MW for each blade was required. The old MW was crossed-out and a new one was vibro-peened into the bottom of the blade dovetail in accordance with the SB. This is the second MW applied to the blades (first one was at new make). The fan blade set was then installed on fan disk PN 340-000-410-0 (fan disk configuration still **did not** require shims) SN BA483782 on ESN 874112.

On July 27, 2001, the ESN 874112 was installed on airplane N742SW in the left position (No. 1) where it stayed until December 2001. On December 22, 2001, ESN 874112 was removed from N742SW due to HPC damage and the engine was sent to the GE Strother engine facility in Arkansas City, Kansas for repair. No fan blade relubes were performed while the engine was installed on N742SW. The fan blades were removed to facilitate the engine repair and they had accumulated 13,765 hours TSN and 9,493 CSN at that time. After the engine repair, a new static MW for each blade was measured, a new MW was vibro-peened into the bottom of the blade dovetail (third new MW), and all the fan blades were installed on a different fan disk, PN 340-000-410-0 (fan disk configuration still **did not** require shims) SN DB495970 but was installed back on ESN 874112.

On April 13, 2002, the engine was installed on airplane N705SW in the left position where it remained until July 2006. On July 9, 2006, ESN 874112 was removed from N705SW due to a compressor stall and the engine was sent to GE On-Wing Support (GEOWS) engine facility in Grand Prairie, Texas for repair. The fan blades had accumulated 29,323 hours TSN and 18,299 CSN at that time. Between April 2002 and July 2006, six³⁰ fan blade set relubes were performed and no fan blade hardware anomalies noted. As part of these maintenance actions, the fan blades were removed, and a new MW was vibro-peened into the bottom of the blade dovetail (forth new MW), and all the fan blades were reinstalled on the same fan disk SN DB495970. Although no specific paperwork could be found for a fan blade repair that would warrant a new MW, SWA fan blade inspection requirements call for: remove, clean, inspect, perform minor repairs as required, lubricate, and reinstall. Had a minor repair been performed, such as an airfoil blends, then a new MW would be appropriate. A borescope inspection (BSI) of the

²⁸ The 7,452 cycles represent that longest duration the fan blade set experienced between dovetail lube/relube. Review of the on-wing fan blade relube documentation provided by SWA from the last engine overhaul in October 2007 showed that the fan blade set had been relubed well below the recommended 3,000 cycles interval on average every 1,274 cycles. According to SWA, this was done due to convenience and on was performed coincident with other maintenance tasks.

²⁹ For brevity GKN/Chem-Tronics and GKN Aerospace will be referred to simply as GKN.

³⁰ Fan blade set relubes were performed in July 2003, March 2004, July 2004, March 2005, October 2005, and June 2006.

engine found no anomalies or findings and a water wash was performed before the engine was returned-to-service.

On August 21, 2006, ESN 874112 was installed on N766SW left position where it remained until June 2007. On June 29, 2007, ESN 874112 was removed from N766SW for a compressor stall and damage to compressor blades; fan blades had accumulated 32,614 hours TSN and 20,196 CSN at that time. Between August 2006 and June 2007, two³¹ fan blade set relubes were performed and no fan blade hardware anomalies were noted. The engine was sent to the GE Celma engine shop for repair. The fan blades were removed and sent to GKN Aerospace/Chem-Tronics facility in El Cajon California for inspection and repair (See Section 6.3.2 GKN 2007 REPAIR OF THE EVENT FAN BLADE SET for details). Review of the WOs for each of the 24 individual fan blades in the event set revealed that they were all processed and repaired alike and had the same tasks as performed back in 2001 – Inspection 001 and Repairs 001, 002, and 004. The only difference between the 2001 and 2007 GKN repairs was that in 2001 SB 72-0253 was complied with. After being inspected and repaired, the fan blades were shipped back to GE Celma where they were installed in fan disk PN 340-000-420 (this fan disk configuration **required** shims and spacers) SN DD763459 and the fan disk was installed back on the same engine, ESN 874112. As part of GKN repair, a new MW was vibro-peened into the bottom of the blade dovetail (fifth and final new MW).

On October 17, 2007, ESN 874112 was reinstalled on N766SW left position where it remained until the fan blade separation event (August 27, 2016). Between October 2007 and August 2016, 13³² fan blade set relubes were performed and no fan blade hardware anomalies were noted except for the January 2012, May 2013, and November 2015 relubes. On the January 2012³³ relube, two fan blade shims were found cracked and replaced; fan blade positions for the cracked shims were not identified on the paperwork.³⁴ Also, all eight of the fan blade retaining ring elastomer seals were damaged and sixteen axial stop pins were deteriorated; all this hardware was replaced. On the May 2013 relube, eight fan blade platform seals were found torn and were replaced; no blade positions were recorded. On the November 2015³⁵ relube, four fan blade shims were found cracked and replaced; no blade positions were recorded. At the time of the fan blade separation event, the fan blades had accumulated 2,397 hours and 1,396 cycles since the last relube and 62,851 hours TSN and 38,152 CSN. Since the engine was installed on the incident aircraft (AC), N766SW, in October 2007, the fan blades were relubed on average every 1,274 cycles, which appears from the data to be consistent through the lubrication life of this fan blade set.

Examination of the SWA task card *CFM56-7 Fan Blade, Disk and Disk Dovetail Slot Cleaning, Inspection and Relubrication* for all the fan blade set relubes showed that no individual fan blades were removed from service (all the fan blades stayed together as a set for their entire operational life), and no flight extensions were given for chipped or missing coating on the blade dovetail.

All the shims for the event fan blade set were the older style configuration PN 340-252-306-0 except for shims for fan blades Nos. 22-24 which were the new configuration PN 340-252-307-0. Shims were replaced for cracks during the January 2012 (two shims) and November 2015 (four shims

³¹ Fan blade set relubes were performed in October 2006 and June 2007.

³² Fan blade set relubes were performed in February 2008, October 2008, June 2009, January 2010, September 2010, May 2011, January 2012, October 2012, May 2013, December 2013, August 2014, April 2015, and November 2015.

³³ The fan blade set relube occurred on January 8, 2012 and the fan blades had accumulated 38,371 hours TSN and 23,697 CSN as of January 7, 2012.

³⁴ The fan blade set relube process does not require recording damage location relative to blade position in the aircraft records.

³⁵ The relube occurred on November 18, 2015 and the fan blades had accumulated 60,453 hours TSN and 36,756 CSN as of November 17, 2015.

replaced) relubes. Since the shims are not serialized, their position in the fan disk was not tracked like the fan blades. It is permissible to intermix the old and new shim configuration, and the new and old configurations were both available during the 2012 and 2015 shim replacements. It could not be positively confirmed during which shim replacement (January 2012 or November 2015) the new shim configuration was installed.

TABLE 4: FAN BLADE HISTORY – ENGINE AND FAN DISK LOCATION										Color Coding: YELLOW are cracked blade roots, Red is the failed blade, and Green are from the same material lot as failed blade	
Blade SN	Crack (Y/N)	ON	OFF	ON	OFF	ON	OFF	ON	OFF	ON	OFF
786555	Y	March 98 ESN 874177 Engine Pos (2) Blade Pos 1 disk BA599558 No Shim Req'd 1 Relube	June 2001 7,754 CSN	July 2001 ESN 874112 Engine Pos (1) Blade Pos 13 disk DB483782 No Shim Req'd 0 Relubes	Dec 2001 9,493 CSN	April 2002 ESN 874112 Engine Pos (1) Blade Pos 1 disk DB763459 No Shim Req'd 6 Relubes	July 2006 18,299 CSN	Aug 2006 ESN 874112 Engine Pos (1) Blade Pos 2 disk DB763459 No Shim Req'd 2 Relubes	June 2007 20,196 CSN	Oct 2007 ESN 874112 Engine Pos (1) Blade Pos 1 disk DD763459 Shim Installed 13 Relubes	Aug 2016 38,152 CSN
725664	Y	March 98 ESN 874177 Engine Pos (2) Blade Pos 19 disk BA599558 No Shim Req'd 1 Relube	June 2001 7,754 CSN	July 2001 ESN 874112 Engine Pos (1) Blade Pos 18 disk DB483782 No Shim Req'd 0 Relubes	Dec 2001 9,493 CSN	April 2002 ESN 874112 Engine Pos (1) Blade Pos 7 disk DB763459 No Shim Req'd 6 Relubes	July 2006 18,299 CSN	Aug 2006 ESN 874112 Engine Pos (1) Blade Pos 2 disk DB763459 No Shim Req'd 2 Relubes	June 2007 20,196 CSN	Oct 2007 ESN 874112 Engine Pos (1) Blade Pos 2 disk DD763459 Shim Installed 13 Relubes	Aug 2016 38,152 CSN
725666	N	March 98 ESN 874177 Engine Pos (2) Blade Pos 6 disk BA599558 No Shim Req'd 1 Relube	June 2001 7,754 CSN	July 2001 ESN 874112 Engine Pos (1) Blade Pos 7 disk DB483782 No Shim Req'd 0 Relubes	Dec 2001 9,493 CSN	April 2002 ESN 874112 Engine Pos (1) Blade Pos 19 disk DB763459 No Shim Req'd 6 Relubes	July 2006 18,299 CSN	Aug 2006 ESN 874112 Engine Pos (1) Blade Pos 3 disk DB763459 No Shim Req'd 2 Relubes	June 2007 20,196 CSN	Oct 2007 ESN 874112 Engine Pos (1) Blade Pos 3 disk DD763459 Shim Installed 13 Relubes	Aug 2016 38,152 CSN
788424	N	March 98 ESN 874177 Engine Pos (2) Blade Pos 2 disk BA599558 No Shim Req'd 1 Relube	June 2001 7,754 CSN	July 2001 ESN 874112 Engine Pos (1) Blade Pos 14 disk DB483782 No Shim Req'd 0 Relubes	Dec 2001 9,493 CSN	April 2002 ESN 874112 Engine Pos (1) Blade Pos 5 disk DB763459 No Shim Req'd 6 Relubes	July 2006 18,299 CSN	Aug 2006 ESN 874112 Engine Pos (1) Blade Pos 4 disk DB763459 No Shim Req'd 2 Relubes	June 2007 20,196 CSN	Oct 2007 ESN 874112 Engine Pos (1) Blade Pos 4 disk DD763459 Shim Installed 13 Relubes	Aug 2016 38,152 CSN
725665	N	March 98 ESN 874177 Engine Pos (2) Blade Pos 8 disk BA599558 No Shim Req'd 1 Relube	June 2001 7,754 CSN	July 2001 ESN 874112 Engine Pos (1) Blade Pos 8 disk DB483782 No Shim Req'd 0 Relubes	Dec 2001 9,493 CSN	April 2002 ESN 874112 Engine Pos (1) Blade Pos 23 disk DB763459 No Shim Req'd 6 Relubes	July 2006 18,299 CSN	Aug 2006 ESN 874112 Engine Pos (1) Blade Pos 5 disk DB763459 No Shim Req'd 2 Relubes	June 2007 20,196 CSN	Oct 2007 ESN 874112 Engine Pos (1) Blade Pos 5 disk DD763459 Shim Installed 13 Relubes	Aug 2016 38,152 CSN
725668	N	March 98 ESN 874177 Engine Pos (2) Blade Pos 20 disk BA599558 No Shim Req'd 1 Relube	June 2001 7,754 CSN	July 2001 ESN 874112 Engine Pos (1) Blade Pos 20 disk DB483782 No Shim Req'd 0 Relubes	Dec 2001 9,493 CSN	April 2002 ESN 874112 Engine Pos (1) Blade Pos 11 disk DB763459 No Shim Req'd 6 Relubes	July 2006 18,299 CSN	Aug 2006 ESN 874112 Engine Pos (1) Blade Pos 6 disk DB763459 No Shim Req'd 2 Relubes	June 2007 20,196 CSN	Oct 2007 ESN 874112 Engine Pos (1) Blade Pos 6 disk DD763459 Shim Installed 13 Relubes	Aug 2016 38,152 CSN
824504	N	March 98 ESN 874177 Engine Pos (2) Blade Pos 5 disk BA599558 No Shim Req'd 1 Relube	June 2001 7,754 CSN	July 2001 ESN 874112 Engine Pos (1) Blade Pos 5 disk DB483782 No Shim Req'd 0 Relubes	Dec 2001 9,493 CSN	April 2002 ESN 874112 Engine Pos (1) Blade Pos 24 disk DB763459 No Shim Req'd 6 Relubes	July 2006 18,299 CSN	Aug 2006 ESN 874112 Engine Pos (1) Blade Pos 7 disk DB763459 No Shim Req'd 2 Relubes	June 2007 20,196 CSN	Oct 2007 ESN 874112 Engine Pos (1) Blade Pos 7 disk DD763459 Shim Installed 13 Relubes	Aug 2016 38,152 CSN
788424	N	March 98 ESN 874177 Engine Pos (2) Blade Pos 23 disk BA599558 No Shim Req'd 1 Relube	June 2001 7,754 CSN	July 2001 ESN 874112 Engine Pos (1) Blade Pos 23 disk DB483782 No Shim Req'd 0 Relubes	Dec 2001 9,493 CSN	April 2002 ESN 874112 Engine Pos (1) Blade Pos 18 disk DB763459 No Shim Req'd 6 Relubes	July 2006 18,299 CSN	Aug 2006 ESN 874112 Engine Pos (1) Blade Pos 8 disk DB763459 No Shim Req'd 2 Relubes	June 2007 20,196 CSN	Oct 2007 ESN 874112 Engine Pos (1) Blade Pos 8 disk DD763459 Shim Installed 13 Relubes	Aug 2016 38,152 CSN
824502	N	March 98 ESN 874177 Engine Pos (2) Blade Pos 3 disk BA599558 No Shim Req'd 1 Relube	June 2001 7,754 CSN	July 2001 ESN 874112 Engine Pos (1) Blade Pos 15 disk DB483782 No Shim Req'd 0 Relubes	Dec 2001 9,493 CSN	April 2002 ESN 874112 Engine Pos (1) Blade Pos 22 disk DB763459 No Shim Req'd 6 Relubes	July 2006 18,299 CSN	Aug 2006 ESN 874112 Engine Pos (1) Blade Pos 9 disk DB763459 No Shim Req'd 2 Relubes	June 2007 20,196 CSN	Oct 2007 ESN 874112 Engine Pos (1) Blade Pos 9 disk DD763459 Shim Installed 13 Relubes	Aug 2016 38,152 CSN
824435	N	March 98 ESN 874177 Engine Pos (2) Blade Pos 9 disk BA599558	June 2001	July 2001 ESN 874112 Engine Pos (1) Blade Pos 16 disk DB483782	Dec 2001	April 2002 ESN 874112 Engine Pos (1) Blade Pos 8 disk DB763459	July 2006	Aug 2006 ESN 874112 Engine Pos (1) Blade Pos 10 disk DB763459	June 2007	Oct 2007 ESN 874112 Engine Pos (1) Blade Pos 10 disk DD763459	Aug 2016

TABLE 4: FAN BLADE HISTORY – ENGINE AND FAN DISK LOCATION

Color Coding: **YELLOW** are cracked blade roots, **Red** is the failed blade, and **Green** are from the same material lot as failed blade

Blade SN	Crack (Y/N)	ON	OFF	ON	OFF	ON	OFF	ON	OFF	ON	OFF
		No Shim Req'd 1 Relube	7,754 CSN	No Shim Req'd 0 Relubes		No Shim Req'd 6 Relubes	18,299 CSN	No Shim Req'd 2 Relubes	20,196 CSN	Shim Installed 13 Relubes	38,152 CSN
770956	N	March 98 ESN 874177 Engine Pos (2) Blade Pos 21 disk BA599558 No Shim Req'd 1 Relube	June 2001 7,754 CSN	July 2001 ESN 874112 Engine Pos (1) Blade Pos 21 disk DB483782 No Shim Req'd 0 Relubes	Dec 2001 9,493 CSN	April 2002 ESN 874112 Engine Pos (1) Blade Pos 16 disk DB763459 No Shim Req'd 6 Relubes	July 2006 18,299 CSN	Aug 2006 ESN 874112 Engine Pos (1) Blade Pos 11 disk DB763459 No Shim Req'd 2 Relubes	June 2007 20,196 CSN	Oct 2007 ESN 874112 Engine Pos (1) Blade Pos 11 disk DD763459 Shim Installed 13 Relubes	Aug 2016 38,152 CSN
746470	N	March 98 ESN 874177 Engine Pos (2) Blade Pos 10 disk BA599558 No Shim Req'd 1 Relube	June 2001 7,754 CSN	July 2001 ESN 874112 Engine Pos (1) Blade Pos 10 disk DB483782 No Shim Req'd 0 Relubes	Dec 2001 9,493 CSN	April 2002 ESN 874112 Engine Pos (1) Blade Pos 2 disk DB763459 No Shim Req'd 6 Relubes	July 2006 18,299 CSN	Aug 2006 ESN 874112 Engine Pos (1) Blade Pos 12 disk DB763459 No Shim Req'd 2 Relubes	June 2007 20,196 CSN	Oct 2007 ESN 874112 Engine Pos (1) Blade Pos 12 disk DD763459 Shim Installed 13 Relubes	Aug 2016 38,152 CSN
786481	N	March 98 ESN 874177 Engine Pos (2) Blade Pos 13 disk BA599558 No Shim Req'd 1 Relube	June 2001 7,754 CSN	July 2001 ESN 874112 Engine Pos (1) Blade Pos 1 disk DB483782 No Shim Req'd 0 Relubes	Dec 2001 9,493 CSN	April 2002 ESN 874112 Engine Pos (1) Blade Pos 13 disk DB763459 No Shim Req'd 6 Relubes	July 2006 18,299 CSN	Aug 2006 ESN 874112 Engine Pos (1) Blade Pos 13 disk DB763459 No Shim Req'd 2 Relubes	June 2007 20,196 CSN	Oct 2007 ESN 874112 Engine Pos (1) Blade Pos 13 disk DD763459 Shim Installed 13 Relubes	Aug 2016 38,152 CSN
783812	N	March 98 ESN 874177 Engine Pos (2) Blade Pos 18 disk BA599558 No Shim Req'd 1 Relube	June 2001 7,754 CSN	July 2001 ESN 874112 Engine Pos (1) Blade Pos 6 disk DB483782 No Shim Req'd 0 Relubes	Dec 2001 9,493 CSN	April 2002 ESN 874112 Engine Pos (1) Blade Pos 3 disk DB763459 No Shim Req'd 6 Relubes	July 2006 18,299 CSN	Aug 2006 ESN 874112 Engine Pos (1) Blade Pos 14 disk DB763459 No Shim Req'd 2 Relubes	June 2007 20,196 CSN	Oct 2007 ESN 874112 Engine Pos (1) Blade Pos 14 disk DD763459 Shim Installed 13 Relubes	Aug 2016 38,152 CSN
786482	N	March 98 ESN 874177 Engine Pos (2) Blade Pos 24 disk BA599558 No Shim Req'd 1 Relube	June 2001 7,754 CSN	July 2001 ESN 874112 Engine Pos (1) Blade Pos 12 disk DB483782 No Shim Req'd 0 Relubes	Dec 2001 9,493 CSN	April 2002 ESN 874112 Engine Pos (1) Blade Pos 9 disk DB763459 No Shim Req'd 6 Relubes	July 2006 18,299 CSN	Aug 2006 ESN 874112 Engine Pos (1) Blade Pos 15 disk DB763459 No Shim Req'd 2 Relubes	June 2007 20,196 CSN	Oct 2007 ESN 874112 Engine Pos (1) Blade Pos 15 disk DD763459 Shim Installed 13 Relubes	Aug 2016 38,152 CSN
783814	N	March 98 ESN 874177 Engine Pos (2) Blade Pos 12 disk BA599558 No Shim Req'd 1 Relube	June 2001 7,754 CSN	July 2001 ESN 874112 Engine Pos (1) Blade Pos 24 disk DB483782 No Shim Req'd 0 Relubes	Dec 2001 9,493 CSN	April 2002 ESN 874112 Engine Pos (1) Blade Pos 17 disk DB763459 No Shim Req'd 6 Relubes	July 2006 18,299 CSN	Aug 2006 ESN 874112 Engine Pos (1) Blade Pos 16 disk DB763459 No Shim Req'd 2 Relubes	June 2007 20,196 CSN	Oct 2007 ESN 874112 Engine Pos (1) Blade Pos 16 disk DD763459 Shim Installed 13 Relubes	Aug 2016 38,152 CSN
783810	N	March 98 ESN 874177 Engine Pos (2) Blade Pos 7 disk BA599558 No Shim Req'd 1 Relube	June 2001 7,754 CSN	July 2001 ESN 874112 Engine Pos (1) Blade Pos 19 disk DB483782 No Shim Req'd 0 Relubes	Dec 2001 9,493 CSN	April 2002 ESN 874112 Engine Pos (1) Blade Pos 15 disk DB763459 No Shim Req'd 6 Relubes	July 2006 18,299 CSN	Aug 2006 ESN 874112 Engine Pos (1) Blade Pos 17 disk DB763459 No Shim Req'd 2 Relubes	June 2007 20,196 CSN	Oct 2007 ESN 874112 Engine Pos (1) Blade Pos 17 disk DD763459 Shim Installed 13 Relubes	Aug 2016 38,152 CSN
783811	N	March 98 ESN 874177 Engine Pos (2) Blade Pos 14 disk BA599558 No Shim Req'd 1 Relube	June 2001 7,754 CSN	July 2001 ESN 874112 Engine Pos (1) Blade Pos 2 disk DB483782 No Shim Req'd 0 Relubes	Dec 2001 9,493 CSN	April 2002 ESN 874112 Engine Pos (1) Blade Pos 21 disk DB763459 No Shim Req'd 6 Relubes	July 2006 18,299 CSN	Aug 2006 ESN 874112 Engine Pos (1) Blade Pos 18 disk DB763459 No Shim Req'd 2 Relubes	June 2007 20,196 CSN	Oct 2007 ESN 874112 Engine Pos (1) Blade Pos 18 disk DD763459 Shim Installed 13 Relubes	Aug 2016 38,152 CSN
767285	N	March 98 ESN 874177 Engine Pos (2) Blade Pos 17 disk BA599558 No Shim Req'd 1 Relube	June 2001 7,754 CSN	July 2001 ESN 874112 Engine Pos (1) Blade Pos 17 disk DB483782 No Shim Req'd 0 Relubes	Dec 2001 9,493 CSN	April 2002 ESN 874112 Engine Pos (1) Blade Pos 17 disk DB763459 No Shim Req'd 6 Relubes	July 2006 18,299 CSN	Aug 2006 ESN 874112 Engine Pos (1) Blade Pos 19 disk DB763459 No Shim Req'd 2 Relubes	June 2007 20,196 CSN	Oct 2007 ESN 874112 Engine Pos (1) Blade Pos 19 disk DD763459 Shim Installed 13 Relubes	Aug 2016 38,152 CSN
725669	N	March 98 ESN 874177 Engine Pos (2) Blade Pos 11 disk BA599558 No Shim Req'd 1 Relube	June 2001 7,754 CSN	July 2001 ESN 874112 Engine Pos (1) Blade Pos 11 disk DB483782 No Shim Req'd 0 Relubes	Dec 2001 9,493 CSN	April 2002 ESN 874112 Engine Pos (1) Blade Pos 6 disk DB763459 No Shim Req'd 6 Relubes	July 2006 18,299 CSN	Aug 2006 ESN 874112 Engine Pos (1) Blade Pos 20 disk DB763459 No Shim Req'd 2 Relubes	June 2007 20,196 CSN	Oct 2007 ESN 874112 Engine Pos (1) Blade Pos 20 disk DD763459 Shim Installed 13 Relubes	Aug 2016 38,152 CSN

TABLE 4: FAN BLADE HISTORY – ENGINE AND FAN DISK LOCATION										Color Coding: YELLOW are cracked blade roots, Red is the failed blade, and Green are from the same material lot as failed blade	
Blade SN	Crack (Y/N)	ON	OFF	ON	OFF	ON	OFF	ON	OFF	ON	OFF
824505	N	March 98 ESN 874177 Engine Pos (2) Blade Pos 15 disk BA599558 No Shim Req'd 1 Relube	June 2001 7,754 CSN	July 2001 ESN 874112 Engine Pos (1) Blade Pos 3 disk DB483782 No Shim Req'd 0 Relubes	Dec 2001 9,493 CSN	April 2002 ESN 874112 Engine Pos (1) Blade Pos 10 disk DB763459 No Shim Req'd 6 Relubes	July 2006 18,299 CSN	Aug 2006 ESN 874112 Engine Pos (1) Blade Pos 21 disk DB763459 No Shim Req'd 2 Relubes	June 2007 20,196 CSN	Oct 2007 ESN 874112 Engine Pos (1) Blade Pos 21 disk DD763459 Shim Installed 13 Relubes	Aug 2016 38,152 CSN
767290	N	March 98 ESN 874177 Engine Pos (2) Blade Pos 4 disk BA599558 No Shim Req'd 1 Relube	June 2001 7,754 CSN	July 2001 ESN 874112 Engine Pos (1) Blade Pos 9 disk DB483782 No Shim Req'd 0 Relubes	Dec 2001 9,493 CSN	April 2002 ESN 874112 Engine Pos (1) Blade Pos 4 disk DB763459 No Shim Req'd 6 Relubes	July 2006 18,299 CSN	Aug 2006 ESN 874112 Engine Pos (1) Blade Pos 22 disk DB763459 No Shim Req'd 2 Relubes	June 2007 20,196 CSN	Oct 2007 ESN 874112 Engine Pos (1) Blade Pos 22 disk DD763459 Shim Installed 13 Relubes	Aug 2016 38,152 CSN
824478	Y	March 98 ESN 874177 Engine Pos (2) Blade Pos 16 disk BA599558 No Shim Req'd 1 Relube	June 2001 7,754 CSN	July 2001 ESN 874112 Engine Pos (1) Blade Pos 4 disk DB483782 No Shim Req'd 0 Relubes	Dec 2001 9,493 CSN	April 2002 ESN 874112 Engine Pos (1) Blade Pos 20 disk DB763459 No Shim Req'd 6 Relubes	July 2006 18,299 CSN	Aug 2006 ESN 874112 Engine Pos (1) Blade Pos 23 disk DB763459 No Shim Req'd 2 Relubes	June 2007 20,196 CSN	Oct 2007 ESN 874112 Engine Pos (1) Blade Pos 23 disk DD763459 Shim Installed 13 Relubes	Aug 2016 38,152 CSN
824482	Y	March 98 ESN 874177 Engine Pos (2) Blade Pos 22 disk BA599558 No Shim Req'd 1 Relube	June 2001 7,754 CSN	July 2001 ESN 874112 Engine Pos (1) Blade Pos 22 disk DB483782 No Shim Req'd 0 Relubes	Dec 2001 9,493 CSN	April 2002 ESN 874112 Engine Pos (1) Blade Pos 14 disk DB763459 No Shim Req'd 6 Relubes	July 2006 18,299 CSN	Aug 2006 ESN 874112 Engine Pos (1) Blade Pos 24 disk DB763459 No Shim Req'd 2 Relubes	June 2007 20,196 CSN	Oct 2007 ESN 874112 Engine Pos (1) Blade Pos 24 disk DD763459 Shim Installed 13 Relubes	Aug 2016 38,152 CSN

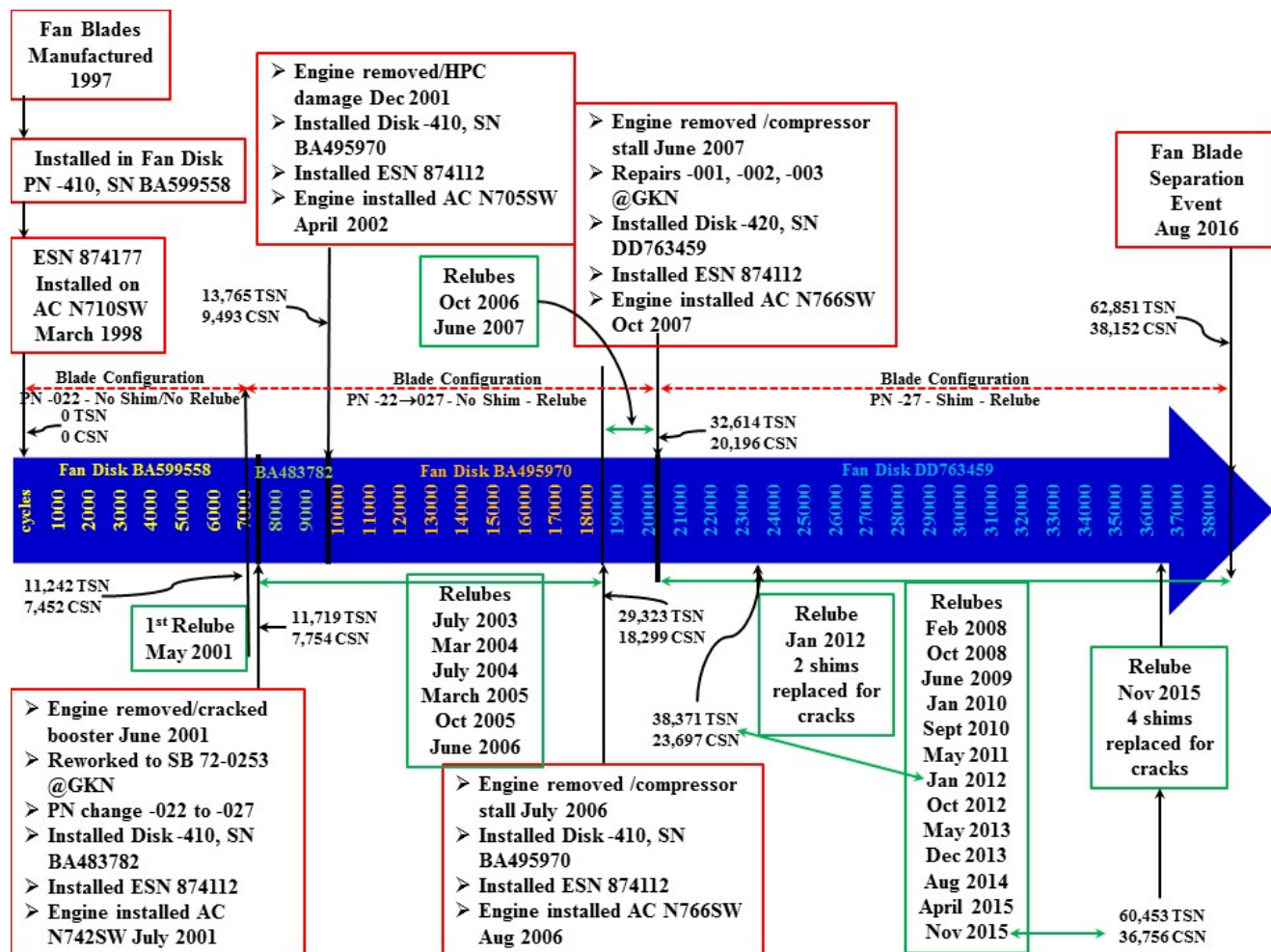


FIGURE 23: EVENT FAN BLADE SET TIMELINE

6.3 REPAIR HISTORY OF THE EVENT FAN BLADE SET

The event fan blade set was repaired in 2001 and again in 2007; both times at a GKN facility. As mentioned previously, all 24 fan blades remained together from when they were installed new in 1998 until the fan blade separation event in August 2016.

The Airworthiness Group convened at the GKN facility in El Cajon, California in November 2016 and again in March 2017 to: (1) review the 2001 (GKN Tulsa) and 2007 (GKN El Cajon) repair records for the event fan blade set, (2) review and observe the GKN's process for reconditioning the fan blade dovetail coating, (3) review of the coating test procedures, and (4) to examine the test coupons for the 2007 repair of the event blades. The GKN Tulsa facility closed in 2001 and the equipment and records have since been centralized at their El Cajon facility.

6.3.1 GKN 2001 Repair of the Event Fan Blade Set

At the time of the GKN/Chem-Tronics 2001 repair, the Cu-Ni-In coating reconditioning repair was not a substantiated process and the instructions for the repair were in the ESM. According to GKN, the procedures described below were typical of how the fan blades were processed in 2001. First, a single standalone preliminary receiving pre-traveler WO for all 24 fan blades was created. This pre-traveler specified the incoming general visual inspection and cleaning procedures. After the pre-traveler tasks were completed, individual traveler WOs for each of the fan blades were created; this paperwork traveled along with the part through all the maintenance tasks. The traveler provided the tasks that needed to be performed, the instructions on how that task was to be performed, and the signature box that showed that the task was completed, by whom, and when. Included in the travelers were specified GKN standalone datacards that provided the detailed instructions on how to perform a specific task. For example, clean for FPI per datacard *Clean-CFM-30*. All the GKN datacards were reviewed and compared with the same instructions specified by CFM ESM repairs, SPM, SB, or standalone repair documents; the workcards were consistent with CFM repair requirements. Any deviation, variations, or anomalies in the GKN traveler WOs were highlighted. Review of the traveler WOs for each of the 24 individual fan blades in the event set revealed that they were all processed and repaired alike.

Attached to all of the traveler WOs was an Aviation Repair – Inspection Report (IR) document No. GY5300 that stated that the Cu-Ni-In coating material test request No. **T2375** failed due to “failed microstructure test” with no specific reject criteria listed.³⁶ GKN, when asked for typical reasons for failures during that time speculated that if “failed microstructure test” was accurate, then the coating was rejected due to either oxides, unmelts, voids, or linear indications. GKN also noted typically, during that time, coating failed due to thickness or interface issues as well. Since the coating was deemed unacceptable, the coating was removed from the entire event fan blade set and reapplied. REP 15-93387 allows for two coating removal options; one is a chemical strip and the other an abrasive strip requiring shot-peening again. GKN performed the chemical strip; therefore, no additional shot-peening was required. The coating was reapplied and the material test request No. **T2381** was stamped on the paperwork indicating that the coating had been retested and the coating was deemed satisfactory. The test coupons for request Nos. **T2375** and **T2381** could not be located so no reevaluation could be performed.³⁷

³⁶ GKN performed the repair in accordance with REP 15-93387 Revision 1, which according to the document, twelve test specimens [coupons] must be coated at the same time as the blades to check if the coating parameters meets the specification.

³⁷ REP 15-93387 does not provide any specific guidance for retention of the work cards, test coupons, or testing results; however, there are applicable FAA regulations for maintenance record retention. Part 145 relates to the requirements of the certified repair station (GKN) and Part 121 relates to the air carrier (SWA). Section 145.219 *Repair Station Recordkeeping*

On July 20, 2001, an FAA Form 8130-3 Airworthiness Approval Tag Authorized Release Certificates was issued for each of the fan blades and the following information was provided in the Remarks section: Revision 16³⁸ of the ESM, inspection 001, repairs 001, 002, and 004, and SB 72-0253 were performed. Below is the description of the tasks performed on the fan blades.

TASK 72-21-01-200-001 *Fan Rotor Blade – Inspection* - As part of this repair a new static MW for the blade is required

TASK 72-21-01-300-001 *Fan Rotor Blade - Repair 001 - Blend Repair of the Blade Airfoil*

TASK 72-21-01-300-002 *Fan Rotor Blade - Repair 002 - Reconditioning of Cu-Ni-In Thermal Spray Coating and The Dry Film Lubricant Varnish on the Fan Rotor Blade Root*

TASK 72-21-01-300-004 *Stage 1 Fan Rotor Blade - Repair 004 - Shot-Peening of the Fan Rotor Blade*

SB 72-0253 titled *Engine – Fan and Booster (72-21-00) Introduction of a New Fan Blade and Rework of the Fan Blade Pressure Face Undercut* - Although not stated explicitly in the SB, a new static NW for the blade is required and is called out in the actual procedures used to comply with the SB.

6.3.2 GKN 2007 Repair of the Event Fan Blade Set

Since the metallurgical evaluation of the event fan blade set found variations with the dovetail coating thickness and inconsistent and low RS values, a close examination of the 2007 travelers as they relate to the Cu-Ni-In coating reconditioning repair was performed. The event fan blades were repaired by GKN in the El Cajon facility in August 2007 at which time they had already been source approved for the Cu-Ni-In coating reconditioning repair and were performing the repair in accordance with REP 15-93387 Revision 004. Review of the 2007 traveler WOs revealed that all the fan blades received the same inspection and repairs as in 2001 shop visits; inspection 001 and repairs 001, 002, and 004. The only difference was that during the GKN 2001 shop visit SB 72-0253 had been incorporated.

Repair Process Paperwork Review

Comparison of the actual WO travelers from the 2001 and the 2007 repairs found that they were principally the same except for some minor changes relating to paperwork and the addition of repairs not previously authorized or available in 2001. For example, in 2001 GKN had a single pre-traveler for incoming visual inspection and cleaning and then individual travelers for each fan blade for the rest of the process. In 2007, GKN incorporated the pre-travel requirements into the standard traveler and used batched travelers, six fan blades on one traveler instead of individual travelers. According to GKN, the CFM56-7 fan blade transportation cart used to route the fan blades from one work station to another hold six blades and all six travelers as a unit through each of the operations together, hence the use of a single traveler.³⁹

that a certificated repair station must retain records in English for at least 2 years from the date the article approved for return to service that demonstrate compliance with requirements of Part 43 in a format acceptable to the FAA; this is to mean the FAA Form 8130-3 Airworthiness Approval Tag Authorized Release Certificates. Section 121.380 *Maintenance Recording Requirements* that each certificate holder must retain records for the last complete overhaul of each airframe, engine, propeller.... Until the work is repeated or superseded by other work or for the one year after the work is performed.

³⁸ Revision 16 of the CFM56-7B ESM was issued on May 15, 2001.

³⁹ If any of the fan blades within the batched traveler require additional attention, the routing stops for all six fan blades; those blades that need additional processing are pulled out from the batch and split-travelers are created to document the additional

Blade Nos. 4, 14, 15, 17, 20, and 21 were routed initially on the same router WO G16922; however, No. 17 required additional repair and a split-traveler was created.⁴⁰ Blade Nos. 1, 5, 7, 8, 10, and 12 were routed initially on the same traveler WO G16923; however, blade No. 8 required additional repair and a split-traveler was created. Blade Nos. 2, 6, 11, 13, 16, and 19 were routed on the same traveler WO G16924; no anomalies, failed test, or reprocessing were noted on the traveler. Blade Nos. 3, 9, 18, 22, 23, and 24 were routed on the same traveler WO G16925; no anomalies, failed test, or reprocessing were noted on the traveler.

Blade Nos. 8 and 17 required an additional repair and split-travelers were created for each blade.⁴¹ Scratches were noted on the concave side of the airfoil on both blades and hand polishing was used to reduce the metal using a stoning and polishing wheel on the localized area. After the polishing, the blades were visually inspected, cleaned, and FPIed before they were routed back into the normal processing/repair cycle.

As previously mentioned, the event fan blade set had a total of seven cracked blades including the fractured fan blade No. 23; the cracked blades were Nos. 1, 2, 11, 17, 19, 23, and 24. Comparing the cracked blades against the 2007 GKN WOs in which they were processed revealed that the cracked blades spanned all 4 WOs; G16922 (blade No. 17), G16923 (blade No. 1), G16924 (blade Nos. 2, 11, and 19), and G16925 (blade Nos. 23 and 24). A review of all the WOs for the 2007 repair found no anomalies related to the coating application.

On August 13, 2007, an FAA Form 8130-3 Airworthiness Approval Tag Authorized Release Certificates was issued for each of the fan blades and the following information was provided in the Remarks section: Revision 34 of the ESM, inspection 001, repairs 001, 002, and 004 were performed. No additional information was noted in the remarks section of the FAA 8130-3 Airworthiness Approval Tag Authorized Release Certificates for the two blades (Nos. 8 and 17) that required additional processing and for which a split-traveler was created ([ATTACHMENT 2](#)).

Shot-Peening Process and Test Review

As previously mentioned, the shot-peening operation was not a proprietary process within the Cu-Ni-In coating reconditioning repair when GKN last repaired the event blades. Instead ESM TASK 72-21-01-300-004 provided the specific requirements for the shot-peening repair: preparation, location, intensity, and testing. Both the airfoil and blade root were shot-peened; the blade root was shot-peened at a higher intensity. For the blade root, a specific steel shot is specified and Almen strip intensity saturation time required is 0.006A-0.009A (F15A to F23A) with 125% coverage per SPM TASK 70-52-12 with a surface roughness of 63 microinches (1.6 micrometers) maximum. The fan blade airfoil shot-peen requires a different saturation time but same steel shot and coverage requirement. Review of the

work. The blade(s) that require additional work are removed from the original batch of blades and are routed on their own split-traveler that has all the same tasks as the original router along with added unique repair/process task. Once all the task has been completed, the blades that had additional processing are issued their own FAA Form 8130-3 Airworthiness Approval Tag Authorized Release Certificates. The remaining blade(s) that did not require additional processing continue on the original router through the rest of the repair cycles (unless another split traveler is required) and are issued one FAA Form 8130-3 Airworthiness Approval Tag Authorized Release Certificates with all the blade SNs listed that were worked under that traveler WO.

⁴⁰ Only the last 6 digits of the GKN traveler/work order will be used for identification purposes.

⁴¹ For blade No. 8, it was originally processed on WO G16922 and a split-travel WO G16964 was created for the additional work. For blade No. 17, it was originally processed on WO G16922 and a split-travel WO G16963 was created for the additional work.

GKN *shot-peen process log sheet* for the event blade WOs revealed that they adhered to the specified requirements.

GKN demonstrated the shot-peening operation on the same automated shot-peening machine used to process the event blade set in 2007. Before any actual blades are processed, a simulated blade test piece (Almen strip is too large for an actual -7B blade root) with multiple Almen strips attached was first processed. If acceptable, the entire blade set was processed individually. For shot-peening the root, the blade was placed upside down in a protective barrel that isolates the airfoil from the root since they are shot-peened to different intensities. The barrel rotates with the blade root exposed while the multiple peening guns aimed at differing angles translates up and down. When the blade set was completed, another test of the shot-peening intensity was conducted with the simulated blade test piece with Almen strips attached. The before and after shot-peening testing is performed in accordance with the repair instructions. The only deviation from the repair procedures was the use of slightly harder grit media than what is called out. According to GKN, only one type of blast medium is used for all applications to eliminate any cross-contamination of the media. Adjustments to the machine setting are made to meet the peening requirements.

Abrasive Blast Process

REP 15-93387 Revision 004 (May 2005), which was in effect at the 2007 GKN shop visit, introduced several changes to the grit blast procedure: (1) changed the maximum wait time between the grit blast operation and the coating application without covering the blasted area from 4 hours to 2 hours, (2) added a measured roughness requirement, and (3) a visual inspection for uniformity and appearance. Currently, GKN uses a manual process, same as in 2007. In 2007, three coupons per side were placed on a test -7B blade to assess the grit blast quality; the current requirement is for only one coupon per side. If the quality of the grit blast was acceptable when visually compared to the standard, actual blades were processed. To ensure that grit blast operation only occurs where the Cu-Ni-In coating would be applied, the repair specifically calls out dimensions where the grit blasting is allowed; GKN employs a hand-held gage as a guide to ensure masking of the blade was done correctly. Review of the process revealed no anomalies or deviations from the repair specification.

Plasma Spray Cu-Ni-In Coating Process and Test Review

The application of the Cu-Ni-In coating is a proprietary process, so the instructions are only available in the repair document REP 15-93387 and a source approval was required. Revision 4 of that repair was what GKN used during the 2007 shop visit. The TASK for applying the coating provides instructions for masking the areas not to be sprayed, thermal spray application of the coating, and testing and metallographic examination of the coating to determine if it is acceptable. The coating was applied to the blade and test specimens. The test specimens are then subjected to: (1) a thickness measurement 0.12-0.18mm (0.0047-0.0071-inches), (2) a tensile bond test per SPM TASK 70-48-03, (3) a metallographic examination and visual comparison with a reference stand per SPM TASK 70-48-04, and (4) a hardness test per SPM TASK 70-31-12 with the hardness parameters specified in the repair document. The testing requirements used today are the same as those used in 2007 except for the hardness requirement. In 2007 (Revision 4), the Vickers hardness was only a lower limit of 120HV0.3 whereas the current revision (Revision 9) has the hardness requirement as 140HV0.3-180HV0.3. Revision 5, released in March 2008, changed the hardness to 140HV0.3-180HV0.3.

GKN demonstrated the automated plasma spray coating operation in the same spray booth as the event blade set was processed back in 2007; however, the spray gun and the holding fixture

were not the same. The spray gun used back in 2007 has since been replaced and periodically it is replaced as they wear. The -7B fan blade holding fixture had been modernized since the event fan blade set was processed back in 2007. The differences between the 2007 and the current version of the -7B blade holding fixtures was in the amount of fan blades it could hold and how the blades are held in the fixture. The current version holds more blades than its predecessor back in 2007 and the current version uses indexing pins for alignment whereas its predecessor used recesses in the shape of the bottom of the blade root to hold the blade in place. In either fixture, the blades are not locked into position but rather held in place by their own weight and each blade location is designated for either the concave or the convex side, the fixture geometry prevents blades from being inadvertently installed in the wrong location.

Since the fan blades are not secured to the fixture, there can be a potential that any minor misalignment (the fan blade not fully seated against the index pins) could potentially result in coating thickness variation and localized heating of the blade root resulting in a potential loss of residual compressive stress as seen during the metallurgical evaluation of the event fan blade set (See Section 3.2 DOVETAIL RESIDUAL STRESS PROFILES for details) (FIGURE 24, this figure is for representation purposes only and does not represent the number of fan blades processed at one time by GKN). To address overheating of the blades, GKN stated that periodic temperature tests using temperature strips are used to verify that the blade temperature does not exceed the specified temperature listed in REP 15-93387; there were no documented temperature reading anomalies for the event fan blade set. Also, GKN stated that they periodically verify and calibrate the spray gun travel (speed and location) relative to the blade position.

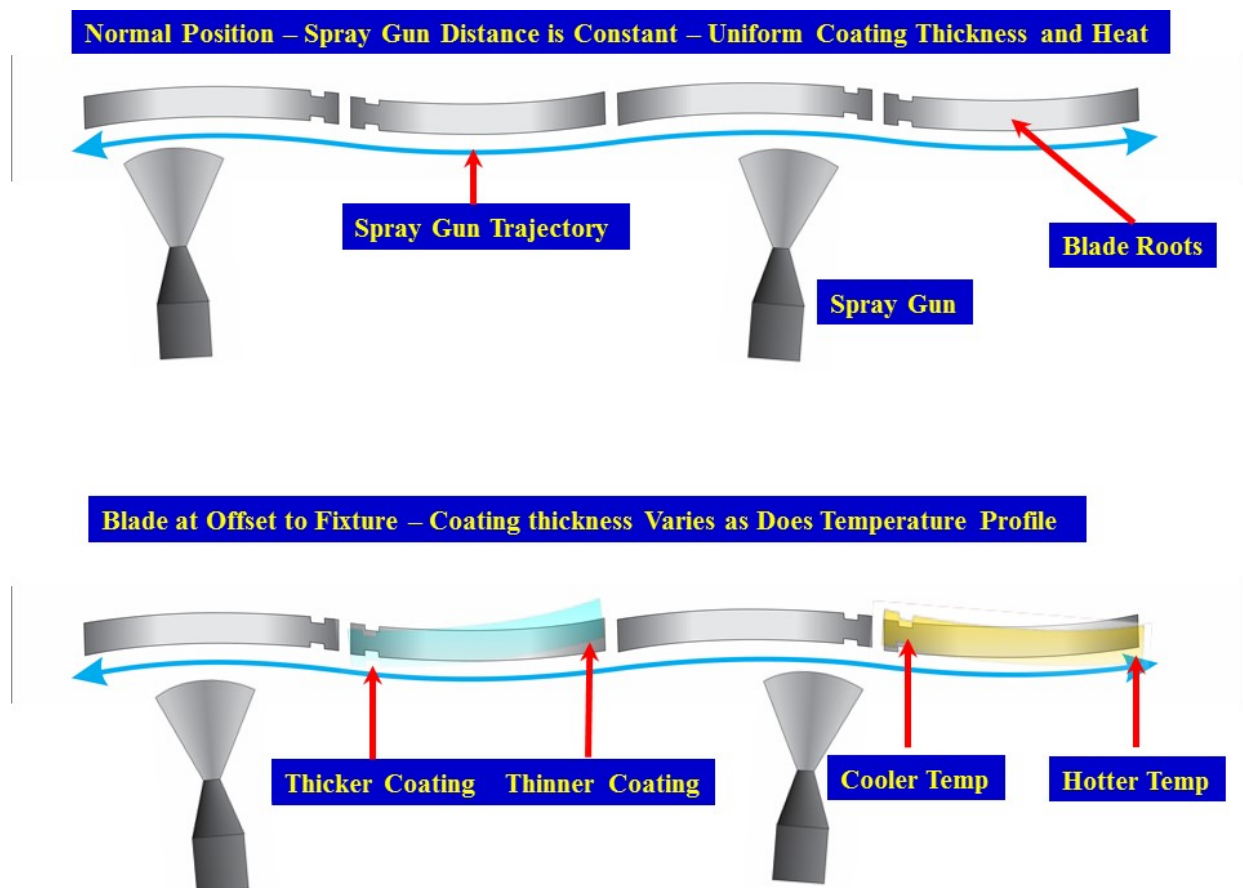


FIGURE 24: PLASMA SPRAY COATING FIXTURE AND POSSIBLE PROJECTIONS

To show consistency throughout the coating of the fan blade set, a test blade was sprayed at the beginning and at the end of the fan blade set. The test fan blade was fitted with three test coupons, one located at the LE, middle, and TE of the root, to ensure consistency of the coating along the length of the dovetail.

Coating Material Testing Review

GKN stated that all the test coupons for shot-peen and coating application were not evaluated after each individual process step (except the grit blast that is a visual inspection of coverage); instead, all the blades go through the entire process first, then the coupons were tested, and if the coupons fail any of the testing requirements, as in the 2001 repair, all the fan blades are reprocessed. Review of all the WOs for the 2007 repair showed the same material test request No. **B28790**. The test sheet No. **B28790** along with the test coupons with the same number were available for review and inspected by the Airworthiness Group. Review of the GKN *Snecma Laboratory Result Certificate – Plasmas Coating – CFM56-7B* and *Material Testing Request* sheets for request No. **B28790**, showed that six test coupons were created at the beginning and another six test coupons at the end of the fan blade set coating reconditioning were tested for hardness microstructure, and thickness. No material anomalies were noted when the Airworthiness Group reinspected the test coupons. The only anomaly or variation found had to do with the method for determining the coating hardness.

According to the *Laboratory Result Certificate*, each test coupon was tested three times using a 100-gram load instead of the specified 300-gram load and average of the measurements was used to categorize the overall hardness. The hardness ranged from 145HV0.1-239HV0.1; which was acceptable in accordance to REP 15-93387 Revision 4 Vickers minimum of 120HV0.3. GKN stated that due to the coating thickness that a 100-gram load was more appropriate and therefore used the 100-gram load instead of the 300-gram load. Hardness readings would be the same whether a 100-gram or 300-gram load was used, the impressed size would be different, but hardness reading correlated to weight and the impression surface area would create the same hardness. The test coupons were retested for hardness at GKN (See *NTSB Materials Laboratory Factual Report No. 17-043* Section 15 for additional details) using a 300-gram load. The hardness values for this retest ranged from 158HV0.3-219HV0.3; all well above the specified lower limit of 120HV0.3.

According to REP 15-93387, the coating thickness range is between 0.12-0.18mm (0.0047-0.0071-inches); the GKN *laboratory result certificate* has the same requirement for coating thickness as specified in the repair and in the NOTES section, a minimum and maximum thickness value is recorded as well as the predominant value; the predominant value is derived as the visual average across the majority of the length of the test coupon. The recorded minimum coating values ranged from 0.0039-0.0055-inches, the recorded maximum values ranged from 0.0066-0.0078-inches, and the predominant value ranged from 0.0051-0.0066-inches. All the predominant values were within the specified range. The test coupons were sent to the NTSB and spot checks were performed for coating thickness. The NTSB values were consistent with the GKN recorded values on their *laboratory result certificate*.

7.0 FAN BLADE STRESS ANALYSIS

CFM conducted a fan blade stress and vibratory analysis considering dovetail nominal coating thickness and local overthickness. For the as-specified configuration of a shim and nominal dovetail coating thickness, the static critical stress area is the dovetail LE convex side, forward of the location where the No. 23 fan blade crack initiated and eventually fractured. The vibratory analysis showed that both the convex side (where the No. 23 blade crack initiation) and concave LEs are critical areas with the

concave LE more critical. But when the local overthickness coating (CFM used +0.2mm (0.008-inches) above the maximum coating thickness value of 0.18mm (0.0071-inches)) was applied to the analysis, there was an increase in the calculated static stress with a shift of the critical area from the LE convex side to the area where the No. 23 fan blade crack initiated and fractured.

8.0 CFM56-7B FAN BLADE SEPARATION HISTORY

The CFM56-7B fan blade has experienced four partial or full FBO events in operation over its entire service history. The first two were partial fan blade separation at about 70% span (tip loss) resulting from a bird strike during takeoff. In each case, the remaining fan blades exhibited airfoil damage, remained intact, and secured with the fan hub. Neither the fan case nor the inlet was breached, the inlet remained attached, and no injuries were reported. Specifically, in both events, the containment shield was not breached by the fan blade fragment that was liberated as a result of each bird strike. The first full FBO event was SWA flight 3472, the basis of this report. The second full FBO event occurred on SWA flight 1380 from New York, New York to Dallas, Texas on April 27, 2018.⁴² Fragments from the engine inlet and cowlings struck the wing, fuselage, and one cabin window, resulted in a depressurization. The flight crew conducted an emergency descent and diverted to the Philadelphia International Airport (PHL), Philadelphia, Pennsylvania. Of the 144 passengers and five crewmembers onboard, one passenger received fatal injuries and eight passengers received minor injuries. The remaining fan blades exhibited airfoil damage, remained intact, and secured within the fan hub and the fan case did not exhibit any through-hole exit penetrations. The containment shield was not breached by the fan blade tip fragment. However, the inlet and the fan cowls sustained sufficient collateral damage such that portions of the inlet D-duct and well as large portions of the outboard and inboard fan cowl separated from the engine.

9.0 CONTAINMENT REQUIREMENTS

Uncontained engine failures can never be eliminated entirely; therefore, the FAA provides regulations and guidance on how engine and airplane manufacturers should try to prevent, mitigate, and address engine uncontained failures when they occur.⁴³ When the CFM56-7B and Boeing 737-700 were certificated (December 1996 for the engine and November 7, 1997 for the airframe), an uncontained engine failure event was generically divided into two major phases or categories. The first phase is the actual failure event where internal engine debris can be released from the engine; this is typically referred to as blade release which can include the impact phase, surge pressure load, and the rundown loads as the engine rapidly decelerates. With regards to FBO, what is included or accounted for in this first phase is the fan blade impact load, fan blade trajectories and energies, engine torque, engine thrust decay, and imbalance loads. The second phase is the effect the failed engine has on the airframe, systems, and occupants on board and the ability for a continued safe flight and landing. This is typically referred to as the windmill or 'Fly Home' phase and it includes the imbalance loads from a continuously windmilling engine and structural damage assessment of the inlet.⁴⁴ The basic windmilling speeds and loads are determined through flight test and analysis. The windmilling loads due to rotor imbalance after a fan blade separation are determined through the various rig and engine tests and also correlated with the flight test data. The approach used to analyze the FBO for the CFM56-7B and Boeing 737-700 combination was a blend of test data and analytically predicted loads using a Finite Element Model (FEM) to generate the on-wing predicted loads jointly developed by CFM and Boeing.

⁴² NTSB accident number DCA18MA142.

⁴³ Since the Boeing 737 and all its variants are defined as a transport category aircraft, only FAR Part 25 will be discussed.

⁴⁴ Since the CFM56-7B and Boeing 737-700 combination was certified, FBO analysis has progressed based on new information and technology to enhance the analysis of the fan case displacement wave due to the initial impact beyond what was accounted for in the 1990s.

Although the CFM56-7B24 is a dual certificated engine (United States and European), only the applicable FAA regulations and guidance will be discussed in detail unless there is a significant difference between the United States FAA Airworthiness Standards and the European JARs for a specific regulation pertaining to engine containment. According to the CFM56-7B24 TCDS No. E00055EN and the Boeing 737-700 TCDS No. A16WE, the engine and airplane TC approval was granted on December 17, 1996 and November 7, 1997, respectively. The certificate basis for the CFM56-7B24 was 14 CFR Part 33, effective February 1, 1965 with Amendments 33-1 through 33-15⁴⁵ while the certification basis for the Boeing 737-700 was 14 CFR Part 25 Amendments 25-1 through 25-77 with exceptions as stated in the TCDS. None of the exceptions as stated in TCDS No. A16WE relate to uncontained engine failures. For clarity and simplicity, when referring to specifying CFRs that pertain to aviation, they will be referenced as a FAR, then the applicable part or section will be referenced; for example, 14 CFR Part 33 AIRWORTHINESS STANDARDS: AIRCRAFT ENGINES will be referred to as FAR Part 33 and 14 CFR SUBPART B - DESIGN AND CONSTRUCTION; GENERAL Section 33.19 DURABILITY will be referred to as FAR Section 33.19; this also applies to the applicable FAR Part 25 AIRWORTHINESS STANDARDS: TRANSPORT CATEGORY AIRPLANES.

9.1 FAR PART 33 AIRWORTHINESS STANDARDS: AIRCRAFT ENGINES - TURBINE ENGINE CONTAINMENT REQUIREMENTS AND GUIDANCE MATERIAL

As it relates to turbine engines, the initial engine failure event and possible subsequent release of engine debris is addressed in engine requirements, FAR Section 33.19 DURABILITY and FAR Section 33.94 BLADE CONTAINMENT AND ROTOR UNBALANCE TESTS; both of these regulations pertaining to containment are the same now as they were when the CFM56-7B24 was certificated. FAR Section 33.19 requires the engine manufacturer (CFM in this case) "...design of the compressor and turbine rotor cases must provide for the containment of damage from rotor blade failure." Although not stated specifically, historically this has been interpreted that engine debris cannot penetrate and pass radially through any of the engine cases, but debris can exit axially out the front or back of the engine. FAR Section 33.19 also addresses engine debris exiting out the front and back of the engine by requiring that "... Energy levels and trajectories of fragments resulting from rotor blade failure that lie outside the compressor and turbine rotor cases must be defined." FAA advisory circular (AC) 33-5 titled *Turbine Engine Rotor Blade Containment/Durability* was issued on June 18, 1990 to provide guidance on acceptable methods for compliance (design and test) to meet FAR Sections 33.19, 33.94, and 33.75 (SAFETY ANALYSIS) requirements with regards to turbine engine rotor blade containment requirements.

According to AC 33-5 'contained' was defined to mean "that no fragments are released through the engine structure, but fragments may be ejected out of the engine air inlet or exhaust" and defined 'engine structure' to mean "the engine structure surrounding the main rotors and extending from the forward-most case flange through the rear most flange, as defined by the type design". AC 33-5 also provides the engine configuration, conditions, and acceptance for engine testing for blade containment (FAR Section 33.94). The engine configuration must meet the type design for those items and features of those items being tested, such as case thickness, retention of external components, blade design, rotor and rotor support structure, etc. and the rotor blade when released should provide the most challenge to the containment structure and the most rotor imbalance. A typical aircraft inlet and exhaust nozzle/ducting or equivalent (those having the same attachment loads and reactions which influence engine case deflections, containment capability and engine vibration response) should be used. The conditions by which the engine is tested for containment may be at (1) nominal sea level conditions, (2) the release of the 'critical rotor blade' at a maximum permissible operating rotor speed, (3) static structure at the worst

⁴⁵ Amendment 10 to FAR Section 33.19, FAR Section 33.75, and FAR Section 33.94 was applicable at the time CFM56-7B24 was certificated. Current versions as of July 2018 of FAR Sections 33.19 and 33.94 are the same as they were back in 1996 when the CFM56-7B24 was certificated; additional requirements have been included in FAR Section 33.75 since 1996.

associated operating temperature and stresses, or (4) any other realistic combination thereof agreed to be more critical, **and** (5) no engine control may be adjusted by the operator for at least 15 seconds after indication of excessive vibration or other evidence that would be available to the pilot in order to simulate the crew recognition and reaction time. The ‘critical rotor blade’ was defined to mean “the compressor or fan blade, and the turbine blade, which when released provides the most challenge to the containment structure...and the most rotor imbalance”. This ‘critical rotor blade’ is typically interpreted and demonstrated to mean fan blade as it generally has the greatest kinetic energy associated with it when there is a full blade release.

A successful engine FBO test (FAR Section 33.94) is one in which the engine cases are capable of containing damage (no radial pass-thru of the engine debris through the structure) without catching fire and without failure of its mounting attachments when operated for at least 15 seconds after the event and can be successfully shut down or the resulting damage results in a self-shutdown of the engine.

To meet the FAR Section 33.19 requirement to define the trajectories and energy levels of fragments that lie outside the engine cases, CFM performed a series of specialized hardware specific tests, commonly referred to as rig tests, to test and validate the capability and performance of the test hardware prior to an actual engine FBO certification test(s). Although both FAR Section 33.19 and AC 33-5 state that “...Energy levels and trajectories of fragments resulting from rotor blade failure that lie outside the compressor and turbine rotor cases must be defined”, neither define exactly in what form that information must be in⁴⁶ and how and at what level of detail is that information passed from the engine manufacturer to the airframe manufacture in order for the airframer to design and size the inlet structure to satisfy the FAR Part 25 containment requirements (See Section 9.2 TRANSPORT CATEGORY AIRPLANE FAR PART 25 CONTAINMENT REQUIREMENTS for details). CFM, Boeing, and UTAS through a series of meetings, memorandums, technical documents, etc. agreed upon what data was needed, how it should be transmitted, and how it should be correlated to ensure that it was accurate and representative of the test results. The resulting data was transmitted by Boeing to UTAS in the form of a Boeing Specification Control Document (SCD).

The type of information passed from CFM to Boeing, and by extension to UTAS as needed, included the size, shape, velocity, and exit trajectory of the fan blade fragments as it passed across the interface between the fan case A1 flange and the inlet case attach flange, as well as engine torque loads, imbalance loads, shear and moment loads, and surge pressure loads. The fragment exit velocity at the A1 flange is variable and is dependent on the low pressure rotor (N1) speed at separation, the size of the fragment, the distance from the center of rotation to the center of gravity of the fragment, and the fan case configuration, such as its energy absorption capability, distance from fan blade rotational plane, and other interference or drag inducing features.

9.2 TRANSPORT CATEGORY AIRPLANE FAR PART 25 CONTAINMENT REQUIREMENTS

As previously mentioned, the FAA approaches engine and airplane certification, with regards to rotor blade failure and its impact on continued safe operation of the airplane, as two separate phases. The first phase is the initial rotor blade failure, impact, and the containment thereof (commonly referred to as FBO) which is covered by FAR Part 33 AIRCRAFT ENGINES certification standards (FARs

⁴⁶ FAR Section 33.5 INSTRUCTION MANUAL FOR INSTALLING AND OPERATING THE ENGINE requires that installation instructions be documented in the installation manual. In Appendix 1 of AC 33-2B (this was the current version of the AC at the time the CFM56-7B was certificated) provided guidance on the minimum information that should be supplied at the time of engine certification.

Sections 33.19, Section 33.75, and Section 33.94 discussed in the Section 9.1 above). Furthermore, FAR Part 25 TRANSPORT CATEGORY AIRPLANES certification standards Section 25.361(b) specifies the requirements for engine mounts and supporting structure to withstand engine torque loads, including sudden engine stoppage due to malfunction or structural failure (such as compressor jamming). FAR Section 25.903(d)(1) requires that design precautions be taken to minimize the hazards to the airplane in the event of an engine rotor failure so that critical aircraft systems required for continued safe flight are protected from uncontained rotor fragments. In the case of the CFM56-7B and Boeing 737-700 combination, SPECIAL CONDITIONS (SC) was used in lieu of Section 25.361(b), or the more recently promulgated Section 25.362 for engine failure loads and Sections 25.571(e)⁴⁷, 25.629(d), and 25.903(d)(1) for protection of the airframe structure and systems from uncontained fan blades.

The second phase is the airplane's response following the fan blade failure (commonly referred to as windmill or 'Fly Home' – this will be discussed in greater detail below). FAR Sections 25.901(c) and Section 25.903(c) apply to this phase. FAR Section 25.901(c) (Subpart E– POWERPLANTS: INSTALLATION) requires that no single failure or malfunction or probable combination of failures in the powerplant installation will jeopardize the safe operation of the airplane. FAR Section 25.903(c) (Subpart E– Powerplants: ENGINES) requires means of stopping the rotation of an engine where continued rotation could jeopardize the safe operation of the airplane. These FAR Part 25 regulations address the consequences of a fan blade failure, which could include imbalance loads, damage tolerance capability, and critical systems functionality. Furthermore, the structural capabilities of the aircraft to withstand the effects of rotor blade failure during those two phases are addressed by FAR Part 25 regulations, such as FAR Sections 25.362 and 25.571(e). In essence, all of the applicable FAR Part 25 regulations are intended to evaluate potential engine failures and their consequences to the airplane, to ensure the airplane's continued safe flight and landing. This ability to continue to operate safely and land successfully is what is generally called the 'Fly Home' or 'windmilling' analysis. As described below for the Boeing 737-600/700/800 certification program, the FAA applied a SC 25-ANM-132, in lieu of FAR Section 25.361(b), to address engine torque loads imposed by sudden engine stoppage due to structural failure, including fan blade failure. For the assessment of windmilling following the fan blade release, Boeing agreed, via FAA Issue Paper A-2, to use the same process and requirements as had been used under the Boeing 777 certification program.

FAR Section 25.361 (Subpart C – Structures Supplementary Conditions: ENGINE AND AUXILIARY POWER UNIT TORQUE) requires that engine mounts, pylons and adjacent supporting airframe structure be designed to withstand 1g flight loads combined with transient dynamic loads and vibrations resulting from the specified engine structural failure condition. FAR Section 25-571(e) requires the airframe structure to be capable of continued safe and landing following an uncontained engine failure or impact from an uncontained fan blade. FAR Section 25.629(d) (Subpart D-DESIGN AND CONSTRUCTION: AEROELASTIC STABILITY REQUIREMENTS) requires aeroelastic stability evaluations including any undue loss of stability and control as a result of structural deformation resulting from failure of any single element of the structure supporting any engine, or large externally mounted aerodynamic body or any single failure of the engine structure that would reduce the rigidity of the rotational axis. The objective of these requirements is to ensure that the airplane is capable of continued safe flight and landing after sudden stoppage, engine rotor failure, or engine structural failure, including ensuing damage to other parts of the engine. The airplane structure, including the inlet and fan cowls, was designed to withstand the initial fan

⁴⁷ FAR Section 25.571 specifically addresses primary structure such as the pylon, wing, fuselage, etc. and FAR Section 25.571(e) addresses predicted fan blade impacts with the primary structure. The nacelle is categorized as secondary structure, which includes the inlet and fan cowls; therefore, FAR Section 25.571 did/does not specifically apply. However, Boeing did assess the inlet against the impact of the defined fan blade fragments provided from CFM.

blade release and subsequent damage as a result of a fan blade failure, and ensure that the airplane was capable of safe flight and landing. FBO test data and loads analysis was assessed as part of the design.

The FAA issued AC 20-128, titled *Design Considerations for Minimizing Hazards Caused by Uncontained Turbine Engine and Auxiliary Power Unit Rotor and Fan Blade Failure*, on March 9, 1988 to provide guidance on the methods for compliance applicable to design precautions intended to minimize hazards to the airplane in the event of an uncontained engine failure as required by FAR Section 25.903(d)(1). When the CFM56-7B/24 and Boeing 737-700 combination were certificated, AC 20-128 was relevant guidance.⁴⁸ Similar to AC 33-5 for the engine, AC 20-128 included guidance regarding protection of the airplane and examples of acceptable methods of compliance for the airplane to protect against uncontained blade fragments, not in strict terms relative to penetration as stated in AC 33-5, but in terms of resulting potential hazards to the airplane. AC 20-128 included definitions for the uncontained fragment threats and the design considerations that should be addressed:

For the purposes of airplane evaluations in accordance with this AC, uncontained failure of a turbine engine is any failure which results in the escape of rotor fragments from the engine or APU that could result in a hazard. Rotor failures which are of a concern are those where released fragments have sufficient energy to create a hazard to the airplane.

AC 20-128 defined impact areas for various engine internal failures, rotor disk failures, and fan blade failures (**FIGURE 25**) based on historical engine failure events. The airplane designer should mitigate hazards within those impact areas by locating critical components or systems outside the likely debris impact areas if possible, duplicate and separate critical components within the likely impact areas or provide suitable protection or provide redundant designs. For the purpose of this discussion, only the fan blade failures and the defined fragment axial spread angle of $\pm 15^\circ$ fore and aft will be discussed. The AC also makes a point to account not only for the initial fan blade release fragment but also notes that the airplane designs should “consider the likely damage extent caused by multiple smaller fragments or shrapnel in the $\pm 15^\circ$ spread angle areas.” When defining the uncontained fragment threat to the airplane and the mitigation plan to address that threat, the AC suggests that the “engine installer should consider the engine manufacturer’s data on fragment energies and trajectories in relation to the engine installation for the particular location on the airplane”. Along with design considerations for the initial defining event, the initial fan blade release, AC 20-128 also includes guidance for continued safe flight and landing after the event, but this relates to the initial separation event and not directly to subsequent damage or imbalance loads from the engine imparted to the airframe.

⁴⁸ The most current version of AC 20-128 is revision A, AC 20-128A, issued March 25, 1997.

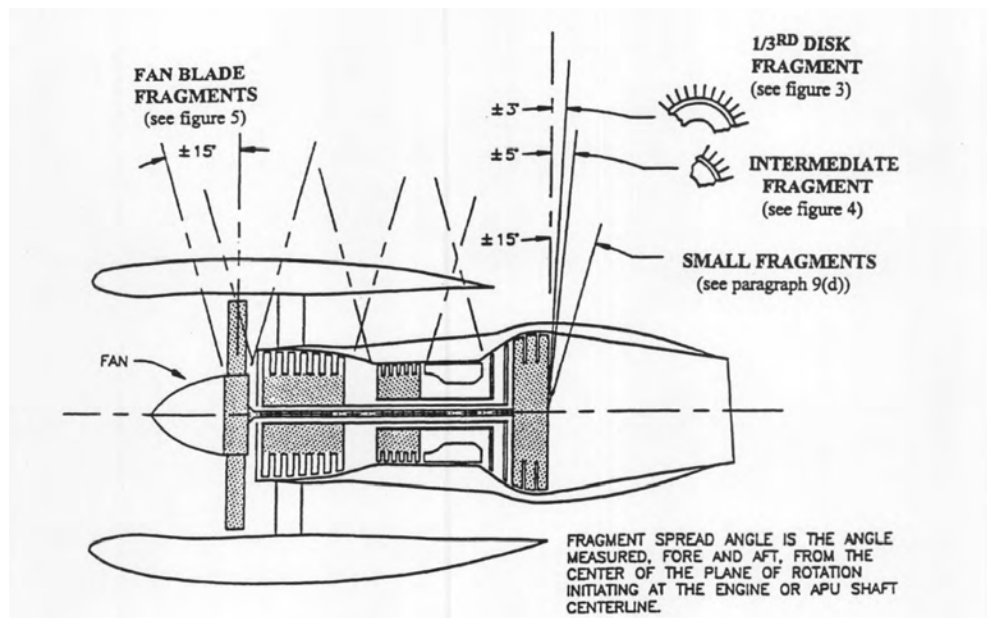


FIGURE 25: FRAGMENT SPREAD ANGLE EXCERPTED FROM FAA AC 20-128⁴⁹

Boeing demonstrated compliance with FAR Section 25.903(d)(1) requirements by: (1) adding a containment doubler in the inlet following the FBO rig 4 test that resulted in a fan blade tip fragment penetrating the inlet (See Section 9.3 ENGINE AND INLET CONTAINMENT TESTING AND DESIGN CONSIDERATIONS for details); (2) providing the appropriate airplane systems-separation such that a single fragment could not contact each redundant element of a critical system; and (3) demonstrating that the smaller fan blade fragments ejected forward and aft of the engine were not a threat to the airplane structure.

In cases when existing regulations are inadequate or inappropriate for the applicable design feature being certified or when proposed changes to the certification standards have not been finalized, the FAA may determine a SC is necessary. FAR Section 21.16 SPECIAL CONDITION states that: “if the airworthiness regulations of this subchapter do not contain adequate or appropriate safety standards for an aircraft, aircraft engine, or propeller because of a novel or unusual design feature of the aircraft, aircraft engine, or propeller” the FAA can prescribe a SC that establishes a level of safety equivalent to the existing regulations. Novel or unusual features that are unique to a specific design are compared and judged relative to the existing standards and are treated on a case-by-case basis. When necessary, an SC is developed through an issue paper process, with participation of the appropriate FAA certification office, the applicant, and other relevant participants. Before being finalized, an SC is published in the Federal Register as a Notice of Proposed Rulemaking (NPRM) for comment.

Under the Boeing model 737-600/-700/-800 certification program, the FAA issued SC 25-ANM-132 titled *Boeing Model 737-600/-700/-800; High Intensity Radiated Fields (HIRF)/Engine Stoppage* with an effective date of September 17, 1997, to address the structural capability of the aircraft to withstand engine torque loads imposed by sudden engine stoppage due to fan blade failure. This SC was issued because the FAA found that the Boeing model 737-600/-700/-800 airplanes would incorporate sufficiently novel or different design features, such that applicable FAR Part 25 airworthiness regulations as amended at that time might not provide adequate or appropriate safety standards. With regards to engines, the discussion section of SC 25-ANM-132 stated:

⁴⁹ The fragment spread angle was unchanged between AC 20-128 and AC 20-128A and is defined as the angle measured, fore and aft, from the center of the plane of rotation of an individual rotor stage, initiating at the engine or APU shaft centerline.

For the CFM56-7B engine, the limit engine torque load imposed by sudden engine stoppage due to malfunction or structural failure (such as compressor jamming) has been a specific requirement for transport category airplanes since 1957. The size, configuration, and failure modes of jet engines has changed considerably from those envisioned in 14 CFR 25.361(b) when the engine seizure requirement was first adopted. Relative to the engine configurations that existed when the rule was developed in 1957, the present generation of engines are sufficiently different and novel to justify issuance of a special condition to establish appropriate design standards. The FAA is developing a new regulation and new advisory circular that will provide more comprehensive criteria for treating engine loads resulting from structural failures. In the meantime, a special condition is needed to establish appropriate criteria for the Boeing 737-600/-700/-800 airplanes.

SC 25-ANM-132 stated that “In lieu of compliance with Sec. 25.361(b), compliance with the following must be shown (TABLE 5, Column 2). TABLE 5 shows the comparison between FAR Section 25.361(b) applicable when the Boeing 737-700 was certificated and the FAA approved SC 25-ANM-132 modifying that requirement.

TABLE 5: COMPARISON OF FAR SECTION 25.361(b) AND SC 25-ANM-132 APPLICATION	
FAR Section 25.361(b) issued June 26, 1990	SC 25-ANM-132 issued September 17, 1997
(b) For turbine engine installations, the mounts and local supporting structure must be designed to withstand each of the following: (1) A limit engine torque load imposed by sudden engine stoppage due to malfunction or structural failure (such as compressor jamming). (2) A limit engine torque load imposed by the maximum acceleration of the engine.	(b) For turbine engine installations, the mounts and local supporting structure must be designed to withstand each of the following: (1) The maximum torque load, considered as limit, imposed by: (i) sudden deceleration of the engine due to a malfunction that could result in a temporary loss of power or thrust capability, and that could cause a shutdown due to vibrations; and (ii) the maximum acceleration of the engine. (2) The maximum torque load, considered as ultimate, imposed by sudden engine stoppage due to a structural failure, including fan blade failure. (3) The load condition defined in paragraph (b)(2) of this section is also assumed to act on adjacent airframe structure, such as the wing and fuselage. This load condition is multiplied by a factor of 1.25 to obtain ultimate loads when the load is applied to the adjacent wing and fuselage supporting structure.

On January 22, 1997, the FAA closed Issue Paper A-2 titled *Limit Engine Torque Loads for Sudden Engine Stoppage*. This issue paper described how to treat a sudden engine stoppage condition resulting from a structure failure as an ultimate load condition. The issue paper acknowledges that:

The size, configuration, and failure modes of jet engines has changed considerably from those envisioned by 25.361(b) when the engine seizure requirement was first adopted. For current turbofan engines, fan blade technology has led to the use of wide chord fan blades resulting in fewer more massive fan blades. This combination of fewer more massive fan blades coupled with the lower fan speed results in greater "out of balance" loads in the event of engine failures that result in releasing fan blades... The latest generation of high bypass fan engines are capable of producing engine torque loads and

fan blade out of balance loads that are significantly higher than previous generations of engines.

The issue paper also acknowledges, at the time of issuance, that the FAA was in the process of developing new regulations and new AC material that would provide more comprehensive criteria for treating engine loads resulting from structural failure. In doing so, Boeing accepted SCs in lieu of FAR Section 25.361(b). **TABLE 5** provided the language for the adopted SC as written in SC 25-ANM-132 and was the same as the proposed conditions listed in Issue Paper A-2 except for one additional condition, which according to Issue Paper A-2, would “not be included in the text of the Special Condition” but that “Boeing has agreed to comply with...”. The additional condition provided in Issue Paper A-2 is stated as follows:

All significant structural responses developed during engine shutdown, resulting from the failure event, defined by the Special Condition below, up to V_c/M_c^{50} and considering appropriate pilot action, must be investigated. Any structural response due to the engine imbalance must not constitute a hazard to the airplane, including the operation of essential systems and flight controls, at all speeds associated with engine out operations. The structure, essential systems, and pilot's ability to perform flight duties must also be evaluated for the effects of continuous oscillatory or vibratory conditions that may exist from the time of failure to the end of flight.

The requirements of SC 25-ANM-132 (Issue Paper A-2) specifically pertain to the engine mounts, pylon and wing attachment structure to sustain certain loads associated with FBO and Windmilling conditions; however, Boeing expanded the application of SC 25-ANM-132 to the nacelle structure design as well. This expanded condition was in agreement with the Boeing SCD inlet and fan cowl assemblies, a section of which states: “Failure of an engine fan blade and subsequent damage shall not cause separation of the inlet or fan cowl from the engine or separation of components from the inlet cowl or fan cowl which would be a hazard to the aircraft.”

9.3 ENGINE AND INLET CONTAINMENT TESTING AND DESIGN CONSIDERATIONS

To meet the applicable FAR Part 33 requirements, and to obtain data that would be used to meet FAR Part 25 certification requirements, CFM performed a total of eight development FBO rig tests and an engine FBO containment certification test, followed by a second engine FBO certification test specifically to demonstrate the fan blade axial retention capability (**PHOTO 125** and **TABLE 6**). For brevity, the engine FBO containment certification test may be referred to as build 2 or test 002/2 and second engine FBO certification test may be referred to as build 4 or test 002/4. The purpose of these FBO tests were to: (1) understand the behavior of the fan blade after separation (kinematics), (2) test the fan case containment capability (radial containment - kinetic energy of each impact), (3) define the loads from the ensuing imbalance, (4) calculate the speed and weight of ejected fragments (forward containment), if any, and (5) demonstrate the proposed production hardware configuration. Each test had a specific set of objectives and based on those objectives various configurations of the fan blade, fan case, and inlet were used. The FBO rig tests were conducted as spin-pits where the fan rotor was driven by electric motors in a vacuum chamber. The first four FBO rig tests were used to define the fan blade break-up fragment properties (kinematics) for fan case radial containment capability, fan blade axial retention, and fan blade interaction. This included gathering data on the fan blade fragments relative size, weight, speed, and forward exit trajectory (the helical angle of travel from the center of rotation as the fragment departs the

⁵⁰ The FAA in FAR Section 1.2 define V_c to mean design cruising speed. M stand for Mach number and M_c is taken to mean design cruising speed in Mach number.

fan case A1 flange and crosses over into the inlet). In FBO rig 4 test, the penetration of a fan blade fragment through the inlet structure identified the need for a containment feature in the inlet.

FBO rig 4 test included a full set of production representative forged fan blades, the production representative fan case, and a combination of actual production and production representative accessories attached to the engine, such as the AGB & TGB, oil tank, and ECU. Also included in this test was a Boeing production representative inlet which included the crushable spacers on the fan case-to-inlet attachment bolts and represented the desired size, shape and stiffness of the intended production inlet. This first Boeing production representative inlet did not have a containment doubler, as it was not known at that time whether inlet containment was needed. For FBO rig 4 test, the fan was driven to the red line speed of 5,382 revolutions per minute (rpm) which matches the maximum permissible engine rotor speed of the N1⁵¹ as stated on the TCDS for all CFM56-7B models and agrees with the testing guidance provided in AC 33-5 and AC 20-128 and the fan blade was released at the 12:00 o'clock position. The fan blade release point is chosen by the engine manufacturer and varies from engine manufacturer to engine manufacturer based on their specific needs and test set-up; there is no specific guidance or regulation pertaining the fan blade release point, just the release speed and blade geometry (full airfoil). According to the UTAS's report titled *CFM56-7 Blade Out Spin Test and Blade Containment* and CFM's report titled *CFM56-7B Fan Rotor - Spin Pit Blade Out Test 18ER Results*, the FBO rig 4 test revealed that: (1) the separated fan blade fractured into 5 pieces with the tip panel spiraling out of the fan case at an estimated speed of 260 meters per second (m/s)/853 feet per second (fps) at an estimated 10°-15° helix angle⁵² and continued to spiral around the inlet before it penetrated through the inner and outer barrel of the inlet about 15-inches forward of the fan blade rotational plane (between the 6:30-9:00 o'clock position outer barrel skin of the inlet was missing), (2) fan case exhibited a small penetration hole and a long crack from the A1 flange back to almost the last stiffening rib, (3) the fan blade axial retention featured allowed some blades to slide forward, (4) the inlet case remained attached, (5) all the fan case-to-inlet attachment bolts were intact, and (6) the inlet attach ring exhibited a long circumferential tear at the forward radius. The inlet inner barrel penetration worked out to be approximately 26° forward of the fan blade rotational plane which was beyond the ±15° spread angle (impact area) as referenced in AC 20-128 where practical design considerations should be used to minimize the hazards of the uncontained fragments.

Based on the results of the FBO testing, CFM redesigned the fan case to strengthen it to prevent future through-case penetrations and changed the fan blade axial retention feature and Boeing instructed UTAS to design a containment shield to stop an FBO fragment from penetrating the outer barrel of the inlet and to modify the inlet attach flange. Since all the fan case-to-inlet attachment bolts were intact and the inlet remained attached, the testing of the crushable spacers was considered a success and no further modification in this area was needed. The fan kinematics as well as the impact loads (sometimes referred to as the forcing function to be used to computationally model the fan case and inlet interaction), were provided to Boeing, and Boeing used this information to generate interface loads that were provided to its inlet supplier UTAS to design an inlet that would comply with the applicable FAR Part 25 requirements.

Trade studies were conducted to determine the best option for containment; Kevlar belt or a shield made of Ti or Al. An Al containment shield was down-selected using cost, weight, and acoustic metrics. The inlet was designed by UTAS in accordance with the Boeing SCD which included the addition of a containment shield and a stronger and axially longer attach ring. According to the Boeing SCD and

⁵¹ According to the engine TCDS, 100% N1 = 5,175 rpm whereas the red line speed of 5,382 rpm represents 104% N1; 100% N2 = 14,460 rpm.

⁵² This 10°-15° helix trajectory angle is the angle where the fragment crosses the A1 flange and is not to be confused with the ±15° spread angle referenced in AC 20-128 (See [FIGURE 25](#)).

relevant UTAS internal correspondence discussing their analysis, the new inlet design provides containment capability for a forward spiraling fan blade tip panel fragment with no pass-through penetration of the outer barrel of the inlet; secondary “batted” fan blade fragments were not considered. According to Boeing, secondary fragments were not considered because none had been reported by CFM from FBO rig 4 test; review of CFM and UTAS FBO rig 4 test reports that only the tip panel traveled forward of the fan case. Secondary fragments are those additional fragments created during the FBO and subsequent fan blade break-up sequence such as mid-panel pieces and other fan tip and root pieces. The fan blade tip panel fragment properties were established by rig test results performed and provided by CFM which provided the energy level, fragment size, and trajectory of fragment that traveled forward of the fan case A1 flange. Revision B of the SCD specified the fragment properties for containment to be slightly more (11% heavier) conservative than the FBO rig 4 test, the total velocity was as provided from the FBO rig 4 test data; these are the same requirements stated in the UTAS Boeing 737/CFM56-7 Qualification Report for the Inlet and Fan Cowl, stating a means of compliance with the Boeing SCD. Initial sizing of the containment shield was performed by UTAS using a combination of FBO rig 4 test data as well as fragment kinematic formulas (fragment velocity vs axial distance forward of the A1-flange) provided by CFM, Boeing SCD requirements, and publicly available FAA ballistic test data and analysis methodology literature.

The FAA issued report No. FAA-RD-77-44 titled *STUDY TO IMPROVE AIRFRAME TURBINE ENGINE ROTOR BLADE CONTAINMENT* dated July 1977. The intent of the study was to evaluate the inlet material containment requirements needed for high bypass engines for wide body transport airplanes using semi-empirical correlations based on actual events and direct empirical results based on testing data for Ti blade fragments. Specifically, the study established a method for determining whether or not additional radial protection (containment) was necessary, and if so, what that containment feature would be (length, thickness, location, and material) and the penalties associated with adding that protection. The FAA report contained predicted fan blade fragments total velocity (along the fragment trajectory) curves for Pratt & Whitney JT9D and GE CF6-50 engine (fan blades for both engines are categorized as narrow chord). The curves showed fragment trajectory velocities as a function of distance from fan blade centerline. The distance the fan blade fragment travels forward along the spiral path depended on the initial energy of the fragment. This included the mass and velocity at the time of the fracture, geometry, and frictional coefficient of the surface that fragment contacts/surfs over. Based on the FAA predictive fragment velocity curves, fragments contained within the inner barrel of cylindrically shaped inlets would come to rest (axial velocity would be zero) between about 18-21-inches forward of the fan blade centerline (Gunderson, July 1977). The curves were intended to provide guidance on how to determine the appropriate thickness of the containment shield by determining the impingement velocity and kinetic energy of the fan blade fragment at any given point along its axial path. Since the inlet for the Boeing 737-700 is not cylindrical but locally conic and diffusive in the airflow direction (inner barrel initially becomes smaller just forward from the engine attachment flange), a further reduction of the fragment velocity occurs as it spirals forward around the inlet.

UTAS, accounting for the average conic angle of the inlet and considered the fan blade tip panel fragment velocity, plotted points on the Gunderson plots using kinematic formulas provided by CFM and it was found that the shape and slope of the fragment velocity curve for the CFM56-7B was similar to those for the JT9D and CF6-50 engines. Using the CFM provided velocity vs axial distance forward of the A1 flange formula, UTAS calculated the CFM56-7B fan blade tip panel fragment was estimated to come to zero forward velocity about 17-inches forward of the inlet attachment A1 flange. The CFM56-7B velocity curve was then used to estimate the velocity of the fan blade fragment and then calculate the fragment kinetic energy where it was predicted to penetrate the inlet inner barrel and impact the containment shield. This information was then used to calculate the minimum thickness of the

containment shield that can absorb the kinetic energy of the fragment using the FAA shear energy equation (**EQUATION 1** (Gunderson, July 1977)) using the dynamic shear modulus for aluminum.

$$t = \sqrt{\frac{E_A \cos^2 \theta}{s E_M}}$$

t = thickness (inch)
 E_A = Energy absorbed by containment shield (in-lbs)
 θ = Fragment impingement angle (°) – calculated value based on helix angle and geometry (radius) of the inlet
 s = Perimeter of impact face of fragment (inch)
 E_M = Dynamic shear modulus (psi) – values provided by FAA-RD-77-44
 = 30,450 psi for aluminum
 = 145,500 psi for titanium
 = 188,500 psi for steel

EQUATION 1: FAA SHEAR ENERGY EQUATION

The containment shield was designed so that the fan blade tip panel fragment would come to rest within the containment shield length. An 11-inch aluminum doubler (containment shield) was added to the inner barrel of the inlet to address a fragment that traveled beyond the $\pm 15^\circ$ spread angle. The inlet aluminum attach ring length was increased by approximately 2-inches to optimize the design for the addition of the containment shield and address discrete damage observed in FBO rig 4 test. Adding the fan case and inlet containment features resulted in about a 37° forward spread angle; as previously mentioned in AC 20-128 which defines the fan blade fragment spread angle as $\pm 15^\circ$ (**FIGURE 26**).



FIGURE 26: CONTAINMENT SPREAD ANGLE CFM56-7B FBO RIG AND ENGINE FBO CONTAINMENT CERTIFICATION TEST INLET CONFIGURATION

FIGURE COURTESY OF CFM

UTAS not only used the FAA study and the CFM56-7B engine FBO test results, but also used their design experience on a previously certificated similar inlet which had successfully arrested a blade fragment during FBO testing, to finalize their design: The FAA shear energy equation was used to calculate an alternate CFM56-7B shield thickness by scaling the similar inlet shield thickness to account for differences in fan diameter, fan blade fragment size and speed, containment shield material, acoustic honeycomb, etc. Because the FAA methodology utilized conservative assumptions, a mid-point between the FAA methodology thickness and the scaled similar engine inlet thickness was selected as the final CFM56-7B containment shield thickness. Furthermore, because the fragment velocity slows as the fragment moves axially forward, and the kinetic energy decreases as a result, the FAA report showed that the containment shield thickness can be tapered to account for the slowing fragment; the final design of the inlet containment shield thickness was tapered in this fashion (See Section 1.5.1 INLET DESCRIPTION – INLET PN 314-2100-2 for details).

Prior to the CFM56-7B engine FBO containment certification test, there were multiple coordination meetings between CFM and Boeing to review trade studies, FBO rig test results, review and agree upon testing and FBO event assumptions and model description, review the various structural and transient dynamic load analyses, review design parameters used to determine the proposed production representative inlet, and to establish the test parameters for the FBO certification test. The conclusions were that production equivalent hardware would be used such that results can be used towards certification. The engine FBO containment certification test is a FAR Part 33 test and its primary purpose is to demonstrate that the engine was capable of containing the released fan blade without catching fire, without failure of its mounting attachments, and that the engine could be shutdown and not to validate the inlet design. However, the FAR Part 33 test data generated was used by Boeing, and as accepted by the FAA, to correlate the FBO loads model, and use as input into the loads development for certification of the Boeing 737-700 airplane FBO loads.

The CFM56-7B engine FBO containment certification test (build 2 or 002/2 for brevity) was conducted in April 1996 on a test stand at a fan speed slightly higher than the redline speed used for FBO rig 4 test. Based on the FBO rig 4 test results, the 002/2 test configuration included a newly modified fan case, fan blade, and redesigned representative inlet configuration with a containment shield (doubler) and a representative TR. No fan cowls were installed on the certification test, per standard industry practice, to permit observation of the fan case and fan case installed components during the test. High-speed film was used along with engine instrumentation to record the FBO event. According to CFM, the test results showed that after the fan blade release, the fan case sustained the initial impact with no penetrations (the fan case did exhibit 2 non-open tears near a stiffening rib and near the oil tank position), the engine did not catch fire⁵³, did not fail its mounting hardware, and was able to be shutdown. The test result also confirmed that no fan fragments penetrated the inlet containment doubler as a result of the initial blade release. Although the fan stopped on its own about 5 seconds after fan blade release, to satisfy the FAR Part 33.75 requirement to be able to command a shutdown of the engine, the high pressure fuel shutoff valve (HPSOV) in the hydromechanical unit (HMU) was shuttled CLOSED indicating that the fuel supply could be cutoff to the engine thus effectively shutting down the engine if it had not shutdown on its own. After the initial fan blade release, an additional five consecutive fan blades, approximately 90° from the initial released fan blade, failed their axial retention feature. These five additional blades slide out from the fan disk with some exiting through the fan case while others exited radially forward doing considerable damage to the inlet. The inlet damage consisted of pass-through penetrations and missing inner and outer barrel material. According to CFM, although the fan blade tip panel fragment was not recovered, a review of the CFM56-7B engine FBO containment certification test high-speed film and forensic analysis of the recovered fan blade fragments and fan case and inlet case damage, showed that the fragmentation and trajectories were consistent with prior FBO rig test experience; a fan blade tip panel exited forward, mid-section panels traveled forward and aft, and the root traveled aft into the fan OGVs.

As mentioned before, the primary objective of the engine FBO containment certification test was centered around the fan blade kinematics and satisfying the applicable FAR Part 33 requirements. However, to help satisfy the various applicable FAR Part 25 requirements as they relate to engine debris containment, additional information beyond forensic inlet damage, such as the loads the engine, inlet, and supporting structure experienced from the initial FBO release to shutdown, was also needed for Boeing and UTAS to model and predict the FBO transient/ultimate loads, fatigue load, and damage tolerance for

⁵³ There was a small tailpipe fire after shutdown and it was extinguished by the fire crew; this was not considered significant for certification purposes.

the FBO and ‘Fly-Home’ analysis (UTAS was provided interface loads). CFM and Boeing jointly correlated the engine FBO containment certification test results, the transient forcing function that defined the various FBO related events and loads, such as initial fan impact loads, surge loads, rundown loads, torque loads⁵⁴, imbalance loads, and windmilling loads for Boeing and UTAS to use in their analysis of the inlet and its capability. Since the FAR Section 33.94 containment testing requirements involve showing that the engine can be successfully shutdown after the FBO, windmilling loads are not obtained as part of the test; instead, windmilling loads are predicted based on analytical loads models and the correlated model and input from CFM experience.

According to CFM, the predicted 3D analysis of the CFM56-7B FBO loads and torque load were consistent with the initial engine FBO certification test, as well as their experience with other CFM56 models. The FBO loads and torques were back-calculated by CFM based on the FBO test results. The FBO impact loads were derived in both the radial and tangential directions and were applied over the area where the hard impacts were observed. The torque loads were a combination of imbalance loads and deceleration rate and were applied uniformly around the circumference of the fan case; according to CFM the highest engine deceleration rate and hence the maximum torque load occurs within the first several revolutions. The high deceleration rate and maximum torque loads are mostly a mechanical phenomenon and less an aerodynamic phenomenon due to interacting rubbing/breaking loads of the fan blades with the fan case immediately after the initial fan blade release and loss of power due to engine stall. The high deceleration rate and maximum torque loads were as-predicted and similar to the engine FBO certification test data; the loads analysis is primarily focused on the first initial revolutions after release because this is the time frame of the highest deceleration and hence the maximum torque. The calculated FBO impact load, along with the deceleration rates, imbalance loads, windmilling loads etc. were introduced in a shared CFM and Boeing NASTRAN⁵⁵ finite element analysis program and scaled using data from other engine model FBO tests to create a transient response forcing function that simulates the physical response of the structure due to the FBO event. CFM transmitted the fan blade kinematics along with the transient forcing function to Boeing so that they could analyze the airplane’s structural response to the FBO event; how the inlet, fan cowl, pylon, mounting structure, etc. respond. Included in this analysis were not only the FBO loads from CFM but also airplane maneuvering loads (limited to a 1g maneuver) and gust loads developed by Boeing.

Based on the engine FBO containment certification test 002/2 test results and the analysis of the predicted loads provided in the CFM56-7B18, -7B20, -7B22, -7B24, 7B26, -7B27 Fan Blade Containment Certification Report, CFM concluded that: (1) no hardware failure during the initial FBO release affected engine containment certification and the subsequent damage to the fan case, accessory hardware, and inlet, and higher than typical imbalance loads were due to failure of fan blade axial retention feature and subsequent release of five consecutive blades, and (2) based on an analysis of predicted loads and flight profiles, the CFM56-7B type design was capable of sustaining ‘Fly-Home’ loads without creating a hazard to the aircraft. In terms of the inlet, the containment shield was deemed adequate for

⁵⁴ Torque loads are a function of deceleration and the fan blade-to-fan case interaction load. As clearance increases with the removal of the fan case abradable and /or loss of fan blade, torque load decreases.

⁵⁵ NASTRAN is an open source finite element analysis program originally developed for NASA in the late 1960s and is an abbreviation of NASA STRuctural ANalysis program. NASTRAN is an industry established computation tool to evaluate and model the FBO event; the model contains engine, inlet, pylon, wing structure etc. and predicts the structural loads for all phases of the FBO event. The current state-of-the-art includes modeling the impact phase of the FBO event using LS-DYNA® (developed by Livermore Software Technology Corporation) software which uses explicit and implicit time integration to analyze nonlinear transient response of structures. In addition to using LS-DYNA®, modern FBO analytical techniques divide up the FBO event into more specific phases to refine the calculation: initial impact (now called a displacement wave), blade fragmentation, surge, rundown (deceleration), and windmilling. LS-DYNA® was not used for the CFM56-7B certification in 1996 and the information allowing for the further segmenting of FBO phases was not available either.

the initial fan blade release based upon CFM input that the damage to the inlet and containment shield was due to the subsequent release of five fan blades. Boeing and UTAS concluded that despite the amount of inlet damage, the inlet remained attached to the engine, so it withstood the initial FBO impact, the surge pressure loads, and the rundown loads. Boeing and UTAS agreed a redesign was not necessary and additional testing was not required even if there was a possibility to reduce weight. Furthermore, according to a UTAS Technical Exchange Memo: Rohr Blade Out - Meeting Minutes, UTAS analyzed the inlet damage provided by Boeing from the initial impact, and through analysis concluded that damage would not change the dynamic behavior of the stems and predicted windmilling loads and the inlet had sufficient loads margin for continued safe flight.

To validate and demonstrate the proposed fan blade axial retention design and a minor increase in fan case thickness, CFM conducted additional FBO rig tests and the second engine FBO certification test. For this engine FBO certification test, an aerodynamic bellmouth with similar weight, stiffness, and flow path shape as a production inlet was installed using airplane mounting hardware (production bolts, crushable spacers). CFM conducted an analysis to determine the impact of using the bellmouth on the FBO loads and concluded, based on fan case and inlet properties, fan blade tip rub forces, and impact loads, that the differences between the production inlet and the bellmouth would not impact the test objectives. The results of the second engine FBO certification test revealed similar fan blade fragmentation as observed on previous tests and validated the fan blade axial retention configuration. Based on the results of the second engine FBO certification test, the fan case was subsequently refined, minimum thickness was increased and was substantiated through an analysis (no testing of the redesigned fan case occurred). The fan case installed on the SWA event engine was a new reference (initial 340-059-818-0 then reworked in -824) thicker than this initial certification (815) design

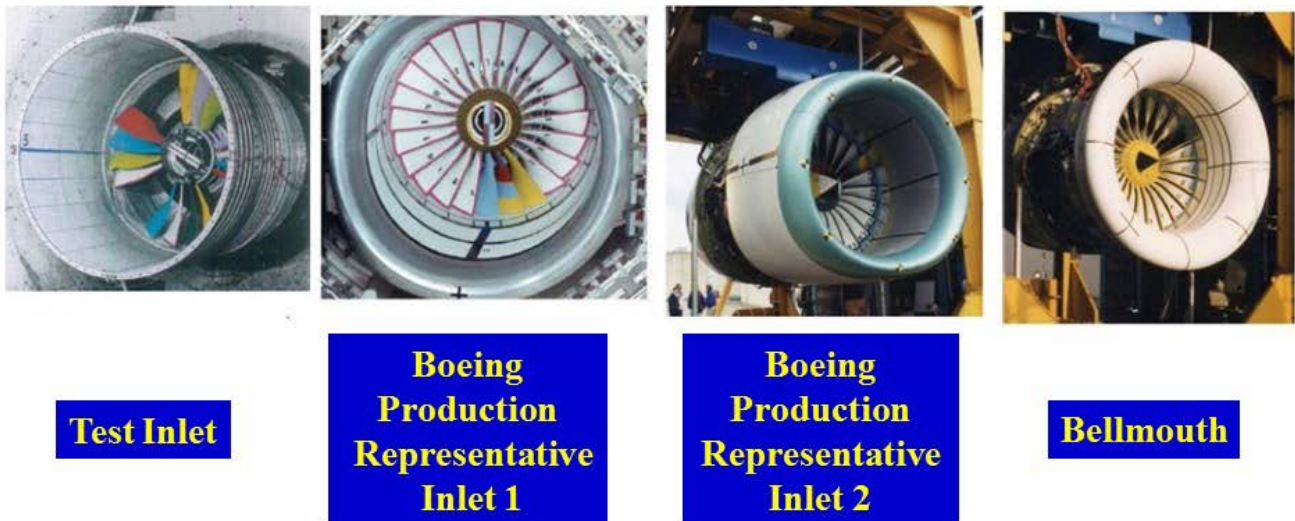


PHOTO 125: CFM56-7B FBO RIG AND ENGINE CONTAINMENT CERTIFICATION TEST INLET CONFIGURATION
PHOTO COURTESY OF CFM

9.4 CONTINUED SAFE OPERATION POST FAN BLADE SEPARATION REQUIREMENTS – ‘FLY HOME’ SAFE ANALYSIS

The Boeing inlet design philosophy was, and remains currently, that the failure of an engine fan blade and subsequent damage should not cause separation of the inlet or fan cowl from the engine or separation of components from the inlet cowl or fan cowl that would be a hazard to the aircraft. These requirements were developed in part based on CFM FBO testing and in part by analysis (in particular windmilling forces).

Boeing developed the final acceptable inlet damage tolerance condition to be used for the assessment of the ‘Fly Home’ requirements and provided them to UTAS based on the engine FBO containment certification test 002/2 damage results following a 12:00 o’clock⁵⁶ fan blade release: (1) outer barrel remain intact, (2) containment shield and inner barrel splice plates remain intact, (3) attach ring and all bolts remain intact, (4) lip skin not buckled and forward bulkhead is undamaged, (5) aft bulkhead-to-attach ring connection severed 360° around the circumference but the remainder of the aft bulkhead and outer T-frame remain intact, (6) the aft 11-inches of the inner barrel face sheet and acoustic core, not including the backskin damaged up to around a 100° continuous arc⁵⁷ and (7) the inner barrel at the forward closeout with the forward bulkhead damaged at two 90° arcs located 180° apart (inner barrel forward land-to-forward bulkhead joint location).

After all the design, testing, and analysis was completed, the critical features for the inlet and fan cowl to withstand the FBO and windmilling loads were determined to be the inlet attach ring end pad lug and the fan cowl’s forward hinge clevis, forward hinge, and the forward latch assembly. UTAS issued a report documenting the means of compliance with the Boeing SCD for the Boeing 737-700/CFM56-7B inlet and fan cowl in June 1998 that stated: (1) failure of a fan blade and subsequent damage will not cause a hazard to the aircraft, (2) inlet and fan cowl assemblies are capable of successfully completing a flight in which structural damage occurs as a result of bird strike, fire, FBO, or uncontained engine failure, (3) the inlet and fan cowl assemblies are capable of sustaining the FBO loads in combination with the aircraft lift off loads, and (4) the structural integrity of the inlet and fan cowl assemblies meets the FAR Part 25 [FAR Section 25.903(d)(1)]⁵⁸ throughout the flight and ground spectra. Furthermore, UTAS concluded that there was positive margin (structural capacity of the inlet or inlet component beyond the Boeing expected and defined loads) for the expected post-FBO inlet damage and windmilling that the inlet would be expected to remain attached to successfully complete the ‘Fly Home’ requirement, which includes a successful safe landing. Simply put, based on this analysis, Boeing and UTAS determined that the failure of a fan blade and subsequent damage, based on the results of the FAR Part 33 engine FBO containment certification test, would not cause a hazard to the aircraft.

⁵⁶ According to UTAS, even though the inlet is asymmetric about its longitudinal axis (the inlet attach ring and engine fan case are symmetric), for “a fan blade release at any other clock location the damage would be rotated by a corresponding amount”, essentially indicating that the asymmetric configuration did not affect the loads analysis.

⁵⁷ No documentation could be found to confirm why a 100° continuous arc was selected as one of the inlet design criteria, after the engine FBO containment certification 002/2 test. Boeing and UTAS met to define the ‘Fly-Home’ windmilling damage and agreed that the continuous last 11-inches of the inner barrel perforate skin and acoustic core up to the attach ring along a 100° arc at approximately the 2:00 o’clock position would define the damage to be used in the certification analysis of the inlet.

⁵⁸ FAR Section 25.903(d)(1) states: “Design precautions must be taken to minimize the hazards to the airplane in the event of an engine rotor failure or of a fire originating within the engine which burns through the engine case.”

9.5 DAMAGE AND EVENT CONDITION COMPARISON BETWEEN FBO TESTING AND SWA FLIGHT 3472 INCIDENT

Comparing the SWA flight 3472 incident inlet damage with the data from the engine FBO certification tests and the 'Fly Home' design requirements showed similarities that the inlet design accounted for, such as the containment shield was not breached by the FBO, the splice plates remained intact and attached, and all the rivets were sheared at the aft bulkhead. However, there were several significant physical differences that were not present in the engine FBO certification tests.

The first difference was the FBO release point, initial fan case impact location, and amount of fan blade of fragments that exited forward of the fan case. For FBO testing, the fan blade release point was targeted for the 12:00 o'clock position (actual release was closer to 12:30 o'clock) with the fan case initial impact location between the 1:30-2:00 o'clock position, and the inlet designed for around a 2.0 pound forward exiting fragment. For the SWA flight 3742 flight incident, the fan blade release point was estimated to be between the 12:30-1:00 o'clock position, the fan case initial impact location was about the 2:00 o'clock position, and the entire fan blade airfoil and root was missing and not recovered and, based on the lack of fan OGV damage, was consistent those the all the fan blade fragments exiting forward of the fan case.

The second difference was the amount of subsequent damage to the inlet. During the engine FBO containment certification test 002/2, a production representative inlet was installed and (1) no separation of the inner and outer barrels occurred as a result of the initial FBO and (2) the entire inlet remained attached. However, the inlet was severely damaged consisting of through-penetrations and missing inner and outer barrel material; this damage was attributed to the released of additional fan blades due to inadequate fan blade axial retention. Subsequent FBO rig tests and a second engine FBO certification test 002/4 addressed and validated the redesigned fan blade axial configuration.

The third difference was related to operation variations such as:

(1) altitude (FBO rig and engine FBO certification tests were conducted either in a vacuum (FBO rig tests) or at sea level (engine FBO certification); the SWA flight 3472 incident occurred at about FL 310),

(2) airspeed/air loads (FBO rig and engine FBO certification tests were performed on static stands at sea level, not in wind tunnels; therefore, no airspeed or air loads effects; the SWA flight 3742 incident occurred at about 289 knots),

(3) engine shut-down (engine FBO certification tests require ability to shutdown the engine 15 seconds after FBO (engines typically self-shutdown within 1-2 seconds); the SWA flight 3772 incident fan continued to windmill for about 18 minutes and 14 second until landing (since the FDR stopped and did not record the landing, this time is based on the pilots statement that the landing occurred at about 9:40 local time), and

(4) airplane induced loads from maneuvering (FBO rig and engine FBO certification tests were performed on static stands so no g-loads were applied; the SWA flight 3472 incident airplane continued to fly, descended, and maneuvered for landing), and

(5) installation of fan cowls (FBO rig and engine FBO certification tests were conducted with no fan cowls installed; the SWA flight 3742 incident airplane had fan cowls installed)

As of this report, whether the difference between FBO test condition and the flight conditions caused, contributed, or had no or negligible influence on the inlet separation is still be investigated.

TABLE 6: COMPARISON OF FAN-BLADE-OUT TEST RESULTS AND EVENT DATA

	Engine Certification Test 002/2	Engine Certification Test 002/4	ESN 874112
Event	FAR Part 33 engine FBO containment certification test	FAR Part 33 engine FBO certification test – validate the new fan blade retention features and fan case thickness adjustment	Fan blade separation event
FBO Conditions	On test stand N1 slightly above redline @ 5,411 rpm Sea level static	On test stand N1 slightly above redline @ 5,416 rpm Sea level static	31,259 feet 99% N1 5,123 rpm 289 knots
Fan Blade Release Position	12:00 o'clock	12:00 o'clock	12:30-1:00 o'clock (estimated based on rig and engine cert tests clocking)
First Fan Case Impact	1:30-2:00 o'clock	1:30-2:00 o'clock	2:00 o'clock
Fan Blade Fragmentation	Initial FBO -tip forward not recovered, small mid-panels forward & aft, root aft	Tip forward, small mid-panels aft, and root aft	Unknown number of fragments–no pieces of the failed airfoil or root recovered; fan case impact marks consistent with FBO testing
Fan Case Damage	Initial FBO contained. Five released fan blades penetrated and torn fan case	FBO contained, no additional fan blade release, axial retention feature validated. Small fragment penetrated fan case	Fan case exhibited three tears and no pass-through penetrations. No additional fan blades were released.
Inlet	Redesigned production representative with containment shield, longer attach ring, Boeing Inlet 2	Bellmouth - Similar weight and stiffness of production inlet	314-2100-2 ⁵⁹ (Revision AF)
Fan Cowl	Not installed	Not Installed	314-2200-3 (L) (Revision AV) 314-2200-11 (R) (Revision AW)
Inlet Damage Results	Remained attached, attachment bolts intact, damage to inlet due to secondary fan blade separations	NA	Inlet lip, outer barrel, forward bulkhead and inner barrel forward of the containment shield missing & not recovered. Aft bulkhead rivets sheared, T12 probe flattened. Containment shield and splice plates intact
Fan Cowl Damage Results	NA	NA	Fan cowls remained attached and forward most latch was secure; aft two found unlatched and hanging

⁵⁹ The current version of the inlet was issued in 2013. Certain enhancements unrelated to containment were introduced. This new inlet version .changed the aft bulkhead configuration and attachment fasteners.

10.0 POST EVENT FAN CASE AND INLET DOCUMENTATION & ASSESSMENT

After the on-scene documentation of the engine and airplane was completed, what remained of the inlet, PN 314-2100-2 (Revision AF), SN 1084001, and the fan cowls PN 314-2200-3 (Revision AV), SN 2177001 (left) and PN 314-2200-11 (Revision AW), SN 2178001 (right) were removed from the engine and shipped to the Boeing Equipment Quality Analysis (EQA) laboratory in Renton, Washington for initial visual evaluation. After Boeing completed their initial review and documentation, the inlet and fan cowls were shipped to the UTAS facility in Chula Vista, California, the inlet and fan cowl manufacturer, for detailed examination. The No. 1 engine, SN 874112, was removed from the airplane and shipped to the GE facility in Evendale, Ohio to remove parts from the fan module for further evaluation. The fan disk and fan case were removed from the engine and shipped to the CFM Villaroche facility near Réau south of Paris France for detailed examination.

The Airworthiness Group convened at the CFM Villaroche facility in Réau from November 28 – December 1, 2016 to attend a meeting on the status of the investigation as well as to review the fan case damage. The documentation of the fan case damage included: (1) the various impact locations, (2) potential fan blade fragmentation, (3) exit trajectories (forward exit angle), and (4) orientation as the various potential fan blade fragments passed across the fan case forward flange A1. This information was documented and provided to UTAS to assist them in performing their inlet ballistic damage and trajectory assessment. The Airworthiness Group also convened at the UTAS facility in Chula Vista, California from November 7-11, 2016, from February 27-March 3, 2017, and from June 18-21, 2018 to perform a similar examination and assessment of the inlet as was performed for the fan case in Villaroche.

Since no airfoil fragments of the failed blade No. 23 were recovered, the documentation on the fan case and inlet impact damage, and fan blade fragment exit trajectories conducted by CFM and UTAS was based entirely on the documented observed damage on the event fan case and inlet and correlating that with the fan case and inlet damage observed on the various FBO rig and engine FBO containment certification tests as a function of the fan blade fragment size, geometry, and dispersion patterns from those tests.

10.1 CFM FAN CASE DAMAGE AND FRAGMENT TRAJECTORY ASSESSMENT

The SWA event fan case, PN 340-059-815-0 (~~813~~ was marked out), SN DA072657-5, was shipped to CFM and a detailed examination was performed to document fan blade fragment impact marks and forward exit trajectory. Note that all references to clock references are estimates to closest clock position within about 15-30 minutes and are not intended to be absolute values (See [ATTACHMENTS 3](#) and [4](#) for complete details of the CFM fan case impact damage assessment). The diameter of the front flange A1 was measured at several locations to assess the roundness of the case; based on the data, the fan case front flange was ovalized and the greatest amount of deformation was along the 4:00-10:00 o'clock arc. Rub marks were noted along the entire circumference in-plane with the fan blade running position where the blue abrasable shroud had once been, except for the area between the 3:00-5:30 o'clock positions. Based on the various impact marks on the fan case, CFM concluded that fan blade No. 23 had released and fractured into at least three, but possibly four pieces, creating four major and distinctive fan case impact marks and several different exit paths and trajectories ([ATTACHMENT 3](#), pages 3-4). To differentiate the impacts marks identified on-scene to those identified by CFM during their visual fan case exam, the impact marks identified on-scene will remain the same (numerical) and the CFM marks will be identified alphabetically. Since none of the event fan blade airfoil fragments were recovered, the CFM impact and blade trajectory assessment was a combination and comparison of historical FBO rig and

engine FBO containment certification test data along with the observed event fan case damage ([PHOTO 126](#)).



PHOTO 126: CFM DOCUMENTATION OF FAN CASE DAMAGE

PHOTO COURTESY OF CFM

The first fan case impact, impact **A**, was a shallow blade tip scuff mark located at about the 2:15 o'clock position at a blade angle of about 35° (this angle is measured in reference to the plane of rotation and is close to the expected blade tip angle at climb according to CFM); this impact mark was not documented during the on-scene examination because it was not identified as being an impact at that time ([ATTACHMENT 3](#), page 7). According to CFM, this initial fan blade impact bends and snaps the blade tip (CFM sometimes refers to the blade tip as the top panel) off and it travels forward out of the engine. This is consistent with FBO rig and engine FBO containment certification test results ([ATTACHMENT 3](#), pages 4). CFM considered this initial impact as a 'low energy' event due to the lack of radial clearance between the blade tip and fan case resulting in low radial release velocity; hence the lack of a deep gouge or any corresponding distortion of the fan case in this area. The terms 'low energy' or 'high energy' are used not as an absolute value or as a quantitative basis, but instead qualitatively to describe the fan blade fragmentation. Comparing trailing blade (No. 22) pressure and suction side damage from the SWA event with engine FBO containment certification test results revealed similar markings: (1) tip curled in the direction opposite rotation, (2) spanwise mark on the pressure side consistent with contact with the released blade, and (3) spanwise mark on the suction side consistent with contact with trailing blade.

After the tip panel fractures, the remaining fan blade portion continues to travel towards the fan case creating a second and third 'high energy' deep crescent-shaped gouges/impact mark, impacts **B** and **C** respectively ([ATTACHMENT 3](#), page 4), located between the 3:00-4:00 o'clock positions. These marks were identified on-scene as impact marks **1** and **2** respectively (See [PHOTO 19](#)). The impact marks coincided with the fan case distortion and tears observed in [PHOTO 23](#) ([ATTACHMENT 3](#), pages 14-15) and was in the vicinity of the greatest measured front flange distortion and ovalization. CFM determined that the blade airfoil fractured at both the second and third impacts, thus each impact was created by a different fan blade fracture surface. CFM also determined that the deep scuffing damage, impact **C'**, located just forward in the direction of blade rotation relative to the on-scene mark **2** could possibly represent an additional impact and airfoil fracture ([ATTACHMENT 3](#), page 14).

The last definitive impact, 'high energy' impact **D**, was located at the 5:00 o'clock position in what appears to be the outline/imprint of the blade root TE profile ([ATTACHMENT 3](#), page 23) represented by on-scene impact marks **3** and **4** (See [PHOTO 20](#) red-outline of blade root). This impact mark coincided with the case distortion and tearing observed in [PHOTO 24](#) and [ATTACHMENT 3](#), pages

25). CFM considered all the fan blade fragment impacts as ‘high energy’ except for the initial ‘low energy’ tip impact.

When CFM compared the impact marks on the event fan case to those observed during the engine FBO containment certification tests, very similar markings (size, shape, and location) were noted such as the light initial contact of the tip and the subsequent two deep crescent-shaped impact gouges very close together followed by a root impact profile ([ATTACHMENT 3](#), page 4). Although no fragments from the separated No. 23 fan blade airfoil were recovered, CFM determined that, based on the damage and impact marks to the event fan case, this event was consistent in terms of blade fragmentation when compared with the engine FBO containment certification tests with two notable exceptions. First, the inlet, although heavily damaged during the engine FBO containment test, remained attached while in the SWA event a large portion of the inlet separated from the engine. Second, the only part of the fan blade that exited forward of the fan case during the FBO rig and engine FBO containment certification tests was a fan blade tip and pieces of mid-section; in the SWA event, the blade root also traveled forward and was not recovered.

In the SWA event, the fan blade root was determined to have exited out the front of the fan case and into the inlet. This assessment was based on two factors. One, some of the event fan case and inlet impact/damage marks and trajectories were consistent with the blade root exiting forward (see follow-on paragraphs for more clarification of this point) and two, the lack of missing or heavily damaged fan OGVs to indicate that the blade root had ever traveled aft (See [PHOTO 25](#)). During the engine FBO containment certification tests, fan OGVs loss was noted, while on the SWA event only some minor fan OGV LE impact damage was observed.

After documenting the initial fan case impact marks, CFM traced the trajectories of the various fan blade fragments by following spiraling scuff, scrape, and impact marks with similar size and shape as those documented during the FBO rig and engine FBO containment certification test around the circumference of the fan case and forward as they crossed over the A1 flange. The blade tip (impact **A**) initially contacted the fan case at about the 2:00 o'clock position and fractured creating the tip panel. Based on the size and orientation of the fan case scuff marks, the tip panel fracture face rotated about 90° in the direction of rotation from its spanwise axis ([ATTACHMENT 4](#), pages 4-6) and laid relatively flat on the fan case creating two distinctive scrape marks as it exited across the A1 flange between the 4:00-5:00 o'clock position ([PHOTO 127](#) and [ATTACHMENT 4](#), pages 5-7). The tip panel, after fracturing from the rest of the fan blade, did not appear to exit along a straight line but rather in an arcing motion where the trajectory of the tip was estimated to have changed from an initial trajectory of about 28° to 14° when the tip crosses the A1 flange; this change in trajectory according to CFM may have been caused by the distortion of the fan case during the subsequent **B** and **C** impacts ([ATTACHMENT 3](#), pages 8-10).

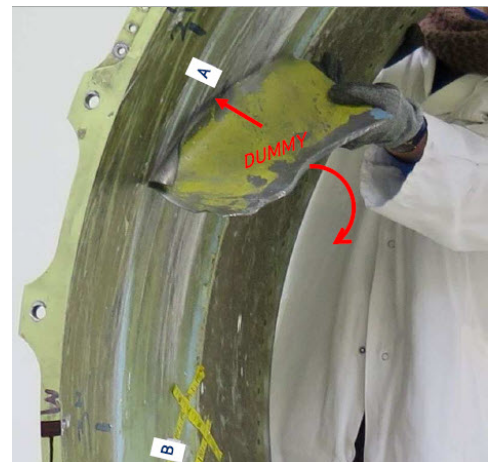


PHOTO 127: TIP PANEL ROTATION

PHOTO COURTESY OF CFM

Impact marks on the fan case seem to support that the mid-span fragments (impacts **B**, **C**, and possibly **C'**) impacted between 3:00-4:00 o'clock positions and followed a spiral pattern and crossed over the A1 flange at about the 8:00 o'clock position. The estimated forward exit trajectory of one of the mid-span fragments as it crossed over the A1 flange ranged from about 6°-8° ([ATTACHMENT 3](#), pages 16-20 and [ATTACHMENT 4](#), pages 12-13). Similar to the tip fragment, local fan case distortion

likely caused this mid-span fragment to change trajectory, hence the 6°-8° range. Additional marks were observed around the A1 flange at the 4:30 o'clock position that CFM attribute to other mid-span fragments. The trajectory of these fragments created deep marks about 37° as it crossed over the A1 flange. This mid-span trajectory was close in proximity to tip panel as it crossed over the A1 flange ([ATTACHMENT 3](#), pages 19-20).

The blade root, impact **D**, initially contacted the fan case at about the 5:00 o'clock position with a corresponding outline of the blade root TE profile ([ATTACHMENT 3](#), pages 22-26). The location and orientation of the blade root TE profile seems to suggest that as part of the impact sequence the root immediately laid flat with the pressure side facing up creating the imprint of the fan case impact mark **③** seen in [PHOTO 20](#). CFM determined that the trailing blade (No. 22) struck/kicked the root fragment standing it up and creating impact gouge impact mark **④** seen in [PHOTO 20](#) ([ATTACHMENT 4](#), page 24). Matching up other circumferential impact gouges and scuff marks around the circumference of the fan case with key geometrical features on the blade root, CFM concluded that the root fragment flipped over once, spun around, slid along its pressure side, and estimated that it finally exited across the A1 flange between from about 7°-9° ([ATTACHMENT 3](#), pages 28-29 and [ATTACHMENT 4](#), pages 15-23). CFM noted that there was no distortion of the fan case in the area of the root exit location. [PHOTO 128](#) provides a graphical presentation of the CFM blade fragment trajectories assessment for the various fragments as they cross the A1 flange.

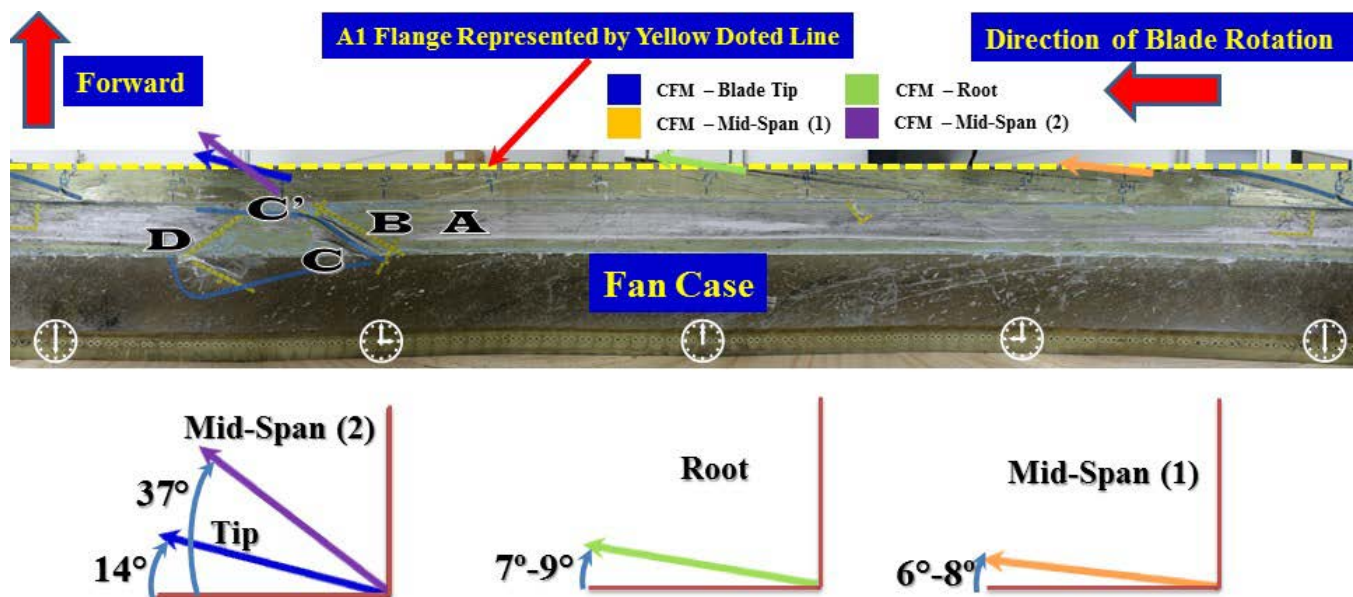


PHOTO 128: CFM DOCUMENTATION OF FAN CASE DAMAGE & FAN BLADE FRAGMENT TRAJECTORY
PHOTO COURTESY OF CFM

10.2 UTAS INLET DAMAGE AND FRAGMENT TRAJECTORY ASSESSMENT

UTAS documented the damage to the inlet and performed metallurgical assessment along multiple circumferential regions which included the following structures: (1) inner barrel assembly consisting of the acoustic core, perforated face sheet, backskin, containment shield (doubler), and (2) splice plates at the 3:00 and 9:00 o'clock positions. UTAS made the following preliminary observations for the inlet inner barrel: (1) Ti-6-4 transfer was observed in the forward edge of the containment shield and at locally bonded back skin and locally collapsed core locations all consistent with fan blade interaction, (2) the perforated face sheet within the inner barrel containment shield was damaged approximately nearly 360° circumferentially, (3) the backskin was completely separated forward of the

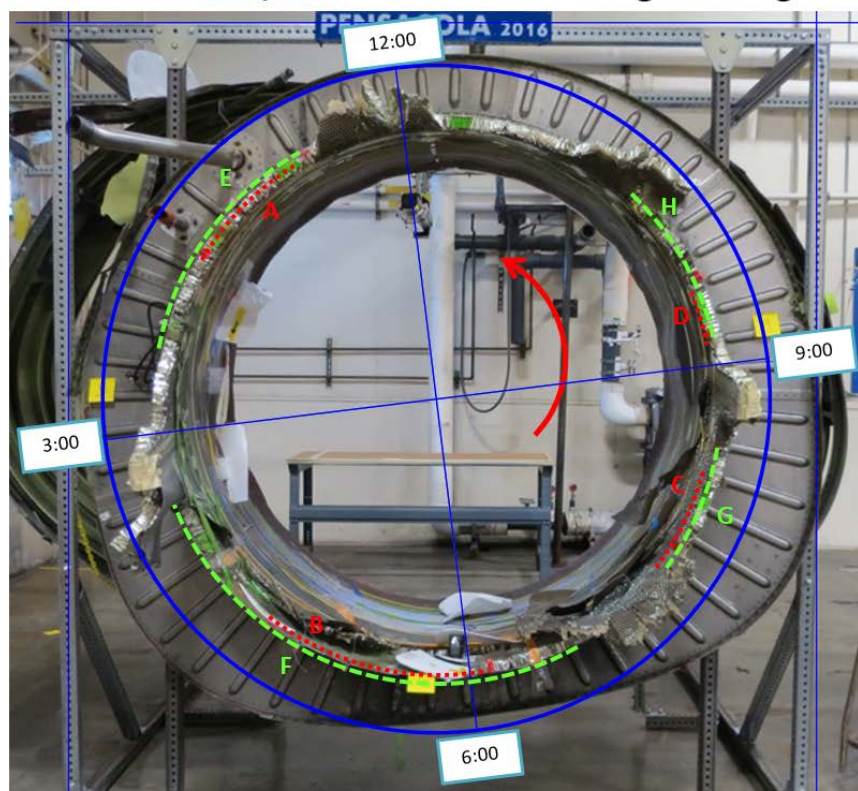
containment doubler (containment shield), (4) no evidence of fatigue was found on the backskin or core, (5) the fracture surfaces exhibited tension, shear, and/or tension shear overload, (6) the containment shield exhibited minor impact damage at approximately the 6:00 o'clock position; no other significant damage was observed, and the green primer paint was intact, and (7) no Ti particles were found on the backskin fracture surface ([ATTACHMENT 5](#)).

After further review of the inlet damage, it was determined that some of the damage could not be positively differentiated between damage caused by exiting fan blade fragments, engine run-down, or damage caused by the front part of the inlet (D-duct assembly) departing the airplane. Thus, instead of a single continuous estimated arc length of the inner barrel backskin damage, a more accurate method of describing the inner barrel damage was developed by the investigative team. This method defined a minimum and maximum inner barrel backskin damage ([PHOTO 129](#)).

The minimum inner barrel backskin damage was defined by evaluation of fan blade fragment trajectories, inlet inner barrel backskin metallurgical assessment. The maximum inner barrel backskin damage was defined as the minimum inner barrel backskin damage plus damage supported by ductile rupture Mode I/III progression⁶⁰ and possible blade debris damage of the inner barrel. The damage associated with the minimum/maximum inner barrel back skin damage was then defined in terms of an estimated inner barrel arc length. At the time this report was published, the investigation is still on-going and final determination of the inlet inner barrel damage was still in-work. However, as of now, the minimum amount of damage that that could be positively identified either visually or metallurgically as having been cut or damaged by exiting fan blade fragments, for the inner barrel backskin damage was an arc length of about 112.5° and for the perforated skin about 360° of perforate skin. The maximum amount of inner barrel damage, approximately 210°, consisted of four segments of varying lengths that included the minimum amount and those additional segments where the damage may have been caused by exiting fan blade fragments but could not be positively identified as such. The metallurgical evaluation for determining the fan blade fragment damage was based on inlet back skin fracture surfaces that exhibited shear and Ti-6-4 embedded particles in the inner barrel doubler. The damaged segments of the inner barrel not identified as part of the minimum/maximum definitions were determined to have occurred due the subsequent tensile and shear loading of the D-duct assembly during liberation of the assembly from the airplane. The original 'Fly Home' analysis was conducted based on FBO testing showing inner barrel damage of about 100° with the inner barrel backskin modeled as being intact (See Section 9.4 CONTINUED SAFE OPERATION POST FAN BLADE SEPARATION REQUIREMENTS – 'FLY HOME' SAFE ANALYSIS for details).

⁶⁰ Mode I refers to opening and tearing and Mode III refers to out-of-plane shear and sliding.

— Min/Max Inner Barrel Through Damage from Fan Blade Fragments



— — — Min Damage

Min Damage Segments

A - 12:45-1:45, $\Delta = 1:00$

B - 4:30-6:15, $\Delta = 1:45$

C - 7:30-8:15, $\Delta = 0:45$

D - 9:15-9:30, $\Delta = 0:15$

Min Total Δ Damage

$= 1:00 + 1:45 + 0:45 + 0:15$

$= 3:45$

$= 112.5^\circ$

— — — Maximum Damage

Max Damage Segments

E - 12:45 to 2:30, $\Delta = 1:45$

F - 3:30 to 6:45, $\Delta = 3:15$

G - 7:30 to 8:30, $\Delta = 1:00$

H - 9:15 to 10:15, $\Delta = 1:00$

Max Total Δ Damage

$= 0:15 + 1:45 + 3:15 + 1:00 + 1:00$

$= 7:15$

$= 210^\circ$

Clocking is based on nacelle coordinate system and is in alignment with blue tape clocking on ID of attach ring

PHOTO 129: INLET INNER BARREL DAMAGE ASSESSMENT

PHOTO COURTESY OF UTAS

UTAS used their inlet damage assessment along with the estimated SWA event fan blade fragment sizes, shapes, exit location, and trajectory provided by CFM to develop their trajectory assessment for the inlet, similar to what CFM performed on the fan case. UTAS traced the trajectories of the various presumed fan blade fragments by following spiraling scuff, scrape, and impact marks with similar size and shape fragment geometries provided by CFM and, based on their trajectory assessment, conducted the inlet failure/departure sequence ([ATTACHMENT 6](#)). [TABLE 7](#) shows the comparison of the CFM and UTAS trajectories for the various fan blade fragments: tip panel, mid-span fragments, and root.

UTAS estimated that the tip panel entered the inlet at about the 4:00-4:15 o'clock position based on a deep gouge in the inlet attach ring; this is consistent with the CFM estimated exit location. The tip panel continued into the inlet and penetrated the perforated skin at about the 5:30 o'clock position and finally came to rest at about the 6:30 o'clock position within the acoustic core. The tip panel was stopped from exiting the inlet by the containment shield. UTAS interpreted the various tip panel scrape marks from the fan case to the inlet case differently than CFM. The UTAS assessment included the tip panel fracture face rotating greater than 90° in the direction of rotation about its spanwise axis then rotating back as its center of gravity traveled at a 30° spiral angle along the direction of travel to account for the various scrape marks. This created a scratch that began at 30° but finished at 15° while the fragment fell flat onto the fan case. When the tip panel struck the inlet attach flange its center of gravity trajectory slide across the fan case A1 flange, continued at about a 30° trajectory ([ATTACHMENT 6](#), pages 11-14). The tip panel then penetrated the perforate skin and sideways collapsed the acoustic core, impacted the containment shield, and continued to sideways collapse acoustic core until arrested at 6:00 o'clock position. UTAS used the CFM provided angular relationship between the fan case scrape marks, along with inlet damage associated with the tip fragment's location of arrest within the acoustic core to support the 30° trajectory conclusion ([PHOTOS 130-131](#)). CFM assessment differs in that they concluded that the marks associated with a 30° trajectory path are more consistent with mid-span fragments than a tip panel.

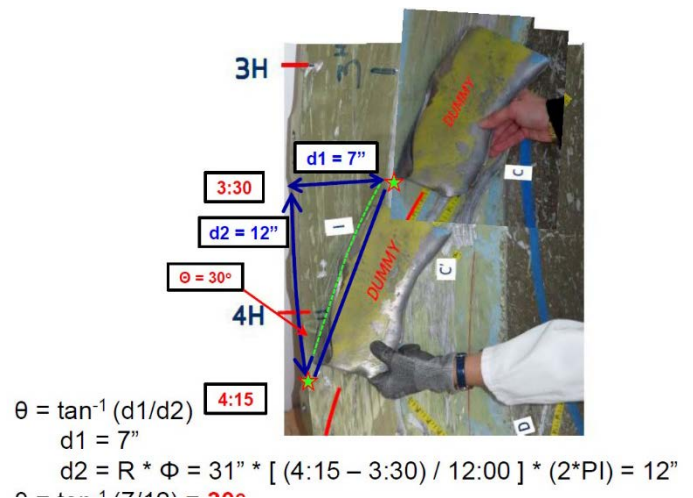


PHOTO 130: UTAS TIP PANEL EXIT TRAJECTORY

PHOTO COURTESY OF UTAS

The mid-span fragments were estimated to have entered the inlet at about the 8:00 o'clock position at about a 6°-8° trajectory angle, slid circumferentially along the aft flange and densified core radially crushing the acoustic core, struck and flattened the T12 temperature probe, penetrated the perforated skin at about the 3:30 o'clock position, and sheared the inner barrel skin at the forward edge of the containment shield from about the 3:30-6:45 o'clock position. After impacting the T12 temperature probe, some of the mid-span fragments were deflected forward and some rearward. One of the mid-span fragments was thought to have deflected aft after the T12 temperature probe impact and slid under the honeycomb core eventually impacting and lifting the fan tip panel that had previously been stopped in the core ([PHOTO 131](#) and [ATTACHMENT 6](#), pages 24-25).

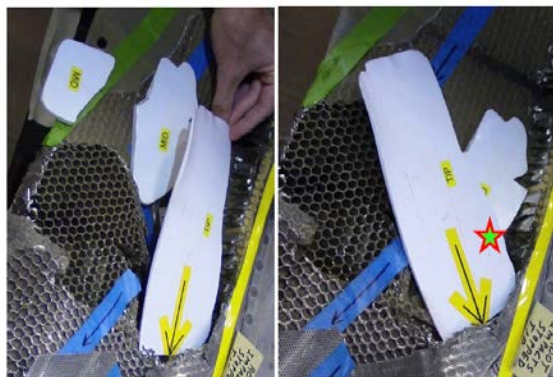


PHOTO 131: MID-SPAN AND TIP PANEL CO-LOCATED IN THE INLET CORE AT THE 6:00 O'CLOCK POSITION

PHOTO COURTESY OF UTAS

The blade root was estimated to have entered the inlet at about the 11:00 o'clock position at about a 5° trajectory angle and continued to slide around the inlet aft flange and densified core making more than a complete revolution. UTAS concluded that the blade root struck the tip panel that was thought to have been momentarily stopped in the acoustic core at the 6:30 o'clock. Based on new trajectory scrape marks and damage to the perforated skin and acoustic core near the blade root and tip collision, UTAS proposed that the collision created new fragments (**PHOTO 132**). UTAS concluded that new fragments along slid across the perforated skin then penetrated the perforated skin, acoustic liner, and sheared the inner barrel from the 7:30-8:30 o'clock position and from the 9:15-10:15 o'clock position. What remained of the original root continued to slide and travel around the inlet almost $\frac{3}{4}$ of another revolution where it sheared the inner barrel from 12:45-2:30 o'clock positions (**ATTACHMENT 6**, pages 32-36). Therefore, based on both the CFM and UTAS assessments, the fan blade root traveled almost two complete revolutions within the fan case and inlet before it departed.

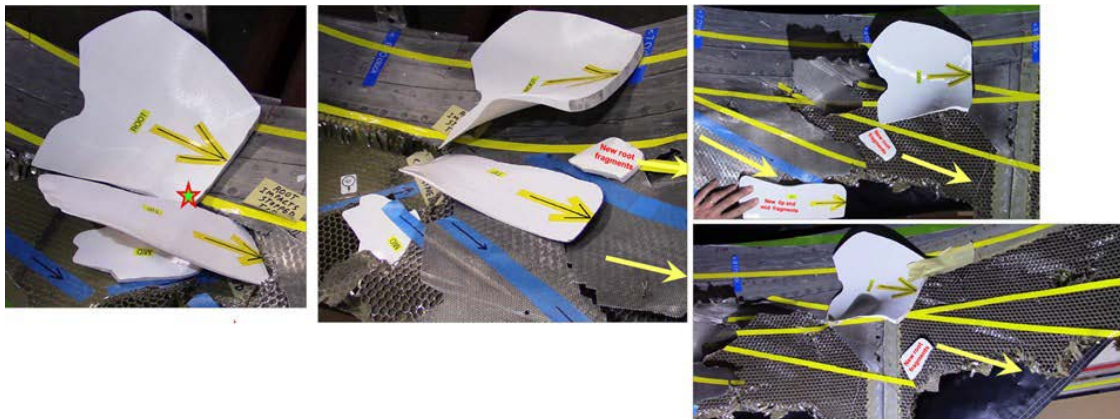


PHOTO 132: ROOT PATH WITH INLET AND COLLISION WITH TIP PANEL

PHOTO COURTESY OF UTAS

Based on the UTAS trajectory and damage assessment of the inlet, the fan blade fragments sheared the inner barrel between about 112.5°-210° circumferentially (See **PHOTO 129**), which is beyond what the inlet was designed to withstand (See Section 9.4 CONTINUED SAFE OPERATION POST FAN BLADE SEPARATION REQUIREMENTS – ‘FLY HOME’ SAFE ANALYSIS for details). **PHOTO 133** provides a graphical presentation and comparison of the CFM and UTAS blade fragment trajectories assessment for the various fragments as they cross the A1 flange.

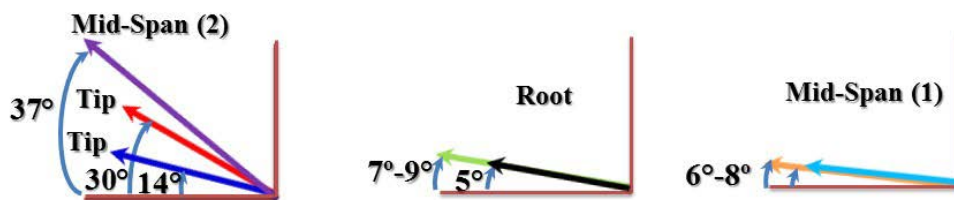
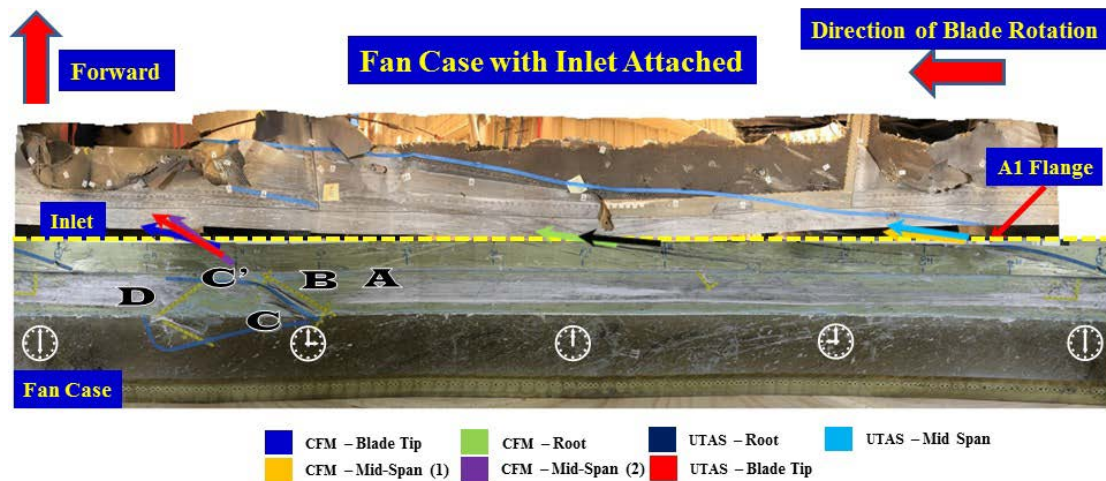


PHOTO 133: CFM AND UTAS FAN BLADE FRAGMENT EXIT TRAJECTORY ASSESSMENT

PHOTO COURTESY OF CFM AND UTAS

TABLE 7: FAN BLADE TRAJECTORY ASSESSMENT						
Blade Fragment	Fan Case CFM Assessment			Inlet Case Inner Barrel UTAS Assessment		
	Initial Impact (clock position)	Exit A1 Flange (clock position)	Exit A1 Flange (degree)	Entry into Inlet (clock position)	Entry into Inlet (degree)	Remarks
Blade Tip (original)	2:15	4:00-5:00	14°	4:00-4:15	30°	Imbedded at 6:30 o'clock position
Mid-span 1 (original)	3:00-4:00	8:00	6°-8°	8:00	6°-8°	Sheared inner barrel skin from about the 3:30-6:45 o'clock position and impacts T12 temperature probe
Mid-span 2 (original)	3:00-4:00	4:30	37°			
Root (original)	5:00	12:00	7°-9°	11:00	5°	Made a complete revolution before impact embedded blade tip at 6:30 o'clock position.
Blade Tip and Root Collision						
New Blade Root and Tip Fragments						Sheared inner barrel skin from about the 7:30-8:30 and 9:15-10:15 o'clock positions
Original Root Fragment						Sheared inner barrel skin from about the 11:30-11:45 and 12:45-2:30 o'clock positions

11.3 FLIGHT DATA RECORDER DATA

The FDR was readout by the NTSB's Vehicle Recorders Division in Washington DC and several specific data plots and data sets from the event flight and the prior flight were created (See the FLIGHT DATA RECORDER SPECIALISTS FACTUAL REPORT for details). The FDR recorded approximately 27 hours of flight data; therefore, the event flight and previous flights were examined to determine if any parameter shifts had occurred prior to the fan blade separation event to indicate the onset of the fan blade failure or if there were any unique operation conditions that could have caused or contributed to the event.

A sequence of events timeline was created based on data from the FDR and the ECU and is provided in **TABLE 8. PLOT 2** shows the airplane flight path based on positioning data from the FDR. It should be noted that some parameters are sampled at different rates based on the need for the fidelity of that data and are recorded at different times. Also, each second is partitioned into multiple fractions of a second so that even if two parameters are recorded in that same whole second, the order that the parameters were recorded may be chronologically different. Due to variations in sample rates and recording times, it was not possible to align all the parameters perfectly on a common time, instead the information is provided in chronological order and represents the best estimate of the occurrence.

A review of the Weight-on-Wheel (WOW) parameter indicated that takeoff from the New Orleans, Louisiana airport (MSY) occurred at approximately 9:08:22 CDT (96,752 SRN). During the takeoff and initial climb phase, the left engine's fan vibration was within limits and reached a peak of about 1.5-1.6 aircraft units (AU)⁶¹ and then decreased to about 1 AU after engine power reduction and where it remained until the engine event occurred later in the flight. The left engine fan vibration was well below the vibration guidelines provided in the Boeing AMM (TASK 71-00-00-910) from takeoff to the fan blade separation; the maximum vibration is 5 AU, sometimes referring to as cockpit units (CU), with certain inspection required to be performed if the vibration level exceeds 3 AU. While climbing, the left and right engine N1 and N2 were closely matched and were consistent with the auto-throttles engaged.⁶²

At 9:21:45 CDT (97,555 SRN), approximately 13 minutes and 23 seconds after takeoff, a discontinuity occurred in the FDR data resulting in about 0.5 seconds of data not being recorded.⁶³ After this occurrence, the FDR continued to operate for an additional 7 minutes 51 seconds before it stopped recording all data (See **PLOT 1**). Just prior to the discontinuity, FDR data indicated that the aircraft was climbing through a pressure altitude of about 31,259 feet at 287 knots (330 miles per hour (mph)), the recorded left engine N1 and N2 speeds were about 99% rpm and the fan, HPC, HPT and LPT vibrations were recorded as 0.75 AU, 0.14 AU, 3.17 AU, and 0.35 AU respectively.

About 1 second after the discontinuity, at 9:21:46 CDT (97,556 SRN), while passing through 31,264 feet, the left engine N1 speed decreased to 63% rpm and the vertical and lateral acceleration parameters began to change; this is considered the first recorded data indicating that the fan blade separation had occurred. About 2 seconds after the discontinuity, the fan vibration parameter increased up to 5 AU; it remained at this value for almost 2½ minutes before decreasing to 0 AU. About

⁶¹ The left engine fan vibration was below the vibration guidelines provided in the Boeing AMM (TASK 71-00-00-910) from takeoff to the fan blade separation; the maximum vibration is 5 AU with certain inspection requirements if the vibration level exceeds 3 AU.

⁶² The rotor speeds and all vibrations are recorded once every second.

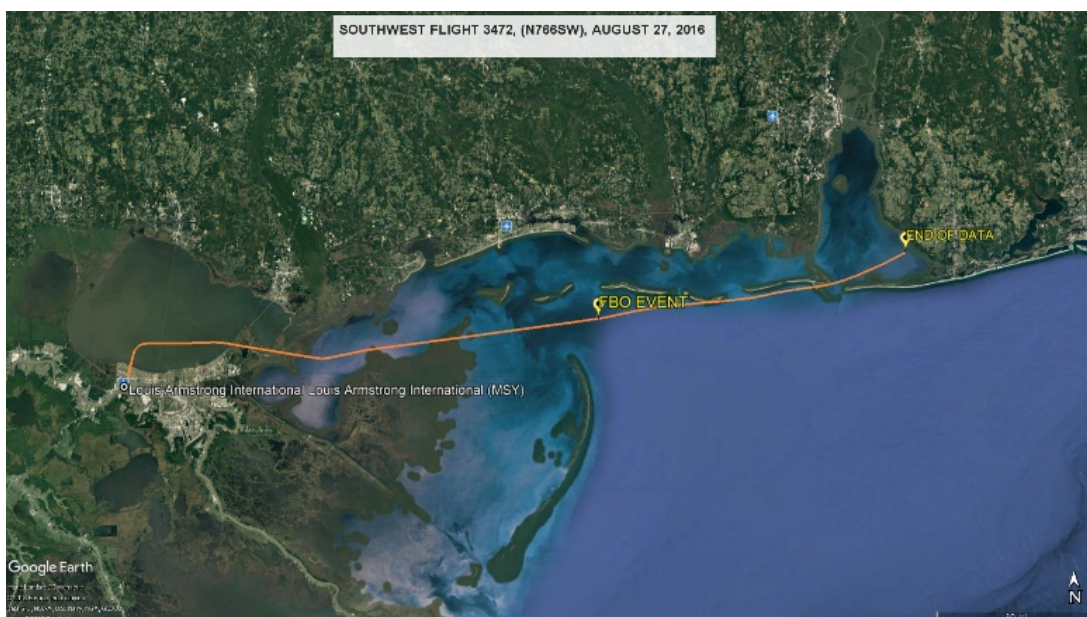
⁶³ At the time of the discontinuity, there was a gap of about 0.5 seconds of data within one subframe (within 1 second). After this subframe, the next sequential subframe was recorded. See Flight Data Recorder Specialist Factual Report, dated March 28, 2018 for details.

3 seconds after the fan blade separation, the left engine N1 speed had decreased and stabilized around 39% rpm; it remained at this value for about 27 seconds at which point the fuel cutoff parameter changed to “CUTOFF”. After the fuel was cutoff to the engine, the N1 decayed to around 21% rpm where it remained until the end of the recording. About 32 seconds after the discontinuity, the cabin altitude warning was triggered. During the engine FBO containment certification tests, the engine shuts down on its own within about 5 seconds after the FBO event. Therefore, the engine FBO containment certification engine did not experience the rundown, run on, or windmilling of the engine post fan blade separation as the SWA flight 3472 event engine experienced.

Since the FDR stopped recording data 7 minutes and 51 seconds after the fan blade separation event, the landing time was not recorded by the FDR. However, based on pilot statements indicating the airplane landed safely at about 9:40 local time, thus the airplane continued to fly for about an additional 18 minutes 14 seconds after the fan blade separation.

The event FDR had recorded multiple flight legs. A review of those flights for fan vibration revealed similar vibration profile; fan vibration ramped up during the takeoff and climb but was still below the recommended guidance and then decreased to well within the recommended limit. On the event flight, the LPT vibration prior to the event was recorded to be 3.17 AU, which was above the recommended limit with no further troubleshooting required. A review of the previous flight showed a similarly high LPT vibration; again, around the 3 AU range.

The various on-scene test scenarios were unable to duplicate the observed in-flight electrical anomaly that caused the FDR to lose power. Subsequent off-wing component tests were also unable to positively identify a specific part or component that could have contributed to the power loss. Prior to the aircraft returning to service, the components removed for additional testing were replaced, and the on-scene test scenarios were accomplished again on the airplane. The additional tests were also unable to duplicate the observed in-flight electrical anomaly that caused the FDR to lose power. The exact cause of the electrical power interruption to the FDR was indeterminate. SWA performed a scheduled FDR download and parameter validation test on February 28, 2017 with no faults found, consistent with a properly functioning FDR.



PLOT 2: FLIGHT PATH FOR INCIDENT FLIGHT 3472

TABLE 8: EVENT TIMELINE		NOTE - All data references are for the No. 1 engine unless otherwise noted
ESTIMATED ELAPSED TIME HOUR:MINUTES:SECONDS SUBFRAME REFERENCE NUMBER	EVENT DESCRIPTION	COMMENTS
T = -00:13:23 96752	Takeoff	Weight-on-Wheels goes from GROUND to AIR mode
T = -00:00:01 97554	Pressure altitude 31,259 feet Computed airspeed 289 knots N2 = 99% N1 = 99% Fan vibration = 0.75 AU HPC vibration = 0.14 AU HPT vibration = 3.17 AU LPT vibration = 0.35 AU	AU = Airplane Unit
T = - 00:00:00 97555	Fan Blade Failure Momentary Power Lose	FDR did not record data for an estimated 0.5 seconds
T = 00:00:01 97556	Pressure altitude of 31,264 feet Airspeed 287 knots N1 = 63% N2 = 89% Low oil pressure caution ALERT Oil pressure redline WARNING Hydraulic system 'A' low oil pressure WARNING Airplane left roll begins	Low oil pressure caution transitions from NORMAL to ALERT and remains at ALERT for the remainder of the recording Oil Pressure redline WARNING transitions from NO WARNING to WARNING (from ECU) Hydraulic system 'A' ENG 1 transitions from NORMAL to LO PRESSURE and remains LO for the remainder of flight Vibrations levels remained same as before event – Possibly vibration level recorded before event Roll to the left (1°)
T = 00:00:02 97557	N1 = 47% N2 = 86% Fan vibration = 5 AU LPT vibration = 0.35 AU	
T = 00:00:03 97558	N1 = 42% N2 = 85%	
T = 00:00:04 97559	N1 stabilized at about 39%-37% N2 = 84% Airplane left roll 16°	N1 remains at this value for 28 seconds Airplane reached maximum roll value
T = 00:00:12 97567	Engine 1 AVM Alert	
T = 00:00:18 97573	Airplane returns to wings level	Airplane left roll returns to 0°
T = 00:00:31 97586	Engine 1 shutdown	Engine 1 fuel cutoff transitions from RUN to CUTOFF
T = 00:00:32 97587	N1 again starts to decay	N1 = 35%, decays for the next 34 seconds
T = 00:00:33 97588	Cabin altitude warning ON	Cabin altitude warning transitions from NO WARNING to WARNING
T = 00:00:43 97598	Oil pressure redline warning OFF	Oil pressure redline warning transitions from WARNING to NO WARNING
T = 00:01:06 97621	N1 stabilizes between to 21%-17%	N1 remains at this level for remainder of flight
T = 00:02:29 97704	Engine 1 AVM alert off	AVM from ALERT to NORMAL

TABLE 8: EVENT TIMELINE		NOTE - All data references are for the No. 1 engine unless otherwise noted
ESTIMATED ELAPSED TIME HOUR:MINUTES:SECONDS SUBFRAME REFERENCE NUMBER	EVENT DESCRIPTION	COMMENTS
T = 00:02:31 97706	Fan Vibration = 0 AU	
T = 00:07:51 98026	Last data set	FDR stopped recording after this time
T = 00:18:14	Landing	Based on pilot statement; landed safely at 9:40 local time. FDR ceased to record data 7 minutes 51 seconds after event

12.0 ADDITIONAL ELECTRICAL HARDWARE TESTING

The APU GCU, PN 762185G, SN 0220, and the BPCU, PN 762187F, SN AAA7001675, were sent to UTAS⁶⁴ in Phoenix, Arizona, the component manufacture, for testing and evaluation. A full bench check was performed per the component maintenance manual (CMM) (CMM-24-23-16 for the APU GCU and CMM 24-23-17 for the BPCU with attention paid to faults that may have caused the units to operate erroneously. In both cases the units passed the incoming and final ATP testing protocol and no faults were confirmed.

The APB PN 1151968-1, SN 119C-3373⁶⁵, located within the power distribution panel (PDP), was sent to the Honeywell (manufacturer of the PDP) facility in Toronto, Canada for a testing and evaluation. The APB was placed in a test PDP and tested in accordance with CMM 24-50-41. The tests were repeated three times and no failures were reported. To perform additional testing of the APB, the APB was sent to Equipment Construction Electrique (ECE), a division of Zodiac Aerospace, Paris, France who is the manufacturer of the APB. Zodiac performed an ATP per the CMM; a vibration test and a visual examination. No significant issues with the breaker were found. Over 1000 ATP setting (closing) and ATP resetting (tripping) cycles/tests were accomplished utilizing voltages under 16 volts (V) applied automatically and manually. The testing resulted in three failures early in the testing process; the failures could not be reproduced on the following cycles/tests. Upon visual inspection, wear was found on one side of the actuator guide and on the magnetic core, along with dust found all around the actuator guide and magnetic core. The wear on the guide and on the magnetic core led Zodiac to surmise that there is a possibility that the actuator might not have set, in particular with, reduced voltage. Zodiac concluded that based on the wear on the guide and on the magnetic core that there was a possibility that the actuator might not have set, in particular with, reduced voltage.

13.0 CORRECTIVE ACTIONS

13.1 PRESS RELEASES AND ALL OPERATOR WIRES

On August 27, 2016, Boeing issued a Multi-Operator Message, No. B-H200-AB-685-ASI, informing Boeing 737-700 operators of the SWA flight 3472 fan blade separation and subsequent inlet departure incident, that the investigation was being led by the NTSB with assistance of the FAA, Boeing,

⁶⁴ Hamilton Sundstrand Corporation was the original designer and manufacturer of the components. Subsequently, Hamilton Sundstrand Corporation was acquired by BFGoodrich/Goodrich Corporation and UTAS respectively.

⁶⁵ The APB is manufactured by Zodiac and their PN and is 231CZ01 and 3265, respectively. The APB is supplied to Honeywell for installation into the PDP. Honeywell affixes their own PN and SN which is listed in the text.

and CFM personnel traveling to the scene, and that Boeing did not recommend nor required any specific actions at that time.

On September 12, 2016, the NTSB issued a press release (<https://www.nts.gov/news/press-releases/Pages/PR20160912.aspx>) informing the general public of the initial on-site findings for both the airplane and engine, initial metallurgical findings performed at the NTSB laboratory in Washington DC, potential future investigative work, and the list of party members.

On September 14, 2016, CFM issued an All Operators Wire (AOW), message No. 16-CFM56-744, informing CFM56-7B customers of the NTSB's press release, that CFM, along with the FAA, Boeing, and SWA were all supporting the investigation, and that currently CFM has no recommended actions for operators.

On December 6, 2016, CFM issued an AOW, message No. 16-CFM56-748, informing CFM56-7B customers of the status of the investigation, that high time fan blades from various operators had been inspected using ECI technique with no cracks found, that the ECI was being added to the ESM piece-part level inspection when the dovetail coating is replaced and provided a list of repair shops currently qualified to perform the Cu-Ni-In coating reconditioning.

On March 10, 2017, CFM issued an AOW, message No. 17-CFM56-752, informing operators of three current or impending actions: (1) repair shops that have been validated to perform the reconditioning of the Cu-Ni-In coating have started to implement the ECI, (2) the fan blade on-wing coating inspection (AMM 72-21-02-200-801) was updated to improve clarity and interpretation of the allowable spalling on the blade root pressure faces (this supersedes the initial guidance outlined in the December 2016 CFM56 Fleet Highlites), (3) impending release of a Category 2⁶⁶ CFM56-7B SB 72-1019 for a one-time inspection of the fan blade root/dovetail pressure faces using a UI technique, and (4) training opportunities for operators intending to perform the on-wing UI.

On June 29, 2017, CFM issued an AOW, message No. 17-CFM56-758, informing CFM56-7B customers of changes to the recommended fan blade maintenance practices, including a revision to SB 72-1019, and the issuance of a new Category 2 SB 72-1024 for inspection of fan blade dovetail. To support future SB applicability, operators were requested to keep track of fan blade PN, SN, and lubrication history. Also, CFM provided a list of regional training seminars open to qualified inspectors from impacted operators who will be performing the UI called out in SB 72-1024.

On September 8, 2017, CFM issued an AOW, message No. 17-CFM56-761, informing CFM56-7B customers that on August 25, 2017, the FAA issued a Notice of Proposed Rule Making (NPRM) 2017-17828 for a proposed Airworthiness Directive (AD) to mandate an UI of fan blades PNs 340-001-022-0 and its field reworked versions (340-001-027-0, 340-001-029-0, 340-001-037-0, or 340-001-039-0) installed on CFM56-7B engines with more than 15,000 CSLSV or on engines with less than 15,000 CSLSV within 18 months or at the next fan blade lubrication.

⁶⁶ According to the CFM56-7B engine manual Category 2 is "Used when CFM International (CFM) technical reasons make the compliance necessary and when an aircraft can stay for suitable time at a line station or maintenance base with the capability to do the procedure. The justification for hour, cycle and end date requirements will be based only on technical considerations (i.e., safety, risk analysis, etc.). May cause non-routine customer action. CFM recommends that you do this Service Bulletin before subsequent flight OR before _____ hours, _____ cycles, (OR) _____ a specified end date (OR) _____ specified interval."

On October 6, 2017, CFM issued an AOW, message No. 17-CFM56-762, informing CFM56-7B customers of EASA's proposal to issue an airworthiness directive (PAD) No. 17-132 to mandate a one-time inspection on all fan blade PN 340-001-022-0 and its field reworked versions (PNs - 027-0, -029-0, -037-0, and -039-0), within 9 months of the effective date of the AD. The EASA will accept comments on the PAD if received by October 27, 2017.

On March 14, 2018, CFM issued an AOW, message No. 18-CFM56-769, providing CFM56-7B customers the same information provided in AOW No. 17-CFM56-762 and that EASA would accept comments on the EASA PAD up to March 19, 2018.

On April 5, 2018, CFM issued an AOW, message No. 18-CF56-770, informing operators that on March 26, 2018, EASA issued AD 2018-0071 requiring a one-time inspection of fan blade PN 340-001-022-0 and its field reworked versions (PNs 340-001-027-0, 340-001-029-0, 340-001-037-0, and 340-001-039-0), within 9 months of the effective date of April 2, 1018.

On April 19, 2018, CFM issued an AOW, message No. 18-CFM56-771, informing CFM56-7B customers of the SWA flight 1380 engine failure and that members of CFM were assisting the NTSB in their investigation.

On April 21, 2018, CFM issued an AOW, message No. 18-CFM56-772, informing CFM56-7B customers that the SWA flight 1380 engine failure was a result of a fan blade release that was contained by the engine; however, had led to damage to the inlet and the eventual separation of the inlet from the engine. The AOW reminded customers of the CFM SB 72-1019 and SB 72-2014 that provide inspection instructions for a one-time on-wing UI of certain fan blades and corrective actions should indications of a crack be detected and that the FAA and EASA issued Emergency Airworthiness Directives (EAD) 2018-09-51 and 2018-0093, respectively to inspect all CFM56-7B fan blades installed in engines with 30,000 CSN or more regardless of PN within 20 days of the issuance of the EAD. To help facilitate the inspections, CFM provided information on how to acquire the necessary tooling and regional training seminars in order to conduct the UI.

On April 24, 2018, CFM issued an AOW, message No. 18-CFM56-773, informing CFM56-7B customers of the FAA's EAD 2018-09-15 mandating an initial fan blade inspection for all CFM56-7B engines with 30,000 cycles or more total accumulated flight CSN using CFM SB 72-1033 inspection instructions. It also stated that CFM had applied for and was approved an Alternate Method of Compliance (AMOC) for the following: (1) the inspection for ne new production blade that was first installed in an engine after January 1, 2017 and (2) provide credit for previous actions, if the fan blade had an ECI per the ESM instructions.

On May 2, 2018, CFM issued an AOW, message No. 18-CFM56-774, informing operators that the FAA would be publishing AD 2018-09-10 on May 2, 2018 mandating a repetitive UI for all CFM56-7B fan blades in full compliance with CFM SB 72-1033 (See Section 13.6.1 SERVICE BULLETINS for details) and to inform operators of the FAA's and EASA's acceptance of an Alternate Method of Compliance (AMOC) allowing operators to take credit for complying with EAD 2018-0093 (EASA) and EAD 2018-09-51 (FAA) if the blades had been previously inspected in accordance with the ESM ECI procedure.

On May 3, 2018, the NTSB issued a Press Release providing the public a status of the ongoing investigation into the SWA flight 1380. The Press Release provided an initial damage assessment to the engine and airplane, preliminary results of metallurgical examination of the fractured No. 13 fan

blade, a brief history of the accumulated fan blade operational and repair history at the time of the accident, the immediate corrective actions implemented by the FAA and EASA to inspect CFM56-7B fan blades, and the flight and cabin crews handling of the emergency. The NTSB Press Release can be viewed at this URL:

https://www.nts.gov/investigations/Pages/DCA18MA142.aspx#utm_source=release&utm_medium=release&utm_campaign=mr&utm_content=aviation.

On May 10, 2018, CFM issued an AOW, message No. 18-CFM56-776, informing operators that Revision 1 to CFM56-7B SB 72-1033 was issued on May 9, 2018 and the objective of the revision was to prioritize the initial ultrasonic fan blade dovetail inspection for high cycle fan blades. The revision would require all the inspection of fan blades with more than 20,000 cycles be inspected before June 30, 2018 whereas the original version of CFM56-7B SB 72-1033 had those inspections completed by August 31, 2018. CFM also informed operators that although the current FAA and EASA ADs address the original version of CFM56-7B SB 72-1033 inspection requirements and timeline and it was anticipated that the FAA and EASA would also mandate the updated timeline per CFM56-7B SB 72-1033 Revision 1.

On May 22, 2018, CFM issued an AOW, message No. 18-CFM56-777, informing operators that the FAA issued AD 2018-10-11 with an effective date of June 1, 2018 that would supersede AD 2018-09-10 and that EASA issued AD 2018-0019 with an effective date of May 18, 2018 supersedes AD 2018-0093-E; the ADs require inspections per CFM56-7B SB 72-1033 revision 1.

On August 7, 2018, AOW, message No. 18-CFM56-779, informing operators that CFM56-7B SB 72-1033 revision 2 was issued on July 27, 2008 that reduced the ultrasonic fan blade dovetail inspection interval from 3,000 flight cycles (FC) to 1,600 FC.

On August 27, 2018, CFM issued an AOW, message No. 18-CFM56-783, informing operators that SB 72-1033 revision 2 was released on July 27, 2018 that reduced the ultrasonic repetitive inspection interval from 3,000 to 1,600 FC.

On October 5, 2018, CFM issued an AOW, message No. 18-CFM56-785, informing operators that the FAA and EASA had issued AD 2018-18-01 and AD 2018-0211, respectively requiring the reduction of the CFM56-7B fan blade repetitive inspection as per SB 72-1033 revision 2 from 3,000 flight cycles to 1,600 flight cycles.

13.2 CFM FLEET HIGHLIGHTS

CFM issues a monthly electronic newsletter called *CFM56 Fleet Highlights* to provide CFM56 operators articles of interest, updates on SBs, changes to maintenance requires, status and findings from investigations, and any other useful, pertinent, and timely information for the operators.

The first *CFM56 Fleet Highlights* issued after the SWA flight 3472 fan blade separation incident was in the December 2016 edition. The article was titled *CFM56-7B – Fan blade dovetail root inspection during scheduled fan blade relubrication (16-12-7221-03)* and it provided guidance, pictures, and figures on how to properly interpret and inspect fan blades dovetails using the AMM TASK 72-21-02 with a focus on clarifying what areas of missing coating are acceptable or not acceptable. Since the that article, there has been 19 additional *CFM56 Fleet Highlights* articles dealing fan blade. The articles provided additional guidance and changes on the inspection and relube instructions as well as updates on

proposed and issued SBs and ADs. **TABLE 9** provides a list of those *CFM56 Fleet Highlites* relating to the fan blade.

TABLE 9: CFM56 FLEET HIGHLIGHTS FAN BLADE ARTICLES	
March 2017	CFM56-7B – Fan blade inspection SB 72-1019 (17-03-7221-03)
May 2017	CFM56-7B – Revised fan blade inspection plan (17-05-7221-01)
June 2017	CFM56-7B – CFM56-7B fan blade inspection changes (17-06-7221-01)
September 2017	CFM56-7B – FAA-2017-17828 NPRM: fan blade inspection (17-09-7221-01)
October 2017	CFM56-7B – EASA PAD17-132 on CFM56-7B fan blade inspection (17-10-7221-01)
March 2018	CFM56-7B – EASA PAD17-132 on CFM56-7B fan blade inspection (18-03-7221-02)
April 2018	CFM56-7B – Emergency AD 2018-09-51 AMOC for ECI (Eddy Current Inspection) (18-04-7221-05)
April 2018	CFM56-7B – Fan blade inspection SB 72-1033 – training sessions and inspection kits (18-04-7221-04)
April 2018	CFM56-7B – New Service Bulletin for fan blade inspections (18-04-7221-03)
April 2018	CFM56-7B – Southwest Airlines flight 1380 (18-04-7200-02)
April 2018	CFM56-7B – EASA AD 2018-0071 on CFM56-7B fan blade inspection (18-04-7200-01)
May 2018	CFM56-7B – SB 72-1033 fan blade ultrasonic inspection added to Workslope Planning Guide (18-05-7221-06)
May 2018	CFM56-7B – FAA AD 2018-10-11, EASA AD 2018-0109 on CFM56-7B fan blade inspections (18-05-7221-03)
May 2018	CFM56-7B – Revision to SB 72-1033 Category 1A for CFM56-7B fan blade inspection (18-05-7221-02)
May 2018	CFM56-7B – FAA AD 2018-09-10, EASA and FAA AMOC approvals and fan blade inspection updates (18-05-7221-01)
August 2018	CFM56-7B – EASA PAD 18-117 and FAA AMOC on fan blade inspection (18-08-7221-04)
August 2018	CFM56-7B – Service Bulletin 72-1033 new revision for fan blade inspection (18-08-7221-03)
October 2018	CFM56-7B – EASA AD 2018-0211 and FAA AD 2018-18-01 on CFM56-7B fan blade inspection (18-10-7221-05)

13.3 CFM56-7B HIGH TIME FAN BLADE INITIAL FLEET SAMPLING CAMPAIGN

Once multiple SWA fan blades from the event set were found to have cracks in the dovetail, CFM started a proactive fleet sampling campaign in September 2016 to inspect high time fan blades with similar cycles as the SWA event set; typically fan blades with similar fan blade configuration evolution and with 30,000 cycles or greater but some lower time blades as well for comparison. The intent was to better understand and collect data on the in-service condition of the Cu-Ni-In coating and if cracks were present in the dovetail on high cycle fan blades. This data was compared with what was observed on the SWA event fan blade with similar flight cycles to determine any similarities.

Fan blades from various operators, including SWA, with similar fan blade configuration evolution and similar high time/cycles since last overhaul were evaluated. Those fan blade sets identified for the sampling program were removed from engines and were sent to either Propulsion Technologies International (PTI) or Safran Aircraft Engines, Saint-Quentin-en-Yvelines for evaluation. Also, any additional fan blade sets easily available for evaluation (already removed from service and awaiting inspection and repair) were also included. The blades were subjected to a combination of various crack inspection techniques; FPI plus ECI or UI. By November 2016, over 700 fan blades were inspected, about half of them over 30,000 cycles. Of the 700 blades inspected, roughly 80% were SWA fan blades (SWA is the single largest operator of the CFM56-7B engine), none were found to have cracks in the dovetail. Furthermore: (1) local spalling of the dovetail coating was found on some fan blade dovetails between 20,000-35,000 cycles, (2) about 10% of all blades showed some dovetail spalling, spalling was located primarily at the TE and LE, more so on the TE concave side, (3) dovetail spalling was found in the areas defined as contact and non-contact, (4) no microcracks were found, and (5) some dovetail coating overthickness was noted and was associated with coating spalling on TE concave side. The RS measurements were conducted on selected fan blades repaired by GKN and by PTI; none showed lower

than the desired compressive stress. None of the sampled fan blades showed spalling in the area where the fractured SWA No. 23 blade exhibited spalling. According to CFM, the spalling that was observed in the sampled fan blades was consistent with their previous field service history.

13.4 NEW FAN BLADE INSPECTION AND REPAIR OVERHAUL REQUIREMENTS AND TECHNIQUES

Prior to the SWA flight 3472 blade separation and subsequent inlet departure event that occurred on August 27, 2016, during overhaul the fan blades were inspected for cracks using an ultra-high sensitivity level 4 FPI. Following the SWA flight 3472 incident, CFM developed an ECI procedure and specialized probes to inspect the blade dovetail; this ECI was in addition to the FPI as the FPI covers the entire blade not just the dovetail. ECI utilizes magnetism to detect discontinuities such as cracks at the surface or near-surface of the part. The ECI developed by CFM is a manual scan of the dovetail on both the concave and convex sides of the dovetail at overhaul and with the Cu-Ni-In coating removed (**FIGURE 27** and **PHOTO 136**). In November 2016, the ECI procedures were validated and CFM visited to the various CFM56-7B fan blade overhaul shops to provide them training and tooling to perform the inspection. Also, in November 2016, the ESM TASK 72-01-200-01 *Fan Rotor Blade – Inspection* was changed to include the ECI inspection. According to GKN, they started implementing the ECI around the middle of December 2016.

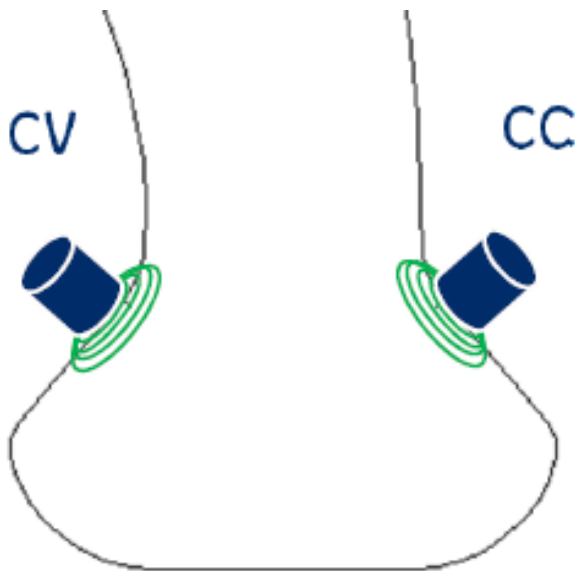


FIGURE 27: LOCATION OF THE ECI SCAN⁶⁷

FIGURE COURTESY OF CFM



PHOTO 136: ECI TOOL

PHOTO COURTESY OF CFM

In July 2018, CFM sent out to various authorized repair shops and proposed revised Cu-Ni-In reconditioning REP 15-93387 repair document for review and validation. The repair restricts abrasive blasting and coating overspray in the dovetail radius; changes the requirement for the abrasive blast surface roughness, changed the lower hardness limit from 140HV0.3 to 120HV0.3, included a requirement to check the coating thickness on individual parts using two gauges that CFM developed; and the shot-peening repair that was once in the ESM was removed and is now a RSS repair to be performed in accordance with REP 15-93389; and deleted the alternative method of using abrasive dry blast to remove newly reapplied plasma coating that did not meet the specification in preparation for recoating

⁶⁷ CV stands for convex and CC stands for concave.

(chemical removal is now the only approved process). As of this report, the validation process is continuing and a final revision to REP 15-93387 has yet been published.

13.5 CHANGES TO FAN BLADE ON-WING INSPECTION REQUIREMENTS

In June 2017, the on-wing visual fan blade dovetail Cu-Ni-In coating inspection AMM SUBTASK 72-21-02-200-001-F00 was revised by: (1) removing any reference to “Area Y” and “More than Area Y”, (2) simplify the “Area X” inspection and (3) removing the continue-in-service restriction of 1,500 flight cycles or 2,500 flight hours condition for the contact area inspection. The previous “Area X” trapezoid figures at the top of **FIGURE 28** have been replaced with simplified rectangles, bottom of **FIGURE 28** with the black revision bar.

Prior to the SWA flight 3472 fan blade separation incident, field inspections of the fan blades dovetail occur only typically within the recommended fan blade dovetail relubrication of every 3,000 cycles or 5,000 hours whichever came first and was only a visual inspection of the Cu-Ni-In coating per AMM TASK 72-21-02. Materials laboratory analysis performed by the NTSB and CFM on the event fan blade set found cracks in the dovetail of several blades that were not visually apparent through the coating; essentially the coating masked the surface presence of the crack. The cracks were only discovered after the coating had been removed. CFM developed an UI technique that could be performed with the coating still on the dovetail; hence it could be utilized during the field inspection during fan blade relube. UI utilizes a mechanical induced ultrasonic wave to detect discontinuities. Similar to ECI, UI is a manual scan of the dovetail on both the concave and convex sides of the dovetail but the orientation of the scan is slightly different. The UI scans from the bottom of the blade so that the coating does not interfere whereas the ECI scans at the interface of the root and dovetail and the coating must be removed first (**FIGURE 29** and **PHOTO 137**). CFM issued Category 2 SB 72-1019 in March 2017 for a one-time inspection of the fan blade root/dovetail pressure faces using a UI technique.

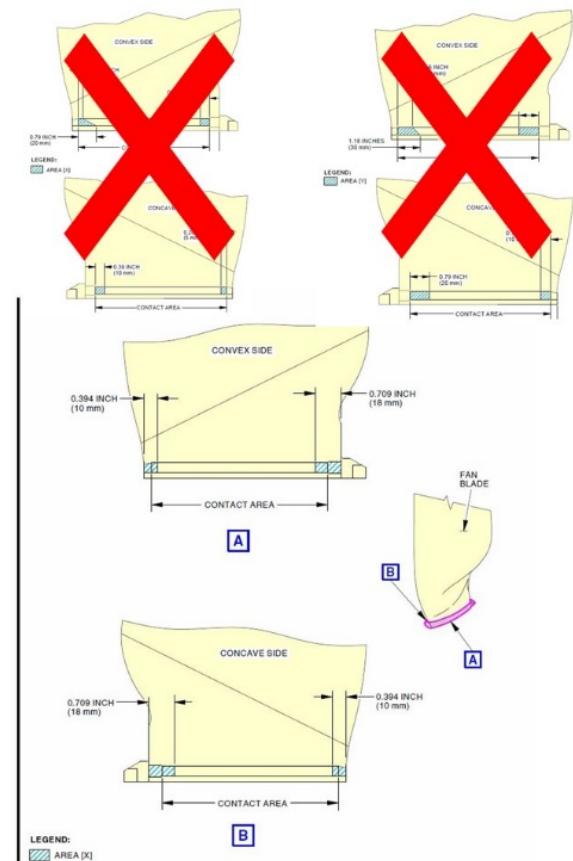


FIGURE 28: NEW FAN BLADE DOVETAIL COATING INSPECTION

FIGURE COURTESY OF CFM

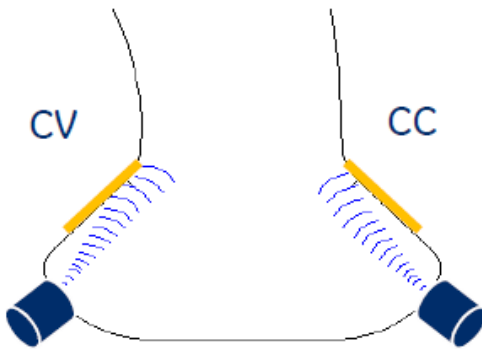


FIGURE 29: LOCATION OF THE UI SCAN

FIGURE COURTESY OF CFM



PHOTO 137: UI TOOL

PHOTO COURTESY OF CFM

13.6 SERVICE BULLETINS AND AIRWORTHINESS DIRECTIVES

TABLE 10 provides, in chronological order, corrective actions, that includes SBs, ADs, and ESM changes dealing with the CFM56-7B fan blade dovetail inspection and relube interval.

13.6.1 Service Bulletins

On March 24, 2017, CFM issued compliance category 2 impact B⁶⁸ CFM56-7B SB 72-1019 titled *ENGINE - Fan and Booster Assembly (72-21-00) - One Time Ultrasonic Inspection of High Time Fan Blades Dovetails* to perform a one-time UI of fan blade root/dovetail pressure faces for cracks on all fan blades, PNs 340-001-022-0, -026-0, -027-0, -028-0, -029-0, -036-0, -037-0, -038-0, and -039-0, installed on CFM56-7B engines (**FIGURE 30** – fan blade configuration flow chart). This inspection was to be performed on-wing; to complete the task the fan blade dovetails are relubed before reinstallation. CFM recommended that the SB be performed as soon as possible without effect on revenue service based on **engine CSLSV not fan blade CSN or CSLSV**, as not all operators track fan blades time, nor are they required to, in accordance with the following inspection plan:

For engines that have accumulated more than 15,000 FC since last shop visit at the issue date of this service bulletin, do this service bulletin not later than 6 months after the issue date of this service bulletin.

For engines that have accumulated more than 12,000 FC since last shop visit at the issue date of this service bulletin, do this service bulletin not later than 12 months after the issue date of this service bulletin.

⁶⁸ CFM defines Category 2 as “Used when CFM International (CFM) technical reasons make the compliance necessary and when an aircraft can stay for suitable time at a line station or maintenance base with the capacity to do the procedure. The justification for hour, cycle and end date requirements will be based only on technical considerations (i.e., safety, risk analysis, etc.). May cause non-routine customer action.”

CFM defines impact B as “This recommendation is to address a condition that may result in an increased Rate of In-Flight Shutdowns (IFSD), Take-Off Aborts (TOA), Air Turn Backs (ATB) or Diversions (DIV).”

For engines that have accumulated more than 9,000 FC since last shop visit at the issue date of this service bulletin, do this service bulletin not later than 18 months after the issue date of this service bulletin.

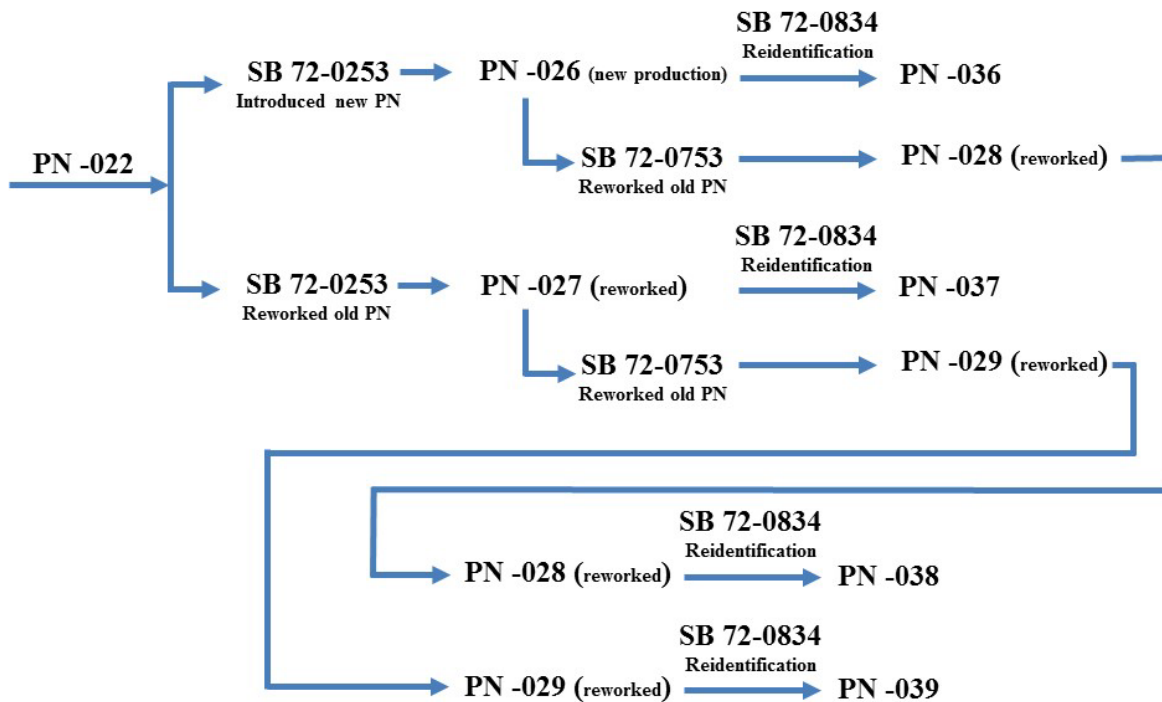


FIGURE 30: CFM56-7B FAN BLADE CONFIGURATION CHANGES

As previously mentioned, this SB can be accomplished on-wing and CFM provided training opportunities for operators to learn to use the UI equipment. On June 20, 2017, SB 72-1019 Revision 1 was issued that removed the 12-month and 18-month compliance sections and maintained only the first set of inspections on engines that have accumulated more than 15,000 CSLSV. Inspections would have to be completed by September 2017 (6 months since SB issuance)⁶⁹ as follows:

For engines that have accumulated more than 15,000 FC since last shop visit at the issue date of this service bulletin, do this service bulletin not later than 6 months after the issue date of this service bulletin.

On July 28, 2017, CFM issued compliance category 2 impact B CFM56-7B SB 72-1024 titled *ENGINE – Fan and Booster Assembly (72-21-00) – Ultrasonic Inspection of Fan Blade Dovetails* to perform ultrasonic inspection on fan blades PNs 340-001-022-0, -027-0, -029-0, -037-0, and -039-0 (fan blades that were modified from the PN -022-0), installed on CFM56-7B engines as soon as possible and no later than December 31, 2018. The significant difference between SB 72-1019 and 72-1024 are that: (1) SB 1024 UI does not apply to new production blades introduced by SB 72-0253 or later modified, for example this SB does not apply to fan blades PNs -028-0 and -038-0 whereas SB 72-1019 does and (2) unlike SB 72-1019 revision 1 that required inspection of engines within 6 months based on

⁶⁹ According to SWA, 111 SWA engines were specifically identified in SB 72-1019. One hundred seven (107) were inspected before the recommended completion date of mid-September 2017 and the other four were inspected by October 12, 2017. The four engines inspected after the end of September were either granted an internal (SWA) extension for maintenance planning due to Hurricane Harvey's impact on our the SWA Houston maintenance base or were already off-wing for heavy maintenance

having more than 15,000 CSLSV, SB 72-1024 required those fan blade PNs listed be inspected regardless of FC or CSLSV within 18 months.

On April 20, 2018, CFM issued compliance category 2 impact A⁷⁰ CFM56-7B SB 72-1033 titled *ENGINE – Fan and Booster Assembly (72-21-00) – Ultrasonic Inspection* to perform an initial and repetitive ultrasonic inspection on all CFM56-7B fan blades (PN 340-001-022-0, -026-0, -027-0, -028-0, -029-0, -036-0, -037-0, -038-0, and -039-0), based on engine CSN and **not** engine CSLSV as SB 72-1019 and SB 72-1024 had done. The SB established initial inspection intervals as follows:

For engines with greater than or equal to 30,000 CSN as of the issuance date of this service bulletin, perform the initial fan blade dovetail ultrasonic inspection as per this service bulletin with 20 days of the service bulletin issuance.

For engines with less than 30,000 CSN as of the issuance date of this service bulletin, the initial threshold for first inspection depends on the information available on fan blade age:

For fan blades with known CSN, perform the initial fan blade dovetail ultrasonic inspection as per this service bulletin no later than 20,000 CSN on the fan blade or not later than August 31, 2018 whichever occurs later.

For fan blades with unknown CSN, perform the initial fan blade dovetail ultrasonic inspection as per this service bulletin no later than August 31, 2018.

The repetitive part of the SB 72-1033 called for inspections every 3,000 FC, which corresponded with the fan blade relube recommended upper limit interval proposed by CFM. The SB also addressed fan blades not already installed on an engine. For those fan blades, they are to be inspected before engine installation except if: the fan blade cycles are known, and they have less than 20,000 CSN or had been previously inspected per this service bulletin less than 300 cycles ago.

On May 9, 2018, CFM issued compliance category 1⁷¹ impact A CFM56-7B SB 72-1033 Revision 01 titled *ENGINE – Fan and Booster Assembly (72-21-00) – Fan Blade Dovetail Repetitive Ultrasonic Inspection* to perform the same repetitive ultrasonic inspection on all CFM56-7B fan blades as instructed in SB 72-1033 but with a completion date of June 30, 2018 instead of August 31, 2018 for a subpopulation of fan blades with more than 20,000 cycles with a prioritization of the highest cycled fan blades. Revision 1 did not alter the provision for inspecting all fan blades installed on engines with greater than 30,000 CSN by May 10, 2018 (20 days after issuance of the original SB 72-1033) or the repetitive inspection to be performed every 3,000 FC after initial inspection. Essentially, the SB prioritizes inspections of engines with greater than 20,000 CSN. This is based on fan blade operational fleet utilization data that indicates that, even though fan blades are not tracked, engines with greater than 20,000 CSN have a much greater probability of having fan blades greater than 20,000 CSN.

On July 27, 2018, CFM issued compliance category 1 impact A CFM56-7B SB 72-1033 Revision 02 titled *ENGINE – Fan and Booster Assembly (72-21-00) – Fan Blade Dovetail Repetitive Ultrasonic Inspection* changed the repetitive inspection to be performed from every 3,000 FC after initial inspection to every 1,600 FC after initial inspection along with some clarification on concurrence

⁷⁰ CFM defines impact A as “This recommendation is to address a condition that may affect flight safety.”

⁷¹ CFM defines Category 1 as “Compliance is mandatory, generally as a result of the DGAC or FAA action, i.e. Consigne de Navigabilite (CD) Airworthiness Directive (AD), NPRM or pending AD etc. Will cause customer action.”

requirements for SB 72-1024 and clerical issues. The reduction of the repetitive inspection interval from 3,000 FC to 1,600 FC as a result of the additional cracked fan blades found during the fleet inspection campaigns and a revised crack propagation analysis based on those findings (See Section 13.7 CFM56-7B ENHANCED FAN BLADE INSPECTION AND FAILURE ANALYSIS RESULTS for additional details).

As of October 12, 2018, the revised (temporary revision) repetitive fan blade dovetail relube interval in the Boeing 737-600/-700/-800/-900/-900ER MPD has been available electronically on MyBoeingFleet; the revised repetitive relube interval is every 1,600 cycles or 4,000 flight hours within comes first. The next MPD revision is scheduled for February 2019 and it will include the revised fan blade dovetail relube interval.

On November 6, 2018, CFM issued compliance category 1 impact A CFM56-7B SB 72-1033 Revision 03 titled *ENGINE – Fan and Booster Assembly (72-21-00) – Fan Blade Dovetail Repetitive Ultrasonic Inspection* to update the fan blade dovetail inspection threshold, to indicate the inspection differences between the P-8 and other applications, such as the Boeing C-40, Boeing 737 AEW&C, and the Boeing 737 Business Jet, and to provide minor changes and clarifications to the inspection procedures. The changes to the inspection thresholds are as follows and they only apply to the Boeing 737 commercial and business jets, the Boeing C-40, and the Boeing 737 AEW&C (the P-8 has specific requirements that only apply to it and those inspections requirements will not be discussed here)⁷²:

For fan blades with more than 16,000 FC since new as of the issuance date of revision SB 72-1033 revision 3, perform the initial dovetail UI within 1,000 FC or before 20,000 FC since new

For all other fan blades, perform the initial dovetail UI before 17,000 FC since new

Replacement fan blades must have completed the initial dovetail UI if the they more than 17,000 FC since new, or if blade FC since new is unknown, the initial dovetail UI may be deferred no later than November 30, 2018.

Repeat the dovetail inspection every 1,600 FC after the initial inspection (Similar requirement as what was provided in SB 72-1033 revision 2)

For fan blades with more than 1,600 FC since initial inspection, the repeat inspection may be deferred no later than November 30, 2018.

If the initial dovetail UI was completed before 15,400 FC since new, the first repeat inspection may be deferred until 17,000 FC since new

⁷² The Boeing P-8 Poseidon is used by the US Navy as a maritime, patrol, and reconnaissance aircraft; the Boeing C-40 Clipper is used by the US Navy as a transport; and the Boeing 737 airborne early warning & control (AEW&C) is used by several different countries as an airborne warning and control aircraft.

13.6.2 Airworthiness Directives

On August 25, 2017, the FAA issued NPRM 2017-17828 in the Federal Register, docket number 2017-0313, with the comment period closed on October 10, 2017, mandating an ultrasonic inspection as follows: (1) any engines with more than 15,000 cycles-in-service (CIS) since the last engine shop visit must be inspected within six months after the effective date of the AD, and (2) for engines with 15,000 CIS or less and with fan blade PNs 340-001-022-0, -026-0, -027-0, -028-0, -029-0, -036-0, -037-0, -038-0, and -039-0 installed, perform the ultrasonic inspection within 18 months after the effective date of the AD or at the next fan blade lubrication after the effective date of the AD, whichever occurs first. Any discrepancy found during the ultrasonic inspection would require the fan blade to be removed and replaced with a part eligible for installation. If the fan blade had been previously inspected using the ECI inspection in the ESM (TASK 72-01-200-01 *Fan Rotor Blade – Inspection*), the AD would be considered complied with. This NPRM related to CFM SB 72-1019 revision 1. The NPRM document can be viewed at this URL: <https://www.federalregister.gov/documents/2017/08/25/2017-17828/airworthiness-directives-cfm-international-sa-turbofan-engines>.

On March 5, 2018, EASA issued PAD No. 17-132R1⁷³, with the comment period to be closed on March 19, 2018, mandating a one-time ultrasonic inspection of fan blades with PNs 340-001-022-0, -027-0, -029-0, -037-0, or -039-0 (fan blades that were modified from the PN -022-0) within 9 months after the effective date of the AD. Any discrepancy found during the ultrasonic inspection would require the fan blade to be removed before next flight or before the engine was returning to service. If the fan blade had been previously inspected using the ECI specified in the ESM (TASK 72-01-200-01 titled *Fan Rotor Blade – Inspection*) or previous inspection in accordance with SB 72-1019, the AD would be considered complied with. On March 26, 2018, EASA issued AD 2018-0071 with an effective date of April 2, 2018, requiring a one-time ultrasonic inspection of the fan blade PNs listed above with the same compliance time as in the same time period specified above. The AD document can be viewed at this URL: <https://ad.easa.europa.eu/ad/2018-0071>.

On April 20, 2018, the FAA issued a Continued Airworthiness Notification to the International Community advising the Civil Aviation Authorities that the FAA, based on the SWA flight 1380 fan blade separation and subsequent inlet departure accident, that the FAA was considering mandating an inspection of certain CFM56-7B fan blades and was coordinating this with EASA. It further stated that the FAA in the near future intended to issue an AD mandating inspections called out in NPRM 2017-17828, docket number 2017-0313.

On April 20, 2018, the FAA issued an emergency AD (EAD) 2018-09-51 prompted by SWA flight 1380 fan blade separation and subsequent inlet departure accident that occurred on April 17, 2018 (DCA18MA142). The AD required a once-through the CFM56-7B engine fleet fan blade inspection for engines with 30,000 CSN or greater than. The AD also required that within 20 days of issuance that all CFM56-7B all fan blade PNs (PN 340-001-022-0, -026, -027-0, -028-0, -029-0, -036-0, -037-0, -038-0, and -039-0) be ultrasonically inspected for cracks per the instructions provided in CFM SB 72-1033 and if any crack indications were found, the affected fan blade must be removed from service before further flight. Although AD 2018-09-51 based the inspection instructions on SB 72-1033, it acknowledges three significant differences. SB 72-1033 called for a repetitive inspection and the AD is a one-time inspection, SB 72-1033 provides actions for engines with fewer than 30,000 CSN where the AD does not apply to those engines, and SB 72-1033 had a reporting requirement and the AD did not. The

⁷³ PAD No. 17-132 was originally issued on September 29, 2017 with a compliance time of: (1) within 6 months for any PN fan blade installed with engine cycles greater than more than 15,000 CSN and (2) within 12 months for engines with less than 15,000 CSN and having fan blades PNs 340-001-022-0, -027-0, -029-0, -037-0, or -039-0 installed.

AD can viewed at this URL:
[http://rgl.faa.gov/Regulatory and Guidance Library/rgad.nsf/0/6de148fe35ddeddb8625827500759aa4/\\$FILE/2018-09-51_Emergency.pdf](http://rgl.faa.gov/Regulatory_and_Guidance_Library/rgad.nsf/0/6de148fe35ddeddb8625827500759aa4/$FILE/2018-09-51_Emergency.pdf).

On April 20, 2018, EASA issued EAD 2018-0093-E that superseded EASA AD 2018-0071 and required the same ultrasonic fan blade inspections to be performed; however, it has some different compliance requirements. Unlike EASA AD 2018-0071, EAD 2018-0093E included all CFM56-7B fan blade PNs (PN 340-001-022-0, -026-0, -027-0, -028-0, -029-0, -036-0, -037, -038-0, and -039-0) not just those PNs that were modified from the -022-0. Both the FAA and the EASA fan blade inspection EADs required compliance with 20 days of issuance for engines with 30,000 CSN or greater no matter the fan blade cycle; however, the EASA EAD included provisions for engines with fewer than 30,000 CSN as well. For engine with less than 30,000 CSN and fan blades with 20,000 CSN or more, or fan blades with unknown times, then the EAD must be complied with within 133 days (August 31, 2018). For engines with fewer than 30,000 CSN and fan blades with less than 20,000 CSN, the EAD must be complied with within 133 days or before the fan blades reached 20,000 CSN whichever comes first. The other difference between the EASA and FAA fan blade inspection EADs is that the EASA is a repetitive inspection to be performed at intervals not to exceed 3,000 FC whereas the FAA inspection is a once-through the fleet inspection. The EAD document can be viewed at this URL: <https://ad.easa.europa.eu/ad/2018-0093-E>.

On May 2, 2018, the FAA issued AD 2018-09-10 that mandated initial and repetitive inspection of all CFM56-7B fan blade in accordance with SB 72-1033. EAD 2018-09-51 had addressed high time CFM56-7B fan blades by mandating a one-time ultrasonic inspection of all fan blades on engines with more than 30,000 CSN but did not address inspections for engines with less than 30,000 CSN or a need for a repetitive inspection. SB 72-1033 provided instructions and guidance for inspecting fan blades with less than 30,000 CSN and instructions for a repetitive inspection every 3,000 cycles. AD 2018-09-10 addressed those engines with less than 30,000 CSN (EAD 2018-09-51 addressed those engines with 30,000 CSN or greater) and mandated the repetitive inspection consistent with the recommendations provided in SB 72-1033. The AD can viewed at this URL: <https://www.federalregister.gov/documents/2018/05/02/2018-09338/airworthiness-directives-cfm-international-sa-turbofan-engines>.

On May 17, 2018, both the FAA and EASA issued ADs that superseded their ADs issued either in the month to revise the compliance time for the initial inspection of certain high-risk fan blades as called-out in CFM56-7B SB 72-1033 Revision 01. The FAA issued AD 2018-10-11 (superseded AD 2018-09-11) with an effective date of June 1, 2018 and EASA issued AD 2018-019 with an effective date of May 28, 2018. The FAA AD can be viewed at this URL: [http://rgl.faa.gov/Regulatory and Guidance Library/rgad.nsf/0/32e3a4f32c73f31a86258290004757d4/\\$FILE/2018-10-11.pdf](http://rgl.faa.gov/Regulatory_and_Guidance_Library/rgad.nsf/0/32e3a4f32c73f31a86258290004757d4/$FILE/2018-10-11.pdf) while the EASA AD can be viewed at this URL: <https://ad.easa.europa.eu/ad/2018-0109>.

On September 28, 2018, EASA issued AD 2018-0211, with an effective date of October 5, 2018, that superseded EASA AD 2018-0109. EASA AD 2018-0211 retains the requirements in EASA AD 2018-0109 but reduced the interval of the repetitive inspections. The new repetitive UI compliance is at intervals not to exceed 1,600 FC and for those fan blades that have already passed the UI as of the effective date of this AD can defer the next UI inspection until 2 months after the effective date of the AD, but not exceeding 3,000 FC since last inspection of that blade.

On October 1, 2018, the FAA issued AD 2018-18-01, with an effective date of October 16, 2018, that superseded AD 2018-10-11. FAA AD 2018-18-01 requires the same initial and repetitive inspection as previously mandated but revises the compliance time for the repetitive inspection. This AD requires that a repetitive inspection using UI or ECI of the concave and convex sides of the fan blade dovetail occur no later than 1,600 FC since the last inspection or with 450 cycle after the effective date of this AD, whichever occurs later using the instructions in CFM56-7B SB 72-1033 Revision 02.

TABLE 10: CFM56-7B FAN BLADE DOVETAIL INSPECTION CHRONOLOGY				
Date Issued	Document	Inspection	Effectivity	Comply By:
3/24/2017	SB 72-1019 Revision 0	<u>One-time</u> UI of fan blade dovetail concave and convex sides	All CFM56-7B Fan Blades	Engine >15K CSLSV within 6 months of SB issuance (Sept 2017) Engine >12K CSLSV within 12 months of SB issuance (March 2018) Engine >9K CSLSV within 18 months of SB issuance (Sept 2018)
6/20/2017	SB 72-1019 Revision 1	<u>One-time</u> UI of fan blade dovetail concave and convex sides	All CFM56-7B Fan Blades	Engine >15K CSLSV within 6 months of SB issuance (Sept 2017) Removed inspection required for engines with less <15K CSLSV
7/28/2017	SB 72-1024	<u>One-time</u> UI of fan blade dovetail concave and convex sides	CFM56-7B Fan Blades PN 340-001 -022-0 -027-0 -029-0 -037-0 -039-0	As soon as possible, no later than by December 31, 2018 regardless of engine FC or CSLSV
8/25/2017	FAA NPRM 2017-17828	<u>One-time</u> UI of fan blade dovetail concave and convex sides	All CFM56-7B Fan Blades	Engines >15K CSLSV within 6 months of effective date of AD Engines <15K CSN within 18 months of effective date of AD or at next fan blade relube whichever occurs first
11/2016	Revision to CFM56-7B ESM TASK 72-01-200-01	ECI of fan blade dovetail at overhaul	All CFM56-7B Fan Blades	During overhaul of the fan blade with the Cu-Ni-In coating removed
3/26/2018	EASA AD 2018-071 effective date of 4/2/18	<u>One-time</u> UI or ECI of fan blade dovetail concave and convex sides	CFM56-7B Fan Blades PN 340-001 -022-0 -027-0 -029-0 -037-0 -039-0	Initial UI within 9 months of effective date of AD (January 2, 2019) regardless of engine FC or CSLSV
4/20/2018	SB 72-1033 Revision 0	<u>Initial and repetitive</u> UI of fan blade dovetail concave and convex sides	All CFM56-7B Fan Blades	Engine >30K CSN initial UI within 20 days of SB issuance (May 10, 2018) Engine <30K CSN at SB issuance, initial UI by Aug 31, 2018 or no later than 20K if fan blade CSN is known

TABLE 10: CFM56-7B FAN BLADE DOVETAIL INSPECTION CHRONOLOGY

Date Issued	Document	Inspection	Effectivity	Comply By:
				Engine <30K CSN at SB issuance, initial UI by Aug 31, 2018 if fan blade CSN is unknown 3,000 FC repetitive UI
4/20/2018	FAA EAD 2018-09-51 effective at issuance	<u>One-time</u> UI of fan blade dovetail concave and convex sides	All CFM56-7B Fan Blades	Engines >30K CSN initial UI within 20 days of effective date of AD (May 10, 2018)
4/20/2018	EASA EAD 2018-0093E effective at issuance	<u>Initial UI or ECI and repetitive</u> UI of fan blade dovetail concave and convex sides	All CFM56-7B Fan Blades	Engine >30K CSN initial UI within 20 days of EAD effective date (May 10, 2018) Engines <30K and fan blades >20K CSN or unknown time initial UI within 133 days of EAD effective date (August 31, 2018) Engines <30K and fan blades <20K CSN initial UI within 133 days of EAD effective date or before fan blades reach 20K CSN 3,000 FC repetitive UI
5/2/2018	FAA AD 2018-09-10 Effective 5/14/2018	<u>Initial UI or ECI and repetitive</u> UI of fan blade dovetail concave and convex sides	All CFM56-7B Fan Blades	Initial inspection before fan blade reached 20K CSN or within 113 days of AD effective date (September 4, 2018) Initial inspection within 113 days of AD effective date if fan blade cycles are unknown 3,000 FC repetitive UI
5/9/2018	SB 72-1033 Revision 1	<u>Initial and repetitive</u> UI of fan blade dovetail concave and convex sides	All CFM56-7B Fan Blades	Engine >30K CSN compliance unchanged Fan blades >20K CSN Engines <30K and >20K changed inspection compliance from August 31, 2018 to June 30, 2018 3,000 FC repetitive UI
5/17/2018	FAA AD 2018-10-11 effective 6/1/2018	<u>Initial UI or ECI and repetitive</u> UI of fan blade dovetail concave and convex sides	All CFM56-7B Fan Blades	Fan blades >20K CSN Engines <30K and >20K inspection compliance within 30 days of AD effective date (July 1, 2018) Within 90 days of AD effective date initial inspection before fan blades accumulate >20K 3,000 FC repetitive UI
5/17/2018	EASA AD 2018-0109 effective 5/18/2018		All CFM56-7B Fan Blades	Fan blades >20K CSN Engines <30K and >20K Fan blade cycles unknown and 20K<engine<30K CSN inspection compliance within 43 days of AD effective date (July 30, 2018)

TABLE 10: CFM56-7B FAN BLADE DOVETAIL INSPECTION CHRONOLOGY

Date Issued	Document	Inspection	Effectivity	Comply By:
				All others 133 days from April 20, 2018 (August 11, 2018) 3,000 FC repetitive UI
7/27/2018	SB 72-1033 Revision 2	<u>Initial and repetitive</u> UI of fan blade dovetail concave and convex sides	All CFM56- 7B fan blades	Initial UI inspection the same as revision 1 1,600 FC repetitive UI (reduction of interval from revisions 0 & 1)
9/28/2018	EASA AD 2018-0211 Effective 10/5/2018	<u>Initial and repetitive</u> UI of fan blade dovetail concave and convex sides	All CFM56- 7B fan blades	Initial UI inspection the same as FAA AD 2018-10-11 Reduced repetitive UI to every 1,600 FC If passed UI prior to effective date of AD, may deferred next inspection not to exceed 3,000 FC
10/1/2018	FAA AD 2018-18-01 Effective 10/16/2018	<u>Initial and repetitive</u> UI of fan blade dovetail concave and convex sides	All CFM56- 7B fan blades	Initial UI inspection the same as FAA AD 2018-10-11 Reduced repetitive UI to every 1,600 FC
11/6/2018	SB 72-1033 Revision 2	<u>Initial and repetitive</u> UI of fan blade dovetail concave and convex sides	All CFM56- 7B fan blades	Fan blades >16K, within 1K FC or before 20K CSN All other Fan blades before 17K CSN Replacement blades must be inspected if > 17K CSN Replacement blades CSN unknown must be inspected no later than November 30, 2018 Under specific circumstances the repetitive UI every 1,600 FC may be deferred

13.7 CFM56-7B ENHANCED FAN BLADE INSPECTION AND FAILURE ANALYSIS RESULTS

EAD 2018-09-15 was issued on April 20, 2018 and mandated that a once through the CFM56-7B engine fleet fan blade dovetail ultrasonic inspection for engines with 30,000 CSN or greater than within 20 days of issuance. At the time the EAD was issued, SWA had 279 engines installed on-wing or spares available to installation that met that 30,000 cycles criteria and by April 25, 2018, all 279 engines had been inspected. According to CFM, there were approximately 640 engines worldwide that met the 30,000 CSN criteria and as of May 10, 2018, all 640 engines had been inspected. CFM considered the 20-day inspection as the first inspection campaign.

CFM issued SB 72-1033 Revision 01 to accelerate inspection of a subpopulation of fan blades with more than 20,000 cycles with a prioritization of the estimated highest cycle fan blades to be inspected by the end of June 2018; SB 72-1033 Revision 0 originally planned for the inspections to be completed by the end of August 2018. CFM considered those engines to be inspected by June 30 to be the second inspection campaign and estimated that almost 5,400 engines met that criteria. The final campaign was to inspect the rest of the engines by the end of August and it was estimated that almost 4,250 engines met that criteria.

In total, as of this report, there are an estimated 356,000 CFM56-7B fan blades and 14,659 CFM56-7B engines delivered. According to CFM, as of August 22, 2018, all CFM56-7B fan blades targeted by the various ADs have been inspected, either using the on-wing UI or at shop visit using ECI; or cleared through paperwork evidence that the fan blades had less than 20,000 cycles; essentially all fan blades have complied with SB 72-1033, FAA AD 2018-10-11, and EASA AD 2018-019. According to

SWA, as of May 18, 2018, all their CFM56-7B fan blades with greater than 3,000 cycles since new had been inspected using either UI or ECI.

As of this report, there have been a total of 15 confirmed cracked CFM56-7B fan blades; two of the cracked fan blades resulted in an FBO separation, this incident (ESN 874112) and the SWA accident in Philadelphia (ESN 875134).⁷⁴ **TABLE 11** provides a list of confirmed cracked CFM56-7B fan blade dovetails; those highlighted in red are the separated fan blades. Crack indications that were detected by UI were on-wing inspections with the dovetail Cu-Ni-In coating still present and cracks detected by ECI were overhaul shop inspections with the dovetail Cu-Ni-In coating removed. Fan blades that were detected using the on-wing UI were also subjected to an ECI to confirm the crack. The list below is only a list of confirmed cracked fan blade dovetails; additional fan blades were removed from service due to an UI indication but were later confirmed to be a false positive crack indication by ECI.

From the incident engine (ESN 874112), six additional fan blade dovetails were found cracked: three were detected using UI while other three were detected either by FPI and confirmed by ECI or by just ECI. The three incident engine fan blades with UI indication were confirmed to be cracked using ECI which is more sensitive and is performed with the dovetail coating removed. Even though **TABLE 11** lists 15 cracked or failed -7B fan blades, only those that were detected before the April 17, 2018, when the SWA Philadelphia accident occurred will be discussed in detail here.

In May 2017, SWA using on-wing UI discovered an indication on the convex side of a fan blade dovetail about 2-inches from the forward face. The blade was immediately removed from service and sent to CFM for examination. The detailed results of the examination into this cracked blade are captured in the *NTSB Materials Laboratory Factual Report No. 17-043*, Section 16; the blade was identified and labeled as blade J. One additional fan blade, also a SWA blade, was rejected due to coating spalling and laboratory examination later found that it had a very small network of microcracks in the dovetail but no primary crack; this blade will be referred to as D in the *NTSB Materials Laboratory Factual Report No. 17-043*.

Blades J⁷⁵ and D both showed features that were consistent with and dissimilar with event fan blade No. 23. Similarities with event blade No. 23 were that: (1) all were originally PN -022-0 and reworked to PN -027-0, (2) all had accumulated over 30,000 cycles when the indication was detected, (3) all exhibited a network of surface microcracks, (4) all were last inspected during overhaul using FPI with no indications noted, (5) etching and SEM examination of blade J revealed a primary crack with features (LCF and striation density) consistent with No. 23⁷⁶, and (6) blade D exhibited similar coating spalling. Differences with the event fan blade No. 23 were that the cracked blade J did not exhibit visible wear, spalling, or missing coating material and that, although similar spalling was observed, no primary crack was present on blade D. Of significance, blade J had been overhauled 2,876 cycles (at PTI in 2015) prior to crack detection and the striation density estimate for the crack was about 9,000 cycles to reach 0.061-inches (1.55mm), indicative of the crack being present at the last overhaul.

Two additional fan blades were found cracked prior to April 17, 2018 using ECI during overhaul; these fan blades belonged to operators that were not SWA and were identified as blades K and L respectively. The fan blade K PN was -026-0, which is a new production fan blade introduced by SB

⁷⁴ See NTSB accident number DCA18MA142 for details.

⁷⁵ In May 2015, SB 72-0753 was incorporated by PTI and the part number changed to PN 340-001-039-0; the final blade configuration.

⁷⁶ The 23 other fan blades from the same J fan blade set were examined and no ECI indications were noted and no microcracks were found on any of the 12 with the same service history as the cracked blade J.

72-0253; this is not a blade that started life as a -022-0 and reworked. Prior to the SWA Philadelphia event, this was the only cracked fan blade that had not started life as a PN -022-0.

Blade K had accumulated an estimated 19,497 CSN and had been repaired by GKN in 2004 and again in 2008. CFM performed a laboratory analysis of that blade and found that it had a network of microcracks (maximum depth 0.42mm (0.017-inches)) located 20.7mm to 35.3mm (0.828-inches to 1.412-inches) from the LE on the convex side; the crack was slightly more forward than the other cracked blades. The striation density estimated for this crack was about 7,500 cycles. Blade L had accumulated an estimate of 32,000-34,000 CSN and had also been repaired by GKN; GKN performed a repair in 2009.

All the cracked blades, except for two, started life as PN 340-001-022-0 and were converted to PN 340-001-027-0. The other two started life as PN 340-001-026-0; of the two that started as PN -026-0, one was converted to PN 340-001-038-0. Fourteen of the fifteen confirmed fan blade dovetail cracks were on the convex side. Thus far, examination of all the cracked or fractured blades revealed no material anomalies, and the cracks are located in the highest (maximum) LCF predicted stress areas (LE convex or TE concave); the LE convex side is the more critical area of the two.

Based on the results of the additional cracked fan blade dovetail findings, CFM reduced the repetitive on-wing inspection interval from every 3,000 FC to 1,600 FC; this change was recommended in SB 72-1033 Revision 02. Based on a relatively stable crack propagation rate observed in the cracked fan blades dovetail and the UI threshold of 1.5mm (0.06-inches) crack depth, CFM wanted to ensure that at least two UI inspections would be performed with the crack longer than the anticipated detection threshold and sooner than when the crack could reach critical length and separate. On average, the recommended reinspection interval should provide approximately 6 unique or independent inspection opportunities to detect a dovetail crack. This assumes that an UI may be performed with the crack below the detectable threshold but that the follow-on UI would detect the crack before fan blade separation.

CFM reevaluated the fan blade dovetail stresses based on the confirmed cracked fan blade dovetails and made several conclusions. The locations of the cracking observed in the fan blade dovetail were what was expected; however, the dovetail experienced elevated operation stresses that were greater than the originally predicted peak stresses. The higher operation stresses were a combination of coating spalling, higher friction levels when operated without lubrication (prior to the release of CFM56-7B CESM No. 5 in May 2000 providing periodic lube guidance and/or the incorporation of this lube guidance in the Boeing MPD in February 2001) or without shim (fan blades installed in a fan disk with PN 410-0 that did not require a shim), variations in coating thickness, higher dovetail edge loading (the dovetail contact surface-to-fan disk interface – **FIGURE 31**), and loss/relaxation of RS. A significant reduction in LCF capability occurs when sufficient RS is reduced beyond what is designed, accounted, or predicted for. Recall at initial new-make and during any subsequent Cu-Ni-In coating reconditioning repair (if not chemically stripped), the dovetail is shot-peened to ensure that the required compressive layer is applied to the surface of the part. The exact cause(s) for the observed loss

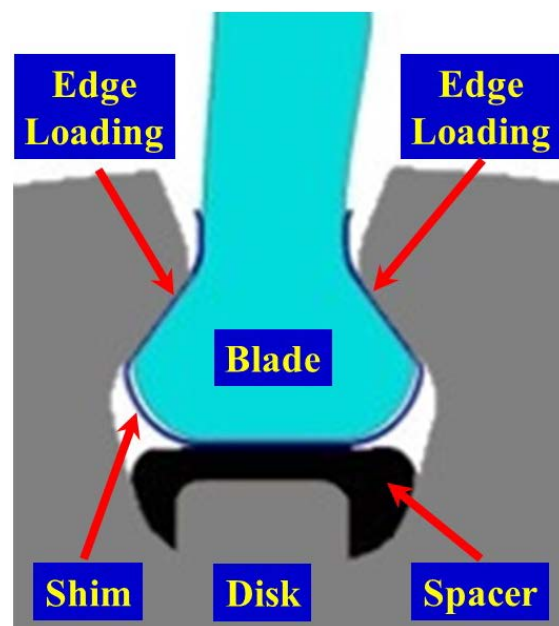


FIGURE 31: EDGE LOAD LOCATIONS

FIGURE COURTESY OF CFM

of RS is unclear and is still being evaluated. Furthermore, CFM determined that some operational events, such as bird strikes or stalls, do not correspond to the cracked blade findings.

TABLE 11: CRACKED (PRIMARY) CFM56-7B FAN BLADE FINDINGS						Concave (CCV), Convex (CVX), ECI-Eddy Current, FPI-Fluorescent Penetrant, UI-Ultrasonic
Cracked Blade	Blade PN	CSN at Crack Detection	Detection Method/ Crack Location	Maximum Crack Depth Inch (mm)	Estimated Cumulative Striation Cycle Count (from 0.0005-inches to maximum depth)	Repairs
SWA 874112 Blade No. 1	PN -022-0 PN 027-0	38,152	Post Event FPI LE CVX	0.009 (0.229)	4,000	GKN (2001 & 2007)
SWA 874112 Blade No. 2	PN -022-0 PN 027-0	38,152	Post Event ECI LE CVX	0.015 (0.381)	4,000	GKN (2001 & 2007)
SWA 874112 Blade No. 11	PN -022-0 PN 027-0	38,152	Post Event UI LE CVX	0.060 (1.52)	10,000	GKN (2001 & 2007)
SWA 874112 Blade No. 17	PN -022-0 PN 027-0	38,152	Post Event UI LE CVX	0.023 (0.584)	5,000	GKN (2001 & 2007)
SWA 874112 Blade No. 19	PN -022-0 PN 027-0	38,152	Post Event FPI LE CVX	0.019 (0.483)	4,000	GKN (2001 & 2007)
SWA 874112 Blade No. 23	PN -022-0 PN 027-0	38,152	Separation LE CVX	0.695 (17.653)	15,000	GKN (2001 & 2007)
SWA 874112 Blade No. 24	PN -022-0 PN 027-0	38,152	Post Event Metallography/UI LE CVX	0.126 (3.200)	16,000	GKN (2001 & 2007)
Blade J	PN -022-0 PN -027-0	33,020	Field UI LE CVX	0.061 (1.55)	9,000	GKN (2001, 2004, 2007) PTI (2015)
Blade K	PN -026-0	19,497	Shop ECI LE CVX	0.016 (0.406)	7,500	GKN (2004 & 2008)
Blade L	PN -022-0 PN -027-0	30K- 34K	Shop ECI LE CVX	0.017 (0.432)	Insufficient data for estimate	GKN (2009)
SWA 875134 Blade 13	PN -026-0 PN -038-0	32,636	Separation LE CVX	0.483 (12.268)	20,000	GKN (2004) PTI (2012)
Blade M	PN -022-0 PN -039-0	22,072	Field UI LE CVX	0.116 (2.946)	9,000	GKN (2007 & 2014)
Blade N	PN -022-0 PN -037-0	31,697	Shop ECI LE CVX	0.033 (0.838)	6,700	GKN (2006) Safran (2013)

TABLE 11: CRACKED (PRIMARY) CFM56-7B FAN BLADE FINDINGS						Concave (CCV), Convex (CVX), ECI-Eddy Current, FPI-Fluorescent Penetrant, UI-Ultrasonic
Cracked Blade	Blade PN	CSN at Crack Detection	Detection Method/ Crack Location	Maximum Crack Depth Inch (mm)	Estimated Cumulative Striation Cycle Count (from 0.0005-inches to maximum depth)	Repairs
Blade O SWA ESN 890978	PN -022-0 PN -039-0	34,770	Field UI LE CVX	0.030 (0.762)	7,800	GKN (2009 & 2017)
Blade P SWA ESN 896763	PN -022-0 PN -027-0	38,936	Shop ECI TE CCV	0.059 (1.4986)	15,000	GKN (2002, 2008, & 2009)

ATTACHMENTS

1	GE Celma Fan Blade Distribution for ESN 874112 for the 2007 Engine Repair
2	GKN Authorized Release Certificate FAA Form 8130-3 Airworthiness Approval Tag Authorized Release Certificates for the SWA ESN 874112 Incident Fan Blade Set dated August 13, 2007
3	CFM CFM56-7B ESN 874112 Event – Fragment Trajectory Analysis, dated October 28, 2018
4	CFM CFM56-7B Southwest ESN 874112 Event – Fan Case Analysis dated January 19, 2017
5	UTAS Aerospace Systems 737NG CFM56-7B24 Southwest Airlines ESN 874112 Pensacola Inner Barrel/Back Skin Investigation Metallurgical Report dated October 26, 2018
6	UTC Aerospace Systems 737NG CFM56-7B24 Southwest Airlines ESN 874112 Pensacola Fan Blade Fragments Fan Case and Inlet Trajectory Analyses dated October 31, 2018

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