Aviation Investigation Final Report

Location:	Pensacola, Florida	**Accident Number:**	DCA16FA217
Date & Time:	August 27, 2016, 09:30 Local	**Registration:**	N766SW
Aircraft:	Boeing 737-700	**Aircraft Damage:**	Substantial
Defining Event:	Powerplant sys/comp malf/fail	**Injuries:**	104 None
Flight Conducted Under:	Part 121: Air carrier - Scheduled		

Analysis

The Boeing 737-700 airplane was climbing through flight level 310 when fan blade No. 23 in the left CFM56-7B engine fractured at its root, with the dovetail (part of the blade root) remaining within a slot of the fan disk. The separated fan blade impacted the engine fan case and fractured into multiple fragments. The blade fragments traveled forward into the inlet and caused substantial damage that compromised the structural integrity of the inlet, causing most of the inlet structure to depart from the airplane. A large portion of the inlet contacted and punctured the left side of the fuselage, creating a hole of sufficient size to cause the cabin to depressurize. The flight crew conducted an emergency descent and landed safely at Pensacola International Airport, Pensacola, Florida, about 21 minutes after the fan-blade-out (FBO) event occurred.

The fan blade fractured due to a low-cycle fatigue crack that initiated in the blade root dovetail under the blade coating near the outboard edge. Metallurgical examination of the fan blade found that its material composition and microstructure were consistent with the specified titanium alloy and that no surface anomalies or material defects were observed in the fracture origin area. The fracture surface had fatigue cracks that initiated close to the dovetail leading edge convex side area, which is where the greatest stresses from operational loads, and thus the greatest potential for cracking, were predicted to occur.

The fan blades were not certified as life-limited parts. The accident fan blade (as well as the six other cracked fan blades in the accident engine) failed with 38,152 cycles since new. Similarly, the fan blade associated with an April 2018 FBO accident (case number DCA18MA142) failed with 32,636 cycles since new. Further, 19 other cracked fan blades on CFM56-7B engines had been identified as of January 2020, and those fan blades had accumulated an average of about 33,000 cycles since new when the cracks were detected.

After this accident, CFM reevaluated the fan blade dovetail stresses and determined that the fatigue cracks initiated in an area of high stress on the dovetail and that the dovetail was experiencing peak stresses that were higher than originally predicted. CFM found that the higher operational stresses resulted from coating spalling, higher friction levels when operated without lubrication or a shim, variations in coating thickness, higher dovetail edge loading than predicted, and a loss or relaxation of compressive residual stress (the stress that is present in solid material in the absence of external forces).

Before the application of the dovetail coating during manufacturing and before the reapplication of the coating that is stripped during each overhaul, the entire blade, including the dovetail, is shot-peened to provide a compressive residual stress surface layer for the material, which increases the fatigue strength of the material and relieves surface tensile stresses that can lead to cracking. A loss of residual stress could be the result of a fan blade's exposure to high temperatures during the application of the dovetail coating as part of the overhaul of a blade set, but no evidence indicated that the accident fan blade dovetail was subjected to an overheat situation during the coating repair process. However, higher-than-expected dovetail operational stresses could also lead to the loss/relaxation of residual stress and premature fatigue crack initiation, which occurred during this event.

Residual stress measurements were taken from multiple areas of the dovetail surface on fan blade No. 23 and eight other blades from the accident engine, including three blades with no identified cracks. All nine blades had abnormal residual stress profiles compared with the reference profile data.

One method that CFM recommended to maintain the fan blade loads within the predicted range and reduce the overall stresses on the blade root in the contact areas is repetitive relubrication of the fan blade dovetails. As part of the relubrication procedure, the fan blades were visually inspected for crack indications. The investigation of this accident found fan blade cracks that had initiated and propagated underneath the dovetail coating. Because such cracks might not be detected during a visual inspection, CFM implemented, in March 2017, an on-wing ultrasonic inspection method to detect cracks with the coating still on the dovetail.

A review of the fan blade overhaul process found that a fluorescent penetrant inspection (FPI) was performed (as specified in the CFM engine shop manual) during the fan blade set's last overhaul in August 2007 to detect cracks. As a result of this accident, CFM implemented, in November 2016, an eddy current inspection (ECI) technique for the fan blade dovetail as part of the overhaul process (in addition to the FPI). An ECI has a higher sensitivity than an FPI and can detect cracks at or near the surface (unlike an FPI, which can only detect surface cracks).

The shims for fan blade Nos. 22 through 24 had a newer design configuration that was introduced after the blades and their associated hardware were first installed on the accident disk. Wear patterns on the shims from blade Nos. 23 and 24 showed that a significant area of coating was missing from the blade dovetails at the time that the shims were installed. The location of the missing coating was in the area where overhaul of the blades was required within 50 cycles. The National Transportation Safety Board could not determine, with the available evidence, how long the dovetail coating damage existed and whether the cracks were large enough to be detected by an FPI during an overhaul in response to the coating damage (when the dovetail coating damage first became significant enough during visual inspections conducted at relubrication to trigger the overhaul).

The damage to the accident inlet and fan case showed that there were significant differences between the accident FBO event and the engine FBO containment certification tests. For example, during the accident FBO event, the fan blade fragments that went forward of the fan case and into the inlet had a greater total mass and a different trajectory (a larger exit angle) and traveled beyond the containment shield. Also, the inlet damage caused by these fan blade fragments was significantly greater than the amount of damage that was defined at the time of inlet certification. Given the results of CFM's engine FBO containment certification tests and Boeing's subsequent structural analyses of the effects of an FBO event on the airframe, the post-FBO events that occurred during this accident could not have been predicted.

Probable Cause and Findings

The National Transportation Safety Board determines the probable cause(s) of this accident to be:

A low-cycle fatigue crack in the dovetail of fan blade No. 23, which resulted in the fan blade separating in flight and impacting the fan case. This impact caused the fan blade to fracture into fragments that traveled farther than expected into the inlet, which compromised the structural integrity of the inlet and led to the in-flight separation of inlet components. A portion of the inlet struck the fuselage and created a hole, causing the cabin to depressurize.

Findings

Aircraft	Compressor section - Fatigue/wear/corrosion
Aircraft	Plates/skins (nacelle/pylon) - Damaged/degraded
Aircraft	Fuselage main structure - Damaged/degraded

Factual Information

History of Flight

Enroute	Powerplant sys/comp malf/fail (Defining event)
Enroute-cruise	Pressure/environ sys malf/fail

On August 27, 2016, about 0922 central daylight time, Southwest Airlines (SWA) flight 3472, a Boeing 737-7H4, N766SW, experienced a left engine failure while climbing through flight level 310 en route to the flight's assigned cruise altitude. Portions of the left engine inlet departed the airplane, and fragments from the inlet impacted the left side of the fuselage, creating a hole. The airplane depressurized, and the flight crew declared an emergency and diverted to Pensacola International Airport (PNS), Pensacola, Florida, where the airplane made an uneventful single-engine landing. The 2 pilots, 3 flight attendants, and 99 passengers were not injured. The airplane was substantially damaged. The regularly scheduled passenger flight was operating under the provisions of Title 14 *Code of Federal Regulations* (*CFR*) Part 121 from Louis Armstrong New Orleans International Airport, New Orleans, Louisiana, to Orlando International Airport, Orlando, Florida.

The flight was the first of the day for the airplane and the flight crewmembers. According to the cockpit voice recorder (CVR), the left engine started at 0903:21, and the right engine started 1 minute later. Air traffic control (ATC) cleared the flight for takeoff at 0907:08. The takeoff and climb were uneventful until 0921:45, when the CVR recorded a "bang" sound followed by the sound of a decrease in engine rpm. The flight data recorder (FDR) showed that the airplane's altitude was 31,259 ft. FDR data also showed that, by 0921:49, the left engine fan speed had decreased from 99% to 39% rpm.

At 0921:52, the captain asked the first officer to declare an emergency. Three seconds later, the first officer contacted ATC to declare an emergency and told ATC that the airplane was descending. At 0922:06, the flight crew completed the SWA Engine Fire or Engine Severe Damage or Separation checklist. At 0922:17, the CVR recorded a sound similar to the cabin altitude warning horn; FDR data showed that the cabin altitude warning parameter had transitioned from "No Warn" to "Warn" 1 second later. (The parameter remained at "Warn" until the FDR data ended at 0929:37 due to a loss of electrical power to the FDR.) At 0922:18, the left engine fan speed had further decreased to 33% rpm.

At 0922:43, ATC told the flight crew that PNS was about 70 miles ahead of the airplane's position. At 0922:51, the CVR recorded the sound of the flight crewmembers using their oxygen masks. Two seconds later, ATC cleared the airplane to flight level 180.

At 0926:47, the captain communicated with the cabin crewmembers about the engine failure and instructed them to secure the cabin. At 0929:22, the flight crewmembers commented about high vibration levels and indicated that they were going to keep the airplane's speed up. The FDR recording ended at 0929:37; the last recorded data (1 second earlier) showed that the airplane was at an altitude of about 17,400 ft and that the left engine fan speed was 17% rpm.

At 0931:38, the flight crew told the cabin crew that there would not be an evacuation. At 0932:10, the CVR recorded the flight crewmembers removing their oxygen masks. At 0935:17, the flight crew made another comment about high vibration levels. At 0937:39, ATC told the flight crew to expect the instrument landing system (ILS) runway 17 approach. Fifteen seconds later, the flight crew selected flaps 1 and 5 to assess the airplane's handling. At 0938:19 and 0938:30, the flight crew selected autobrake level 3 and flaps 15, respectively.

At 0939:10, ATC cleared the airplane for the ILS runway 17 approach. Twelve seconds later, the flight crew completed the descent and approach checklists. At 0940:01, the flight crew reported that the airport was in sight; 12 seconds later, the crew lowered the landing gear. At 0941:40, the flight crew again commented about the vibration. Eleven seconds later, the CVR recorded the sound of the 500-ft automated callout. At 0942:30, the CVR recorded the sound of the airplane touching down on the runway, about 21 minutes after the left engine failure occurred.

This report will discuss that the left engine failure occurred when a fan blade fractured and exited the engine fan case, which is referred to as a fan-blade-out (FBO) event. An FBO event consists of four phases (the impact phase, the engine surge phase, the engine rundown phase, and the windmilling phase) during which the airplane structure is subjected to various loads.

AIRCRAFT INFORMATION

The airplane was equipped with two CFM International CFM56-7B24 turbofan engines, with one engine mounted under each wing. CFM International was established in 1974 as a partnership between General Electric Aviation (GE), a US manufacturer, and Safran Aircraft Engines (Safran), a French manufacturer formerly known as Snecma. (GE manufactured the CF6 engine, and Snecma manufactured the M56 engine; those engine designations were combined to form the new company and engine names.) The CFM56-7B24 engine is one of several CFM56-7B models (other models are the -7B18, -7B20, -7B22, -7B26, and -7B27) installed in Boeing 737 next-generation-series airplanes (the 737-600, -700, -800, and -900). The CFM56-7B references throughout this report apply to all of the engine models.

The inlet, which is part of the Boeing 737-700 airframe, is an aerodynamic fairing that guides air into and around an engine. The inlet is attached to the front of each engine's fan case. The inlet is part of the nacelle, which houses the engine. (Other nacelle components include the fan cowl and the thrust reverser.) The CFM56-7B engine and the Boeing 737-700 inlet are further described in the sections below.

Engine

The CFM56-7B 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). A four-stage low-pressure turbine (LPT) drives the booster assembly, which comprises the engine fan and three-stage low-pressure compressor. The engine rotates clockwise (aft looking forward).

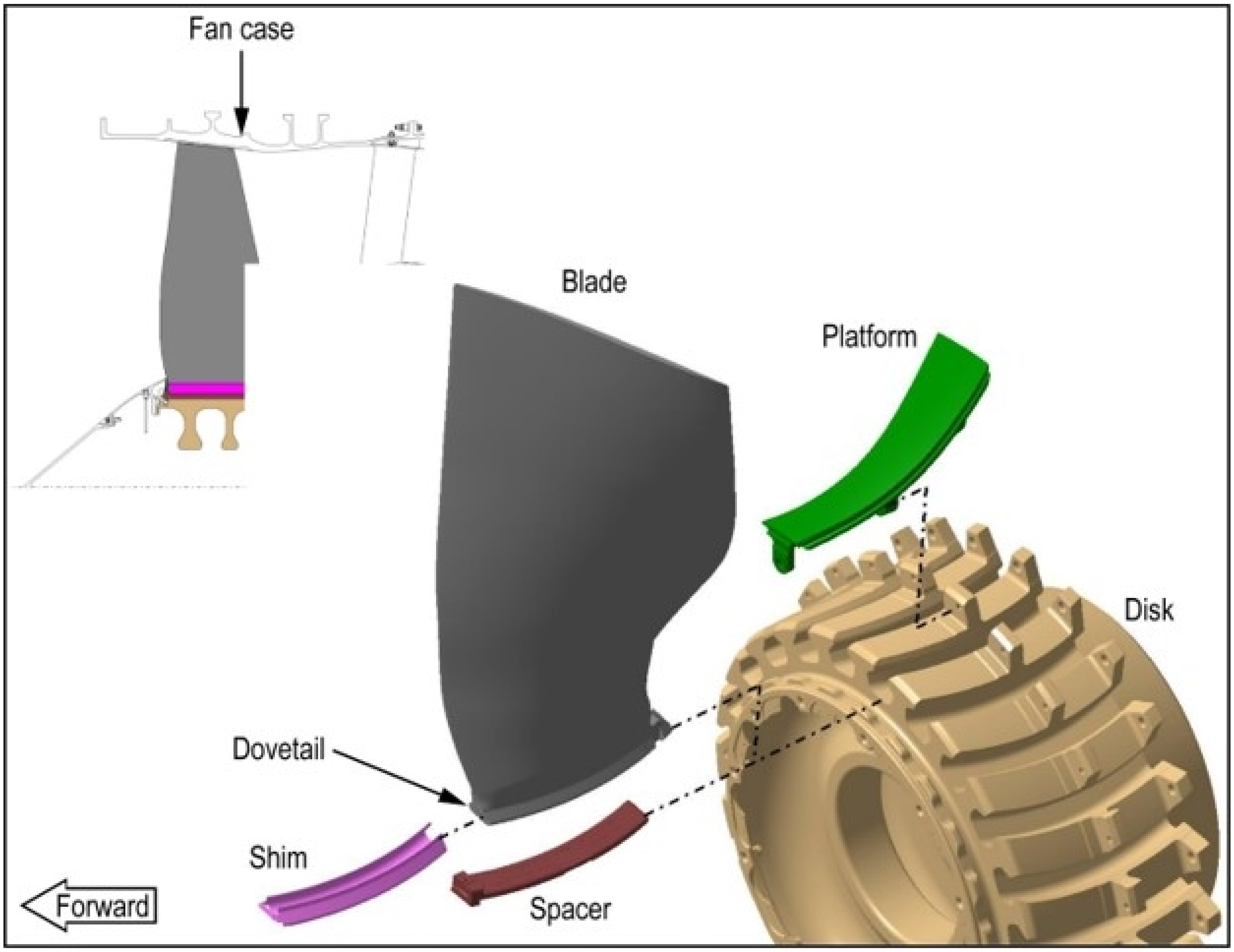
The engine consists of three major assemblies: the fan, engine core, and LPT. GE is responsible for manufacturing the HPC, combustion chamber, and HPT (collectively referred to as the engine core). Safran is responsible for manufacturing the LPT and the booster assembly. Both companies assemble the engines; those assembled by GE are identified by an even engine serial number (ESN) prefix (for example, 874), and those assembled by Safran are identified by an odd ESN prefix (for example, 875). The ESN of the accident engine, 874112, showed that GE assembled the engine.

The fan and booster assembly comprises the front and aft spinner cones, fan disk, fan blades, booster rotor, booster vanes, and associated hardware. The fan disk, which is secured to the booster, has 24 fan blade slots. In accordance with the instructions in Boeing's *737-600/700/800/900 Aircraft Maintenance Manual*, the fan blades are numbered sequentially (1 through 24) in the counterclockwise direction (forward looking aft).

Each CFM56-7B fan blade had a chord (width) of about 11 inches at its widest point. The nominal weight of each fan blade is about 10.83 pounds. The fan blades are made of a titanium alloy (known as Ti-6-4), and the dovetail part of the fan blade, which slides into the fan disk, has a copper-nickel-indium coating to protect the blade from wear and provide a better fit with the fan disk (compared with no coating). Before the application of the fan blade coating, the entire blade, including the dovetail, is shot-peened to increase the fatigue strength of the material and reduce surface tensile stresses that can lead to cracking. (Shot-peening is a process that adds a compressive residual stress surface layer to material, and residual stress is the stress that is present in solid material in the absence of external forces.)

A spacer is installed under each fan blade root primarily to limit fan blade radial (outward) movement. The spacer also ensures that axial (longitudinal) loads would be transmitted to the fan blade axial retention feature if an FBO event or bird ingestion were to occur. A platform is installed on both sides of each fan blade to provide a smooth aerodynamic flow path between the blades. A shim is installed over each fan blade dovetail to prevent fretting (wear) of the fan disk pressure faces and improve lubrication durability, which reduces the amount of stress on the fan blade dovetail and the fan disk pressure faces. The shims for fan blade Nos. 22 through 24 had a newer design configuration than the other shims installed in the fan assembly, which had the previous design configuration. The newer design configuration for the shims was introduced in February 2008 to improve reliability. Both shim configurations are authorized for CFM56-7B fan blades, and both configurations are interchangeable.

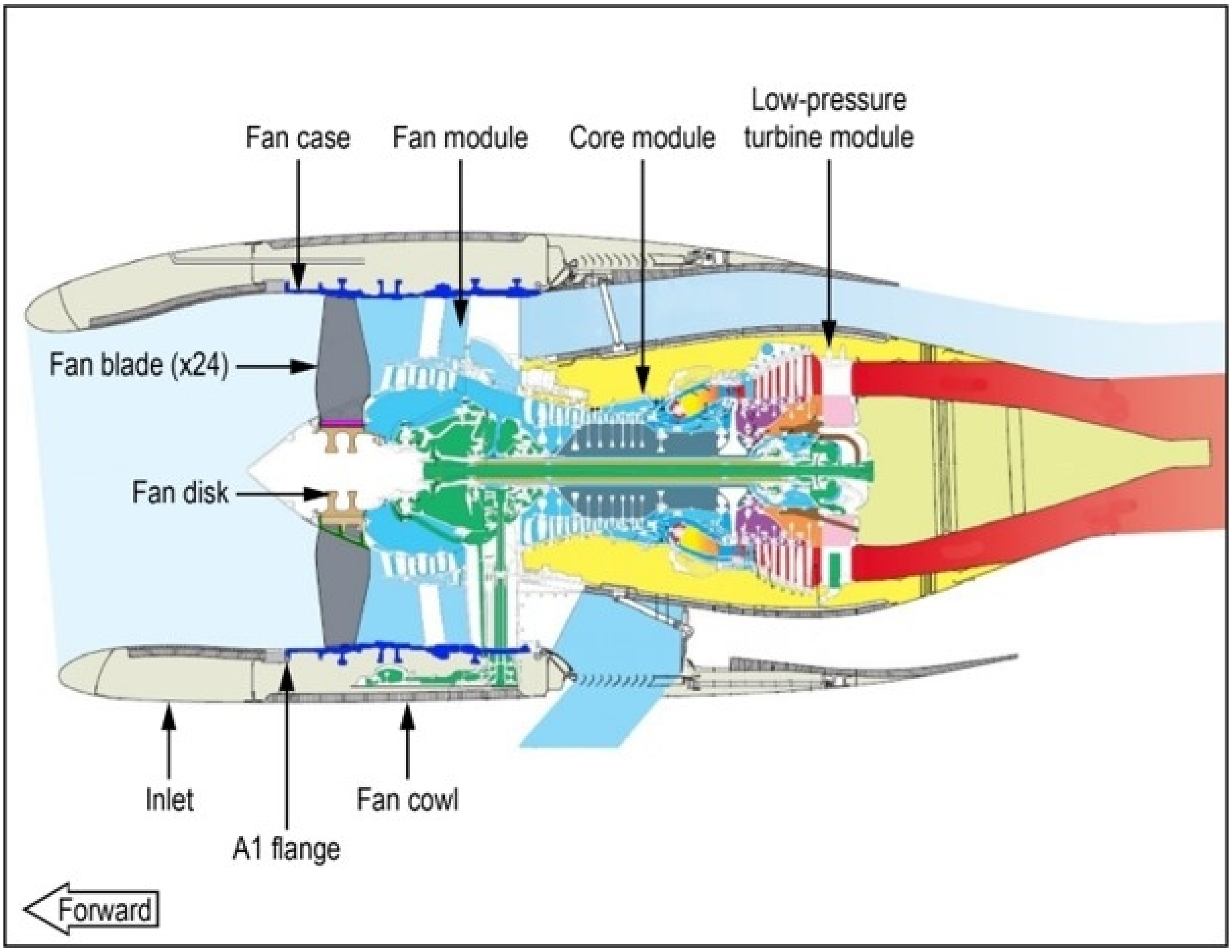
The fan disk and spacers are manufactured from a titanium alloy (Ti-6-4). Both shim configurations are manufactured from a nickel-chromium-iron alloy (alloy 718). The platforms are manufactured from an aluminum alloy. Figure 1 shows the CFM56-7B engine fan assembly.



Source: CFM

Figure 1. Fan assembly.

The fan frame assembly is the main forward support for the installation of the engine to the airframe and includes the fan frame, the fan case, and the fan outlet guide vanes. The fan case, which is made of an aluminum alloy, was designed to provide fan blade radial containment if an FBO event were to occur and transmit FBO loads to the fan frame and the inlet. Although the fan case provides the primary FBO radial containment protection, the inlet, which is attached to the fan case A1 flange, provides additional FBO protection. Figure 2 shows a cross-section of the engine and airframe components.



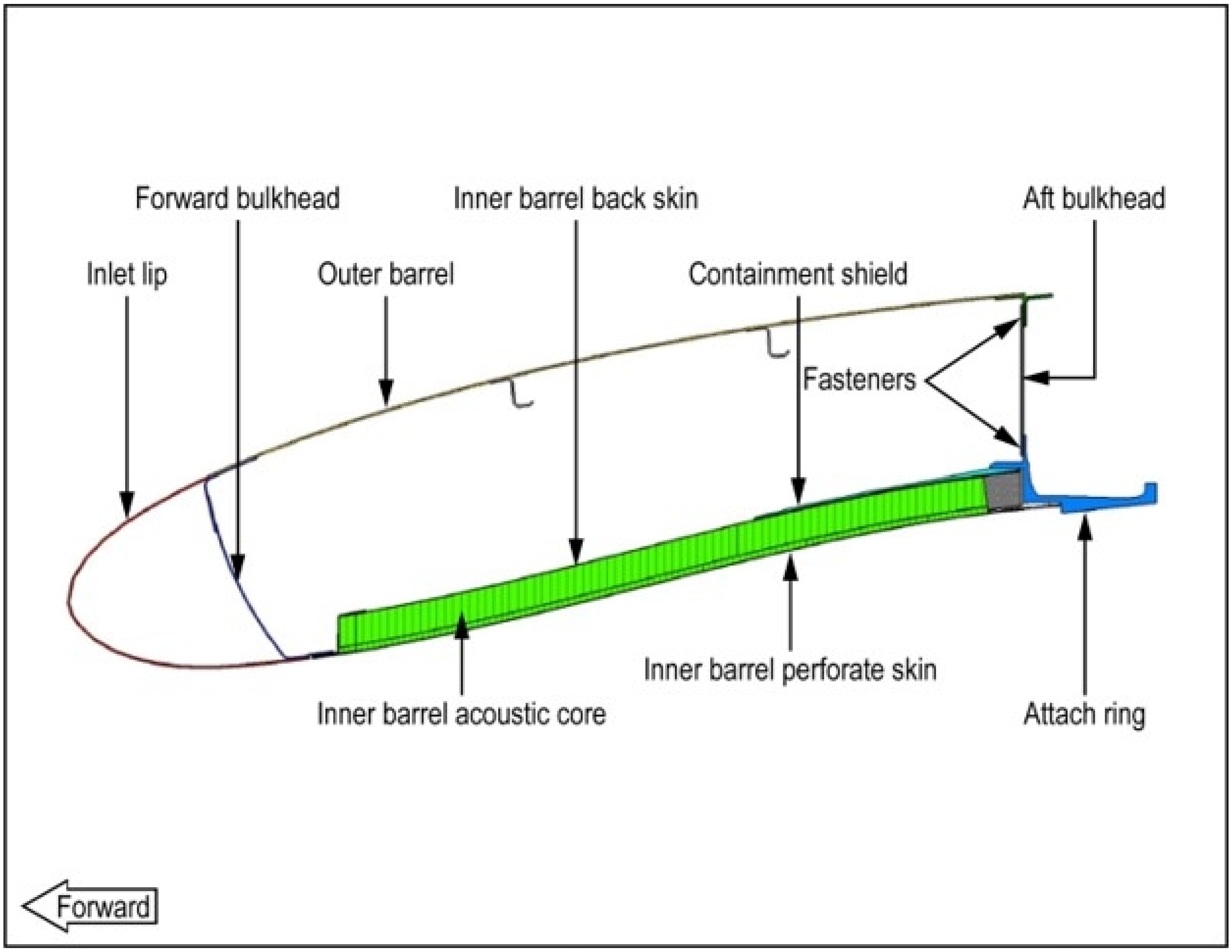
Source: CFM

Figure 2. Cross-section of engine and airframe components.

Inlet

The inlet is attached to the engine at the fan case A1 flange with 24 bolted assemblies (fasteners). The inlet directs smooth uninterrupted airflow into the engine fan and core sections and is an external aerodynamic pressure surface that directs the airflow over other nacelle components. Boeing created the design specifications for the inlet. Rohr Industries, which later became UTC Aerospace Systems (UTAS), was the inlet manufacturer. (In November 2018, UTAS and Rockwell Collins merged and became Collins Aerospace.)

The inlet consists of the inlet lip, inner and outer barrels, forward and aft bulkheads, and attach ring. The inlet lip is an aerodynamic surface to reduce airplane drag. The inner and outer aft ends of the inlet lip are attached to the forward bulkhead. The inner and outer barrels are concentric structures connected to the forward and aft bulkheads. The inner barrel consists of an acoustic honeycomb core; an aluminum containment shield (also referred to as a containment doubler), which is bonded over an area comprising about the aft 11 inches of the inner barrel; two bolted splice plates at the three and nine o'clock positions, which is where the two halves of the inner barrel join together; perforate skin; and back skin. The outer barrel comprises three panels with two support frames. The attach ring is located at the aft end of the inlet and is secured to the engine fan case A1 flange. The interface between the inlet attach ring and the engine fan case A1 flange is considered to be critical to the inlet's FBO capability because engine loads are reacted at that interface. The aft bulkhead and the inner barrel are also secured to the attach ring. Figure 3 shows a cross-section of the inlet.



Source: Boeing.

Figure 3. Inlet cross-section.

Note: The upper fasteners attach the aft bulkhead to the outer barrel. The lower fasteners attach the aft bulkhead to the attach ring, which is attached to the inner barrel and the engine fan case A1 flange. The fan case and the A1 flange are shown in figure 2.

The inlet lip, outer barrel, and attach ring are made of an aluminum alloy. The forward and aft bulkheads are made of a titanium alloy. The inner barrel is primarily made of an aluminum alloy. The 24 bolted assemblies that attach the inlet to the fan case include a crushable spacer that is designed to absorb energy and compress during an FBO event.

Aerodynamic and inertia loads on the outer barrel are transmitted to the inner barrel through the forward and aft bulkheads. Loads on the aft bulkhead are transmitted to the interface between the inlet attach ring and the fan case A1 flange.

Engine and Airframe Certification

The CFM56-7B engine was jointly certificated by the Federal Aviation Administration (FAA) and its counterpart in France, the Direction Générale de L'Aviation Civile (DGAC). The FAA issued the type certificate for the engine model on December 17, 1996, and the certification basis was 14 *CFR* Part 33, Airworthiness Standards: Aircraft Engines (amendment levels 33-1 through 33-15).

The DGAC certificated the engine in December 1996 under Certificat de Type Moteur M21, which was superseded by European Aviation Safety Agency (EASA) type certificate EASA.E.004 in 2006. (In 2004, EASA assumed responsibility for the certification of CFM engines.) There were no significant differences between the FAA and DGAC certification requirements, and the engine met all requirements of both agencies. Because the engine was dual certificated, the certification basis also included *Joint Aviation Requirements* JAR-E Change 8 (dated May 4, 1990).

To meet Part 33 requirements and obtain data for Boeing (as the airframe manufacturer) to use to meet certification requirements under 14 *CFR* Part 25, Airworthiness Standards: Transport Category Airplanes, CFM performed eight development FBO rig tests and two engine FBO containment certification tests. The purposes of the FBO tests were to (1) understand the fan blade fragmentation and kinematics (fragment energy levels and trajectories) after blade separation, (2) determine the fan case containment capability (radial containment), (3) define the loads and displacements from the initial impact and the resulting engine imbalance, (4) calculate the speed and weight of any ejected fragments (forward containment), and (5) demonstrate the proposed production hardware configuration.

Each rig test had a specific set of objectives and used various fan blade and fan case configurations. The first four FBO rig tests were designed to define the fan blade fragmentation and kinematics for fan case radial containment capability, fan blade axial retention, and fan blade interaction. The fourth FBO rig test included a full set of production-representative fan blades, a production-representative fan case, and a combination of actual and production-representative engine accessories. Also included in this test was a Boeing production-representative inlet, which had the same size, shape, and stiffness of the intended production inlet at that time. The production representative inlet included crushable spacers but did not include a containment shield.

After a fan blade release, fan blade airfoil and tip fragments generally travel in a forward spiral/helical pattern around the fan case and inlet. The forward movement is due to several factors, including the slope/conical nature of the fan case and inlet surfaces, the angle of attack of the blades, and aerodynamic loading in the forward direction. According to the results of FBO rig test 4, which was conducted in August 1995, the separated fan blade fractured into five pieces with the blade tip panel traveling forward of the fan case and into the inlet at an estimated 10° to 15° helix angle (where the fragment crossed the A1 flange). The blade tip panel then spiraled around the inlet and penetrated the inlet inner and outer barrels. The inner barrel penetration was about 26° forward of the fan blade rotational plane, which was beyond the ± 15° fragment spread angle (impact area) referenced in FAA Advisory Circular 20-128, "Design Considerations for Minimizing Hazards Caused by Uncontained Turbine Engine and Auxiliary Power Unit Rotor Failure," as a practical design consideration to minimize the hazards of uncontained fragments. The test also revealed a small penetration hole in the fan case. Further, the test revealed that the fan blade axial retention feature allowed some blades to slide forward in the disk, but the fan blades still remained engaged in the fan disk slots.

Given the results of the FBO testing, CFM redesigned the fan case to prevent through-hole penetrations and changed the fan blade axial retention feature. Also, Boeing instructed UTAS to design a containment shield that would prevent a fan blade fragment from penetrating the inlet structure.

CFM conducted an engine FBO containment certification test in April 1996. The purpose of the engine FBO containment certification test was to demonstrate that (1) the engine case would be capable of containing damage (no radial pass-through of engine debris into the structure) without catching fire and without failure of the engine mounting attachments when the engine was operated for at least 15 seconds after the FBO event and (2) the engine could be successfully shut down (or the damage could result in a self-shutdown of the engine). The test configuration included the redesigned fan case and a production-representative inlet with an integral containment shield. Although an inlet (an airframe part) was included in the test, the purpose of the test was to validate and certify engine hardware. A fan cowl was not included in the test configuration, per standard industry practice, to permit observation of the fan case and fan case-installed components during the test.

According to CFM, the test results showed that, after the initial fan blade release, the fan case withstood the initial FBO impact with no penetrations, the engine did not catch fire, the engine mounting hardware did not fail, and the engine was able to be shut down. The test results also showed that the inlet remained attached to the engine and that no fan blade fragments penetrated the containment shield. Further, after the initial fan blade release, five consecutive fan blades moved forward out of the fan disk, indicating that the blades' axial retention capability had failed. The fan case and the inlet were subsequently damaged by pass-through penetrations, and the inlet was further damaged due to missing inner and outer barrel material and the severing (360° circumferentially due to fastener shearing from the displacement wave) of the aft bulkhead-to-attach ring joint. (The displacement wave occurs when an engine fan blade impacts the engine fan case during an FBO event and the fan case deforms locally over a short period of time. This deformation travels both circumferentially and forward and aft.)

Boeing and UTAS determined that, even with the inlet damage, a redesign of the inlet was not necessary and additional inlet testing was not required because (1) the inlet remained attached to the engine; (2) the inlet withstood the initial FBO impact, the surge pressure loads, and the rundown loads; and (3) the containment shield performed appropriately during the initial fan blade release. CFM redesigned the fan blade axial retention feature based on the results of this certification test and conducted four additional rig tests to ensure that the redesigned feature would prevent fan blade axial movement.

CFM conducted a second engine FBO containment certification test in December 1996. Because the inlet was not part of the configuration for the second certification test, an aerodynamic bellmouth with a similar weight, stiffness, and flow path shape as a production-representative inlet was used for the test. The second engine FBO certification test revealed fan blade fragmentation that was similar to that observed during the previous certification test and demonstrated that the fan blade axial retention feature had been successfully redesigned. Thus, the CFM56-7B engine met the fan blade containment certification requirements under Part 33.

CFM provided Boeing with the engine FBO containment certification test results so that Boeing and UTAS could analyze, as part of the airplane's design and certification, how the inlet would respond to an FBO event. Also, CFM and Boeing jointly correlated the engine FBO containment certification test results and defined various FBO-related loads, including initial fan blade impact loads, surge loads, rundown loads, torque loads, imbalance loads, and windmilling loads, which were assessed as part of Boeing's and UTAS' analysis of the inlet. Boeing performed certification structural analyses based on the FBO test results and FBO loads and determined that the failure of an engine fan blade and any resulting damage should not cause the inlet or its components to separate from and present a hazard to the airplane. In December 1997, the Boeing 737-700 was certificated in accordance with the requirements of Part 25.

N766SW Engine and Inlet Information

The left engine on the accident airplane, which was installed in October 2007, had accumulated a total of 60,790 flight hours and 36,329 flight cycles. (A flight cycle is one complete takeoff and landing sequence.) The left engine's last shop visit (overhaul) occurred in August 2007 at the GE Celma facility in Petrópolis, Brazil. The left engine accumulated 30,237 hours of flight time and 17,956 flight cycles between the time of the last shop visit and the accident.

The fan blades in the accident engine were manufactured as part number 340-001-022-0. The accident fan blades were initially in a different engine that was installed new on a different SWA airplane on March 28, 1998. The engine was removed from the airplane on June 21, 2001, due to booster spool damage, and the engine was sent to the GE Celma facility for repair. The fan blade set was removed from the engine and was sent to the GKN Aerospace/Chem-Tronics facility in Tulsa, Oklahoma, for repair. At that time, the fan blades had accumulated 11,719 hours since new and 7,754 cycles since new. Because CFM56-7B fan blades were not certified as life-limited parts, fan blade times and cycles were not required to be tracked. (SWA started to track fan blade time and cycles in 2015.) The fan blades on the accident engine stayed together as a set, so the times and cycles since new were based on the number of hours and cycles that the blades were installed in an engine.

While at the GKN Aerospace/Chem-Tronics facility, the fan blades were reworked according to CFM Service Bulletin (SB) 72-0253 (which was initially issued in July 2000). The fan blade rework included machining of the dovetail profiles to prevent spalling (flaking of material) and wear damage of the blade coating at the forward and aft ends of the contact faces. The fan blades' part number was changed to 340-001-027-0 to indicate that the rework had been performed.

The fan blade set was then installed in the accident engine, which was installed on another SWA airplane on July 27, 2001. The accident engine was installed on N766SW in the left position on August 21, 2006, and remained in that position until June 29, 2007, when it was removed from N766SW due to a compressor stall and damage to the compressor blades. The engine was sent to the GE Celma engine shop for repair. The fan blades were removed and sent to the GKN Aerospace/Chem-Tronics facility in El Cajon, California, for repair. At that time, the fan blades had accumulated 32,614 hours and 20,196 cycles since new. After the fan blades were repaired, they were sent back to the GE Celma facility and were installed in a different configuration fan disk (which required shims and modified spacers). The fan disk was then installed on the accident engine, which was reinstalled in the left position on the accident airplane on October 17, 2007.

The National Transportation Safety Board (NTSB) reviewed the records for the 2001 and 2007 fan blade repairs and found that all 24 fan blades in the set were similarly processed and repaired. The review of the records also found no evidence indicating that the repairs were improperly performed.

The accident engine remained in the left position on the accident airplane through the date of the accident (August 27, 2016). At that time, the fan blades had accumulated 62,851 hours and 38,152 cycles since new.

To maintain the fan blade loads within the predicted range and prevent wear on the fan disk and the fan blade dovetail coating, CFM recommended repetitive on-wing relubrications of the dovetails. As part of the relubrications, the fan blades are visually inspected to assess the blade coating condition and identify any airfoil damage and crack indications. At the time of the accident, the CFM-recommended fan blade dovetail relubrication interval was every 3,000 cycles or 5,000 hours, whichever came first. Between October 2007 (when the fan blades and fan disk were installed on the accident airplane) and August 2016 (when the accident occurred), the fan blade set was relubricated 13 times, the last of which occurred in November 2015. Between that time and the time of the accident, the fan blades had accumulated 2,397 hours and 1,396 cycles.

The shims for fan blade Nos. 22 through 24 were replaced during the last and/or the second-to-last blade relubrications before the accident, which occurred 1,396 and 2,689 cycles, respectively, before the accident flight. SWA maintenance records for the accident inlet (since its installation on the accident airplane in 2007) showed two discrepancies: a lightning strike in February 2008 and a missing sealant on the inlet lip in December 2015. For both of these discrepancies, the damaged area was repaired in accordance with Boeing's structural repair manual.

FLIGHT RECORDERS

The airplane was equipped with a Honeywell 6022 solid-state CVR. The CVR contains a two-channel digital audio recording of the last 2 hours of operation. One channel combines three audio panel sources—the captain, the first officer, and the observer—and the other channel is the cockpit area microphone. The CVR also contains a three-channel digital audio recording of the last 30 minutes of operation. The three channels are the individual audio panels for the captain, the first officer, and the observer. Each of the CVR channels contained either excellent- or good-quality audio information, and the data were extracted normally. A CVR summary of events was prepared. The summary began at 0903:21, when the left engine was started, and ended at 0946:20, when the right engine was shut down.

The airplane was also equipped with a Honeywell 4700 solid-state FDR. The recorder was in good condition, and the data were extracted normally. About 27 hours of operational data were retained on the recording medium, including about 25 minutes of data from the accident flight, from about 0904:34 (before takeoff) to 0929:37 (when the airplane was at an altitude of about 17,400 ft). About 13 minutes 23 seconds after takeoff (0921:46), a 0.5-second gap occurred within one subframe (1 second) of data. This gap was assumed to be associated with the engine failure. The recording ended 21 minutes 15 seconds after takeoff due to a loss of electrical power to the FDR.

The NTSB conducted postaccident on-wing airplane electrical checks to identify possible components that could have caused or contributed to the power loss to the FDR. The test scenarios were unable to determine an in-flight electrical anomaly that would have caused the FDR to lose power. Subsequent off-wing component tests were also unable to identify a specific part that could have contributed to the electrical power loss. The components that were removed for testing were replaced, and the test scenarios were repeated, but these tests were also unable to determine an in-flight electrical anomaly that would have caused the FDR to lose power. On February 28, 2017 (after the airplane had been returned to service), SWA performed a scheduled FDR download and parameter validation test, and no faults were found.

POSTACCIDENT EXAMINATIONS

Airplane

Postaccident examination of the airplane revealed a 16-by-5-inch through-puncture on the left side of the fuselage, as shown in figure 4. The through-puncture was just below the cabin windows by rows 11 and 12, as indicated by the yellow tape in figure 5, with no corresponding damage on the interior cabin sidewall. Numerous impacts were observed along the left side of the fuselage, the left wing, and the left horizontal stabilizer, and the left winglet front spar was partially severed.

Figure 4. Close-up view of through-puncture on left side of fuselage.

Figure 5. Location of fuselage skin puncture (as indicated by yellow tape).

Almost all of the left engine inlet was missing, including the outer barrel, the inlet lip, the forward bulkhead, and the inner barrel forward of the containment shield. The part of the inlet that remained was severely damaged, as shown in figure 6. The aft bulkhead remained in place, but rivets connecting it to the attach ring were sheared 360° around the ring. A 360° section of the inlet inner barrel forward of the containment shield separated from the rest of the inlet about 17 inches forward of the A1 flange. Although both inner barrel splice plates (located at the three and nine o'clock positions) remained intact, the splice plate at the nine o'clock position exhibited impact damage and outward deformation. The inner barrel honeycomb core and the inner skin that remained were torn and flared outward. Gaps between the inlet attach ring and the fan case A1 flange were observed at several locations around the circumference of the interface between the two parts.

Figure 6. Inlet damage.

The part of the inlet that remained attached to the airplane was sent to the Boeing Equipment Quality Analysis laboratory in Renton, Washington, for an initial visual examination. Afterward, the inlet was sent to the UTAS facility in Chula Vista, California, to assess the inlet damage and determine the fan blade exit trajectories (see the Tests and Research section).

The left winglet front spar and part of the left wing that exhibited impact damage (a piece of the leading edge No. 1 slat) were sent to the NTSB's Materials Laboratory in Washington, DC, for examination to determine if fan blade fragments had impacted those airplane structures. The structures were examined using a scanning electron microscope (SEM) and energy dispersive x-ray spectroscopy (EDS). The surfaces adjacent to the fracture on the left winglet front spar were generally deformed outboard and aft. Portions of the surfaces adjacent to the fracture had features consistent with sliding contact with another object. EDS spectrums within the sliding contact areas revealed elements consistent with titanium. (The fan blades were made of a titanium alloy, and titanium is not an element of the winglet front spar material.)

Outside of the sliding contact area, no elements consistent with titanium were observed. The left wing leading edge No. 1 slat exhibited sliding contact marks with an associated inward deformation. The spectrums within and outside of the sliding contact area had no elements that were consistent with titanium.

Engine

All of the fan blades were found in their as-installed (full length) positions in the fan disk except for fan blade No. 23, which had fractured at the blade root with the dovetail remaining in the fan disk. The blade dovetail is shown in figure 7; the blade airfoil was missing and was not subsequently recovered. All of the intact fan blades exhibited hard-body impact damage, tears, missing material, and airfoil curling or distortion. Most of the blade platform for fan blade No. 22 (which was adjacent to fan blade No. 23 in the counterclockwise direction) was missing.

Figure 7. Fan blade No. 23 fracture surface.

The fan case had no through-hole penetrations and showed no evidence of an uncontainment. Circumferential scoring and rubbing were noted along almost the entire fan blade rotational plane, and several hard impact marks were noted along and aft of the fan blade rotational plane.

The 23 intact fan blades and the dovetail of fan blade No. 23 were sent to the NTSB Materials Laboratory for examination. The fan disk and fan case were sent to the Safran Villaroche facility in Réau/Moissy-Cramayel, France, to evaluate the (1) impact locations on the fan case, (2) potential fan blade fragmentation, (3) exit trajectories (the forward exit angles), and (4) orientation of the fan blade fragments as they traveled across the fan case A1 flange. The results of the NTSB's examination and CFM's evaluation are discussed in the Tests and Research section.

For fan blade No. 23, chipped and missing coating on the convex side of the dovetail was observed. Dimensional measurements showed that the coating damage was centered about 2 inches aft of the leading edge of the dovetail and spanned about 1.3 to 2.7 inches from the leading edge. According to Boeing's *737-600/700/800/900 Aircraft Maintenance Manual*, the coating on the dovetail is to be visually inspected for any missing and/or chipped coating in the contact area on both the concave and convex sides of the blades. This visual inspection is typically performed during relubrications of the dovetail. If a coating loss is found during the inspection, the affected fan blades are either removed from service or allowed to continue in service with a restricted inspection interval; this determination depends on the amount and location of the coating loss. The loss of coating on fan blade No. 23 at the last relubrication was such that the blade could continue in service for 50 flight cycles or 75 flight hours, whichever came first, before the blade needed to be removed from service for overhaul. The blade remained in service.

NTSB Metallurgical Examination

The fan blades from the accident engine and their associated hardware—shims, spacers, and platforms—were examined by the NTSB's Materials Laboratory. All of the components, including the fracture surface on the dovetail of fan blade No. 23, were examined visually and under optical magnification.

Part of the fracture surface on fan blade No. 23 was relatively smooth with well-defined curving crack arrest lines, which were consistent with fatigue, as shown in figure 8. (Crack arrest lines generally represent changes in the stress state, environment, or time interval associated with fatigue crack growth.) Fracture features emanated from an origin area centered about 2.1 inches from the leading edge face on the convex side of the blade. Ratchet marks (which are formed when two adjacent fatigue cracks originate on slightly offset planes) were observed in the origin area, consistent with multiple fatigue origins. The fracture had initiated at the contact face just inboard of the outboard edge (relative to the fan disk rotational axis) of the dovetail coating. The coating was missing and spalled in several locations on the convex side of the blade, including an area adjacent to the fracture and a larger area directly inboard from the fracture origin.

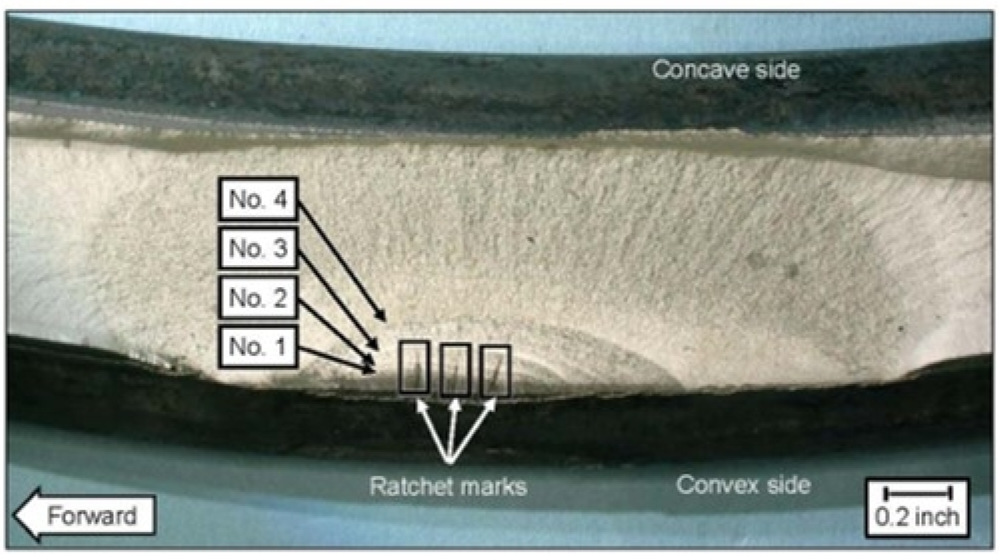


Figure 8. Fatigue region and crack arrest lines.

The progressive crack growth region extended up to 2.277 inches axially and was 0.695 inch deep. Beyond the boundary of the progressive crack growth region, the fracture surface had a lighter gray appearance with prominent shear lips, which was consistent with ductile overstress fracture.

The dovetails for the intact fan blades (Nos. 1 through 22 and 24) were examined visually and with an optical stereomicroscope with the dovetail coating in place. No cracks were detected using these inspection methods. Nine of the 23 blades (including blade No. 24) had areas of missing coating.

The shims for nine of the fan blades, including blade Nos. 23 and 24, underwent a visual examination of the interior contact faces. The interior surfaces of the shims for fan blades Nos. 23 and 24 appeared as a mirror image of the blade dovetail contact face, with contact marks in areas where the coating was intact and no contact marks in areas where the coating was missing and spalled.

<u>Additional examinations of fan blade No. 23</u>

The fan blade No. 23 fracture surface was further examined using a SEM. The four crack arrest lines found during the visual and optical examination (shown in figure 8) were observed on the fracture surface. The depths of crack arrest lines Nos. 1 through 4 were 0.072 inch, 0.093 inch, 0.139 inch, and 0.228 inch, respectively. A possible crack arrest line was observed between arrest lines Nos. 3 and 4, especially near the aft side of the fatigue region, but there was no distinct boundary around the entire front of the crack.

At higher SEM magnifications, striations consistent with low-cycle fatigue crack growth were observed. (Striations are linear features on a fatigue fracture surface that indicate how far a crack advanced with each stress cycle.) The striation spacing generally became larger as the depth of the crack increased. Immediately beyond the crack arrest lines, the striation spacing changed abruptly; the spacing between striations was shorter than the spacing before the arrest lines, which was consistent with a temporary reduction in crack growth rate after each arrest line had formed.

Fan blades Nos. 23 and 24 had similar damage to their dovetail coating, and both blades were from the same forging lot. A metallographic examination of fan blade No. 24 identified a primary crack and secondary cracks that emanated from the surface of the blade under the dovetail coating near the outboard end of the contact face on the convex side, which was comparable with the origin area for the fracture on fan blade No. 23. Fractographic examinations of fan blade Nos. 23 and 24 showed that the estimated total number of flight cycles associated with crack growth from a depth of 0.0005 inch to failure was 15,000 cycles for fan blade No. 23 and 16,000 cycles from a depth of 0.0005 inch to the maximum crack depth for fan blade No. 24. Thus, given that the fan blade set had accumulated 17,956 cycles since the last overhaul, the crack on fan blade No. 23 initiated about 3,000 cycles after the last overhaul.

For fan blade No. 23, a metallographic examination was also conducted using transverse cross-section samples from the aft end of the dovetail. The cross-section samples were polished and etched to reveal the blade microstructure, which was consistent with the expected microstructure for the specified material. No material or manufacturing anomalies were observed on the surfaces.

<u>Inspections and examinations of intact fan blades</u>

No cracks were detected by magnified visual inspections of the intact blades. Three other nondestructive inspections—fluorescent penetrant inspections (FPI), eddy current inspections (ECI), and ultrasonic inspections—were conducted on various dovetails of the intact blades from the accident engine to determine if cracks existed on the convex side near the leading edge, which is where the highest fan blade operational stress is located. Ultrasonic inspections were conducted before and after the dovetail coating on the blades was removed. FPIs and ECIs were performed with the dovetail coating removed.

Fan blade Nos. 11 and 17, with their dovetail coating removed, had an ultrasonic signal response that exceeded the threshold value. Fan blade No. 24 did not undergo an ultrasonic inspection with the dovetail coating removed because the detected crack (which was found during metallographic inspection of the blade) could be examined without removing the coating, but the signal response from blade No. 24 with its dovetail coating was comparable to that of blade No. 11 before the dovetail coating was removed.

In addition to fan blade No. 24, cracks were found in five other intact blades in the fan blade set (Nos. 1, 2, 11, 17, and 19) using one or more nondestructive inspection methods. Cracks in these six blades were opened through laboratory fracture, and each crack fracture surface was examined using a SEM. These examinations found that the crack surfaces were relatively flat with curving crack arrest lines and boundaries, which were consistent with fatigue. All of the cracks had initiated on the contact face of the dovetail under the dovetail coating on the convex side of the blade about 2 inches aft of the forward end of the blade; this location was comparable with the fracture on fan blade No. 23. Table 1 summarizes the length and depth of the fan blade cracks from the accident engine. Each crack showed striations consistent with low-cycle fatigue, and relatively fine striations were observed near the crack origins. The fine striations generally decreased in density as the crack depths increased. The estimated flight cycles accumulated during crack growth (also shown in table 1) ranged from about 4,000 cycles for blade Nos. 1, 2, and 19 to about 16,000 cycles for blade No. 24.

Table 1. Fan blade cracks on accident engine.

Nondestructive inspections of the remaining 17 fan blades in the set found no crack indications. These results were verified through SEM examination of 6 of the 17 blades. During the SEM examination, microcracks were detected on these six blades.

Residual stress measurements

Residual stress measurements on nine fan blades were conducted at Lambda Technologies in Cincinnati, Ohio, and at the Safran Materials and Processes Laboratory in Evry Corbeil and Vernon, France. Lambda Technologies measured residual stresses on fan blade Nos. 3, 7, 8, 17, and 23 parallel to the radial direction, and Safran measured residual stresses on fan blade Nos. 5, 6, 19, and 24 parallel to the longitudinal axis.

Residual stress measurements were taken from multiple areas of each dovetail surface, and the measurements were compared with reference residual stress depth profile data, provided by Safran, for fan blades that were shot-peened in accordance with specified parameters. All of the fan blades, including those with no identified cracks (Nos. 3, 5, and 6 through 8) had abnormal residual stress profiles compared with the reference profile data.

Coating thickness measurements

Coating thickness variations were observed on some fan blade dovetails, including cases in which the thickness in the crack initiation area exceeded the maximum allowable thickness in the coating repair process specification. When local coating overthickness (excess coating) was considered in the stress analysis of the blade root, the calculated static stress of the dovetail increased in the crack initiation area.

Fan Blade Impact Damage and Exit Trajectories

CFM performed a detailed examination of the fan case to assess the impact damage. The greatest amount of deformation on the A1 flange was observed along the four-to-ten o'clock arc. Rub marks were noted along the entire circumference in line with the fan blade rotational plane except for the area between the three and five o'clock positions. CFM determined that, after fan blade No. 23 released and impacted the fan case, the blade fractured into three (or possibly four) pieces, which created the four major and distinct fan case impact marks shown in figure 9 and several different exit paths and trajectories.

Table 1. Fan blade cracks on accident engine.

Blade number	Maximum crack depth (inch)	Maximum crack length (inch)	Number of cycles of crack growth
1	0.009	0.057	~4,000
2	0.015	0.046	~4,000
11	~0.060	~0.640	~10,000
17	0.023	>0.300	~5,000
19	0.019	0.190	~4,000
23	0.695	2.277	~15,000
24	0.126	~0.447	~16,000

Nondestructive inspections of the remaining 17 fan blades in the set found no crack indications. These results were verified through SEM examination of 6 of the 17 blades. During the SEM examination, microcracks were detected on these six blades.

Residual stress measurements

Residual stress measurements on nine fan blades were conducted at Lambda Technologies in Cincinnati, Ohio, and at the Safran Materials and Processes Laboratory in Evry Corbeil and Vernon, France. Lambda Technologies measured residual stresses on fan blade Nos. 3, 7, 8, 17, and 23 parallel to the radial direction, and Safran measured residual stresses on fan blade Nos. 5, 6, 19, and 24 parallel to the longitudinal axis.

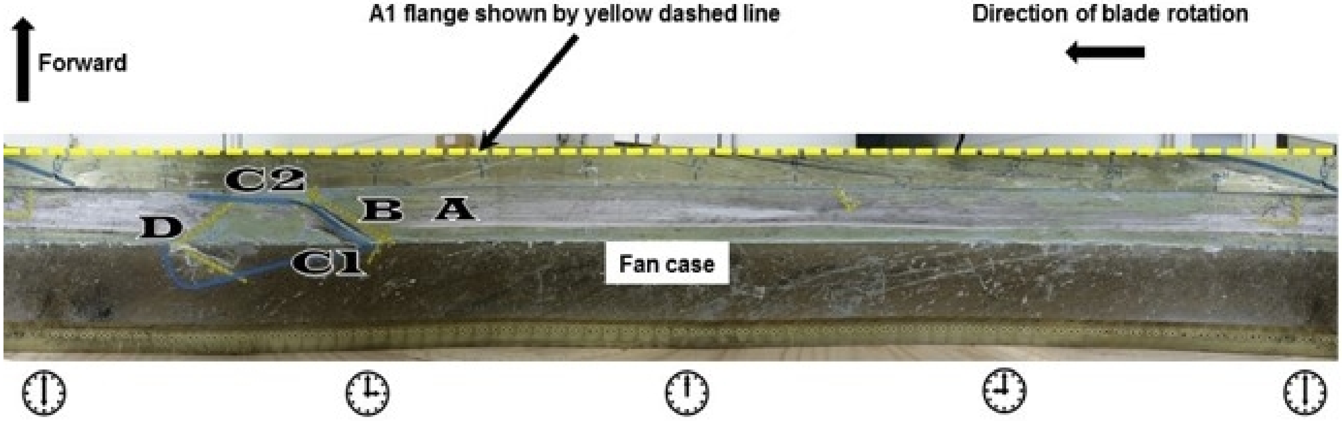
Residual stress measurements were taken from multiple areas of each dovetail surface, and the measurements were compared with reference residual stress depth profile data, provided by Safran, for fan blades that were shot-peened in accordance with specified parameters. All of the fan blades, including those with no identified cracks (Nos. 3, 5, and 6 through 8) had abnormal residual stress profiles compared with the reference profile data.

Coating thickness measurements

Coating thickness variations were observed on some fan blade dovetails, including cases in which the thickness in the crack initiation area exceeded the maximum allowable thickness in the coating repair process specification. When local coating overthickness (excess coating) was considered in the stress analysis of the blade root, the calculated static stress of the dovetail increased in the crack initiation area.

Fan Blade Impact Damage and Exit Trajectories

CFM performed a detailed examination of the fan case to assess the impact damage. The greatest amount of deformation on the A1 flange was observed along the four-to-ten o'clock arc. Rub marks were noted along the entire circumference in line with the fan blade rotational plane except for the area between the three and five o'clock positions. CFM determined that, after fan blade No. 23 released and impacted the fan case, the blade fractured into three (or possibly four) pieces, which created the four major and distinct fan case impact marks shown in figure 9 and several different exit paths and trajectories.



Source: CFM

Figure 9. Fan case impact damage.

The first fan case impact, labeled impact mark A in figure 9, was a shallow blade tip scuff mark located near the two o'clock position. According to CFM, this initial fan blade impact caused the blade tip to bend, snap off, and travel forward and out of the engine. CFM considered this initial impact to be a "low energy" event due to the small radial clearance between the blade tip and fan case (resulting in a low radial release velocity) and the lack of a deep gouge or any related distortion of the fan case in this area.

According to CFM, after the blade tip fractured, the remaining fan blade continued to travel and created two "high energy" crescent-shaped impact marks (labeled B and C1) located between the three and four o'clock positions on the fan case. The impact marks coincided with observed fan case distortion and tears and was near the largest A1 flange distortion. CFM determined that the blade airfoil fractured at these locations, indicating that the impact was created by a different fan blade fracture surface. CFM also determined that the deep scuffing damage by impact mark C1 (labeled C2) could have been the result of an additional impact and airfoil fracture.

Another "high energy" impact mark (labeled D) was located at the five o'clock position on the fan case. This impact mark appeared to have the outline/imprint of the blade root trailing edge (profile view). The impact mark coincided with the observed fan case distortion and tears.

CFM determined that the impact marks on the accident fan case had similar markings (in terms of size, shape, and location) as those observed during the engine FBO containment certification tests. As a result, CFM determined that this FBO event was consistent (in terms of blade fragmentation) with the engine FBO containment certification tests except in two areas. First, even though the inlet was damaged during the engine FBO containment certification tests, the inlet remained attached, but a large portion of the inlet separated from the engine during the accident sequence. Second, during the engine FBO containment certification tests, the only fan blade parts that exited forward of the fan case were the fan blade tip and mid-section pieces, but, during this FBO event, the entire airfoil of the blade, including the root, most likely traveled forward of the fan case (as indicated by impact marks and other damage on the accident fan case and inlet as well as trajectories that were consistent with the blade root exiting forward). No fragments from the airfoil or root outboard of the dovetail were recovered.

After documenting the fan case impact marks, CFM determined the trajectories of the various fan blade fragments by noting, around the circumference of the fan case and toward the A1 flange, spiraling scuff, scrape, and impact marks that were similar in size and shape as those documented during the FBO rig and engine FBO containment certification tests. Although no fragments from the separated fan blade No. 23 airfoil were recovered, CFM determined that, given the damage and impact marks to the fan case, the fan blade fragmentation during this event was likely similar to that observed during the engine FBO containment certification tests.

CFM estimated that the blade tip exited the fan case as it crossed the A1 flange along a trajectory of about 14°, the blade root exited along a trajectory of between 7° and 9°, and one of the mid-span fragments exited along a trajectory of between 6° and 8°. Additional marks were observed around the A1 flange between the four and five o'clock positions; CFM attributed these marks to other mid-span fragments. These fragments created deep marks along a trajectory of about 37° as they crossed over the A1 flange. This mid-span trajectory was in proximity to the tip panel as it crossed over the A1 flange.

Inlet Damage Assessment and Fan Blade Fragment Exit Trajectories

UTAS documented the damage to the inlet and found that the remaining inner barrel exhibited the following damage: (1) Ti-6-4 transfer consistent with fan blade interaction was observed on the forward edge of the containment shield, the back skin, and the acoustic core; (2) the perforate skin within the inner barrel containment shield was damaged almost 360° circumferentially; (3) the back skin was completely separated forward of the containment shield; (4) the fracture surfaces showed evidence of tension, shear, or combined tension and shear overload; (5) the containment shield had minor impact damage near the six o'clock position; and (6) no titanium particles were found on the back skin fracture surface. No evidence of fatigue was found on the inner barrel back skin or acoustic core.

The available evidence did not specifically indicate whether the source of some inlet damage, including the damage to the inner barrel back skin, was caused by forward-traveling fan blade fragments or displacement wave loads. Thus, to better assess the amount of inner barrel back skin damage, estimates of both the minimum and maximum inner barrel back skin damage were developed (instead of a single damage estimate), and both estimates were expressed as the number of degrees around the circumference of the inlet (arc length).

The amount of inner barrel back skin damage (before portions of the inner barrel separated from the inlet) was determined by an evaluation of the fan blade fragment trajectories, a metallurgical analysis of the inner barrel back skin, and the presence of fan blade debris damage to the inner barrel. The inner barrel back skin damage (forward of the containment shield) that could be positively identified (either through visual observations or metallurgical analysis) as having been caused by forward-traveling fan blade fragments had an estimated arc length that ranged from 112.5° to 210° circumferentially. This amount of damage was more than the 100° maximum inner barrel back skin damage estimate that Boeing used to model the inlet to withstand various loads (blade impact loads, surge loads, rundown loads, torque loads, imbalance loads, and windmilling loads) so that the inlet would remain attached to the engine.

Similar to CFM, UTAS conducted a trajectory analysis using the impact damage to the fan case and inlet as well as the scrape and scuff marks. UTAS' exit trajectory estimates for the blade root and the mid-span fragments crossing the A1 flange were consistent with CFM's estimates. UTAS estimated that the fan blade tip crossed the A1 flange at a trajectory of 30°, whereas CFM estimated that the fan blade tip crossed the A1 flange at a trajectory of 14°.

UTAS estimated that the blade tip entered the inlet near the four o'clock position because of a deep gouge in the inlet attach ring that was consistent with CFM's estimated location where the blade tip exited the fan case. The blade tip continued into the inlet, penetrated the inner barrel perforate skin between the five and six o'clock positions, and came to rest in the inner barrel acoustic core between the six and seven o'clock positions. The containment shield prevented the blade tip from exiting the inlet.

UTAS estimated that the mid-span fragments entered the inlet near the eight o'clock position, slid circumferentially along the aft flange, crushed (radially) the acoustic core, struck and flattened the T12 temperature sensor (which was attached to the inlet inner barrel), penetrated the perforate skin between the three and seven o'clock positions, and sheared the inner barrel skin at the forward edge of the containment shield from about the three to seven o'clock positions. After impacting the T12 temperature probe, some of the mid-span fragments were deflected forward, and others were deflected aft.

UTAS estimated that the blade root entered the inlet near the eleven o'clock position and continued to slide around the inlet attach flange, making more than a complete revolution. UTAS determined that, when the blade tip was momentarily stopped in the acoustic core between the six and seven o'clock positions, the blade root struck the blade tip, creating additional blade fragments. UTAS found that these additional blade fragments slid across and penetrated the perforate skin; penetrated the acoustic liner; and sheared the inner barrel in two separate locations, between the seven and nine o'clock positions and between the nine and eleven o'clock positions. The remaining original blade root continued to slide and traveled around the inlet almost three-quarters of another revolution, at which point the blade root sheared the inner barrel between the twelve and three o'clock positions. Thus, according to this assessment, the fan blade root would have made almost two complete revolutions within the fan case and the inlet before the inlet failed and portions of the inlet departed the airplane.

SURVIVAL ASPECTS

A supplemental oxygen system was located above the flight attendant jumpseats, in the passenger service unit above each of the 48 seat sets on the airplane, and in service units in the forward and aft lavatories. The system was designed to provide oxygen to flight attendants and passengers, either manually or automatically, if a depressurization event occurred. (The flight crew received supplemental oxygen from a different source.) Each row of passenger seats (to the left and right of the aisle) had a total of eight masks and four oxygen generators.

The passenger and flight attendant oxygen masks, as well as those in the lavatories, were found deployed. The passenger oxygen activation toggle switch in the cockpit was gated and safety wired, which was consistent with the supplemental oxygen system being automatically (and not manually) activated.

Postaccident Actions—Flight 3472 (PNS)

After this accident, CFM reevaluated the fan blade dovetail stresses and found that the fatigue cracks initiated in an area of high stress on the dovetail and that the dovetail was experiencing peak stresses that were higher than originally predicted. CFM also found that the higher operational stresses resulted from coating spalling, higher friction levels when operated without lubrication or a shim, variations in coating thickness, higher dovetail edge loading (at the dovetail contact surface and fan disk interface) than predicted, and a loss or relaxation of compressive residual stress.

At the time of the fan blade set's last overhaul in August 2007, the overhaul process included an FPI (as specified in the CFM engine shop manual) to detect cracks. After this accident, CFM developed an ECI procedure to be performed on the dovetail at overhaul (in addition to the FPI). The ECI involves a manual scan of both the concave and convex sides of the dovetail with the blade coating removed. An ECI has a higher sensitivity than an FPI and can detect cracks at or near the surface (unlike an FPI, which can only detect surface cracks). In November 2016, this requirement was added to CFM's engine shop manual.

Also, the cracks in the six intact fan blades (Nos. 1, 2, 11, 17, 19, and 24) were discovered only after the dovetail coating had been removed. Because these cracks could not be seen through the dovetail coating, CFM developed an ultrasonic inspection technique that could be performed with the coating still on the dovetail. In March 2017, CFM implemented the first of several ultrasonic inspection requirements to be performed as part of the on-wing fan blade inspection at the time of fan blade relubrication.

Southwest Airlines Flight 1380 Accident

On April 17, 2018, SWA flight 1380, a Boeing 737-700 equipped with CFM56-7B engines, experienced a left engine failure while climbing through flight level 320 en route to the flight's assigned cruise altitude. The flight had departed from LaGuardia Airport, Queens, New York, about 30 minutes earlier. As a result of the engine failure, the flight crew conducted an emergency descent and diverted to Philadelphia International Airport (PHL), Philadelphia, Pennsylvania. Portions of the left engine inlet and fan cowl separated from the airplane, and fragments from the inlet and fan cowl struck the left wing, the left-side fuselage, and the left horizontal stabilizer. One fan cowl fragment impacted the left-side fuselage near a cabin window, and the window departed the airplane, which resulted in a rapid depressurization. The airplane landed safely at PHL about 17 minutes after the engine failure occurred. Of the 144 passengers and 5 crewmembers aboard the airplane, 1 passenger received fatal injuries, and 8 passengers received minor injuries. The airplane was substantially damaged. The regularly scheduled domestic passenger flight was operating under the provisions of 14 *CFR* Part 121 with a destination of Dallas Love Field, Dallas, Texas.

The left engine failure occurred when one of the fan blades fractured at its root as a result of a low-cycle fatigue crack that initiated in the dovetail. At the time of failure, the fan blade had accumulated 55,471 hours and 32,636 cycles since new. The separated fan blade impacted the engine fan case and fractured into multiple fragments. Some of the fan blade fragments traveled forward of the engine and into the inlet. In addition, the fan blade's impact with the fan case caused the fan case to deform locally over a short period of time. This deformation traveled both around and forward/aft of the fan case. The forward-traveling fan blade fragments and the deformation compromised the structural integrity of the inlet, causing portions of the inlet to depart the airplane.

The impact of the separated fan blade with the fan case also imparted significant loads into the fan cowl through the radial restraint fitting, which was located at the bottom of the inboard fan cowl. These loads caused cracks to form in the fan cowl skin and frames near the radial restraint fitting. This damage then propagated forward and aft, severing the three latch assemblies that joined the inboard and outboard halves of the fan cowl, which caused large portions of both fan cowl halves to separate and depart the airplane. For more information about this accident, including the safety recommendations that resulted from the investigation, see case number DCA18MA142 at the NTSB's website.

Postaccident Actions—Flight 1380 (PHL)

On April 20, 2018 (3 days after the PHL accident), CFM issued SB 72-1033, "Fan Blade Dovetail Repetitive Ultrasonic Inspection," to recommend initial and repetitive ultrasonic inspections for all CFM56-7B engine fan blades based on an engine's cycles since new. This SB established specific initial and repetitive inspection intervals and was revised three times (in May, July, and November 2018) to refine the inspection criteria and intervals. The most recent revision indicated that, for fan blades with more than 16,000 cycles since new, the initial dovetail ultrasonic inspection should be performed within 1,000 flight cycles or before 20,000 cycles since new and that, for all other fan blades, the inspection should be performed before 17,000 cycles since new. The most recent revision also repeated the recommendation (introduced in the July 2018 revision) that the dovetail ultrasonic inspection be repeated every 1,600 flight cycles after the initial inspection.

Also on April 20, 2018, the FAA and EASA issued emergency airworthiness directives (AD). FAA emergency AD 2018-09-51 required that, within 20 days (May 10, 2018) of the AD's issuance, all CFM56-7B fan blades with 30,000 or more cycles since new undergo an ultrasonic inspection for cracks. The AD also required that any fan blade with indications of a crack be removed from service before further flight. The FAA superseded the emergency AD with ADs 2018-09-10 (early May 2018), 2018-10-11 (mid-May 2018), 2018-18-01 (October 2018), and 2018-26-01 (December 2018) to refine the inspection criteria and intervals based on the recommendations provided in CFM SB 72-1033.

EASA emergency AD 2018-0093-E required an initial ultrasonic inspection of all CFM56-7B engine fan blades. Similar to the requirements of the FAA's emergency AD, EASA's emergency AD required compliance within 20 days (May 10, 2018) of the AD's issue date for engines with 30,000 or more flight cycles. The AD also addressed the required compliance interval for engines with fewer than 30,000 flight cycles. The AD required repetitive ultrasonic inspections within 3,000 flight cycles. Fan blades with discrepancies were required to be replaced before the next flight or before the engine's return to service. EASA superseded the emergency AD with ADs 2018-0109 (May 2018), 2018-0211 (September 2018), and 2019-0018 (January 2019) to refine the inspection criteria and intervals based on the recommendations provided in CFM SB 72-1033.

CFM56-7B Fan Blade Cracks

As of January 29, 2020, the SWA flight 3472 (PNS) and flight 1380 (PHL) accidents were the only known CFM56-7B full-length fan blade release events. In addition to the fan blades involved in these two accidents, 25 cracked fan blades had been identified as of that date. These 25 cracked fan blades comprised the 6 blades on the PNS accident engine that were identified during metallurgical examinations as part of this investigation and 19 cracked blades that were found during field ultrasonic inspections or shop ECIs. (Three of the 19 blades were identified between the time of the PNS and PHL accidents, and 16 blades were identified after PHL accident through January 2020.) Table 2 provides information, including the number of flight cycles, for the 19 cracked fan blades found during field ultrasonic inspections or shop ECIs.

Table 2. Fan blade cracks detected as of January 2020.

Crack report date	Crack detection method	Number of fan blade estimated cycles since new
May 2017	Ultrasonic inspection	33,032
October 2017	ECI	19,371
March 2018	ECI	30,000 to 34,000
April 2018	Ultrasonic inspection	22,072
April 2018	ECI	31,697
April 2018	Ultrasonic inspection	34,770
June 2018	ECI	38,936
August 2018	ECI	32,500
December 2018	ECI	35,537
December 2018	ECI	21,327
December 2018	ECI	35,660
May 2019	ECI	40,113
June 2019	ECI	29,000 to 34,000
June 2019	ECI	31,000 to 36,000
August 2019	Ultrasonic inspection	42,000
October 2019	ECI	37,000
December 2019	ECI	35,000
December 2019	ECI	35,600
December 2019	ECI	33,600

Information

Certificate:	**Age:**
Airplane Rating(s):	**Seat Occupied:**
Other Aircraft Rating(s):	**Restraint Used:**
Instrument Rating(s):	**Second Pilot Present:**
Instructor Rating(s):	**Toxicology Performed:**
Medical Certification:	**Last FAA Medical Exam:**
Occupational Pilot:	**Last Flight Review or Equivalent:**
Flight Time:	

Aircraft and Owner/Operator Information

Aircraft Make:	Boeing	**Registration:**	N766SW
Model/Series:	737-700	**Aircraft Category:**	Airplane
Year of Manufacture:	2000	**Amateur Built:**	
Airworthiness Certificate:	Transport	**Serial Number:**	29806
Landing Gear Type:	Retractable - Tricycle	**Seats:**	151
Date/Type of Last Inspection:	August 24, 2016 Continuous airworthiness	**Certified Max Gross Wt.:**	154500 lbs
Time Since Last Inspection:		**Engines:**	2 Turbo fan
Airframe Total Time:	58344 Hrs	**Engine Manufacturer:**	CFMI
ELT:	Installed	**Engine Model/Series:**	CFM56-7B24G22
Registered Owner:	SOUTHWEST AIRLINES CO	**Rated Power:**	
Operator:	Southwest Airlines	**Operating Certificate(s) Held:**	Flag carrier (121)

Meteorological Information and Flight Plan

Conditions at Accident Site:	Visual (VMC)	**Condition of Light:**	Day
Observation Facility, Elevation:	KPQL	**Distance from Accident Site:**	
Observation Time:	08:53 Local	**Direction from Accident Site:**	
Lowest Cloud Condition:	Clear	**Visibility**	10 miles
Lowest Ceiling:	None	**Visibility (RVR):**	
Wind Speed/Gusts:	6 knots /	**Turbulence Type Forecast/Actual:**	/ None
Wind Direction:	60°	**Turbulence Severity Forecast/Actual:**	/ N/A
Altimeter Setting:	30 inches Hg	**Temperature/Dew Point:**	27°C / 24°C
Precipitation and Obscuration:	No Obscuration; No Precipitation		
Departure Point:	New Orleans, LA (KMSY)	**Type of Flight Plan Filed:**	IFR
Destination:	Orlando, FL (KMCO)	**Type of Clearance:**	IFR
Departure Time:	09:02 Local	**Type of Airspace:**	

Airport Information

Airport:	Pensacola KPNS	**Runway Surface Type:**	
Airport Elevation:	52 ft msl	**Runway Surface Condition:**	
Runway Used:		**IFR Approach:**	None
Runway Length/Width:		**VFR Approach/Landing:**	

Wreckage and Impact Information

Crew Injuries:	5 None	**Aircraft Damage:**	Substantial
Passenger Injuries:	99 None	**Aircraft Fire:**	None
Ground Injuries:	N/A	**Aircraft Explosion:**	None
Total Injuries:	104 None	**Latitude, Longitude:**	30.473333,-87.186668

Administrative Information

Investigator In Charge (IIC):	English, William
Additional Participating Persons:	
Original Publish Date:	March 30, 2020
Last Revision Date:	
Investigation Class:	Class
Note:	The NTSB traveled to the scene of this accident.
Investigation Docket:	https://data.ntsb.gov/Docket?ProjectID=93897