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

March 26, 2020

POWERPLANT GROUP CHAIRMAN'S FACTUAL REPORT
NTSB No: ENG19IA013

A. INCIDENT

Location:	Orlando International Airport (MCO) – Orlando, Florida
Date:	February 21, 2019
Time:	0729 eastern standard time (EST)
Aircraft:	Boeing 737-924, N30401, United Airlines Flight 1768

B. POWERPLANTS GROUP

Investigator in Charge:	Robert Hunsberger National Transportation Safety Board Washington, DC
Member:	Kyle Gustafson Federal Aviation Administration (ECO) Burlington, Massachusetts
Member:	Kate Keogh United Airlines Chicago, Illinois
Member:	Amy O'Dell United Airlines Chicago, Illinois
Member:	Ken Wolski GE Aviation Cincinnati, Ohio
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Member:	Rob Hous CFM International Cincinnati, Ohio
Member:	Steve Johnson CFM International Strother, Kansas
Member:	Andrew Fabian Boeing Seattle, Washington

C. SUMMARY

On February 21, 2019, about 0729 eastern standard time (EST), United Airlines Flight 1768, a Boeing 737-924, N30401, powered by two CFM International[1] CFM56-7B26 turbofan engines, experienced a No. 2 (right) engine failure during initial climb from Orlando International Airport (MCO), Orlando, Florida. At about 7,000 feet altitude, the flight crew reported an "abrupt loud grinding noise and instantaneous boom," followed by a loss of No. 2 engine power and subsequent uncommanded engine shutdown. The crew initiated quick reference handbook procedures, closed the No. 2 engine fuel shutoff valve, declared an emergency and returned to MCO, where they made an uneventful overweight single engine landing. Airport rescue and firefighting (ARFF) crews met the airplane on the adjacent high speed taxiway, but no fire or smoke was visible, and the airplane was cleared to taxi to the gate under its own power. No passenger or crew injuries were reported. The flight was being operated in accordance with 14 *Code of Federal Regulations* Part 121 and was a regularly scheduled flight from MCO to George Bush Intercontinental Airport (IAH), Houston, Texas.

Maintenance crews at MCO completed a visual inspection of the No. 2 engine and reported high pressure compressor (HPC) case burn through. There was no visible damage to the engine cowling or airplane structure. The No. 2 engine was removed and shipped to GE Aviation in Evendale, Ohio for examination and disassembly.

Party members from United Airlines (UAL), GE Aviation, CFM International, Boeing, the Federal Aviation Administration (FAA), and the National Transportation Safety Board (NTSB) met at GE Aviation to examine and disassemble the engine from February 26- March 2, 2019.

A visual examination of the HPC stator cases was performed after removal of the external engine components, and in addition to case burn through, one HPC stage 1 variable stator vane (VSV) trunnion stem was missing a washer and retaining nut. During engine disassembly, eight HPC stage 2 rotor blades, including the blade dovetails were found separated and missing. The HPC stage 2 blade slots for each of the eight missing blades had post fractures and missing material. There was secondary impact damage observed through the gas path aft of HPC inlet guide vanes and thermal damage was documented in the high pressure turbine (HPT).

The HPC rotor and HPC stator case assemblies were moved to the GE Aviation Materials Laboratory, also located at the Evendale, Ohio facility for additional examination and failure analysis. A

[1] CFM International is a joint venture between GE Aviation and Safran Aircraft Engines.

visual and binocular examination of the HPC stage 1 VSV trunnion stem with the missing washer and retaining nut confirmed that the lever arm was disengaged from the trunnion D-head, allowing the VSV to rotate approximately 31 degrees off schedule relative to the rest of stage 1 vanes. The HPC stage 2 disk post fracture morphology was consistent with a lower alternating stress, high cycle fatigue (HCF) and/or mixed-mode low cycle fatigue (LCF)/HCF mechanisms. There were no material anomalies observed near the fracture origin and the material microstructure was consistent with Ti-17[2] material as specified in the engineering drawing.

1.0 ENGINE INFORMATION

1.1 Engine History/Maintenance

According to United Airlines records, the No. 2 engine had accumulated the following hours and cycles:

Engine Position	Serial Number	Time Since New (hours)	Cycles Since New	Time Since Overhaul (hours)	Cycles Since Overhaul
Right/No. 2	875318	59,393	21,364	12,429	5,233

The last shop visit was an engine overhaul at the GE Aviation Celma maintenance, repair, and overhaul (MRO) facility in Petrópolis, Brazil in July 2014. GE Aviation located photographs taken during the overhaul, but the HPC VSV actuation hardware condition could not be determined from the photos, because many of the VSV trunnions and lever arms were obstructed by external engine components in the photographs. According to the shop records, 22 of the 82 HPC S1 VSVs were replaced and the remaining VSVs were overhauled. Installation position of the replaced and overhauled S1 VSVs were not recorded.

1.2 Engine Information

The CFM International CFM56-7B26 is a dual rotor, axial flow, high bypass ratio turbofan engine; single stage fan, three stage axial low-pressure compressor (LPC), nine stage axial HPC, annular combustion chamber, one stage high pressure turbine (HPT), four stage low pressure turbine (LPT), exhaust nozzle, exhaust center body, starter, and a Full Authority Digital Engine Control (FADEC) system.

According to the type certificate data sheet No. E0056EN Revision 10, dated August 9, 2016, the CFM56-7B26 has a takeoff static sea level thrust rating 26,300 pounds, flat rated at an ambient temperature of 86ºF and a maximum continuous sea level static thrust rating of 25,900 pounds, flat rated at an ambient temperature of 77ºF.

All directional references to front and rear, right and left, top and bottom, and clockwise and counterclockwise are made aft looking forward (ALF). Engine stages are abbreviated sequentially (S1, S2, etc.). A cross section of the engine is shown below in **Figure 1**.

[2] Ti-17, or Allvac® Ti-17 is a beta-rich, alpha-beta titanium alloy.

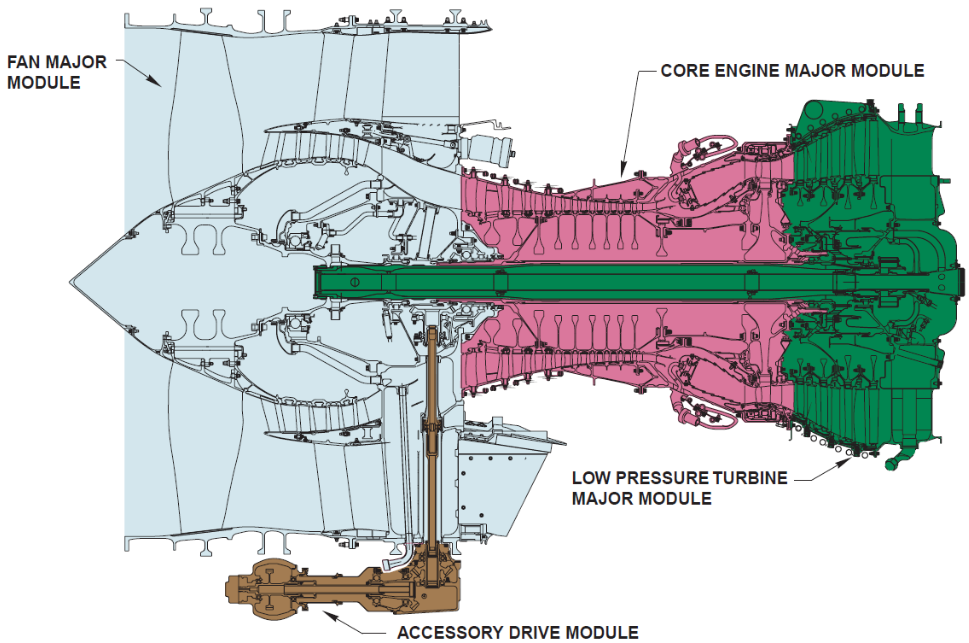

Figure 1-CFM56-7B Cross Section

2.0 ENGINE SERIAL NUMBER 875318 EXAMINATION AND DISASSEMBLY

The engine was shipped from MCO to GE Aviation in Evendale, Ohio with the quick engine change (QEC) hardware and core nozzle installed. When the investigation team arrived, the engine was in a shipping stand and secured in a cordoned off bay in the GE Aviation Development Engine Assembly shop (**Photos 1, 2, and 3**). According to United Airlines, when the cowling doors were opened at MCO to remove the engine from wing, loose metal debris fell to the ground and was collected, bagged, and shipped with the engine. Five loose HPC vane fragments were recovered from the engine. The vane fragments were deformed and missing material. Three of the vanes were resting against the aft fan case frame struts at the bottom of the engine and two vanes were recovered in the variable bleed valve (VBV) exit louvers (one vane located in the exit louver on either side of the strut located at the 6 o'clock position). The aft sump and accessory gearbox (AGB) / transfer gearbox (TGB) sump magnetic chip detector (MCD) plugs did not have metal debris accumulation on the magnet or in the filter screen. There was metal debris and "fuzz" accumulation on the forward sump MCD, and a sample was collected and sent to the GE Aviation Materials Laboratory for analysis (**Photo 4**). The electronic engine control (EEC) was removed and hand carried by GE Aviation Flight Safety to BAE Systems in Fort Wayne, Indiana for download and acceptance test procedure (ATP) on February 28, 2019. There were no errors or anomalies recorded during EEC testing.

Photo 1- Left Side of Engine, as Received

Photo 2- Right Side of Engine, as Received

Photo 3- ESN 875318 Data Plate

Photo 4- Forward Sump Magnetic Chip Detector Plug

The engine had no visible indications of undercowl fire or high energy uncontainment. The LPC fan blades were all intact and in good condition, and the fan spun smoothly with concurrent rotation of the low pressure turbine. A light surface rub was noted on the fan blade rub strip between the 6 and 10 o'clock positions, but the rub marks did not appear fresh. The engine had a circumferential burn through in plane with the HPC S1 rotor. External accessories and wiring obstructed a clear view of all HPC case surfaces, but case burn through was visible between the 9 and 6 o'clock positions. There was metal spray on external components, lines, and ducting in close proximity to the case burn through holes. The components and wiring did not exhibit signs of thermal damage or discoloration (**Photo 5**) with the exception of the J10 harness cloth braid. Very light metal spray was observed on the low pressure turbine case.

Photo 5- Metal Spray on Ducting Adjacent to Case Burn through (12 o'clock)

2.1 Low Pressure Turbine

All LPT S1 nozzle vanes were present, but the vanes exhibited thermal distress, most concentrated between the 11 and 1 o’clock positions. At these locations, the vane trailing edges showed indications of melting and material loss. The remaining vanes had varying degrees of suction side blade surface wrinkling (**Photo 6**).

Photo 6- LPT S1 Nozzle Vane Trailing Edge, (R) Close up at 12 o'clock

The LPT S1 blades were all present and complete. Light/minor impact marks and a small amount of fine metal debris accumulation and metal splatter was observed on the leading edge along the length of the blade span (**Photo 7**). The LPT S2 nozzle vanes visible through the LPT S1 rotor appeared to be intact and in good condition. The trailing edge of the visible LPT S4 blades were also intact and in good condition. The LPT shaft had a light circumferential rub mark 360 degrees around the shaft that appeared shiny, about 12 inches forward of the S1 disk. The rub marks on the shaft were coincident with the axial position of the core air rotor duct. Light impact marks were also noted on the forward end of the LPT shaft, just aft of the coupling splines coincident with the axial position of the booster sump pressurization air ducts (**Photo 8**). Metal flakes/debris were found in the area in and around the No. 4 bearing. Samples were collected by the GE Aviation Materials Laboratory for analysis (**Photo 9**). The No. 4 bearing roller elements were oil wetted and spun smoothly. When the bearing outer race was removed, two pieces of foreign debris were recovered from the race surface (**Photo 10**).

The group decided not to disassemble the LPT into stages due to a lack of investigative value.

Photo 7- LPT S1 Rotor Leading Edge

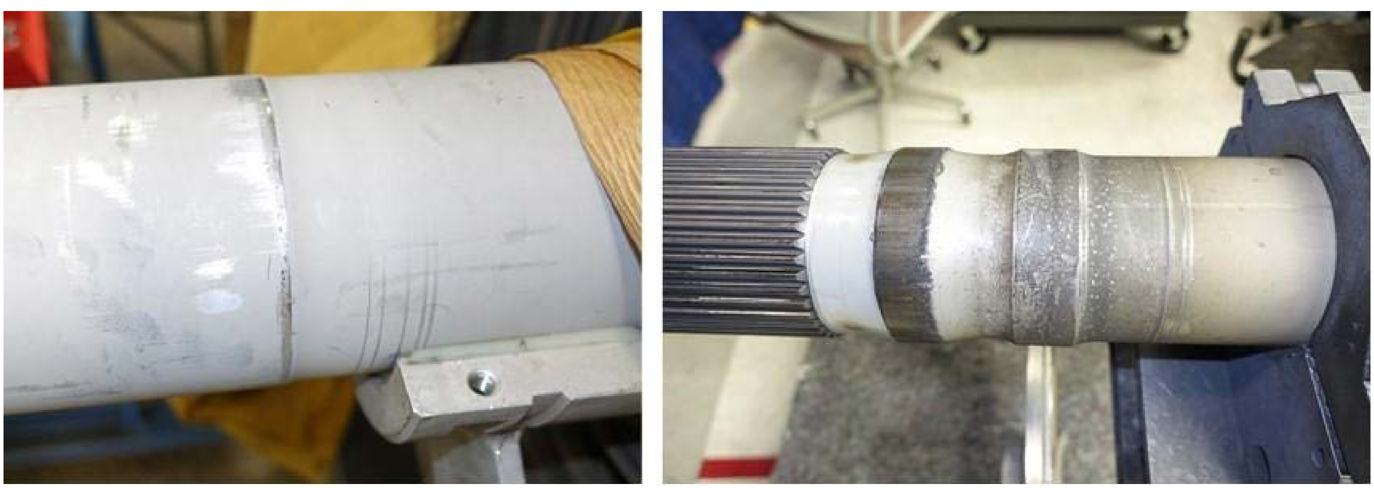

Photo 8- (L) LPT Shaft Rub, (R) LPT Shaft Light Impact Marks

Photo 9- No. 4 Bearing, Metal Debris

Photo 10- No. 4 Bearing Outer Race Debris

2.2 High Pressure Turbine

The HPT S1 nozzle vanes were all complete but exhibited trailing edge thermal distress and impact damage along the length of the vane span (**Photo 11**). Loose metal debris was resting at the inner and outer vane platforms. The HPT S1 rotor blades were all present, but leading edge thermal damage including areas of missing thermal barrier coating was noted. All HPT S1 blade trailing edges also exhibited thermal damage including material loss at the blade tips. Metal spray was present on the convex side of all blades (**Photo 12**).

Photo 11- HPT S1 Nozzle, (R) Trailing Edge

Photo 12- HPT S1 Blades (L) Leading Edge, (R) Trailing Edge

2.3 Combustor

The combustion liner was intact and did not show indications of burn through or localized hot spots. The liner had some discoloration that was considered normal for service run engines according to CFM International (**Photo 13**). Fine metal debris that was accumulated in the liner was removed and examined. Metal fragments were collected in the combustor discharge nozzle. Both combustor igniters were mechanically damaged.

Photo 13- Combustion Liner

The fuel nozzles were all intact and there was no visual evidence of excessive coking.

2.4 High Pressure Compressor

As noted in the initial observations, the HPC stator case had burn through in plane with the S1 rotor blade path. The burn through was observed around the case but was not continuous and was estimated have affected approximately 60% of the circumference. HPC case material remained intact around the rub button pads[3] and at the HPC stator case split flange.

During a visual inspection of the HPC cases after external component removal, one S1 VSV trunnion stem, identified as VSV #33, retaining nut was found missing at the 2 o'clock position (**Photo 14**). A majority of the VSV #33 lever arm was separated near the trunnion stem but the remaining intact portion of the lever arm was displaced and no longer engaged with the trunnion D-head. The trunnion D-head acts as an anti-rotation feature to ensure the VSVs rotate in unison. An alignment mark on the top of the trunnion stem was at a different angle relative to the rest of the trunnion stems in the stage. The threaded trunnion stem exhibited metal splatter on the forward facing side. The HPC stator case bridge connectors were intact and secure. The Inlet Guide Vane (IGV) lever arms were complete and secure on both the top and bottom cases, but multiple IGV lever arms were separated and missing near the vane trunnion stem. All IGV, S1, S2 and S3 VSV lever arms were labeled with a marker beginning at the 9 o'clock split line in the clockwise direction. The missing VSV lever arms are listed below in **Table 1**. Panoramic photos were taken of each stator case half to flatten out the case arc (**Photos 15, 16**). The S1 VSV actuation ring was consumed at multiple points where there was case burn through. The actuation rings for the other stages were complete but exhibited thermal damage in areas in proximity to the HPC case burn through. There was a circumferential crack on the HPC stator case, in plane with the S3 rotor, from the 12 to 2 o'clock positions. The IGV and S1 actuation hardware was removed to better examine and document the case burn through. The HPC stator had three IGV vanes still present in the outer platform and six additional vanes remained in the separated IGV inner shroud. The vanes exhibited deformation, material loss and thermal distress. All remaining HPC vanes were separated and missing and a thick layer of metal slag was adhered to the interior surface of both the HPC stator top and bottom case halves (**Photo 17, 18**).

[3] The rub button pads are metal blocks that act as guides for composite button that are installed to keep the actuation ring round.

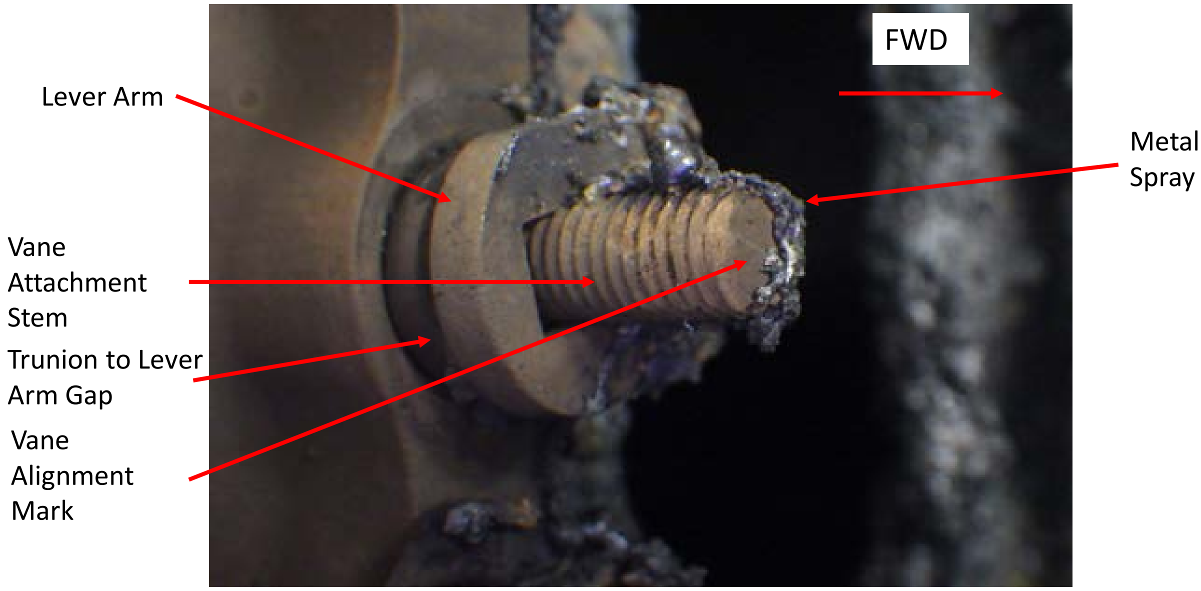

Photo 14- S1 VSV #33 Stem with Missing Retaining Nut

Stage	Total Vanes	Separated Lever Arms	Notes
Inlet Guide Vanes	42	None	*IGV 27 lever arm had minor deformation.
1	82	4, 5, 6, 11, 12, 13, 18, 19, 23, 24, 25, 31, 32, 33*, 34, 35, 39, 40, 41, 44, 47, 53, 54, 55, 80, 81	*VSV 33 lever arm retaining nut missing.
2	84	None	
3	72	None	

Table 1- Missing Lever Arm Information

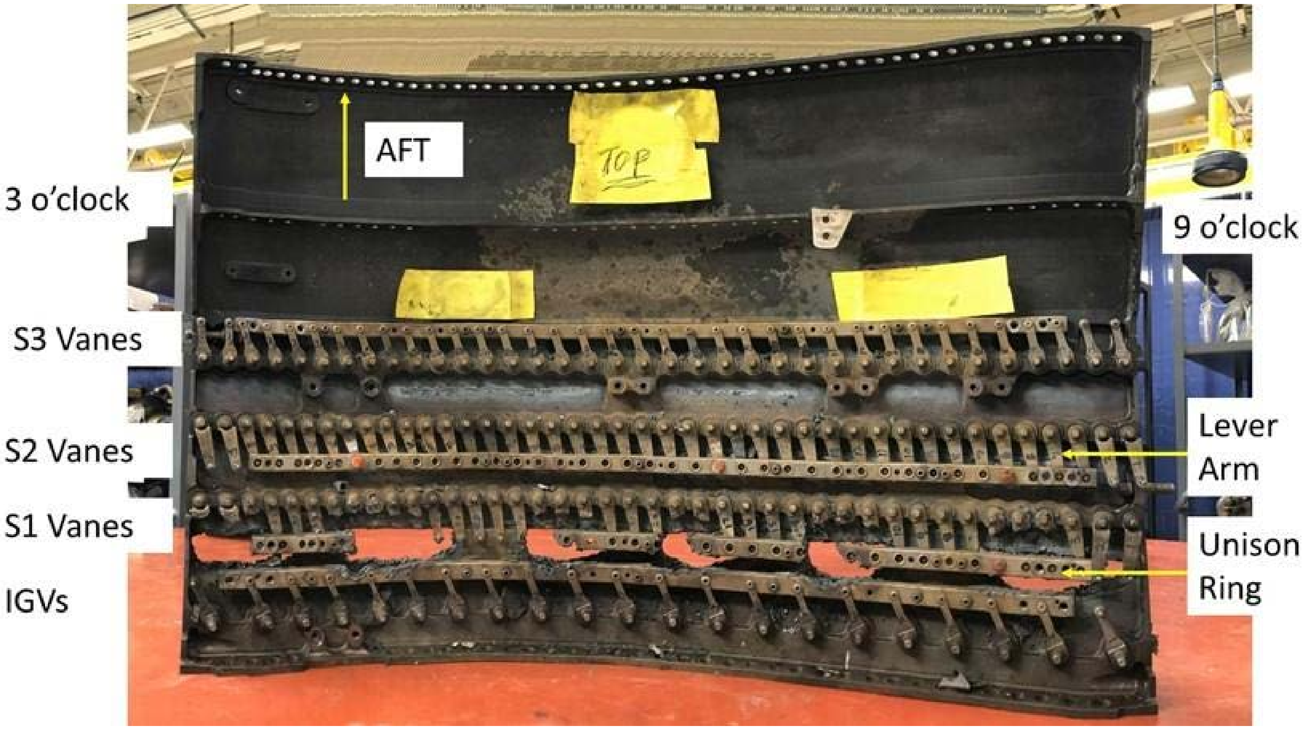

Photo 15- Panoramic Photo of HPC Stator Case (Exterior), Top Half

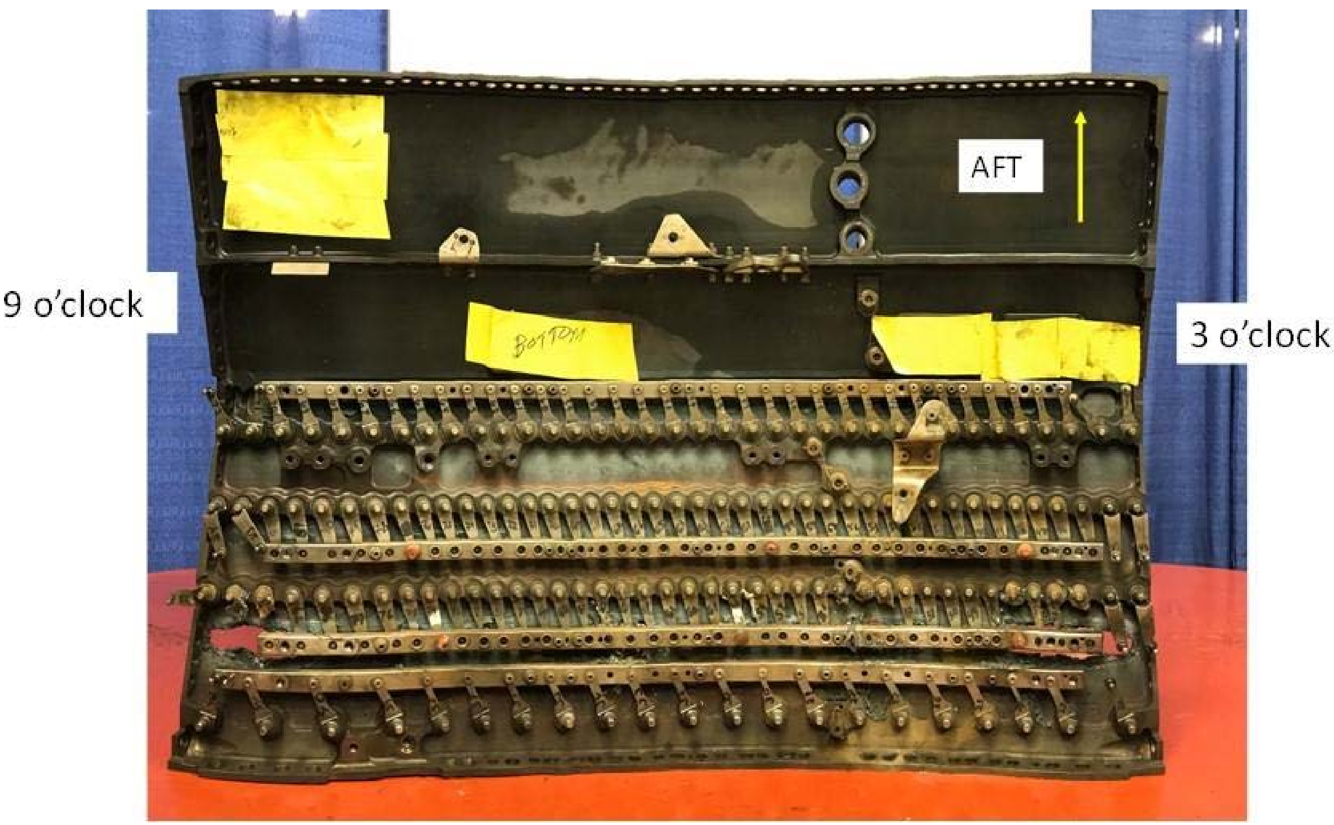

Photo 16- Panoramic Photo of HPC Stator Case, Bottom Half

Photo 17- HPC Stator Case, Top Half

Photo 18- HPC Stator Case, Bottom Half

When the stator case halves were removed to expose the HPC rotor, loose metal debris fell into a collection tarp placed on the ground. The debris that was recovered was mostly fine metal grit, but deformed vanes, vane shrouds and vane trunnions were also identified. The S1 HPC rotor blades were all present but exhibited substantial impact and thermal damage (**Photo 19**). The S1 blade leading edges were missing material leaving a wave like shape from the blade platform to blade tip. All remaining S1 blades were about the same shape around the disk and measured about 1 inch from the platform to the blade tip.

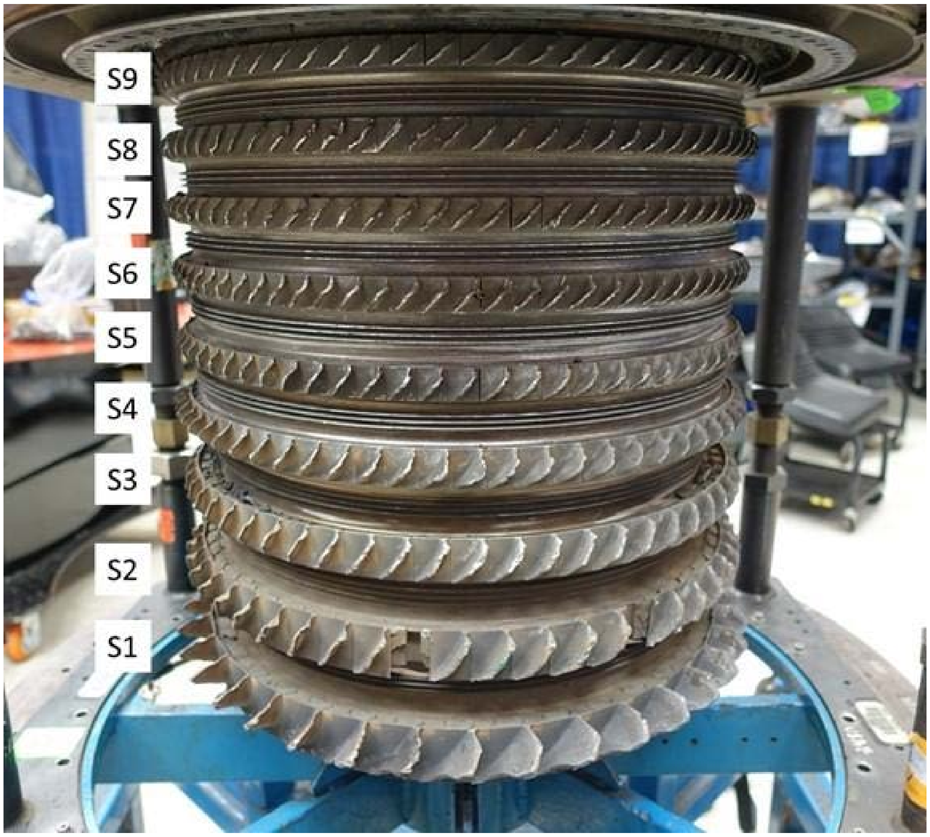

Photo 19- HPC Rotor

The HPC S2 blades also exhibited mechanical and thermal damage. A total of eight blades, including the blade dovetails were separated and missing (**Photo 20**). All remaining blades had a rounded/tombstone shape and measured about ½ inch from blade platform to tip. The separated blades all had fractured disk posts on either one or both sides of the blade dovetail disk slot. The blades were numbered beginning with the blade directly forward of the spool drain hole in the clockwise direction. The missing blade numbers, retaining ring condition, and disk post fracture direction are shown in **Table 2**. A complete documentation of the S2 disk posts fracture condition is included in the GE Aviation Materials Laboratory report.

Photo 20- HPC S2 Missing Blade 40, Fractured Disk Posts

Separated S2 Blade	Forward Retaining Ring	Aft Retaining Ring	Post Fracture
1	Separated/Missing	Intact	R- Fwd to Mid L- Fwd to Mid
3	Intact	Separated/Missing	R- Aft to Mid L- Intact
5	Intact	Separated/Missing	R- Complete Post Fracture L- Complete Post Fracture
6	Intact	Separated/Missing	R- Complete Post Fracture L- Complete Post Fracture
7	Intact	Separated/Missing	R- Complete Post Fracture L- Aft to Mid
34	Intact	Separated/Missing	R- Complete Post Fracture L- Intact
40	Intact/Bent	Separated/Missing	R- Fwd to Mid L- Fwd to Mid
53	Separated/Missing	Intact	R- Minor Leading Edge Damage L- Fwd to Mid

Table 2- Separated S2 Rotor Blade Retaining Ring and Post Damage Descriptions

HPC rotor blade stages 3-9 had impact damage and material loss. All blade platform and dovetail posts were present, but only stubs of the airfoils remained (corn-cobbed condition).

2.5 Low Pressure Compressor (Booster) / Fan Case

The fan frame struts leading to the HPC had metal spray on the trailing edge and loose metal debris resting along the bottom of the frame against the struts. Each of the VBV louvers were removed and metal spray and loose debris were present in the VBV ports, most concentrated along the bottom half of the engine (**Photo 21**). A loose bolt was found in the VBV louver at the 7 o’clock position. The side of the bolt head was flattened/rubbed, and an oval/racetrack pattern was worn into the louver where the bolt was resting (**Photo 22**). The source of the loose bolt was not identified. The VBV doors were in the open position at the start of the exam. An IGV was wedged in the 8 o’clock VBV door. The VBVs were cycled by pressurizing the VBV actuators, they closed fully and opened uniformly.

Photo 21- VBV Louver, 6 o'clock

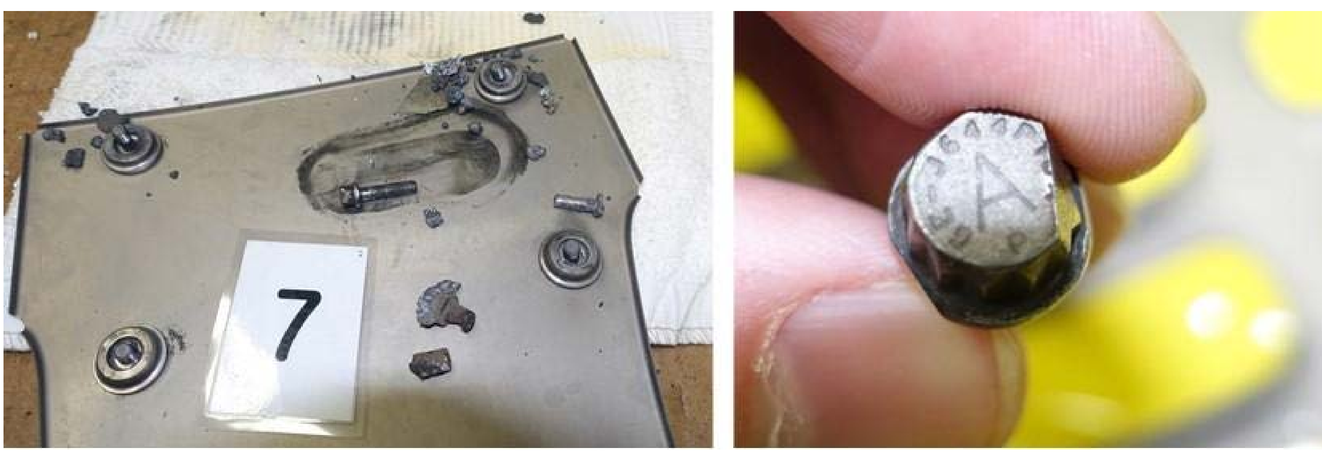

Photo 22- Louver 7 Rub and Loose Bolt

The group determined the booster did not require disassembly. The S4 nozzle vanes (booster outlet guide vanes) visible from the aft end of the booster module assembly had minor impact damage on the trailing edge and impact damage and tearing on the leading edge between the midspan and outer vane platform. The booster S4 blades exhibited impact damage and material tearing along the length of the blade span. The leading edge of the S4 vanes weren’t fully visible but appeared to be in good condition. The S3 vane trailing edges that were visible were intact and in good condition.

3.0 MATERIALS LAB ANALYSIS

The HPC rotor assembly, HPC cases, and the metal debris samples collected from the aft sump adjacent to the No. 4 bearing and the forward sump MCD were moved to the GE Aviation Materials Laboratory in Evendale, Ohio for analysis.

Visual and binocular examination of the HPC S1 VSV (#33) trunnion stem confirmed that the missing washer and retaining nut identified during the engine teardown resulted in disengagement of the lever arm from the trunnion D-head allowing the vane to go off-schedule approximately 31 degrees relative to the other S1 VSVs (**Photo 23**). The VSV #33 trunnion stem had a uniform coating of dirt/debris along the full length of the stem consistent with engine operation over an extended period of time without a washer and retaining nut (**Photo 24**). The laboratory also identified witness marks on VSV #33 indicating that a washer and retaining nut were present at some point, but it could not be determined when the parts separated.

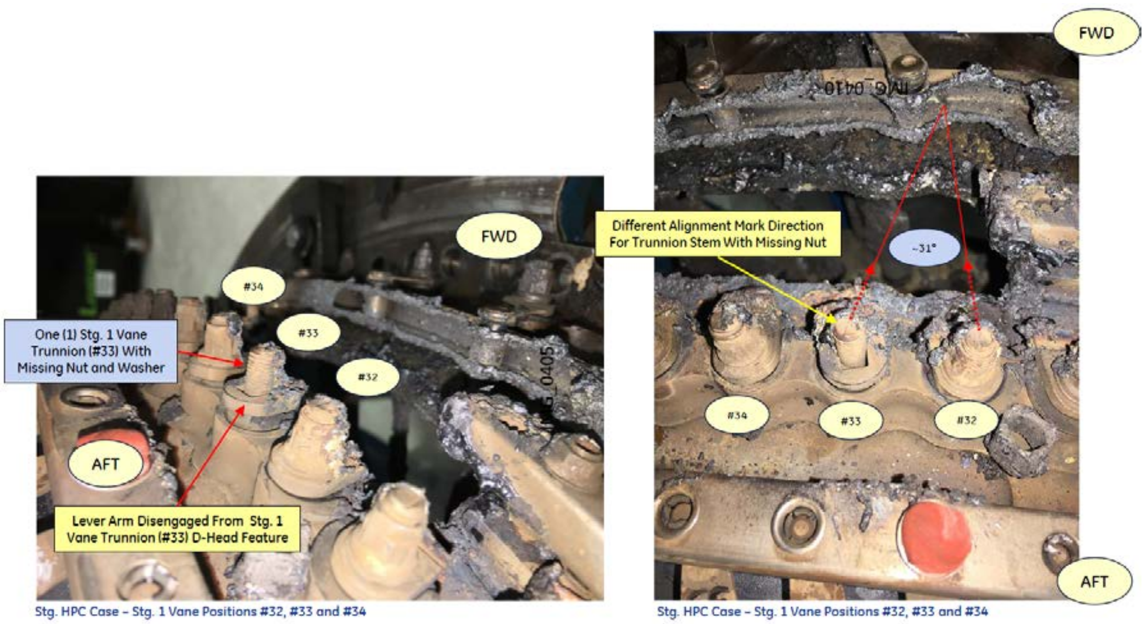

Photo 23- HPC S1 VSV Trunnion #33 (Photos courtesy of GE Aviation)

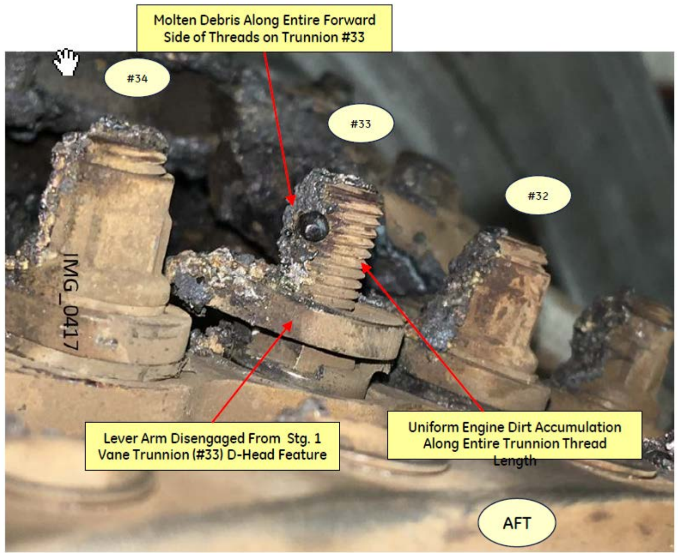

Photo 24- HPC S1 VSV #33 Trunnion Stem (Photo Courtesy of GE Aviation)

The HPC S2 disk posts fracture morphology was consistent with a lower alternating stress, high cycle fatigue (HCF) and/or mixed-mode low cycle fatigue (LCF)/HCF mechanisms. The primary initiation occurred adjacent to the forward right corner (corner #1) of the disk posts (**Photo 25**). A metallographic examination through a crack origin on one of the S2 disk posts did not identify any material anomalies or stress risers. The material microstructure was consistent with properly processed beta forged Ti-17 material as specified in the engineering drawing.

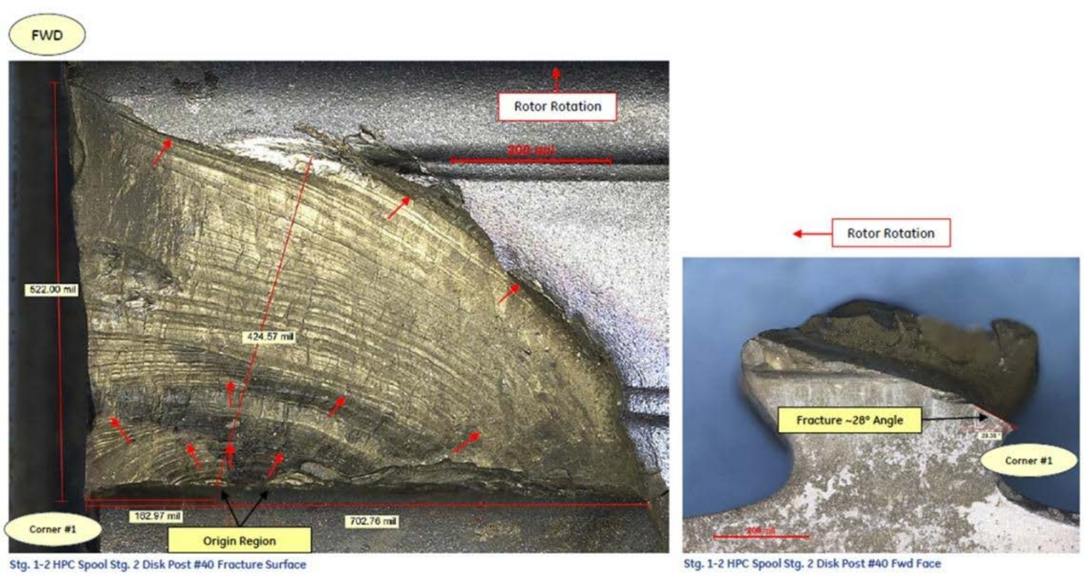

Photo 25- HPC S2 Disk Post #40, (L) Visible Fatigue Beachmarks, (R) Fracture Angle (Photos courtesy of GE Aviation)

The metal flakes collected in the aft sump region were analyzed and RBD[4] modified steel and silver plating were identified. Both of these materials are present in the No. 4 bearing assembly. The metal that was collected on the forward sump MCD was identified as M50Nil, also a bearing material. The complete Metallurgical Investigation Report is available in the investigation docket. The NTSB Materials Laboratory assisted in development of the metallurgy workscope and concurred with the findings and conclusions in the final report.

4.0 CFM56 LEGACY FAILURE MODE EXPERIENCE

According to CFM International records, the CFM56 engine fleet has had a total of 53 field reports of VSV misalignment on engines featuring a steel forward compressor (HPC) case. Three titanium fire events have been attributed to VSV misalignment. The fire events occurred on CFM56-3 and -5A engines. The incident event covered by this investigation is the first compressor titanium fire on record for the CFM56-7 series engine.

Compressor titanium fire risk was addressed during engine certification in the CFM56-7B Fire Prevention and Fault and Safety Analysis Reports. The CR 988-7B Fire Prevention Report includes the following statement: "Engine experience has shown that a compressor titanium fire will cause compressor stall and engine shut down in a time interval less than 10 seconds. As a result of these CFM56-7B… engine features and the expected short duration of titanium fire exposure, the likelihood of an uncontrolled fire resulting from casing burn through is expected to be extremely remote."

5.0 CONTINUED AIRWORTHINESS ASSESSMENT METHODOLOGIES (CAAM) ASSESSMENT

A CAAM analysis was completed under the guidelines outlined by FAA Advisory Circular 39-8. The analysis identified the incident as a "no safety effect" event based on the findings that the engine shutdown occurred above 1,500 feet altitude and the crew did not receive a fire warning indication. The analysis as referenced the findings that the nacelle inner barrel surfaces did not exhibit thermal damage and the fire/thermal damage on the engine was localized to the HPC case.

6.0 CORRECTIVE ACTION REVIEW

Following the incident event, the GE Aviation Celma CFM56 MRO facility incorporated additional control measures to avoid improper VSV lever arm installation. The revised installation procedure requires a trunnion retaining nut torque verification check by a second technician using a torque wrench. The primary mechanic is then instructed to complete a trunnion retaining nut seating check using a 0.001 inch feeler gauge to prevent false torque indications (**Photo 26**). The feeler gauge check ensures there is no gap between the retaining nut and the lever arm surface.

[4] RBD modified steel is a carburized tool steel that was developed in the United Kingdom.

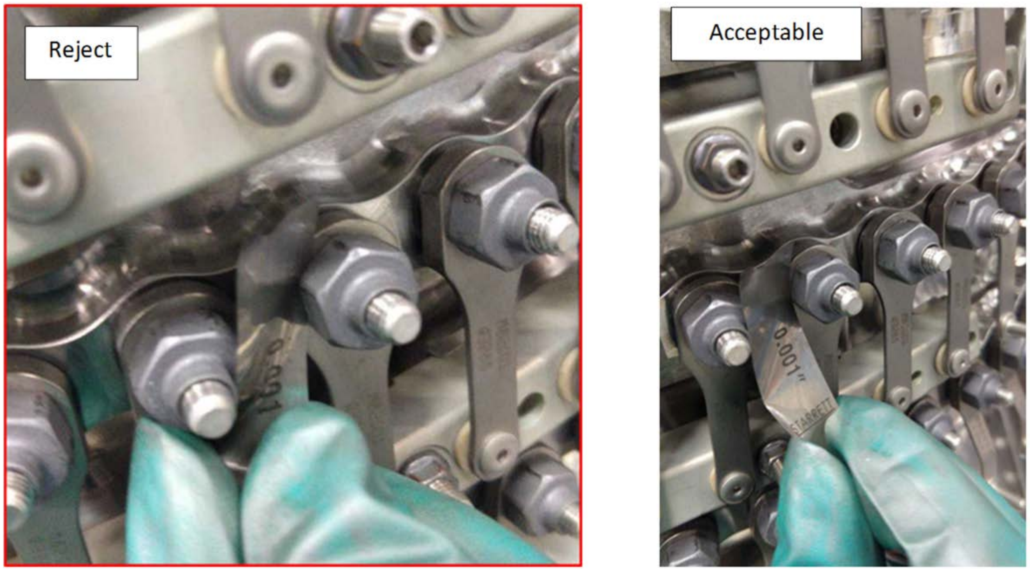

Photo 26- VSV Trunnion Retaining Nut Feeler Gauge Check (Photos Courtesy of GE Aviation)

The GE Aviation Strother, Kansas CFM56 MRO facility also revised their VSV lever arm installation instructions to add a mechanic stamp requirement during engine build to confirm proper thread protrusion on each VSV trunnion.

GE Aviation/CFM International notified the NTSB that they are currently evaluating new torque wrench technology that measures torque and angle. The wrench will alert the technician if the pre-set angle is not applied to achieve the torque values specified in the engine manual.

The missing VSV washer and retaining nut finding was reviewed at all CFM56 engine maintenance, repair, and overhaul (MRO) shops and presented at both the June 2019 Operators Symposium in Brussels, Belgium and the November 2019 All Middle East Operators Conference in Muscat, Oman. Finally, the incident findings were featured in the first quarter (Q1) 2020 edition of the GE Fleet Highlites that is available on the GE portal and can be accessed by all operators and MRO shops.

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Propulsion