



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

February 26, 2018

**POWERPLANT GROUP CHAIRMAN'S FACTUAL REPORT
NTSB No: ENG17IA006**

A. INCIDENT

Location: Kansas, Illinois

Date: November 29, 2016

Time: 0850 central standard time (CST)

Aircraft: Bombardier CRJ700, N367CA, GoJet Airlines LLC, Flight 6210

B. POWERPLANTS GROUP

Group Chairman: Robert Hunsberger
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C. SUMMARY

On November 29, 2016, at about 0850 central standard time (CST), GoJet Airlines, LLC, Bombardier CRJ700, N367CA, equipped with two General Electric (GE) CF34-8C5B1 turbofan engines, experienced an uncontained No. 1 (left) engine failure in cruise flight (FL 340). The crew reported hearing a loud bang from the back of the airplane, immediately followed by No. 1 engine roll back and flame out. The crew declared an emergency, diverted to Lambert-St. Louis International Airport (STL), and made an uneventful single engine landing. There were no reported passenger or crew injuries. The flight was being operated in accordance with 14 *Code of Federal Regulations* Part 121 and was a regularly scheduled flight from Cincinnati/Northern Kentucky International Airport (CVG) - Covington, Kentucky to Denver International Airport (DEN) - Denver, Colorado. The GoJet Airlines flight was operating as a Delta Air Lines connection.

A visual inspection of the No. 1 engine was conducted by maintenance personnel at STL who reported a hole in the turbine case and extensive damage to the low pressure turbine (LPT) hardware visible through the exhaust nozzle. There were no penetrations through the engine cowlings. The No. 1 engine was removed from the airplane and shipped to Delta TechOps in Atlanta, Georgia for examination and disassembly. Party members from Delta Air Lines, GE Aviation, and the NTSB met at Delta TechOps from January 9-11, 2017.

The LPT case penetration was located at the 8 o'clock position, about 1 inch aft of the LPT flange, coincident with the high pressure turbine (HPT) 2nd stage rotor plane of rotation. The metal around the penetration was curled radially out, away from the engine. Seven HPT 2nd stage blades were found separated below the blade platform and the remaining 61 blades exhibited impact damage and were separated above the platform at random locations along the blade span. Impact and thermal damage were observed on all four LPT rotor and nozzle stages.

The complete set of 68 HPT 2nd stage rotor blades were sent to the GE materials laboratory in Lynn, Massachusetts for analysis. The seven blades that separated below the blade platform were all fractured at the shank upper radius and the fracture surfaces exhibited visual evidence of fatigue originating on the convex side from areas of corrosion. The 61 blades that were separated above the platform all failed due to overload, consistent with secondary damage. Of the 61 blades, multiple blades had corrosion and cracks present below the blade platform in the same general location as the blades that separated below the platform.

The LPT case was sent to the GE materials laboratory in Evendale, Ohio for visual examination and material analysis. The case material composition and hardness levels met drawing specification. The LPT case penetration fracture surfaces had ductile dimples consistent with overload.

GE released a redesigned HPT 2nd stage blade featuring minor geometry improvements and a protective coating to prevent corrosion as a category 7¹ service bulletin (SB) in 2010. The SB was upgraded to category 3² in 2013.

¹ A Category 7 service bulletin is for notification purposes and can be completed at the customers convenience.

² A Category 3 service bulletin recommends replacement of the blades with the updated part number at the next shop visit.

D. DETAILS OF THE INVESTIGATION

1.0 ENGINE INFORMATION

1.1 Engine History/Maintenance

According to GoJet maintenance records, the engine had accumulated the following hours and cycles:

Engine Position	Engine Serial Number	Time Since New (hours)	Cycles Since New	Time Since Repair (hours)	Cycles Since Repair
Left/No. 1	E965352	33,206.7	23,814	20,709.2	14,839

The last engine inspection was completed 503.1 hours prior to the incident and included a borescope inspection of the combustion liner and HPT. There were no findings reported during the inspection.

According to GoJet Airlines maintenance personnel engine water washes are performed at 1200 flight hour intervals. Additional water washes are performed if engine trend data shows performance degradation. These maintenance intervals are within GE recommended guidelines.

1.2 Engine Information

The GE CF34-8C5B1 is a two spool turbofan engine that has a single stage high-bypass fan driven by a four stage axial-flow low pressure turbine (LPT). The high pressure section of the engine includes a 10 stage axial flow compressor with inlet guide vanes and variable stator vanes on stages 1-4, an annular combustor, and a two stage axial-flow HPT that drives the high pressure compressor (**Figure 1**). The engine control system is composed of a full authority digital/electronic engine control (FADEC), a fuel metering unit (FMU), a permanent magnet alternator (PMA), engine sensors, variable geometry actuator, and an operability bleed valve (OBV). The engine has a static sea level thrust ratings of 13,790 pounds maximum takeoff (5 minutes) and 13,280 pounds maximum continuous.³

³ Engine ratings are based on standard sea level static conditions of 59°F and 29.92 inches Hg with no accessory loads, air extraction, or anti-ice.

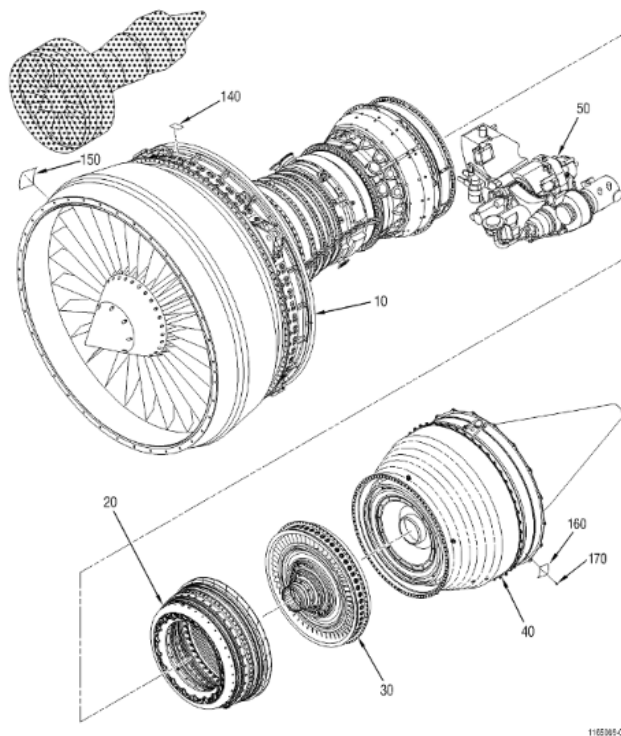


Figure 1- CF34-8C Engine Diagram

All directional references to front and rear, right and left, top and bottom, and clockwise and counterclockwise are made aft looking forward (ALF).

2.0 NO. 1 ENGINE EXAMINATION, ENGINE SERIAL NUMBER E965352

The No. 1 engine was shipped from STL in a transport dolly and was held in the NTSB quarantine cage at Delta TechOps until the investigative group arrived. The engine was then relocated to the TechOps CF34 shop where it was placed in work pedestals for visual examination and disassembly (**Photos 1, 2, and 3**).

The LPT case had a 3 inch circumferential by 1.5 inch axial penetration located 1 inch aft of the LPT flange at the 8 o'clock position (**Photo 4**). The LPT case and baffle material around the penetration was curled radially outward, away from the engine. The exhaust gas temperature (EGT) thermocouple and thermocouple harness adjacent to the penetration were damaged. Fine metal debris was resting on the exhaust case struts. The remainder of the engine exterior was intact and in good condition.



Photo 1- Left Side of the Engine, As Received



Photo 2- Right Side of the Engine, As Received

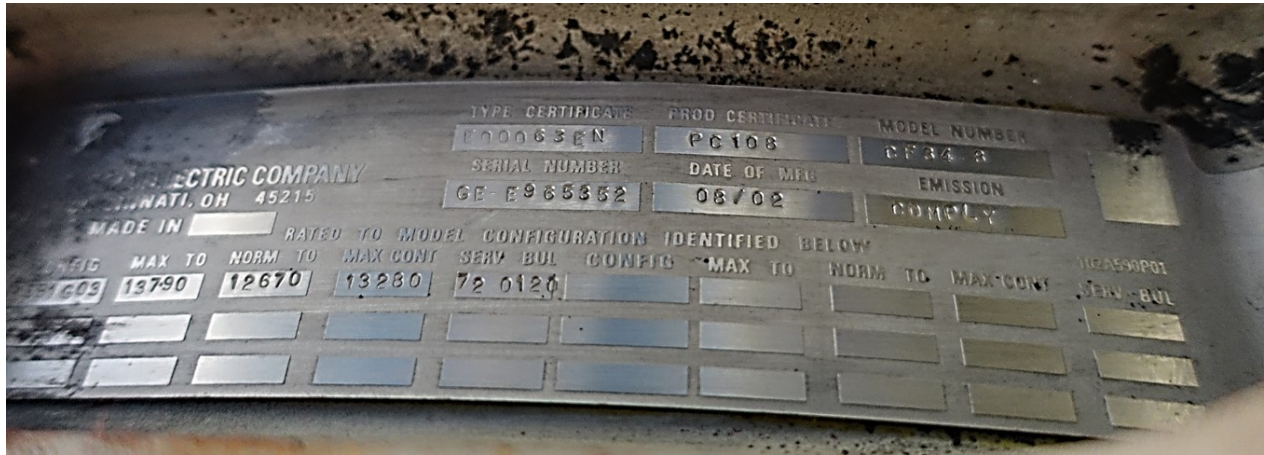


Photo 3- Engine Serial Number E965352 Data Plate

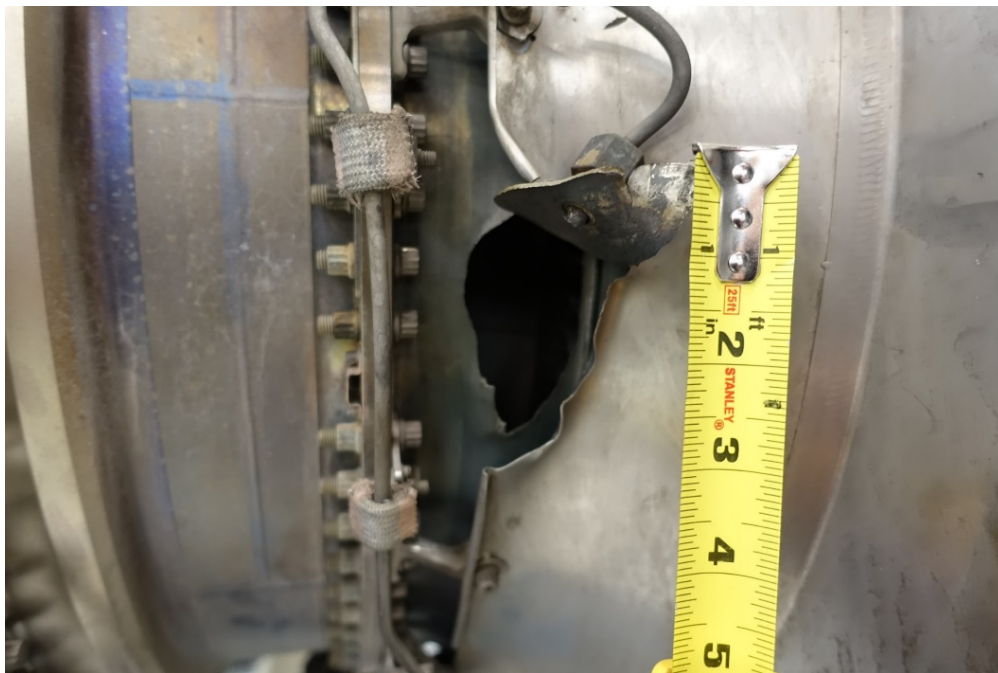


Photo 4- LPT Case Penetration, 8 o'clock Position

2.1 Fan and Compressor

The fan case had rub damage along the fan blade tip path 360 degrees around the case circumference, with areas of heavier localized rub, including wear of the abradable layer to the base metal at the 6 o'clock position (**Photo 5**). The fan could not initially be rotated by hand, but after removal of the LPT, the fan spun smoothly, consistent with functional No. 1 and No. 2 bearings. The compressor was not further disassembled due to a lack of evidence indicating internal damage.



Photo 5- Fan Case Rub

2.2 Combustor

The HPT 1st stage nozzle was removed to examine the combustion liner. The liner was intact, without indications of burn through or fuel nozzle streaking. The fuel nozzles and igniters both had dark discoloration (coking) that is considered normal for service run engines according to GE and Delta.

2.3 High Pressure Turbine

The HPT 1st stage blades were intact but exhibited minor thermal damage and tip rub. Thermal barrier coating (TBC) loss and minor blade trailing edge damage were observed at the blade airfoil midspan over about a 40% section of the disk. The HPT 1st stage rotor shroud exhibited rub damage that resulted in material loss and bluing. The HPT 1st stage nozzle was intact and in good condition.

All HPT 2nd stage disk blades were fractured and separated but the blade dovetails all remained secure within the disk blade slots (**Photo 6**). Sixty one of the 68 blades were fractured between $\frac{1}{4}$ to $\frac{1}{2}$ inch above the blade platform and the remaining seven blades were fractured below the blade platform. The blade stubs exhibited both impact and thermal damage (**Photo 7**). The HPT 2nd stage disk shroud was heavily rubbed and had an area of localized deformation at the 8 o'clock position, coincident with the location of the penetration observed in the LPT case (**Photo 8**). The HPT 2nd stage nozzle was intact but had thermal damage including flaked TBC on the leading edge of multiple vanes. A single vane at the 6 o'clock position had

midspan burn through on the leading edge and on the pressure side, near the trailing edge. Fine sand/dust was packed in crevices throughout the HPT.

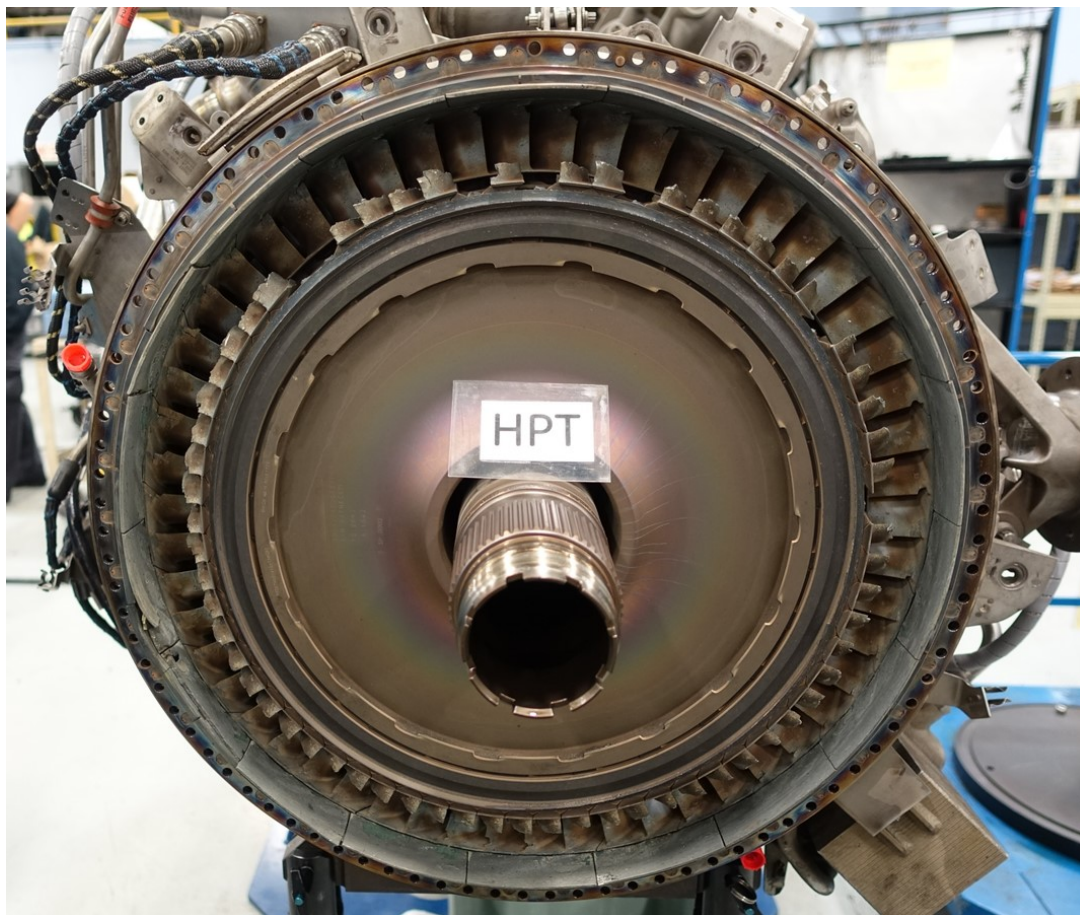


Photo 6- HPT 2nd Stage (ALF)



Photo 7- HPT 2nd Stage Blade Damage



Photo 8- HPT 2nd Stage Blade Shroud

2.4 Low Pressure Turbine

The LPT 3rd stage nozzle exhibited extensive impact damage on the trailing edge of all vanes (**Photo 9**). The midspan of eight consecutive vanes were thermally consumed at the 8 o'clock position, leaving small sections of the vane in place at the inner and outer platforms (**Photo 10**). An unidentified yellow substance was noted on several vanes. The LPT 3rd stage blades were all fractured near the blade platform and the remaining blade stubs had impact damage, including material tearing and smearing (**Photo 11**). The outer transition liner (OTL) segments were fractured and missing from the LPT case. One complete segment of the OTL was still present at the 12 o'clock position. A fragment of one of the OTL segments was adhered to the forward side of the 3rd stage nozzle at the 2 o'clock position. To remove the LPT baffle, the deformed metal around the LPT case penetration was broken off with a hammer. The piece was collected for shipment to the GE Materials Laboratory with the LPT case.

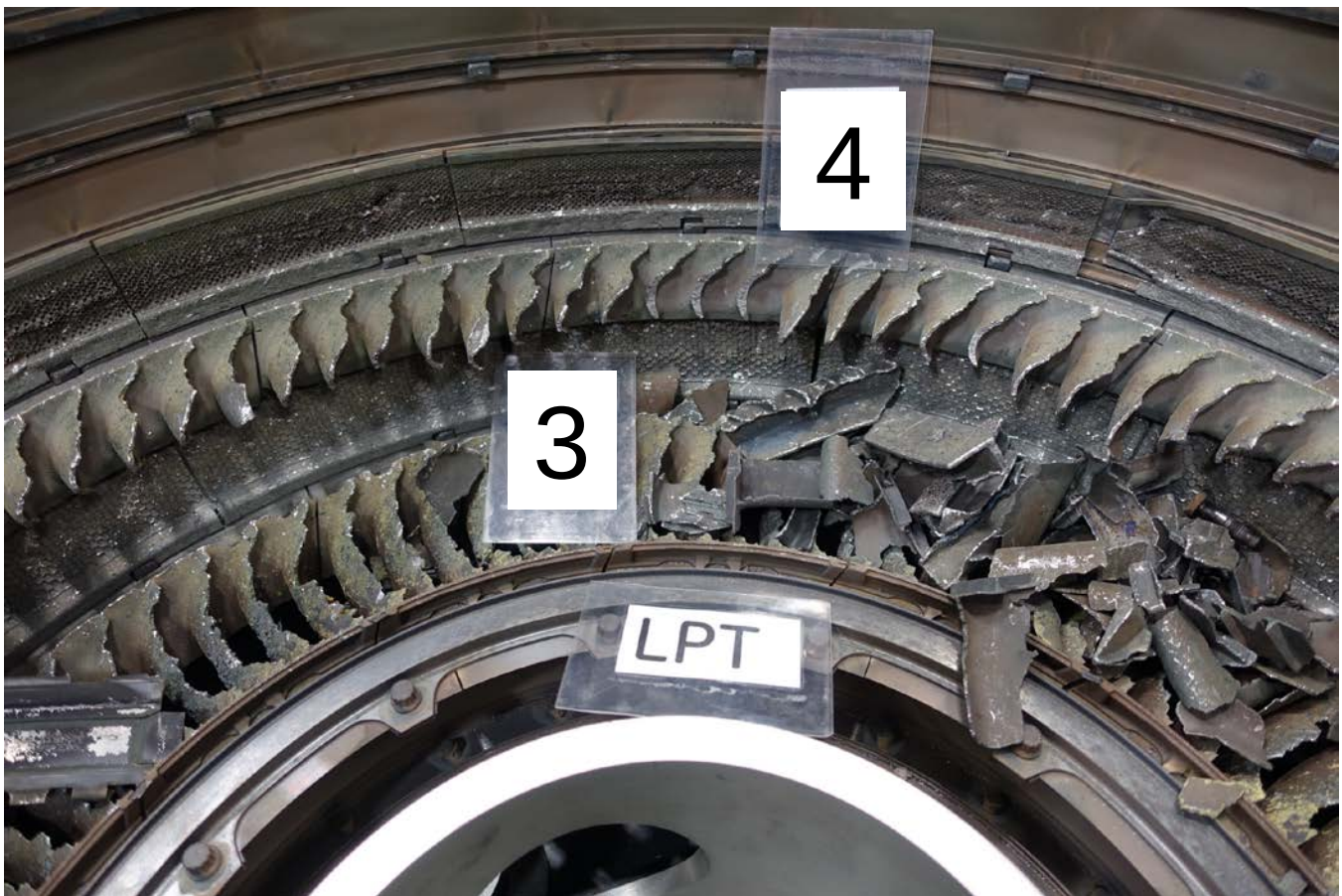


Photo 9- LPT 3rd and 4th Stage Nozzle Vanes



Photo 10- LPT 3rd Stage Nozzle Vane Thermal Damage



Photo 11- LPT 3rd and 4th Stage Rotor Blade Damage

The LPT 4th stage nozzle inner diameter platform was fragmented into pieces and all nozzle vanes were separated near the outer diameter platform. All LPT 4th stage blades were fractured near the blade platform and the remaining blade stubs exhibited impact damage including material tearing and smearing (**Photos 9 and 11**).

The LPT 5th stage nozzle vanes were all fractured, with small pieces of each vane airfoil remaining on the inner and outer diameter platforms. Two vane segments were separated

completely from the case at the 8 o'clock position. A large quantity of metal debris was trapped between the LPT 5th and 6th stages (**Photo 12**).

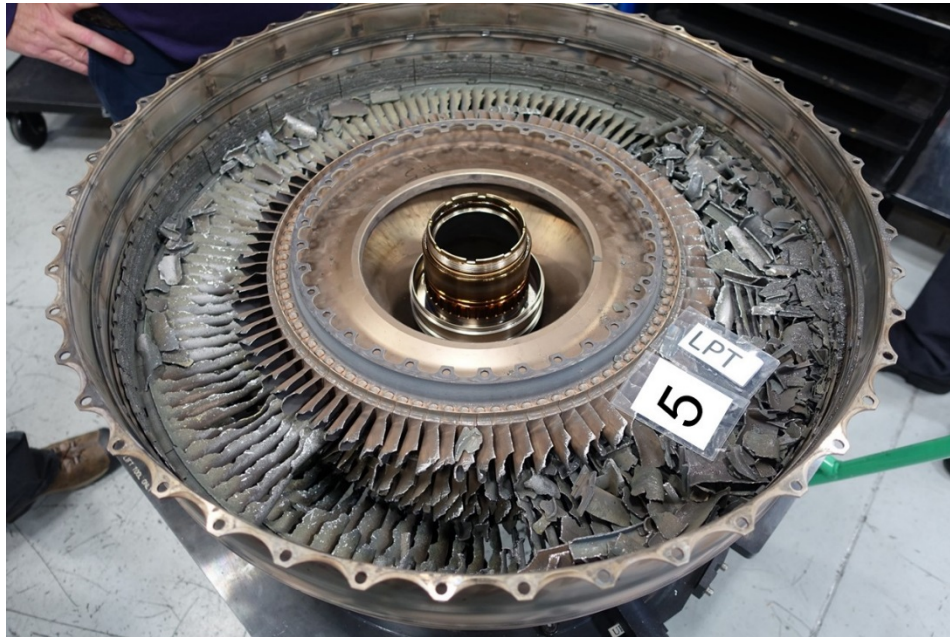


Photo 12- LPT 5th Stage Damage

The LPT 6th stage nozzle vanes were intact but had impact damage on both the leading and trailing edges. An area of localized impact and thermal damage was noted at the 8 o'clock position.

The No. 5 bearing was oil wetted and the bearing elements rotated smoothly.

3.0 METALLURGY

The remaining stubs of all 68 HPT 2nd stage rotor blades were removed from the disk and shipped to the GE materials laboratory in Lynn, Massachusetts for analysis. The laboratory confirmed that seven HPT 2nd stage blades, part number 4125T01, separated under the platform at the shank upper radius, and each fracture surface exhibited visual evidence of fatigue on the convex side originating from an area of corrosion (**Photo 13**). Multiple blades that were separated above the blade platform also had cracks originating from areas of corrosion below the platform in the same general location as the seven blades that separated below the platform. The 61 blades that were separated above the platform failed due to overload consistent with secondary damage. It could not be determined which blade separated first.

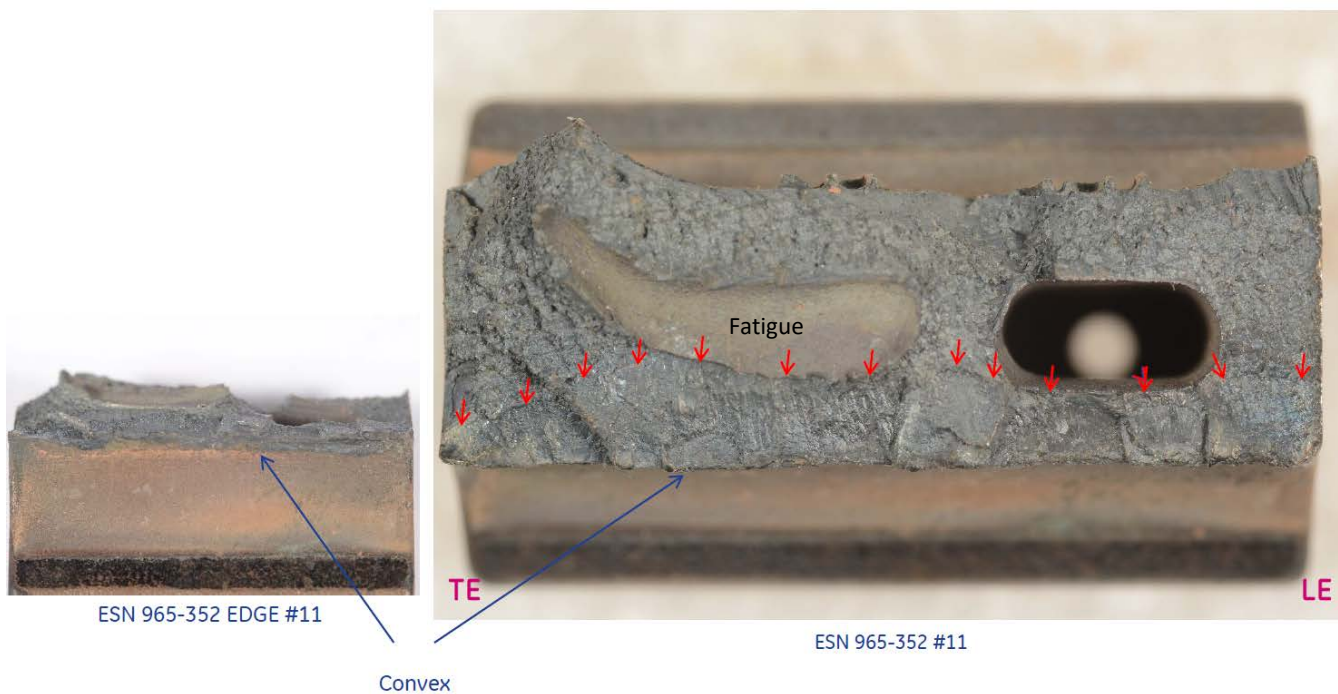


Photo 13- HPT 2nd Stage Blade Under Platform Fracture Surface

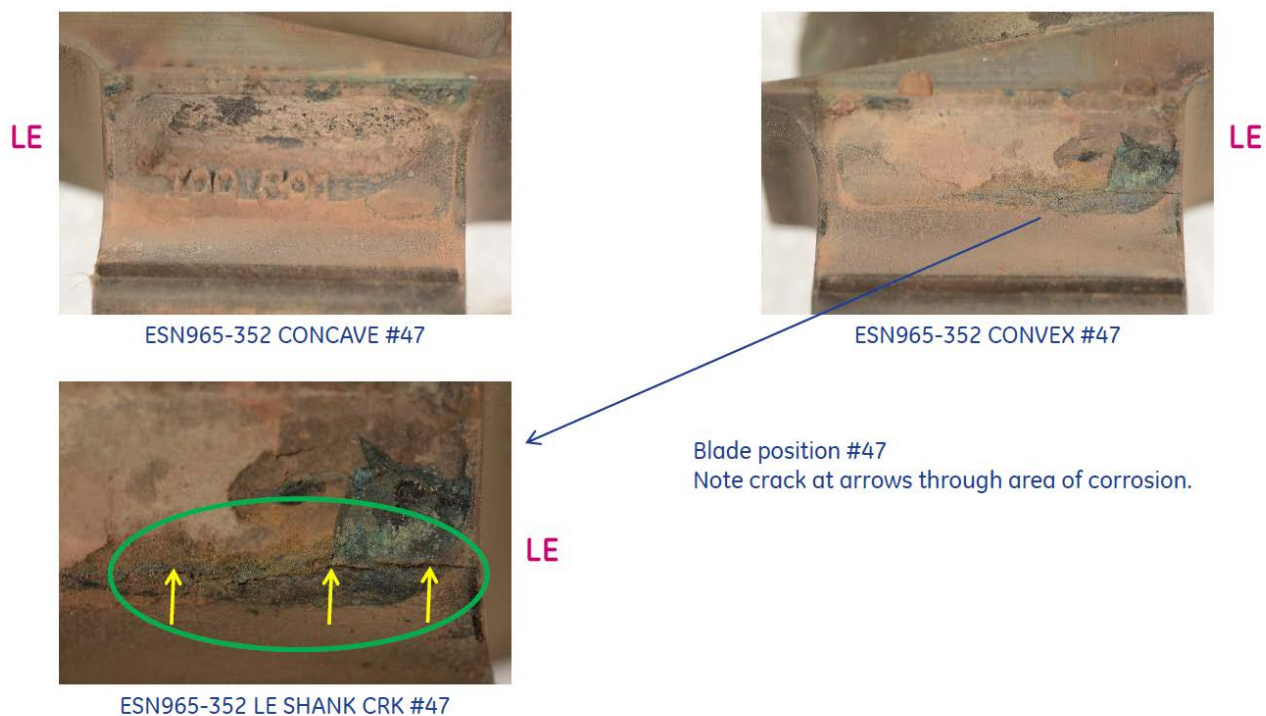


Photo 14- Below Platform Crack on a Blade Separated Above the Platform

Energy dispersive spectroscopy (EDS) analysis of the fracture of one of the blades separated below the platform (Blade #22) confirmed that the blade material met drawing specifications. In addition to the base material, EDX also identified the presence of silicon (Si), aluminum (Al), Calcium (Ca), Magnesium (Mg), Sulfur (S), Sodium (Na), Potassium (K), and Phosphorus (P). The presence of S, Na, K, and P are consistent with corrosion. The complete lab report is available in the docket under “GE Aviation CF34-8 ESN 965352 HPT S2B Under-Platform Separation.”

The LPT case was sent to the GE material lab in Evendale, Ohio for examination. The material around the case penetration was examined with a scanning electron microscope (SEM) and images confirmed that the material had ductile dimples consistent with event-related shear/overstress. EDS analysis confirmed the case material was consistent with drawing specifications and identified material transfer around the penetration that was consistent with HPT 2nd stage blade material. Case hardness measurements met drawing specification. The LPT case report is available in the docket under “Metallurgical Evaluation of a CF34-8C LPT Case from ESN 965352 Operated by GoJet.”

The NTSB materials laboratory reviewed both GE materials laboratory reports and concurred with all findings.

4.0 GENERAL ELECTRIC CF34 HPT 2ND STAGE BLADE SEPARATION BACKGROUND

According to the GE CF34 program, between 2010 and 2016, 27 CF34-3 and CF34-8 series HPT 2nd stage rotor blade failure events were reported. All separated blades were identified as part number 4125T01, including this event. The blade failure on ESN E965352 was the first event where a blade penetrated the LPT case, requiring NTSB notification.

On April 29, 2010, an upgraded HPT 2nd stage blade (part number 4125T65) was released as two category 7 service bulletins (SB), SB 72-0119 for CF34-8E engines, and SB 72-0228 for CF34-8C engines. The upgraded blade features minor geometry improvements and a corrosion preventative coating. The SBs were upgraded to CAT 3 on February 27, 2013 after multiple failures of the 4125T01 blade.

GE performed a risk assessment to determine the probability of a dual inflight shutdown due to HPT 2nd stage blade failure and estimated the probability to be 1.19 E-10. The generally recognized threshold for action is 1.0E-09.

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