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Office of Aviation Safety
Washington, DC 20594



ENG24LA014

POWERPLANTS

Group Chair's Factual Report

July 25, 2025

TABLE OF CONTENTS

TABLE OF CONTENTS.....	2
A. TABLE OF FIGURES.....	3
B. TABLE OF ACRONYMS	5
C. INCIDENT	7
D. ENGINE EXAMINATION GROUP	7
E. SUMMARY	9
F. DETAILS OF THE INVESTIGATION	13
1.0 Airplane and Engine Descriptions.....	13
1.1 Airplane Nacelle and Nacelle Fire Detection.....	13
1.2 Nacelle Fire Detection System	15
1.3 Engine Description	17
1.4 High Pressure Turbine Section Description	19
2.0 No. 1 (Left) Engine - Maintenance and Assembly Information.....	20
3.0 Engine Flight Data and Timeline	21
3.1 GE Evaluation of Recorded Data Sources	21
3.2 Flight Data Recorders.....	22
3.3 Event Timeline.....	22
4.0 Airframe and Nacelle - Documentation and Examination	26
4.1 On Scene Airframe and Nacelle Damage	26
4.2 Examination of Nacelle Damage	27
5.0 Engine Documentation and Examination.....	30
5.1 On Scene Engine Damage	30
5.2 Initial Workscope Performed by GE Aerospace	35
5.3 As Received Documentation by GE Aerospace	36
5.4 Initial Disassembly by GE Aerospace	39
5.5 Engine Examination - June 24-28, 2024	41
6.0 GE Aerospace Fire Assessment.....	62
7.0 Metallurgical Analysis.....	62
7.1 Detailed Analysis & Material Testing.....	67
8.0 Related Events - HPT Stage 1 Aft Blade Retainer (Current Configuration)....	69

8.1	Summary of Operational Event No. 1	70
8.2	Summary of Opportunistic Find at GE Caledonian	70
8.3	Summary of UAL Failed HPT Stage 1 Aft Blade Retainer Information	71
9.0	HPT Stage 1 Aft Blade Retainer (Previous Configuration)	71
9.1	HPT Stage 1 Aft Blade Retainer Related Events	71
9.2	Design/Configuration History	73
10.0	Containment Actions by GE Aerospace	74
10.1	GE Fleet Analytic Tool	74
10.1	Service Bulletins GEnx-1B-72-0546 and GEnx-2B-72-0475	75
10.2	Service Bulletins GEnx-1B 72-0551 and GEnx-2B 72-0478.....	76
10.3	Long Term Containment Plan	76
G.	ATTACHMENTS.....	77

A. TABLE OF FIGURES

Figure 1.	ENGINE NACELLE NOMENCLATURE	13
Figure 2.	ENGINE NACELLE NOMENCLATURE AND LOCATION	14
FIGURE 3.	FIRE/OVERHEAT DETECTOR LOCATION - LEFT SIDE	16
FIGURE 4.	FIRE/OVERHEAT DETECTOR LOCATION - RIGHT SIDE	16
FIGURE 5.	LPT CASE MOUNTED FIRE/OVERHEAT DETECTOR LOCATIONS	17
FIGURE 6.	ENGINE MODULE IDENTIFICATION	18
FIGURE 7.	CONFIG 1 HIGH PRESSURE TURBINE AREA - PART NOMENCLATURE.....	19
FIGURE 8.	HIGH PRESSURE TURBINE ACTIVE CLEARANCE CONTROL MANIFOLD	20
Figure 9.	LEFT HAND INBOARD FLAPERON DAMAGE - LOWER SKIN.....	26
Figure 10.	No. 1 Engine - Outboard Damage	27
Figure 11.	Outboard Thrust Reverser	28
Figure 12.	Lodged Debris in Thrust Reverser	29
Figure 13.	Punctured TR IFS.....	30
Figure 14.	Front View of Fan Blades	31
Figure 15.	NO. 1 ENGINE REMOVED - HPT CASE EXIT HOLE	31
Figure 16.	Damage to Heat Shielding and HPT ACC Ducts	32
Figure 17.	BSI - HPT Stage 1 Blades.....	33
Figure 18.	STAGE 1 HPT AFT BLADE RETAINER FRACTURED AND MISSING.....	34
Figure 19.	Engine Fan Module Removed.....	35
Figure 20.	ENGINE AS-RECEIVED, LEFT SIDE.....	36
Figure 21.	ENGINE AS-RECEIVED, RIGHT SIDE.....	37
Figure 22.	LEFT HAND AND LOWER CENTER FIRE LOOPS	38
Figure 23.	Right HAND AND LOWER CENTER FIRE LOOPS.....	38

Figure 24. HPT CASE EXIT HOLE	39
Figure 25. THRUST LINKS AND REAR ENGINE MOUNTS.....	40
Figure 26. HPT Case with Exit Holes	41
Figure 27. Engine Propulsor Data Plate	42
Figure 28. Engine Hardware Removed	42
Figure 29. BOXED HARDWARE, PORTIONS OF HPT STAGE 1 AFT BLADE RETAINER	43
Figure 30. Engine Propulsor Upon Arrival	44
Figure 31. LPT - IMPACT DAMAGE AND LOSS OF MATERIAL.....	45
Figure 32. Damaged HPTACC Ducts and HPC Stage 4 Cooling Manifold.....	46
Figure 33. FLANGE LOCATIONS.....	47
Figure 34. TCF Removed - Strut Impact Damage	48
Figure 35. AFT SIDE OF TCF	49
Figure 36. HPT Case Removed	50
Figure 37. HPT Stage 1 Nozzle Segments and Combustion Chamber	51
Figure 38. NO. 4 ROLLER BEARINGS AND HPT AFT SEAL.....	52
Figure 39. HPT Rotor Assembly - HPT Case Removed	53
Figure 40. HPT STAGE 2 BLADE DAMAGE.....	54
Figure 41. HPT Stage 1 Blades - Installed.....	55
Figure 42. HPT STAGE 1 BLADES REMOVED - SHORTEST AND LONGEST	56
Figure 43. HPT FORWARD SEAL.....	57
Figure 44. HPT FORWARD SEAL - DISASSEMBLY DAMAGE.....	57
Figure 45. HPT STAGE 2 DISK	58
Figure 46. BLOCKED HPT STAGE 1 DISK AND HPT RIS	59
Figure 47. HPT RIS AND HPT Stage 1 AFT BLADE RETAINER.....	60
Figure 48. HPT RIS AND HPT Stage 1 AFT BLADE RETAINER	61
Figure 49. Stage 1 Disk.....	62
Figure 50. HPT Stage 1 Aft Blade Retainer - Cross Sectional View	63
Figure 51. HPT Stage 1 Aft Blade Retainer.....	64
Figure 52. HPT Aft Blade Retainer Fracture Surfaces.....	65
Figure 53. Inboard Side - Intergranular Fracture (L), OCNC Particle Cluster (R).....	65
Figure 54. Close Up of OCNC Particle Cluster	66
Figure 55. Outboard Side - Crack Origin (L), Rabbet Cyclic Damage (R).....	67
Figure 56. Initial Fault Tree Diagram.....	68
Figure 57. HPT Stage 1 Aft Blade Retainer - Current & Past Fracture Locations.....	72
Figure 58. HPT Rotor Assembly (Configuration 2)	74

B. TABLE OF ACRONYMS

°C	Degree Celsius
°F	Degree Fahrenheit
AAIB	Aircraft Accident Investigation Bureau
ACARS	Aircraft Communication and Reporting System
ACC	Active Clearance Control
AGB	Accessory Gearbox
ALF	Aft Looking Forward
BSI	Borescope Inspection
BTU	British Thermal Unit
CDN	Combustion Discharge Nozzle
CEOD	Continuous Engine Operating Data
CFR	Code of Federal Regulations
CID	Change in Design
CNR	Customer Notification Report
CRES	Corrosion Resistant Steel
CPL	Continuous Parameter Logging
CSLSV	Cycles Since Last Shop Visit
CSN	Cycle Since New
CU	Cockpit Unit
EAFR	Enhanced Airborne Flight Recorder
EDS	Energy-dispersive X-ray spectroscopy
EEC	Electronic Engine Control
EGAT	Evergreen Aviation Technologies Corporation
EGT	Exhaust Gas Temperature
EICAS	Engine Indicating and Crew Alerting System
EMU	Engine Monitoring Unit
ESN	Engine Serial Number
ETH	Ethiopian Airlines
FAA	Federal Aviation Administration
FADEC	Full Authority Digital Control
FCD	Fuel Couple Dynamics
FF	Fuel Flow
FL	Flight Level
FMU	Fuel Management Unit
FOAS	Forward Outer Air Seal
FODS	Fire and Overheat Detection System
GE	GE Aerospace
GVI	General Visual Inspection
HP	High Pressure
HPC	High Pressure Compressor
HPT	High Pressure Turbine

HPTACC	High Pressure Turbine Active Clearance Control
IFS	Inner Fixed Structure
IFSD	inflight shutdown
IPC	Illustrated Parts Catalog
lb.	Pounds
lbs./hr.	Pounds per Hour
ksi	Kilopounds per Square Inch
LPC	Low Pressure Compressor
LPT	Low Pressure Turbine
LPTACC	Low Pressure Turbine Active Clearance Control
MEDC	Main Engine Data Concentrator
min	Minute
MSD	Mass Storage Device
N1	Low Pressure Rotor Speed (fan speed)
N2	High Pressure Rotor Speed (core speed)
NTSB	National Transportation Safety Board
OCNC	Oxide/Carbonitride/Nitride/Carbide
OWS	On-Wing Support
PEEK	Polyether Ether Ketone
PFPS	Propulsion Fire Protection System
PIP	Performance Improvement Package
psia	Pounds Per Square inch (absolute)
QAR	Quick Access Recorder
RIS	Rotating Interstage Seal
R-ratio	Minimum Stress to Maximum Stress
rpm	revolutions er minute
SB	Service Bulletin
sec	Second
SEM	Scanning Electron Microscopy
SPLCF	Sustained Peak Low Cycle Fatigue
TCDS	Type Certificate Data Sheet
TCF	Turbine Center Frame
TR	Thrust Reverser
TRA	Throttle Resolver Angle
TRF	Turbine Rear Frame
TSLSV	Time Since Last Shop Visit
TSN	Time Since New
VFSG	Variable Frequency Starter Generator
VHF	Very High Frequency

C. INCIDENT

Location: Singapore
Date: May 14, 2024
Time: 14:30 UTC
Airplane: United Airlines Boeing 787-9, Registration N27957

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E. SUMMARY

On May 14, 2024, a United Airlines (UAL) Boeing 787-9 airplane, serial number 36409, registration number N27957, powered by two GE Aerospace GEnx™-1B76A/P2 turbofan engines, experienced an uncontained high pressure turbine engine failure and inflight shutdown (IFSD) of the No. 1 (left) engine shortly after reaching their assigned flight altitude of FL310 following departure from the Singapore Changi Airport, Singapore. The airplane was operated as a Title 14 Code of Federal Regulations (CFR) Part 121 air carrier flight as a regularly scheduled passenger flight from the Singapore Changi Airport, Singapore, to the San Francisco International Airport, San Francisco, California (SFO).

The flight crew reported hearing a pop sound and felt the airplane shudder, followed by the annunciation of a No. 1 engine EICAS (Engine Indicating and Crew Alerting System) message for left engine Fire Warning. The flight crew pulled the No. 1 engine fire handle, secured the engine, and performed an inflight air turnback to Singapore Changi Airport where an uneventful single engine landing was performed. There were no reported injuries.

Upon landing, ground inspection of the No.1 engine revealed damage to the airplane, missing material from the left side thrust reverser cowl and a penetration of the engine's HPT case. The damage to the airplane and engine was documented by representatives from UAL and GE Aerospace (GE). The initial on-scene engine examination found two exit holes in the turbine case, centered at approximately the 9:00 o'clock position, the HPT stage 1 blades and stage 2 stators could be seen through the exit holes, and it appeared the HPT stage 1 aft blade retainer was missing material. There was no evidence of an under-cowl fire but localized heat distress was observed on electrical wiring harnesses, fire loop grommets/isolators and some of the HPT active clearance control (HPTACC)/low pressure turbine active clearance control (LPTACC) rubber bellows. Examination of the left side thrust reverser cowl found a large section of material, approximately 3 feet by 6 feet in size, missing from the translating sleeve and metal debris was found lodged into the inner diameter of the thrust reverser. The thrust reverser inner fixed structure contained an exit hole, approximately 13 inches by 15 inches in size, centered at about the 8:30 o'clock position.

The engine was removed from the airplane, the fan module removed from the engine, and the engine propulsor prepared for shipment to the United States for disassembly and examination. The left side thrust reverser cowl, various other components and recovered metal debris were also shipped to GE for further examination.

Since this event took place in international waters outside the territory of a State, the National Transportation Safety Board (NTSB) led the investigation as the State of Registry in accordance with ICAO Annex 13 guidelines. Additional parties to the

investigation include the Federal Aviation Administration (FAA), Boeing, United Airlines, and GE Aerospace.

The event engine was comprised of a UAL owned fan module and a GE Aerospace leased propulsor module (core/combustor/turbine assembly) that entered service in 2014 and was mated to the fan module for installation on the event airplane on May 2, 2023. Prior to the event, the engine propulsor received a performance improvement upgrade in 2016 during a scheduled shop visit and two additional shop visits in 2019 for high pressure turbine (HPT) distress. The 2016 performance upgrade included installation of new hardware that included the HPT stage 1 disk, HPT stage 1 aft blade retainer, HPT interstage seal, HPT stage 2 disk, HPT forward seal, HPT stage 2 blade retainer, and HPT aft air seal. The two engine shop visits in 2019 were due to distress in the propulsor's hot section (HPT and combustor); in both cases the HPT rotor assembly was partially disassembled to remove/replace various HPT components, but the HPT stage 1 aft blade retainer was not removed or replaced.

The airplane was equipped with an enhanced airborne flight recorder (EAFR); the data from the event flight was downloaded and a sequence of engine related events was created. The takeoff and initial climb to flight level (FL)310 was uneventful but at approximately 32 minutes after takeoff, the No. 1 engine performance parameters began to change. The fuel flow (FF) rate dropped significantly, the No. 1 bearing vibration increased to a peak of 7.56 cockpit units, a No. 1 engine failure was annunciated to the flight crew and then a No. 1 engine fire warning was annunciated. The flight crew cutoff the fuel to the No. 1 engine, discharged a single fire suppression bottle, and the airplane made an uneventful single engine landing. The entire flight from wheels up to wheels down was approximately 88 minutes.

The examination team, comprised of party members from GE, Boeing, FAA, UAL, and the NTSB, convened at the GE's, Evendale, Ohio facility to perform the engine disassembly and examination. Disassembly of the engine found significant hard body damage and missing material throughout the HPT module. The HPT case contained two exit holes, and the case wall was distorted outward. There were no signs of fuel nozzle streaking, hot spots or burn throughs in the combustion chamber. Multiple HPT stage 1 nozzle segments were found disengaged. All HPT stage 1 and stage 2 blades were fractured transversely across the airfoils at various span lengths. Downstream damage to the low pressure turbine (LPT) module included one fractured LPT stage 1 blade but the LPT stage 7 blades appeared undamaged.

The HPT stage 1 aft blade retainer was found lodged in place on the forward side of the HPT rotating interstage seal (RIS) and fractured, with the outboard portion of the retainer arm missing. The HPT stage 1 aft blade retainer and the HPT RIS were transported to GE's material systems laboratory, separated, and the retainer further examined.

The HPT stage 1 aft blade retainer presented a 360-degree circumferential fracture surface adjacent to the rabbet undercut fillet and the outboard retainer arm was missing. Fractography revealed the crack initiated in the upper collet radius at a single point of origin which corresponded to a surface connected anomaly that measured 0.005 inches by 0.015 inches by 0.0005 inches. The surface connected anomaly was determined to be a cluster of particles enriched with Titanium, Aluminum, Zirconium, Magnesium, and Oxygen, which is consistent with a cluster of Oxide/Carbonitride/Nitride/Carbide (OCNC) particles. The fracture surface contained an area of intergranular fracture morphology with visible beach marks, consistent with hold-time fatigue growth and the remaining fracture surface was consistent with overload fracture.

The investigation of this retainer supplements the ongoing investigation into two other cracked HPT stage 1 aft blade retainers; one was an opportunistic find during a scheduled maintenance activity and the other was an in flight failure that led to an uncontained failure event similar to the UAL event covered within this report. A fault tree was established to consider various aspects of the component design, material, and operational factors to identify possible causes and contributing factors.

GE's analysis of material test data suggests that surface connected OCNCs have a reduced crack incubation/nucleation life at high stress conditions, and that surface connected OCNCs do not behave according to traditional crack growth threshold behavior. Due to the difference between the original calculated life capability (e.g. crack initiation) and the observed operational cycles, GE performed a material sensitivity study to assess the sensitivity of the collet feature in the HPT stage 1 aft blade retainer to various parameters identified within the fault tree. The event HPT stage 1 aft blade retainer is manufactured from Rene 65, a wrought nickel-base superalloy. GE initiated material test plans to explore the Rene 65 material behavior with focus on low cycle fatigue (LCF) and sustained peak low cycle fatigue (SPLCF) capabilities, by using standard test bar specimens and coupons representative of the collet feature. The studies found that activating surface connected OCNCs can in fact reduce material fatigue life capability and that not every OCNC will initiate a crack, suggesting an unknown bimodal behavior.

The calculated stress conditions (high peak, high mean stress, high stress-ratio (minimum stress to maximum stress), and high temperature) for the GENx HPT stage 1 aft blade retainer collet feature were found to be uniquely high when compared to the stress conditions of other rotating hardware. This higher stress environment is the result of the assembly interference fits between the HPT stage 1 disk, the HPT stage 1 retainer, and the HPT RIS in combination with the operating loads. The collet feature was found to have a significant SPLCF debit (low crack growth life) when a hold-time fatigue condition was taken into account.

Multiple corrective actions have already been taken by GE as a result of this investigation and the findings from three other cracked HPT stage 1 aft blade retainers.

The corrective actions focused on inspection/removal of suspect HPT stage 1 aft blade retainers, and continuous fleet monitoring for specific shifts in core engine vibration trends. The short-term containment actions include service bulletins (SB) for the removal/inspection/replacement of the HPT stage 1 aft blade retainer at the piece part level, SBs for the ultrasonic inspection of the HPT stage 1 aft blade retainers at the module level, and the use of a data analytic tool to monitor core vibration trends in all GENx engines. At the time this report was nearing completion, there were at least 477 HPT stage 1 aft blade retainers removed from service for inspection and at least 55 HPT modules ultrasonically inspected, with zero crack findings to date.

A data analytic tool was created to perform continuous fleet monitoring and evaluation of core engine vibration trends for all engines within the GENx-1B and GENx-2B engine fleets. The data analytic tool was based on the engine performance/vibration data from the three engines that contained the cracked HPT stage 1 aft blade retainers, and the analytical rules established such that the core engine vibration shifts of these engines would have resulted in an alert for their removal. In the event the data analytic tool identifies a specific engine with a core vibration alert, a more thorough assessment would be performed by GE's engineering team on a case-by-case basis. In February 2025, the data analytic tool alerted on an operational engine, the engine was removed and a detailed ultrasonic inspection resulted in a positive crack indication.

The long-term corrective action being pursued by GE will re-establish the manufacturing process for the previously approved HPT stage 1 aft blade retainer configuration made of a Rene 88 alloy, a nickel-based superalloy powder metal. GE intends to make the Rene 88 retainer the primary configuration. The Rene 65 retainer will remain certified for use and listed in the illustrated parts catalog (IPC) but it will no longer be available as a spare part, further diminishing the population of the Rene 65 retainers through attrition.

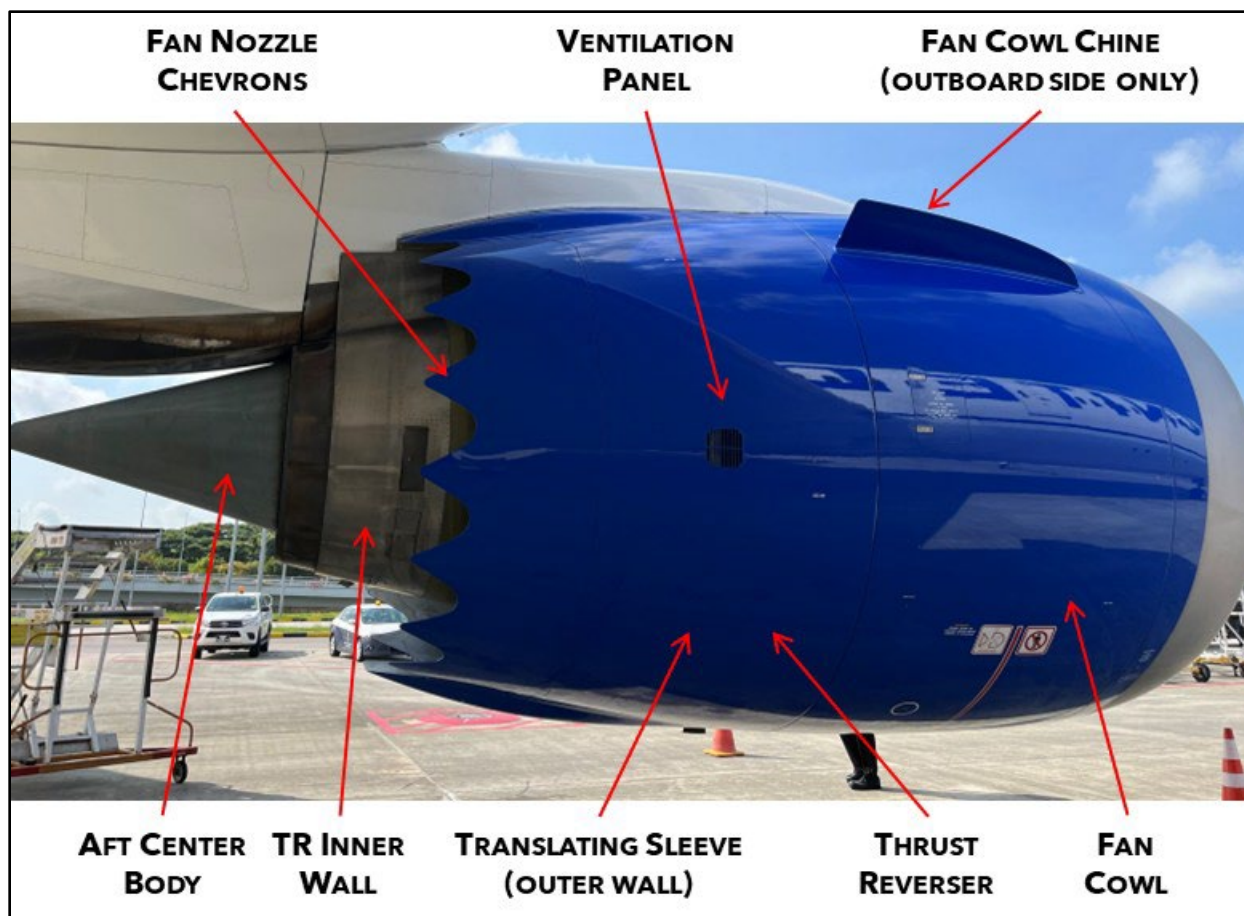


Figure 2. Engine Nacelle Nomenclature and Location

Courtesy of UAL, Modified by NTSB

The TR is a conventional translating sleeve/cascade design where each independent TR half (left and right) is a "C" duct configuration hinged at the engine pylon/strut and has a translating sleeve (outer wall). There are chevrons on the outer panel of the translating sleeves. The inner surface of the translating sleeves and the outer surface of the TR Inner Fixed Structure (IFS) forms the fan duct for the fan air exhaust. The rearward motion of the translating sleeves causes the blocker doors to move down into the fan air stream and block the fan airflow, redirecting it through circumferentially located forward turning vanes (known as cascades) which produce the reverse thrust. The stationary cascades are exposed as the translating sleeves move rearward.

The engine-side (hot-side) of the TR IFS is encased in overlapping thermal blankets for fire protection. Collins Aerospace is responsible for the nacelle package, including the inlet, fan cowl, TR, and exhaust. Boeing is responsible for the integration and installation of the nacelle package on the airplane.

Although the turbine exhaust sleeve and the exhaust plug (exhaust plug is also commonly referred to as the exhaust centerbody) are attached to the engine, they are

both Boeing supplied components. The turbine exhaust sleeves are attached to the outer rear flange of the engine's turbine rear frame (TRF), and the exhaust plug is bolted to the TRF hub's aft flange. The turbine exhaust nozzle is comprised of an inner and outer sleeve, an aft ring at the location where the two skins meet in the back and a seal assembly that seals against the TR IFS. The exhaust components provide a smooth exit path for turbine exhaust. The exhaust sleeve and plug together form a convergent-divergent nozzle that aids in producing thrust.

1.2 Nacelle Fire Detection System

The Boeing 787-9 nacelle design incorporates a fire and overheat detection system (FODS), which consists of a series of detectors throughout the nacelle, each of which has two distinct trigger points and is monitored by the propulsion fire protection system (PFPS) software that notifies the flight crew of a fire/overheat event through alerting displays on the flight deck. Although these fire detectors are installed on the engine, they are Boeing supplied components.

Each engine is equipped with an independent fire detection system consisting of five dual loop detector assemblies that provide fire and overheat detection capability for the core compartment fire zone and are routed where fire/overheat conditions are most likely to occur or be detected, such as at fuel or ignition sources or at ventilation exits. The five dual detector assemblies are wired together in parallel forming two detection loops connected to the main engine data concentrator (MEDC). Each loop, designated as "Channel A" and "Channel B", is routed separately to the MEDC, allowing continuous fire detection in the fire zone even when one of the elements or channels fails. The FODS also has fault isolation capability and continuously monitors for short circuits, open circuits, detector faults, and reports failures as appropriate.

The pylon fire/overheat detector assembly is mounted to the bottom of the pylon strut and provides a prompt fire warning for combustor burn-through or fire in this area (Figure 3 and Figure 4). Three fire/overheat detector assemblies are mounted around the circumference of LPT case (Figure 5). The lower bifurcation detector assembly is mounted to the aft side of the lower splitter tube raceway at the bottom of the fire zone near the pressure relief door.

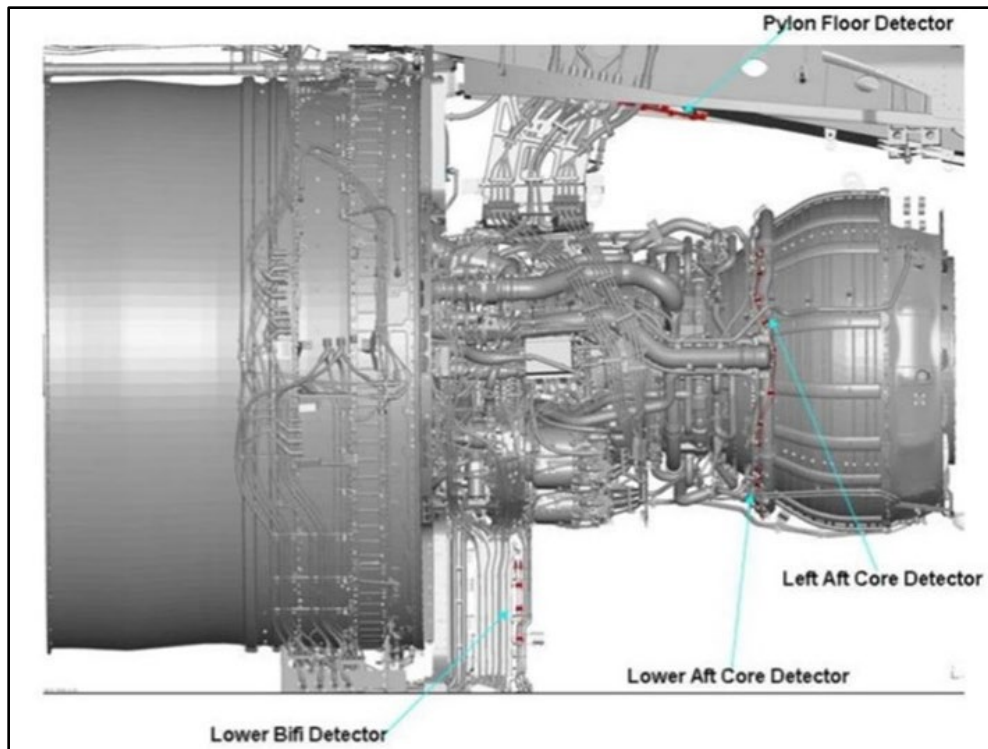


FIGURE 3. FIRE/OVERHEAT DETECTOR LOCATION - LEFT SIDE

Figure Courtesy of Boeing
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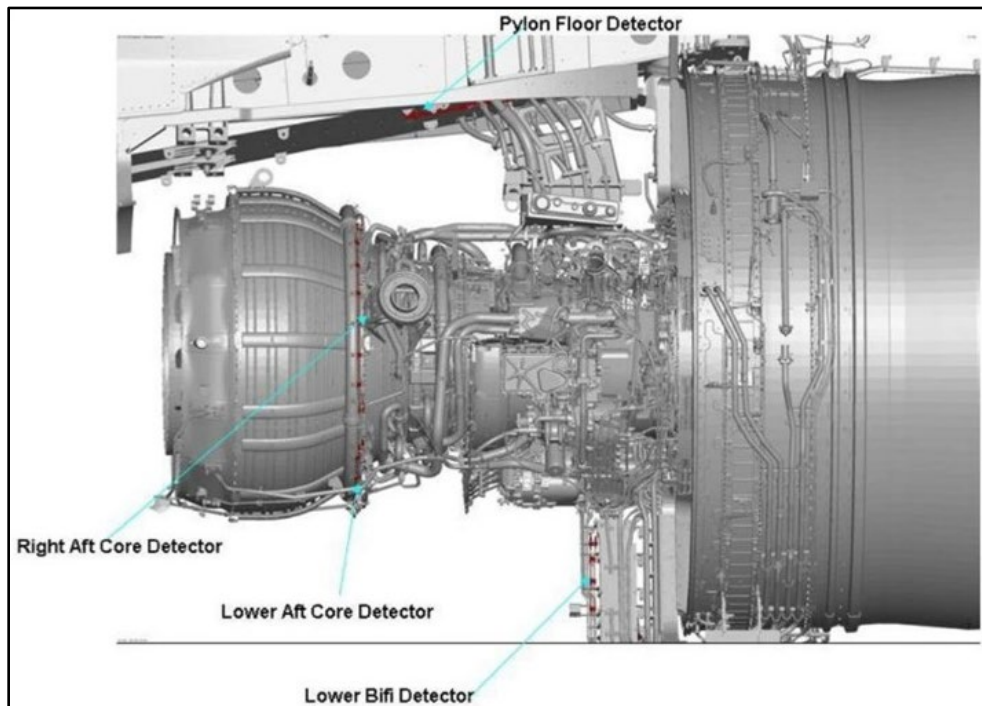


FIGURE 4. FIRE/OVERHEAT DETECTOR LOCATION - RIGHT SIDE

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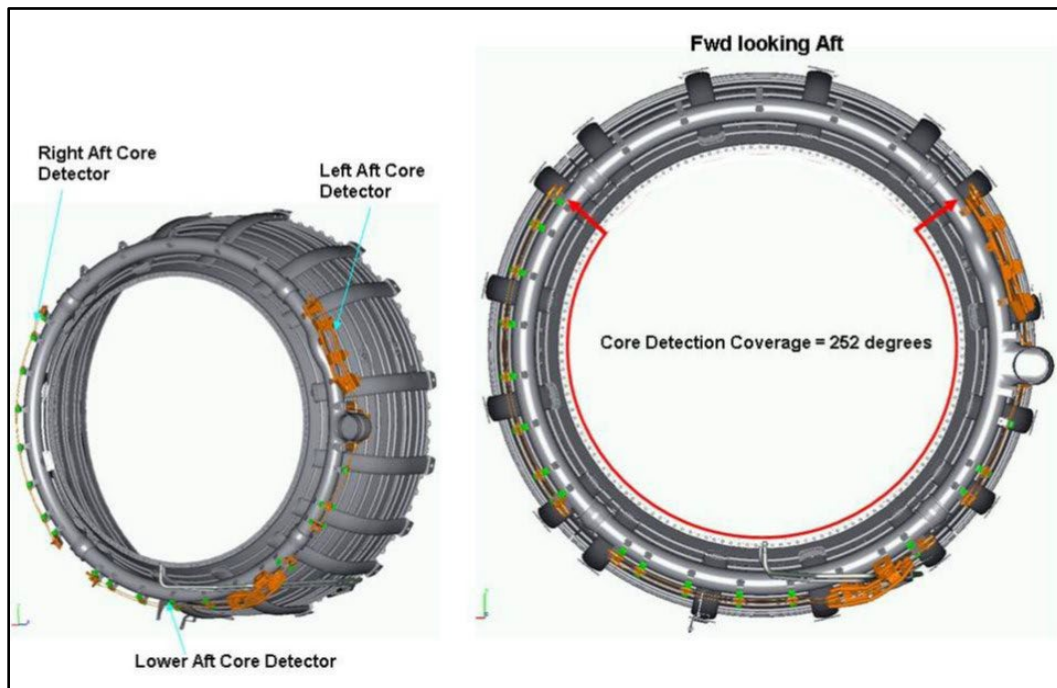


FIGURE 5. LPT CASE MOUNTED FIRE/OVERHEAT DETECTOR LOCATIONS

Figure Courtesy of Boeing
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1.3 Engine Description

The GEnx™-1B engine is a dual rotor, axial flow, high bypass ratio turbofan. The 10-stage high pressure compressor (HPC) is driven clockwise by a 2-stage high pressure turbine (HPT). The single stage fan and 4-stage low pressure compressor (LPC) are driven counterclockwise by a 7-stage low pressure turbine (LPT). The GEnx™-1B is constructed of 6 major modules: fan module, HPC module, combustor diffuser nozzle (CDN), HPT module, LPT module and accessory gearbox (AGB) module (Figure 6).

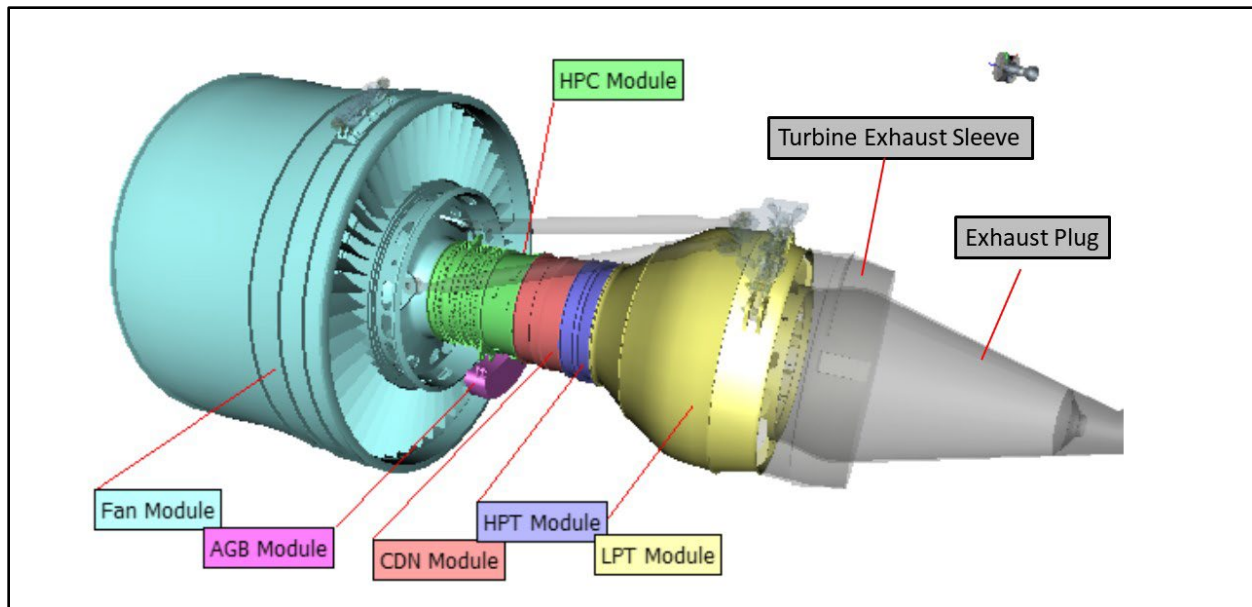


FIGURE 6. ENGINE MODULE IDENTIFICATION

Figure Courtesy of GE Aerospace, Modified by NTSB

The engine control system is essentially composed of an electronic engine control (EEC) also known as a full authority digital engine control (FADEC) that has an airplane connection for digital communication, and a fuel metering unit (FMU) along with other various engine sensors. The EEC controls the engine (operation, performance, and efficiency) in response to thrust command inputs from the airplane and provides information to the airplane for cockpit indication, maintenance reporting and engine condition monitoring. An engine monitoring unit (EMU) provides vibration level signals to the airplane along with engine parameters used to create the continuous engine operating data (CEOD) files.

According to the engine's FAA Type Certificate Data Sheet (TCDS) E00078EN¹, Revision 17, dated December 21, 2020, the GENx™-1B76A/P2 was issued a FAA type certificate on June 20, 2016. The engine has a maximum takeoff (5 minutes) at sea level static thrust rating of 78,500 pounds at a fan speed of 2,375 revolutions per minutes (rpm), flat-rated² to 91°F (32.8°C) and a maximum continuous at sea level static thrust rating of 68,600 pounds at a fan speed of 2,521 rpm flat-rated to 91°F (32.8°C).

¹ The attached engine data plate indicated a maximum takeoff thrust of 72,300 pounds and a maximum continuous rating of 66,500 pounds.

² Flat-rated to a specific temperature indicates that the engine will be capable of attaining the rated thrust level up to the specified inlet temperature. Engine ratings for the GENx™-1B76A/P2 turbofan are based on calibrated test stand performance under the following conditions: 1) sea level static, standard pressure (14.696 pounds per square inch absolute (psia), up to the flat rating ambient temperature °F, 2) no customer bleed or customer horsepower extraction, 3) ideal inlet, 100% ram recovery, 4) production aircraft flight cowlings, 5) production instrumentation and 6) fuel lower heating value of 18,400 British Thermal Unit (BTU)/pound (lb.).

1.4 High Pressure Turbine Section Description

The HPT module consists of 2 stages of rotors, a high pressure turbine active clearance control (HPTACC), an inter-stage nozzle guide vane assembly (HPT stage 2 nozzle guide vane assembly), and cooling ports (Figure 7 and Figure 8). The HPT module case and integral HPT stage 2 nozzle assembly is installed between the compressor discharge nozzle (CDN) case aft flange and the turbine center frame (TCF) forward flange. The HPT stage 1 nozzle guide vane assembly is part of the CDN module and consists of 40 air-cooled nozzle vanes. Both HPT rotor stages have 62 air-cooled blades while the HPT stage 2 nozzle guide vane assembly consists of 42 air cooled nozzle vanes with 1 having a borescope port.

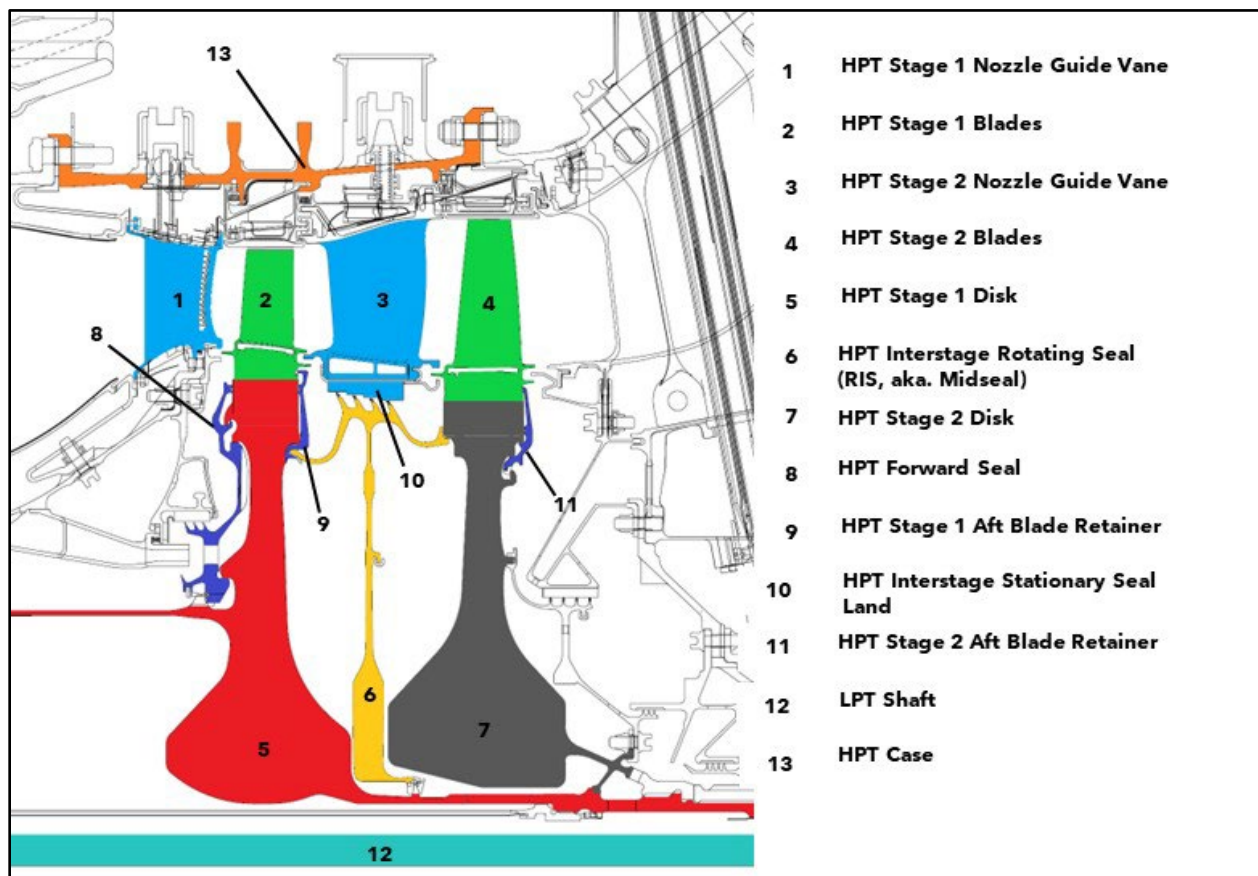


FIGURE 7. CONFIG 1 HIGH PRESSURE TURBINE AREA - PART NOMENCLATURE

Figure Courtesy of GE Aerospace, Modified by NTSB

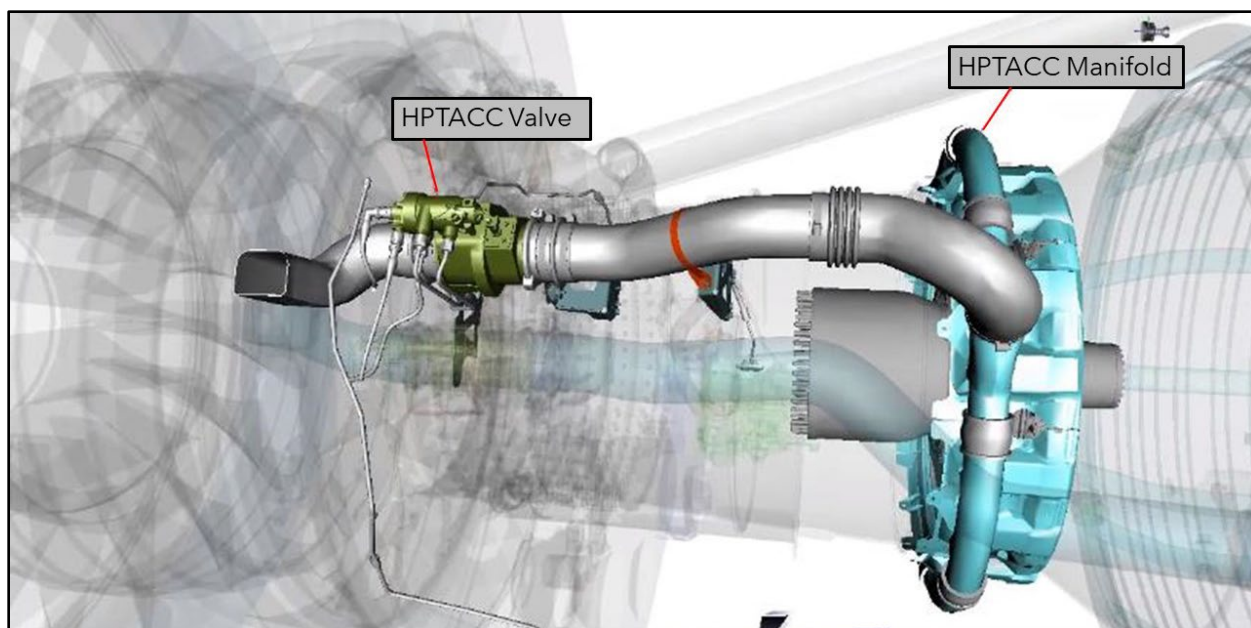


FIGURE 8. HIGH PRESSURE TURBINE ACTIVE CLEARANCE CONTROL MANIFOLD

Figure Courtesy of GE Aerospace, Modified by NTSB

2.0 No. 1 (Left) Engine - Maintenance and Assembly Information

The No. 1 engine installed on the incident airplane, registration number N27957, was a GEnx™-1B76A/P2 turbofan, engine serial number (ESN) 956102. According to GE, at the time of the engine failure, the propulsor module had accumulated 19,298 hours time since new (TSN), 3,320 cycles since new (CSN), 3,736 hours times since last shop visit (TSLSV), and 347 cycle since last shop visit (CSLSV).

According to information provided by GE, the incident engine was comprised of a United Airlines owned fan module and a GE Aerospace leased propulsor (core/turbine) module. The propulsor originally entered service in 2014 and was later mated to the fan module for installation on the event airplane on May 2, 2023. Prior to the incident, the engine propulsor had a performance improvement package (PIP) upgrade in 2016 during a shop visit at the GE Celma facility in Petropolis, Brazil and two additional shop visits in May 2019 and Dec 2019, both of which had high pressure turbine (HPT) maintenance performed.

The 2016 PIP upgrade shop visit involved installation of new hardware throughout the propulsor to upgrade the configuration from GEnx-1B67 to GEnx-1B74/75/P2. Specifically, the HPT rotor assembly was upgraded to meet the GEnx-1B/P2 configuration requirements, which included a new HPT stage 1 disk, HPT stage 1 aft blade retainer, HPT interstage seal, HPT stage 2 disk, HPT forward seal, HPT stage 2 blade retainer, and HPT aft air seal. The engine propulsor had accumulated 8,601 hours TSN and 1,530 CSN at the time of the conversion.

The May 2019 engine shop visit was due to fuel couple dynamics (FCD) distress with the propulsor's hot section (HPT and combustor). The propulsor was sent to GE On-Wing Support (OWS) in Korea for a quick turn shop visit to address the hot section distress. The HPT rotor assembly was partially disassembled to remove the HPT forward seal, HPT stage 1 blades, and HPT stage 2 blades. The remainder of the rotor assembly remained assembled, and a general visual inspection (GVI) was performed. A new set of HPT stage 1 blades, a mixed set of new and used HPT stage 2 blades, and an overhauled HPT forward seal were installed in the HPT rotor. The HPT rotor assembly and the HPT stage 2 nozzle assembly were sent to Evergreen Aviation Technologies Corporation (EGAT) for grind and balance repair processes that were not available at OWS in Korea. The engine had accumulated 15,115 hours TSN and 2,884 CSN at that time. The HPT stage 1 aft blade retainer had 1,314 CSN at that time.

The December 2019 shop visit was again due to FCD distress with the propulsor's hot section (HPT and combustor). The engine was sent to the GE Caledonian engine shop in Prestwick, Scotland for a hot section quick turn shop visit. The HPT rotor assembly was partially disassembled to remove the HPT forward seal, HPT stage 1 blades, HPT stage 2 blades, HPT aft seal, and the HPT stage 2 disk. The HPT stage 2 disk was exposed to the piece-part level to comply with GENx-1B Service Bulletin (SB) 72-0463 for immersion ultrasonic inspection. The remainder of the rotor assembly remained assembled, and a GVI was performed. The same HPT stage 2 disk, HPT aft seal, and HPT forward seal were reinstalled into the HPT rotor assembly along with new HPT stage 2 blades. The propulsor had accumulated 15,397 hours TSN and 2,973 CSN at that time. The HPT stage 1 aft blade retainer had 1,403 CSN at that time.

3.0 Engine Flight Data and Timeline

3.1 GE Evaluation of Recorded Data Sources

GE Aerospace evaluated the engine performance data of the incident engine based on various data sources; one source was engine specific data defined by GE Aerospace, which is provided by the EMU in two different types of data. One type of data is identified as "snapshot" data which is a single recording snapshot of engine data in time and is captured at predefined times commensurate with airplane takeoff conditions, climb conditions, and cruise conditions (a single snapshot at each condition for a total of three snapshots per flight). This snapshot data is recorded by the EMU, which is attached to the engine fan case, and is sent to the airplane where it is typically transmitted via ACARS (aircraft communication and reporting system) in near real-time to ACARS ground stations where it is then transferred to GE Aerospace for engine trend monitoring. The other type of data is referred to as CEOD that is recorded in the EMU buffer and when the buffer is full (512K data blocks) it is transferred to the airplane data storage server where it can be later downloaded.

GE Aerospace examined the CEOD data for the incident flight to better understand the engine's performance prior to the failure event. For the event flight, the CEOD data was contained in two files, the first file contained CEOD data beginning at engine startup to when the airplane reached initial cruise altitude, the second file contained CEOD data beginning when the airplane was at the top of the climb phase through to shortly after the engine blowout indication but is cutoff prior to the fire handle being pulled. Based on the CEOD data evaluation, GE Aerospace concluded that there was no significant core speed, vibration, or exhaust gas temperature (EGT) anomalies until just before the failure event. The time difference between the CEOD cut-off and the event was calculated using other data sources such as data from the airplane's EAFR. GE Aerospace used CEOD to compare the state of the event engine (ESN 956102) with the sister engine installed in the No. 2 (right) position; the No. 2 engine was ESN 958529.

3.2 Flight Data Recorders

The EAFR were removed from the airplane after the event and shipped to the NTSB's vehicle recorder division in Washington D.C. for data download and analysis. Refer to the recorder specialist's factual report that is available in the investigation docket for detailed information regarding EAFR data analysis.

3.3 Event Timeline

A timeline for the sequence of events was created from the data downloaded from the EAFR³ data. The EAFR data is logged every second for a variety of airplane and engine parameters such as fire warning, engine fuel cutoff, airspeed, altitude, engine vibration, engine fan speed (N1), engine core speed (N2), engine EGT, engine fuel flow (FF), and engine fire handle (extinguishing switch). Most parameters are not recorded at the exact same time or sampling rate, so they are time aligned with one another during each second of recorded data from these sources. As a result, some parameters may have changed state within fractions of a second of another parameter.

For this event timeline, times are rounded to the nearest whole second for simplicity. Time T = 0 is defined as the time when the No. 1 engine fire warning was annunciated (ENG-L fire warning); negative time is time prior to the fire annunciation and positive time is time after the fire annunciation. See Table 1 for details.

The takeoff and initial climb-to-altitude of flight level (FL)310 was uneventful with the engine parameters for both engines as expected and consistent with one another.

³ EAFR is equivalent to a flight data recorder (FDR) for the Boeing 787; it is a combined unit for both the FDR and cockpit voice recorder functions.

Approximately 32 minutes after takeoff, a noticeable change in No. 1 engine performance was noted. With no change in throttle position (throttle resolver angle (TRA)), the No. 1 engine FF rate dropped dramatically from about 9,609 pounds/hour (lbs./hr.) to 4,096 lbs./hr. The No. 1 bearing vibration (Eng1VibBBB⁴ - broadband from No. 1 bearing accelerometer) increased from 1.125 cockpit units (CU) to 1.875 CU, and both rotor speeds decreased from 88% to 65% N1 and 103% to 93% N2 respectively. Three seconds later (T = -3 seconds), the No. 1 engine failure was annunciated to the flight crew, No. 1 bearing vibration peaked at 7.56 CU⁵ in parallel with increased vibration across all vibe parameters, TRA increased from a steady 64.25° to 81° and both rotor speeds continued to drop with a corresponding drop in FF rate. Three seconds after that (T = 0 seconds), the No. 1 engine fire warning was annunciated to the crew, both engine rotors speeds and fuel flow rate continued to decrease, and all vibration levels remained elevated.

The flight crew cutoff the fuel to the No. 1 engine 29 seconds after the fire warning was annunciated; the warning lasted for an additional 31 seconds before the warning ceased. Four seconds after fuel was cutoff to the No.1 engine (T = +29 seconds), the No. 1 engine fire handle was pulled (T = +33 seconds) and twelve seconds later (T = +45 seconds) a single fire suppression bottle was discharged. The second No.1 engine fire suppression bottle was not discharged. The airplane made an uneventful and successful single engine landing approximately 56 minutes after the No. 1 engine fire warning annunciated; the entire flight from wheels up to wheels down was approximately 88 minutes.

Table 1. Incident Timeline

Table 1: Incident Timeline (All values relate to the No. 1 engine)				
FDR Data				
T = X	T = 0 is when ENG1 Fire Warning Illuminated			
Time (min/sec)	Comment & Airplane Parameters	Engine Parameters	Vibration Levels (Cockpit Unit)	
T = -32min	Approximate Time of Takeoff			
	Airspeed: 192 kn	ENG1 N1: 98%	Eng1VibBBA	1 CU
	Altitude: 43ft	ENG1 N2: 113%	Eng1VibBBB	1.125 CU
	Fuel Cutoff: Run	ENG1 TRA: 79	Eng1VibHW	1.6875 CU
		Fuel Flow: 22,156 lbs/hr	Eng1VibN1A	0.5 CU
		EGT: 996 C	Eng1VibN1B	1.6875 CU
			Eng1VibN2A	0.9375 CU

⁴ VibBBA - broadband from TCF accelerometer; 2) VibBBB - broadband from No. 1 bearing accelerometer; 3) VibHW - Eng 1/2 selected highest rotor vibration value; 4) VibN1A - N1 (LPT) from TCF Accelerometer; 5) VibN1B - N1 (FAN) from No. 1 Bearing Accelerometer, 6) VibN2A - N2 (N2) from TCF Accelerometer, and 7) VIBN2B - N2 (N2) from No. 1 Bearing Accelerometer.

⁵ On earlier airplane models, engine vibration levels were displayed/recorded with a maximum value of 5 CU. The Boeing 787 records a maximum value of 10 CU; the vibration values are supplied by the EMU.

			Eng1VibN2B	0.625 CU
T = -3min	Nominal Climb			
	Airspeed: 314 kn	ENG1 N1: 100%	Eng1VibBBA	0.8125 CU
	Altitude: 30,700ft	ENG1 N2: 109%	Eng1VibBBB	1.125 CU
	Fuel Cutoff: Run	ENG1 TRA: 71	Eng1VibHW	1.25 CU
		Fuel Flow: 10,585 lbs/hr	Eng1VibN1A	0.6875 CU
		EGT: 951 C	Eng1VibN1B	1.25 CU
			Eng1VibN2A	0.75 CU
			Eng1VibN2B	0.875 CU
T = -69s	Nominal Cruise			
	Airspeed: 312 kn	ENG1 N1: 91%	Eng1VibBBA	0.875 CU
	Altitude: 31,000ft	ENG1 N2: 108%	Eng1VibBBB	1.0625 CU
	Fuel Cutoff: Run	ENG1 TRA: 62	Eng1VibHW	0.875 CU
		Fuel Flow: 7,657 lbs/hr	Eng1VibN1A	0.375 CU
		EGT: 836 C	Eng1VibN1B	0.8125 CU
			Eng1VibN2A	0.8125 CU
			Eng1VibN2B	1 CU
T = -9s	One Second Prior to Vibe increase			
	Airspeed: 313 kn	ENG1 N1: 88%	Eng1VibBBA	0.875 CU
	Altitude: 30,982ft	ENG1 N2: 103%	Eng1VibBBB	1.125 CU
	Fuel Cutoff: Run	ENG1 TRA: 69	Eng1VibHW	0.875 CU
		Fuel Flow: 9,609 lbs/hr	Eng1VibN1A	0.5 CU
		EGT: 957 C	Eng1VibN1B	0.875 CU
			Eng1VibN2A	0.8125 CU
			Eng1VibN2B	1 CU
T = -8s	Vibe begins to increase			
	Airspeed: 313 kn	ENG1 N1: 65%	Eng1VibBBA	2.4375 CU
	Altitude: 30,983 ft	ENG1 N2: 93%	Eng1VibBBB	1.875 CU
	Fuel Cutoff: Run	ENG1 TRA: 70	Eng1VibHW	1.25 CU
		Fuel Flow: 4,096 lbs/hr	Eng1VibN1A	1.25 CU
		EGT: 1,084 C	Eng1VibN1B	0.9375 CU
			Eng1VibN2A	0.8125 CU
			Eng1VibN2B	1.125 CU
T = -5s	ENG_FAIL_L Warning: TRUE			
	Airspeed: 313 kn	ENG1 N1: 38%	Eng1VibBBA	4.125 CU
	Altitude: 31,019 ft	ENG1 N2: 78%	Eng1VibBBB	5 CU
	Fuel Cutoff: Run	ENG1 TRA: 79	Eng1VibHW	1.375 CU
		Fuel Flow: 889 lbs/hr	Eng1VibN1A	1.375 CU

		EGT: 979 C	Eng1VibN1B	0.6875 CU
			Eng1VibN2A	1.0625 CU
			Eng1VibN2B	3.875 CU
T = -2s	Peak BBB Vibe Airspeed: 312 kn Altitude: 31,021 ft Fuel Cutoff: Run	ENG1 N1: 30% ENG1 N2: 65% ENG1 TRA: 81 Fuel Flow: 792 lbs/hr EGT: 874 C	Eng1VibBBA Eng1VibBBB Eng1VibHW Eng1VibN1A Eng1VibN1B Eng1VibN2A Eng1VibN2B	1.25 CU 7.5625 CU 1.9375 CU 1.9375 CU 0.8125 CU 0.0625 CU 6.4375 CU
T = 0s	ENG-L Fire Warning: True Airspeed: 311 kn Altitude: 31,017 ft Fuel Cutoff: Run	Warning persisted for 60 sec ENG1 N1: 29% ENG1 N2: 60% ENG1 TRA: 75 Fuel Flow: 427 lbs/hr EGT: 896 C	Eng1VibBBA Eng1VibBBB Eng1VibHW Eng1VibN1A Eng1VibN1B Eng1VibN2A Eng1VibN2B	1.4375 CU 7.0625 CU 2.4375 CU 2.4375 CU 0.625 CU 0.0625 CU 6.625 CU
T = 29s	Eng1 Fuel Cutoff: False to TRUE Eng1 Fuel Vlv Open: True to False Airspeed: 299 kn Altitude: 30,975 ft	ENG1 N1: 27% ENG1 N2: 36% ENG1 TRA: 34 Fuel Flow: 499 lbs/hr EGT: 819 C	Eng1VibBBA Eng1VibBBB Eng1VibHW Eng1VibN1A Eng1VibN1B Eng1VibN2A Eng1VibN2B	0.125 CU 4.625 CU 0.5 CU 0.5 CU 0.0625 CU 0 CU 4.625 CU
T = 33s	Eng1_Fire_Sw_Pulled			
T = 45s	BOTTLE_1_DISCH_ENG, 45 seconds after fire warning			
T = 60s	ENG-L Fire Warning: False			
T = 56min	Approximate time the airplane lands			

4.0 Airframe and Nacelle - Documentation and Examination

4.1 On Scene Airframe and Nacelle Damage

The following on-scene information was gathered and provided to the NTSB by UAL and GE Aerospace in-country service representatives.

The on ground inspection of the airplane found the left hand inboard flaperon exhibited two penetration holes through the lower skin/surface (Figure 9), where penetration number 1 is approximately 2.6 inches and penetration number two was approximately 1.6 inches in length.

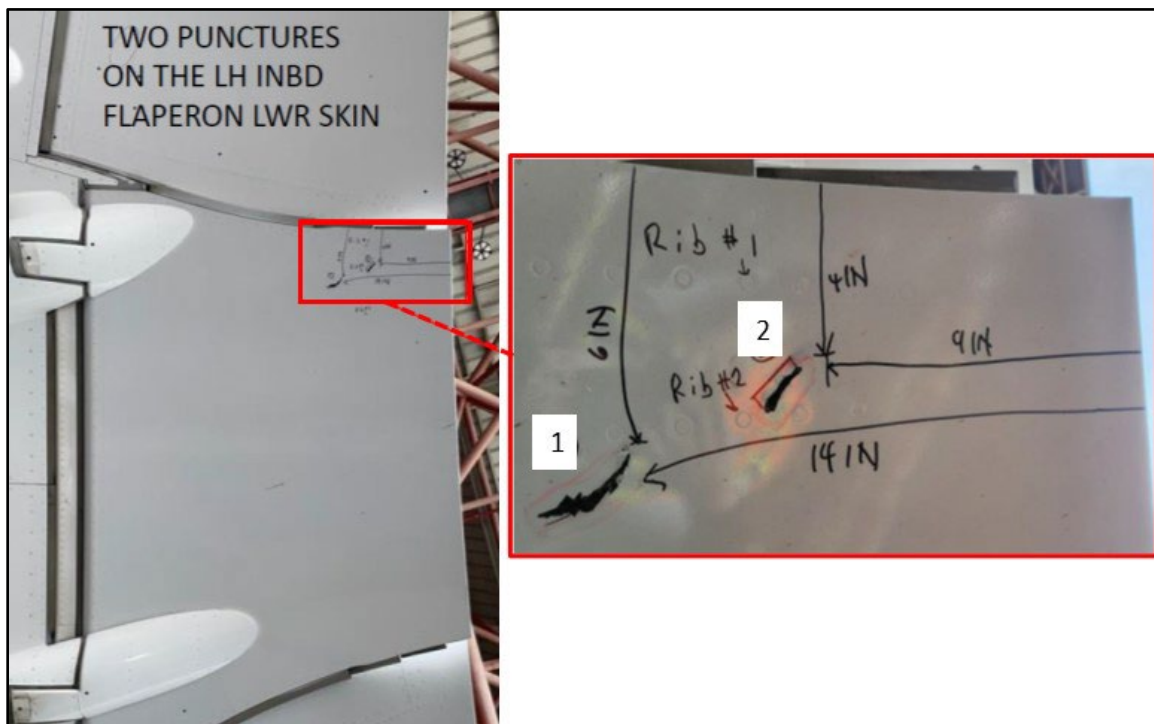


Figure 9. LEFT HAND INBOARD FLAPERON DAMAGE - LOWER SKIN

Photos Courtesy of UAL, Modified by NTSB

Damage and penetration of the outboard TR IFS was found at approximately the 10:00 o'clock position (Figure 10) and the outboard (left) side of the TR (translating sleeve) was missing material between approximately the 8:00 and 11:00 o'clock positions.



Figure 10. No. 1 Engine - Outboard Damage

Photos Courtesy of UAL, Modified by NTSB

4.2 Examination of Nacelle Damage

The investigative team made up party members from UAL, GE Aerospace, Boeing, FAA, and the NTSB convened on June 24, 2024, at the GE's Evendale, Ohio facility to continue with damage documentation and examination of the propulsor module and the outboard TR; documentation and examination were completed on June 28, 2024.

The outboard (left side) TR was missing material from the translating sleeve, with dimensions of approximately 72 inches circumferentially and 36 inches axially, located approximately 21 inches from the top edge of the TR and 65 inches from forward edge of the TR (Figure 11). Forward of the TR missing material was an area of additional deformation to the outer TR skin which was opposite of the lodged debris located on the inner side of the TR. The TR skin puncture appeared pushed outward but did not appear to be a full penetration of the TR skin.

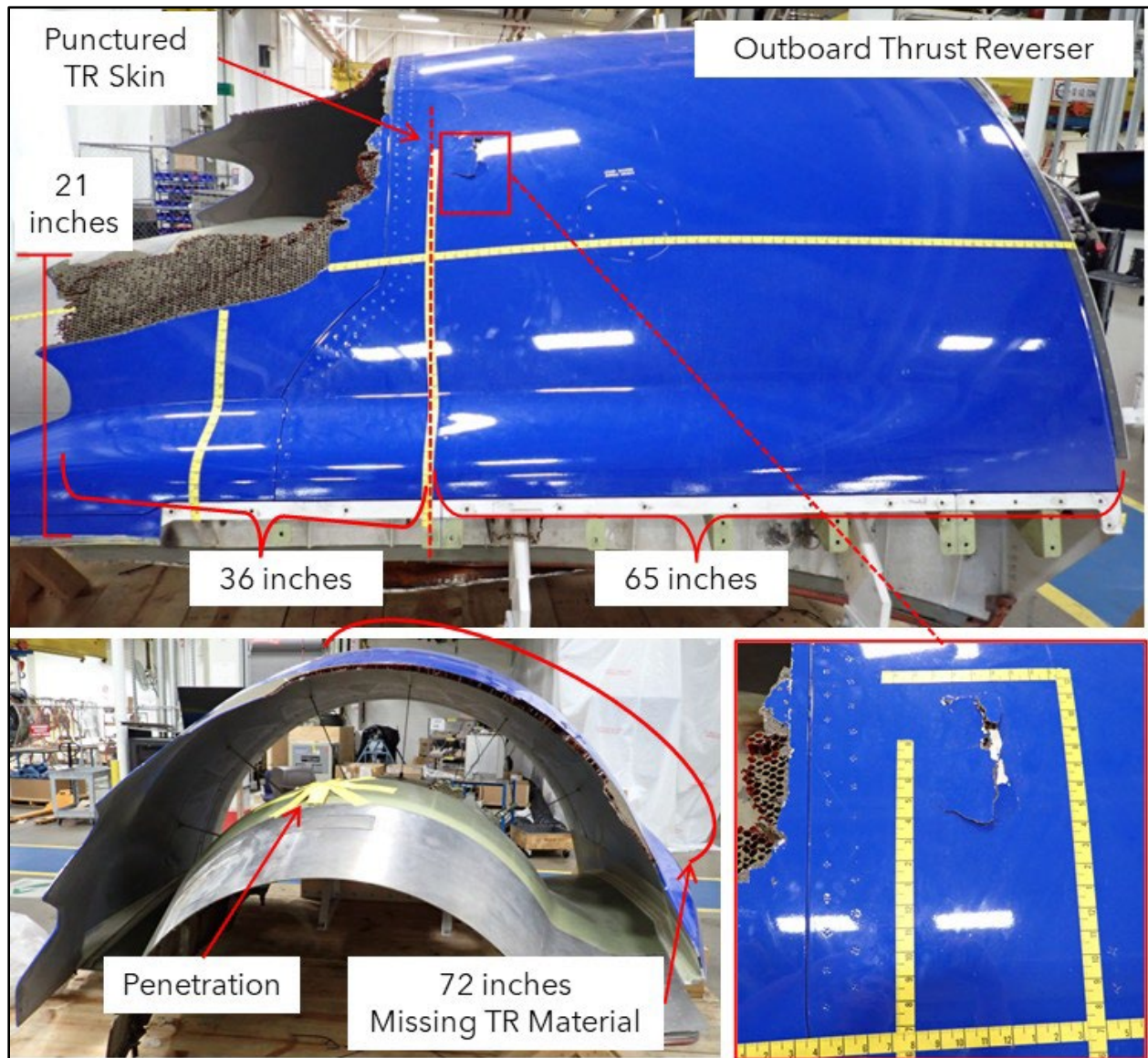


Figure 11. Outboard Thrust Reverser

The TR was found with lodged debris on the inner diameter of the TR, located outboard of the TR IFS exit hole and forward of the missing translating sleeve material. Multiple pieces of debris were recovered and retained for identification and future examination (Figure 12).

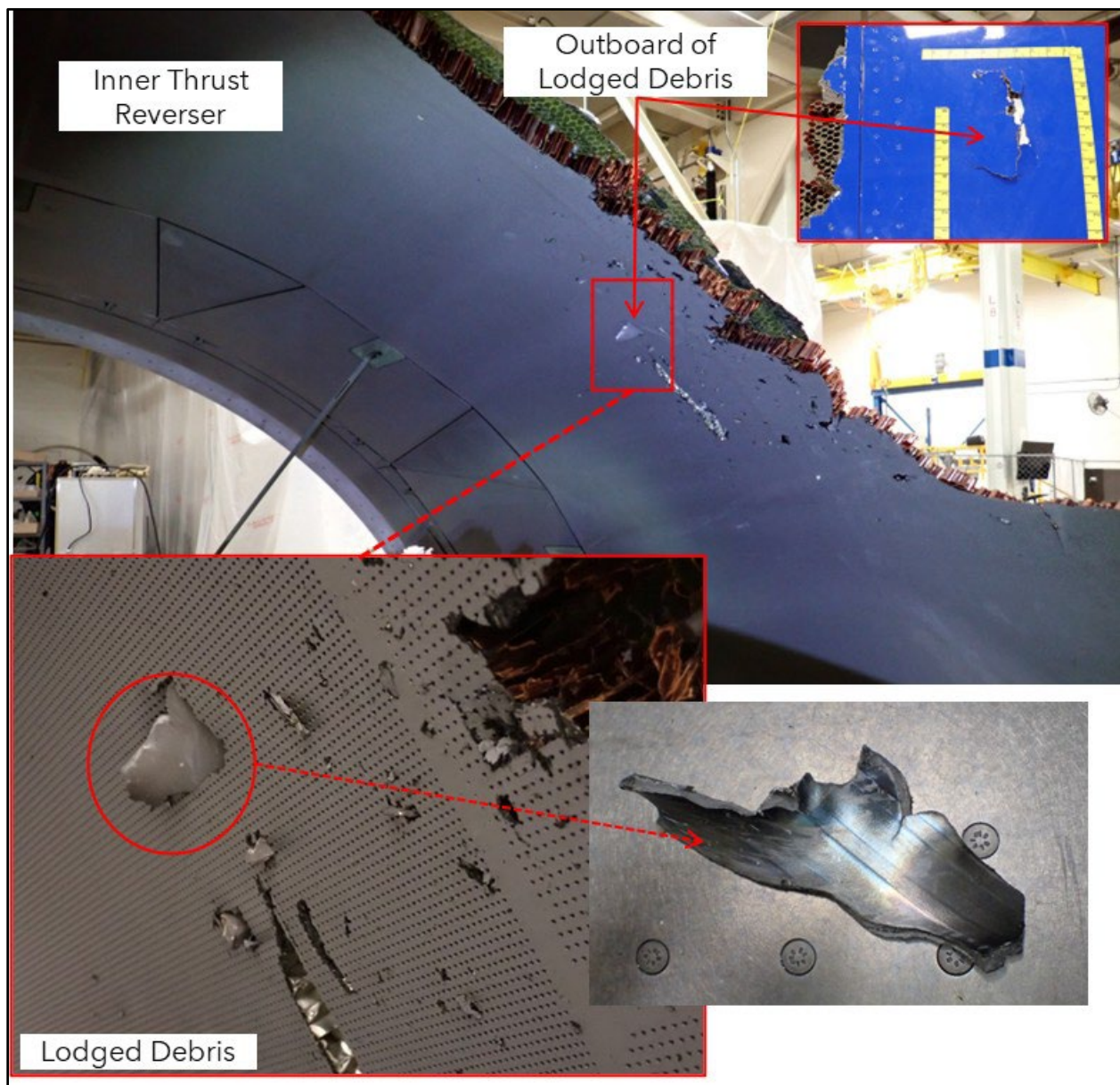


Figure 12. Lodged Debris in Thrust Reverser

The TR IFS contained an exit hole approximately 13 inches axially and 15 inches circumferentially in size, located approximately 54 inches from the forward edge of the TR IFS and 51 inches from the rear edge of the TR IFS (Figure 13). The fireproof thermal blanket⁶ located on the inner diameter of the TR IFS did not appear fire damaged or sooted.

⁶ The thermal blanket is stuffed with 8 pcf insulation, topped with corrosion resistant steel (CRES) foil and backed with Kapton.

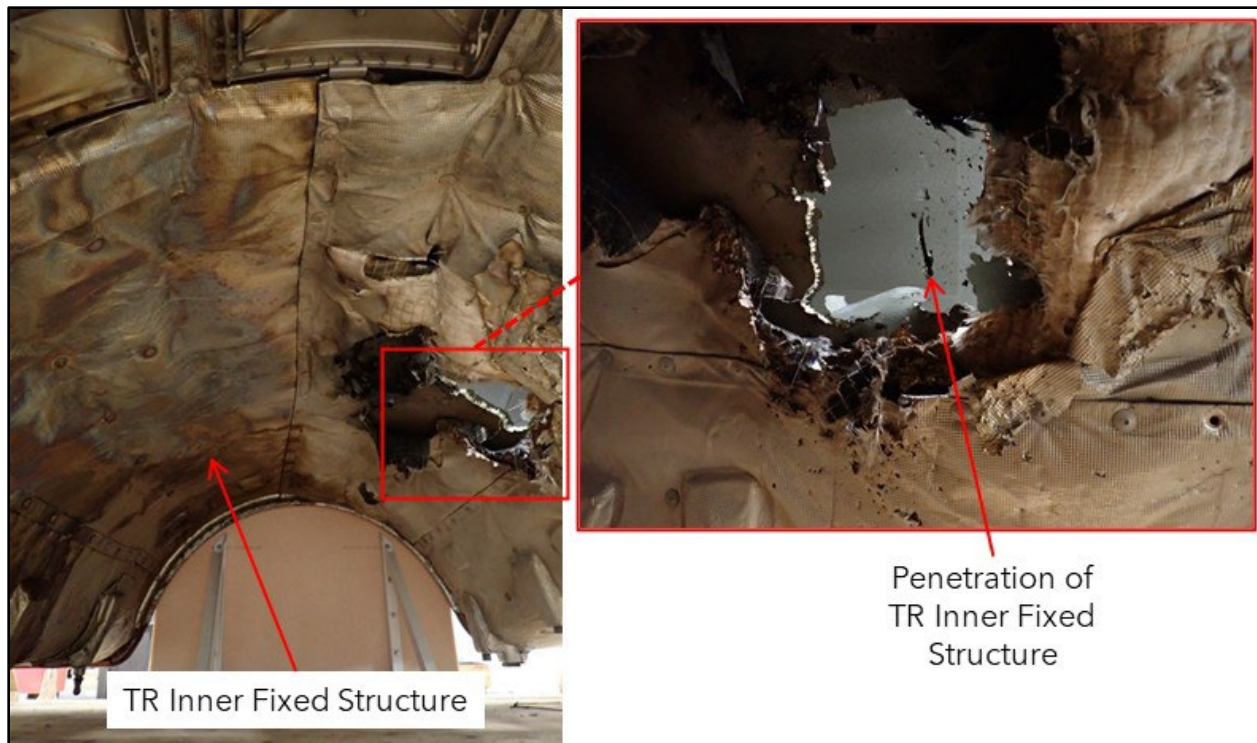


Figure 13. Punctured TR IFS

5.0 Engine Documentation and Examination

5.1 On Scene Engine Damage

The following on-scene information was gathered and provided to the NTSB by UAL and GE Aerospace in-country service representatives.

The fan blades were full length and appeared undamaged (Figure 14).



Figure 14. Front View of Fan Blades

Photos Courtesy of UAL

With the cowls opened, the exterior of the engine was visible. The turbine case was missing a section of case skin between approximately the 8:00 - 10:00 o'clock positions, near the flange that attaches the turbine case to the TCF and axially in-plane with HPT stage 2 nozzles (Figure 15). Hardware visible through the penetration/exit hole in the HPT case included the HPT stage 1 nozzles, HPT stage 1 blades, HPT stage 2 nozzle and the HPT rotating interstage seal (RIS). The HPT stage 1 aft blade retainer was not visible and appeared to be missing.

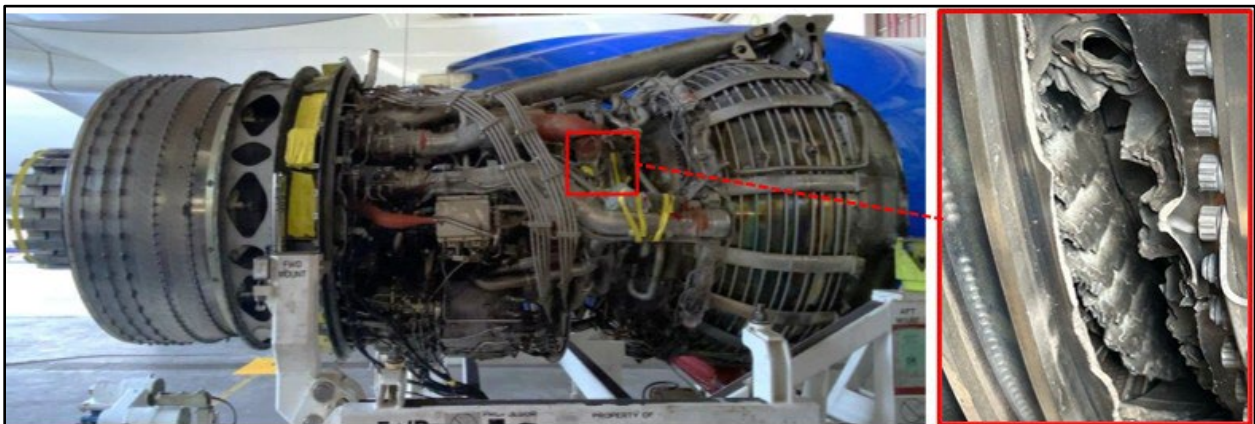


Figure 15. No. 1 ENGINE REMOVED - HPT CASE EXIT HOLE

Photos Courtesy of UAL, Modified by NTSB

A portion of the HPTACC cooling duct/manifold⁷ near the turbine case exit holes was damaged (Figure 16). The heat shielding located between the turbine case and the HPTACC ducts was damaged outboard of the exit holes.

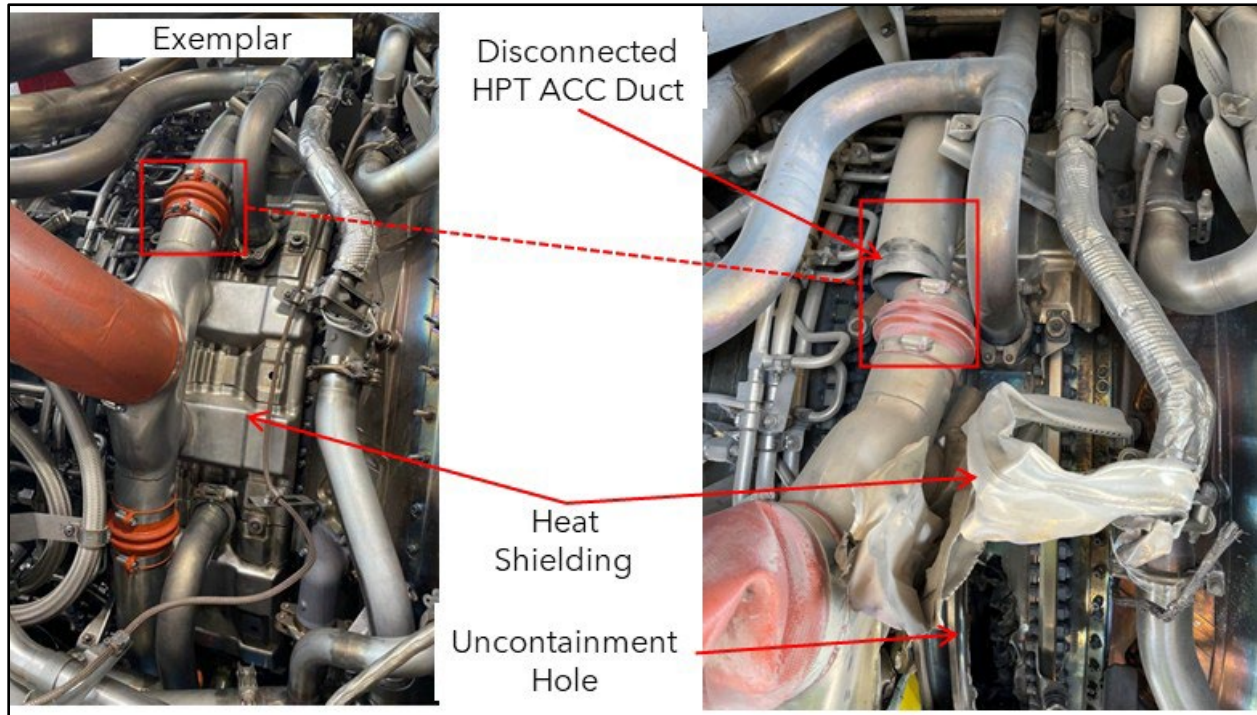


Figure 16. Damage to Heat Shielding and HPT ACC Ducts

Photos Courtesy of UAL, Modified by NTSB

Two borescope inspections (BSI) of the HPT module were conducted to view as much of the circumference at the location of the HPT stage 1 aft blade retainer as possible; it was estimated that about 95% of the circumference was viewable. The following HPT BSI observations are only for what was visible during the BSI and should not be construed as the condition of the entire module. The HPT stage 1 blades were fractured transversely across the airfoil at approximately 25% span but did not appear to exhibit thermal distress/melting of the fracture surfaces (Figure 17). The HPT stage 1 nozzles appeared to be present and intact with some nozzle segments appearing to be displaced. The HPT stage 2 nozzles were present but detached from the turbine case at the exit hole.

⁷ HPT active clearance control is provided by impinging fan air on the high pressure turbine casing. This air is modulated by a servo fuel actuated valve as controlled by the FMU and EEC. The air is distributed by a manifold, consisting of eight shroud segments, which surrounds the HPT case and then passes into the core compartment to exhaust through the core compartment-cooling vent.

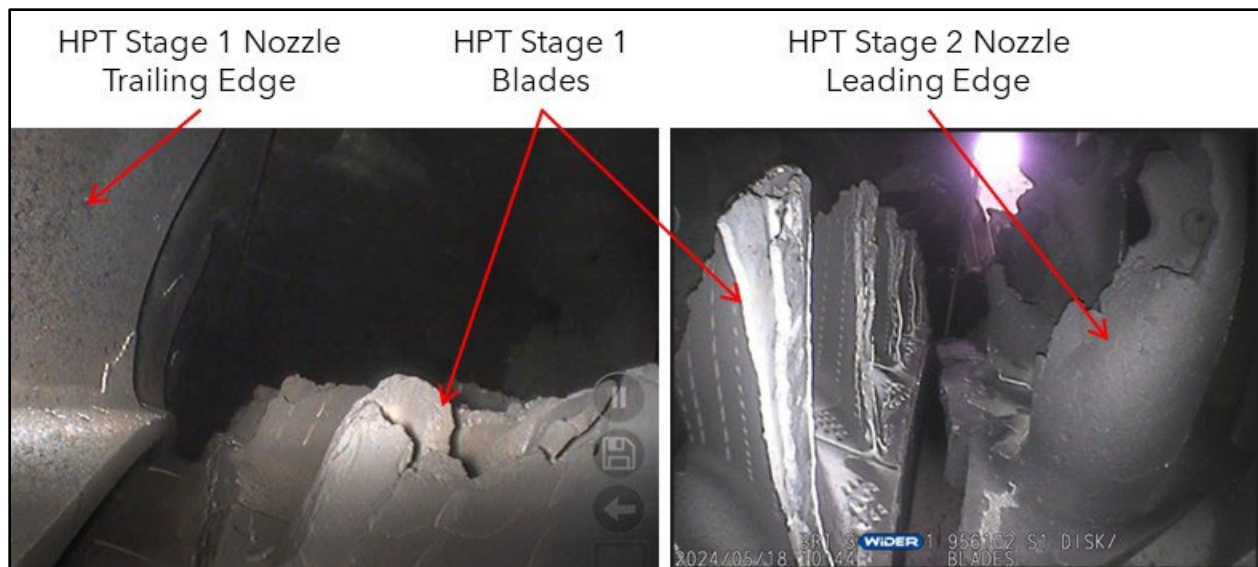


Figure 17. BSI - HPT Stage 1 Blades

Photos Courtesy of GE, Modified by NTSB

The HPT RIS is composed of 4 knife-edge seal teeth and is attached to both HPT stage 1 and HPT stage 2 disks. The HPT RIS provides air flow sealing between the HPT rotor stages by rubbing into the HPT stage 2 nozzle stationary seal honeycomb. The HPT RIS appeared to be intact. The HPT stage 1 aft blade retainer, installed on the aft side of the HPT stage 1 disk, provides axial retention for the HPT stage 1 blades in the disk dovetail and provides air sealing for the blade cooling circuits. The HPT stage 1 aft blade retainer appeared to be missing the outboard portion of the part while viewed with the borescope; the HPT stage 1 disk posts and blade dovetails should not be visible with an intact and present HPT stage 1 aft blade retainer (Figure 18).

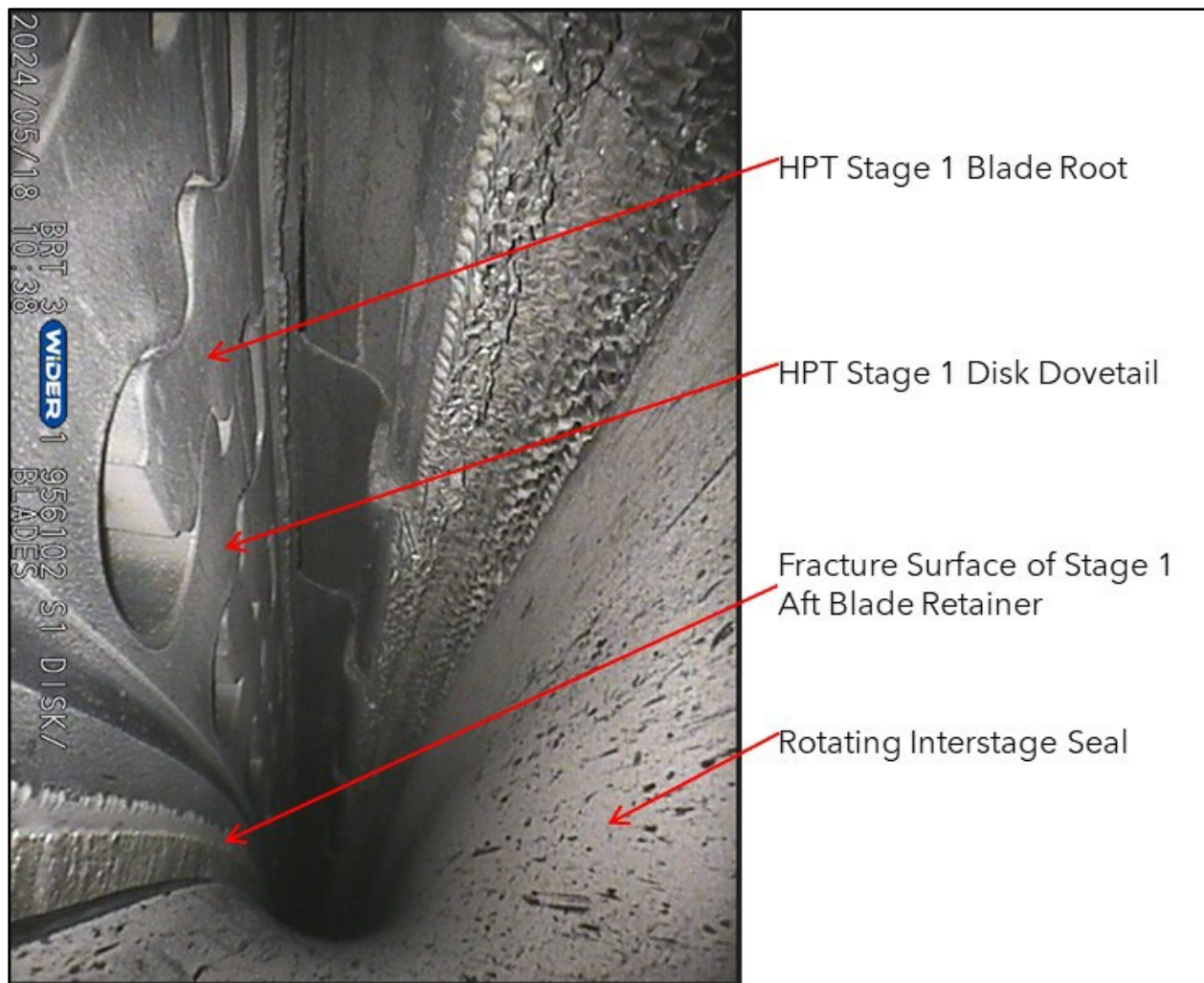


Figure 18. STAGE 1 HPT AFT BLADE RETAINER FRACTURED AND MISSING

Photo Courtesy of GE, Modified by NTSB

With the engine removed from the airplane, the damage was found to be localized to the engine propulsor module, left side TR fan duct and left side thrust reverser hardware. With approval from the NTSB, the fan module, the exhaust nozzle and aft center body were removed from the engine and locally retained by UAL (Figure 19).



Figure 19. Engine Fan Module Removed

Photos Courtesy of UAL

The propulsor module and the outboard (left side) thrust reverser were shipped to the GE Evendale, Ohio, facility for further examination. A total of nine additional boxes, containing various pieces of hardware that were removed from the engine during the engine removal, were shipped with the propulsor. Additional evaluation of the fan blades by UAL personnel found two fan blades with serviceable/repairable damage, later determined to be unrelated to the uncontained failure event, were also shipped for evaluation and disposition.

5.2 Initial Workscope Performed by GE Aerospace

Prior to the engine examination with the investigation team onsite, GE was authorized by the NTSB to remove the propulsor from the shipping container, remove the AGB module and all external hardware from the propulsor, remove the LPT module, place the propulsor into a vertical pit/stand and perform the necessary disassembly work to prepare the TCF for removal. The TCF remained in place and the final separation/removal was performed when the investigation team arrived on site at GE.

5.3 As Received Documentation by GE Aerospace

The propulsor was received in the transportation stand and unwrapped by GE personnel (Figure 20 and Figure 21). The engine did not exhibit any evidence of an undercowl fire but a fire assessment of the propulsor and various hardware was performed during the Group examination.

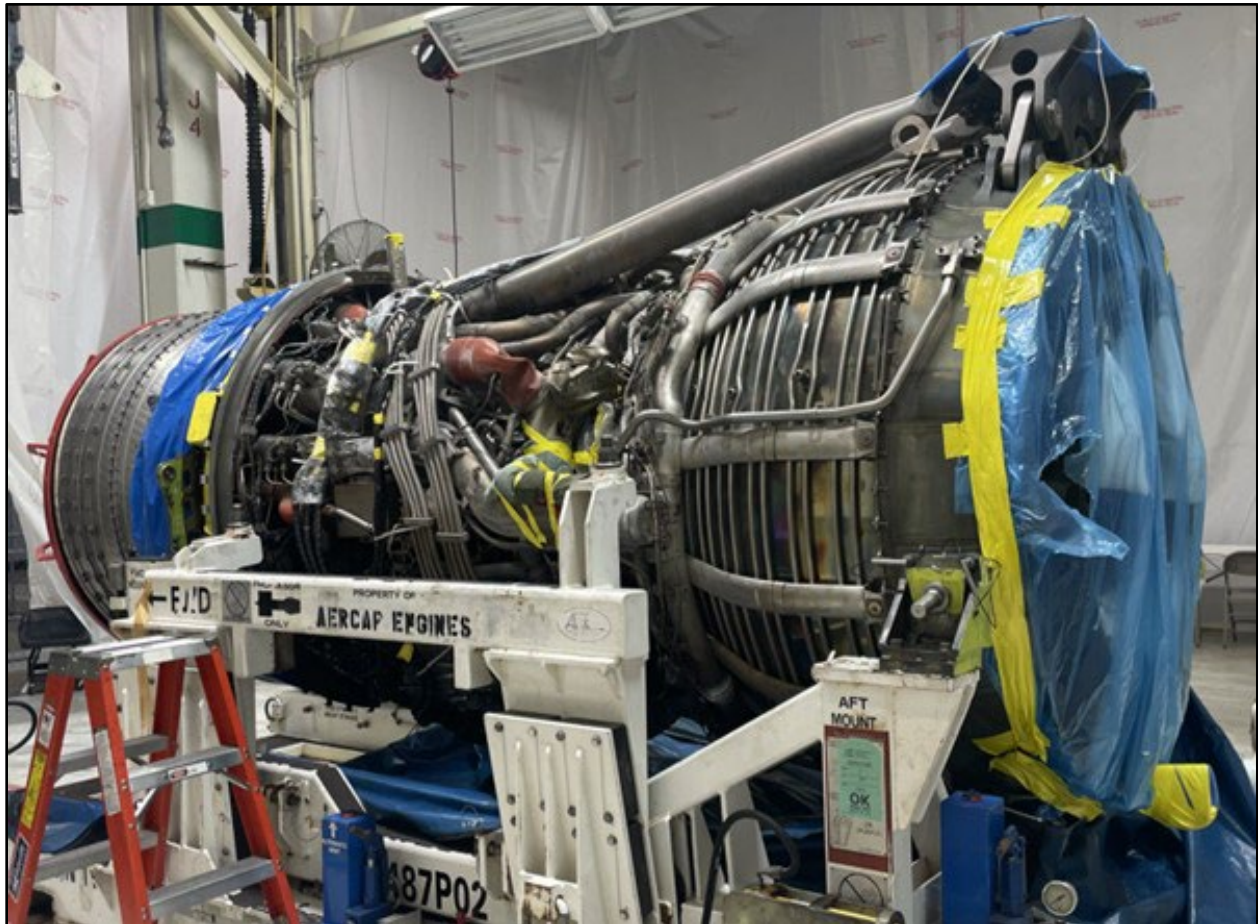


Figure 20. ENGINE AS-RECEIVED, LEFT SIDE

Photos Courtesy of GE Aerospace

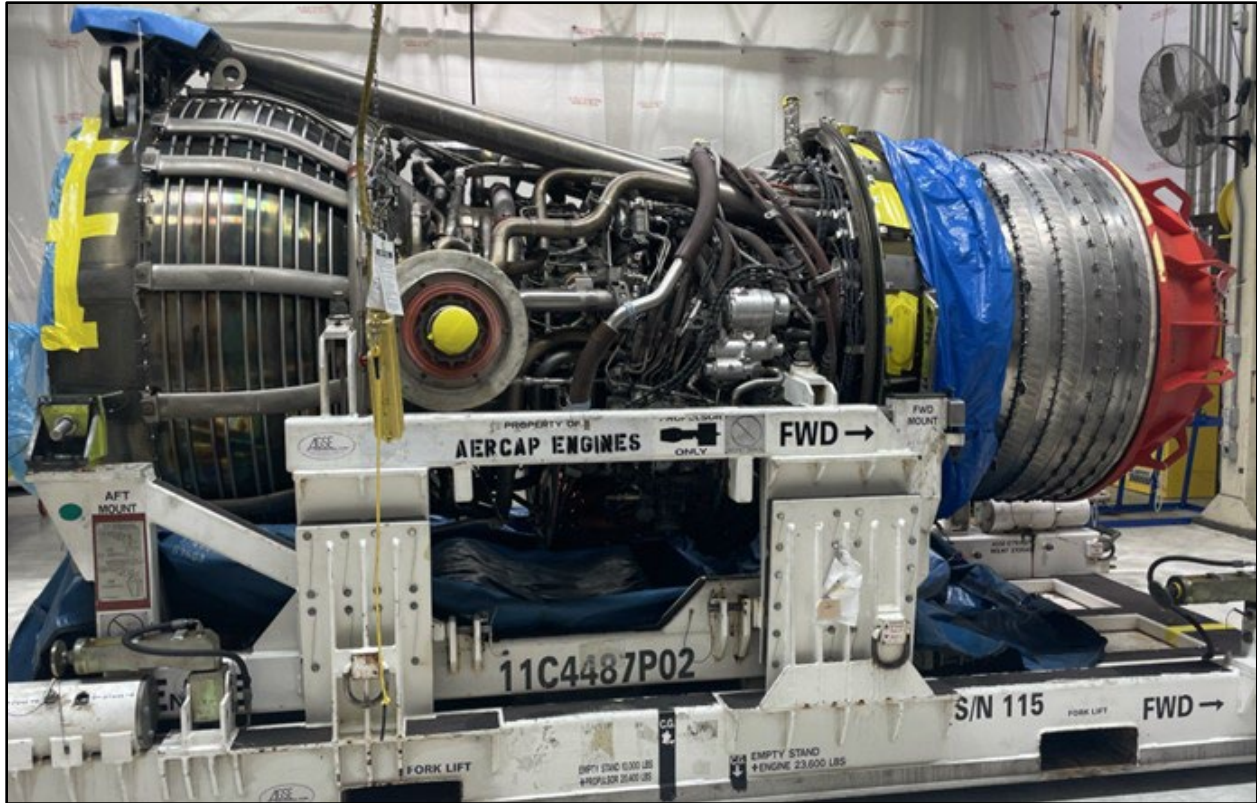


Figure 21. Engine As-Received, Right Side

Photos Courtesy of GE Aerospace

There are three fire loops installed near the HPT module; the left hand loop located circumferentially between approximately the 7:00 o'clock and 10:00 o'clock positions, the right hand loop located circumferentially between approximately the 2:00 o'clock and 5:00 o'clock positions and the lower center loop located circumferentially between approximately the 5:00 o'clock and 7:00 o'clock positions. These three fire loops are connected to the wiring harnesses located at approximately the 12:00 o'clock position on the propulsor which then connects to the airplane wiring harness junction box aft of the fan module at the 12:00 o'clock position.

The left hand, right hand, and lower center fire loops were all present and appeared to be intact. All fire loop isolators were present and appeared undamaged (Figure 22 and Figure 23).

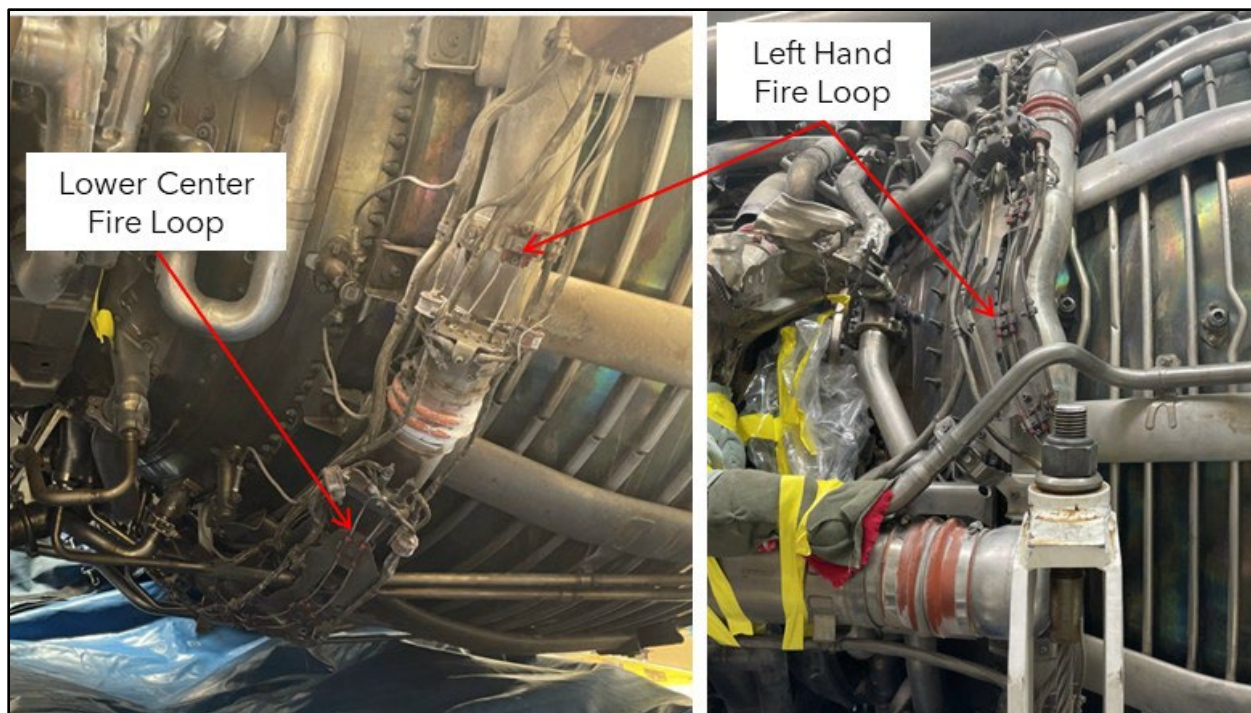


Figure 22. Left Hand and Lower Center Fire Loops

Photos Courtesy of GE Aerospace, Modified by NTSB

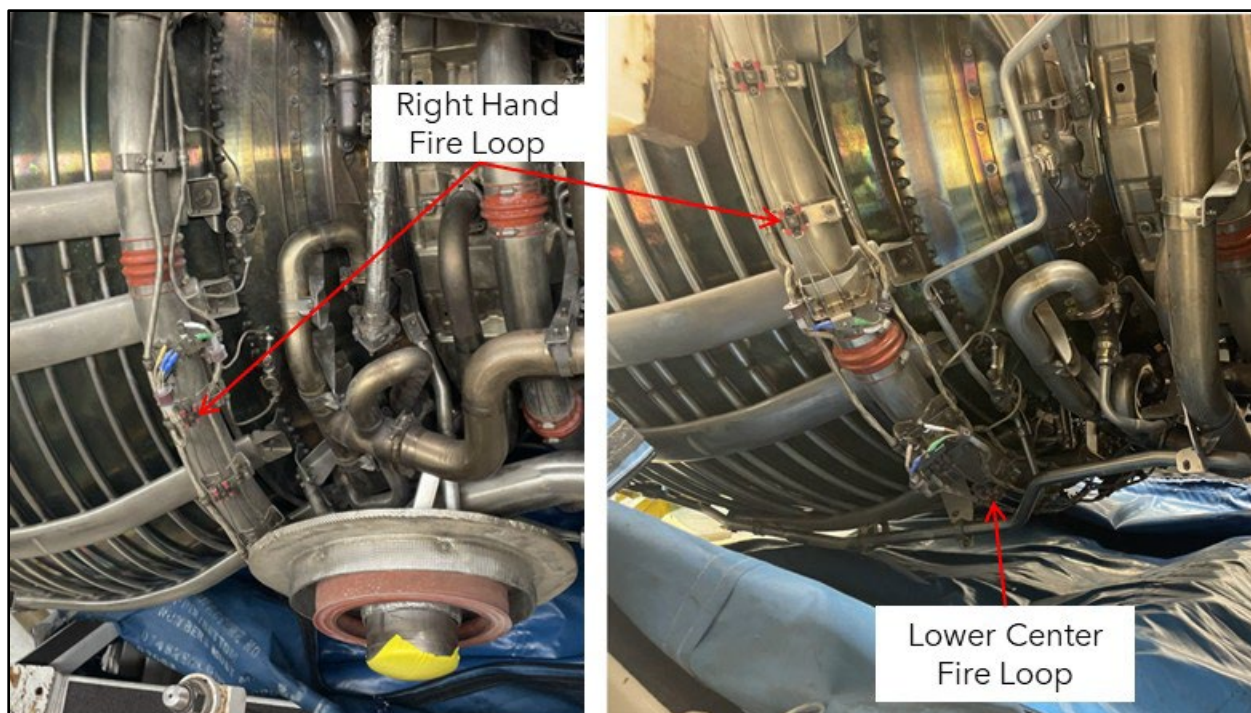


Figure 23. Right Hand and Lower Center Fire Loops

Photos Courtesy of GE Aerospace, Modified by NTSB

The exit holes in the HPT case were located circumferentially between approximately the 7:30 and 9:30 o'clock positions and located axially between the HPT turbine case aft flange and the aft stiffening ring (Figure 24).

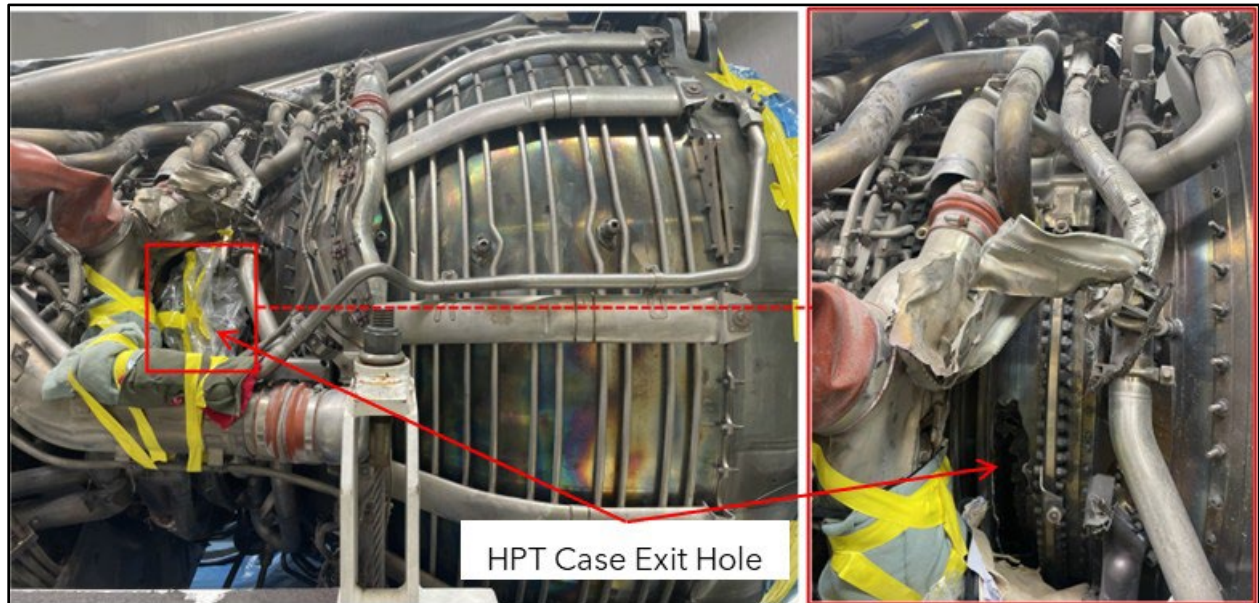


Figure 24. HPT Case Exit Hole

Photos Courtesy of GE Aerospace, Modified by NTSB

5.4 Initial Disassembly by GE Aerospace

The engine thrust links and rear engine mount were removed and found to be intact and undamaged (Figure 25). The spherical bearings between the rear engine mount and the engine-to-rear mount attachment links rotated freely.



Figure 25. Thrust Links and Rear Engine Mounts

All accessory components and external hardware (ex. oil/fuel tubes) were removed and placed in palletized boxes or on tables for evaluation by the investigation team upon arrival. The AGB and LPT modules were removed from the propulsor, no anomalies were observed or reported.

With the accessory hardware removed, two exit holes were observed in the HPT case (Figure 26). The larger hole, located between the TCF/HPT case flange and the aft HPT case stiffening ring, measured approximately 2.25 inches axially by 11 inches circumferentially. The smaller hole, located between the HPT case stiffening rings, measured approximately 2 inches by 2 inches.

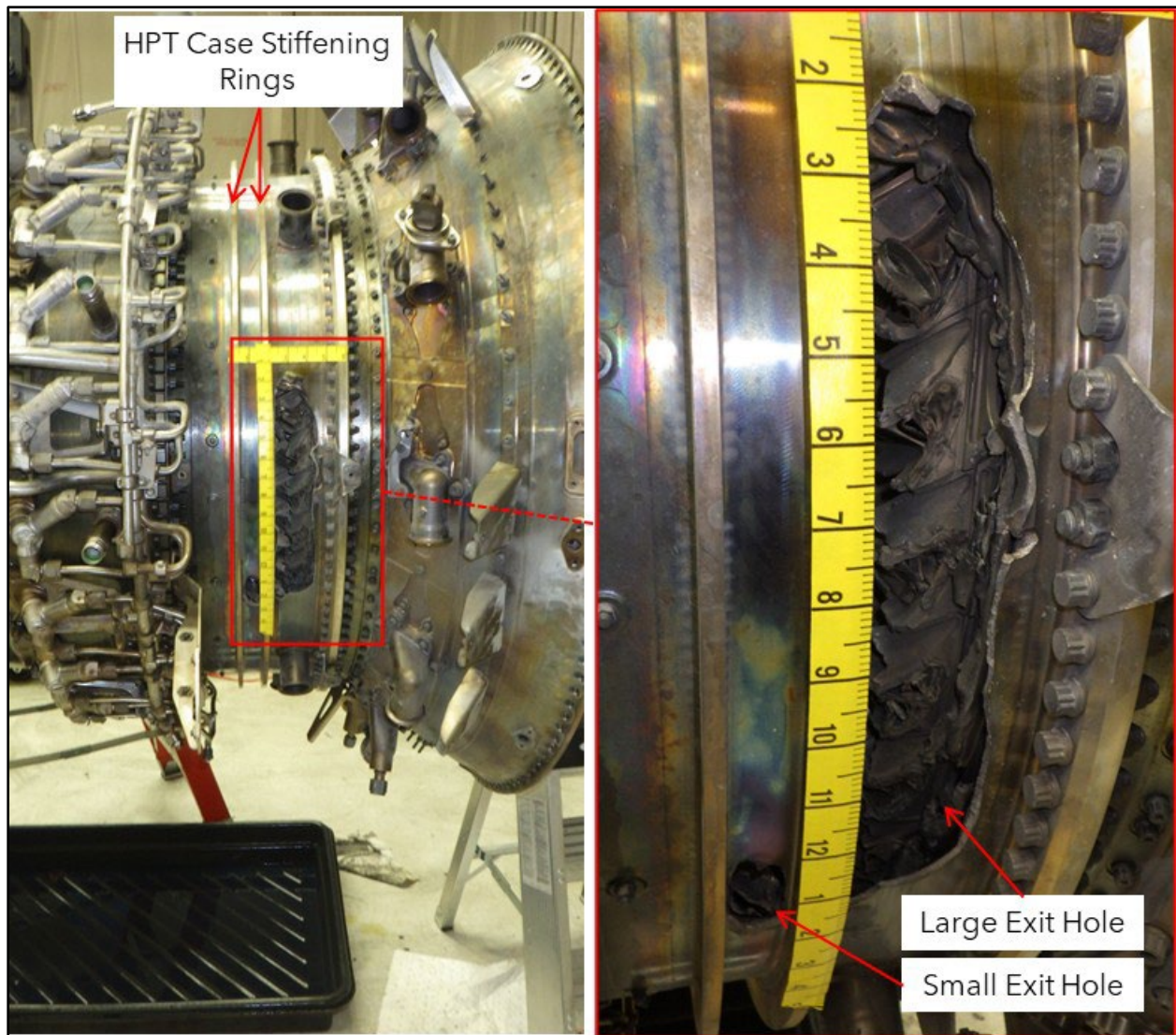


Figure 26. HPT Case with Exit Holes

Photos Courtesy of GE Aerospace, Modified by NTSB

5.5 Engine Examination - June 24-28, 2024

The investigative team made up party members from UAL, GE Aerospace, Boeing, FAA, and the NTSB convened on June 24, 2024, at GE's Evendale, Ohio, facility to continue the engine disassembly and examination, beginning with separation of the TCF from the HPT module. The engine exam and documentation were completed on June 28, 2024.

5.5.1 Engine Hardware Upon Arrival

The engine propulsor data plate, located on the fan hub frame, for ESN 956102 was verified (Figure 27).

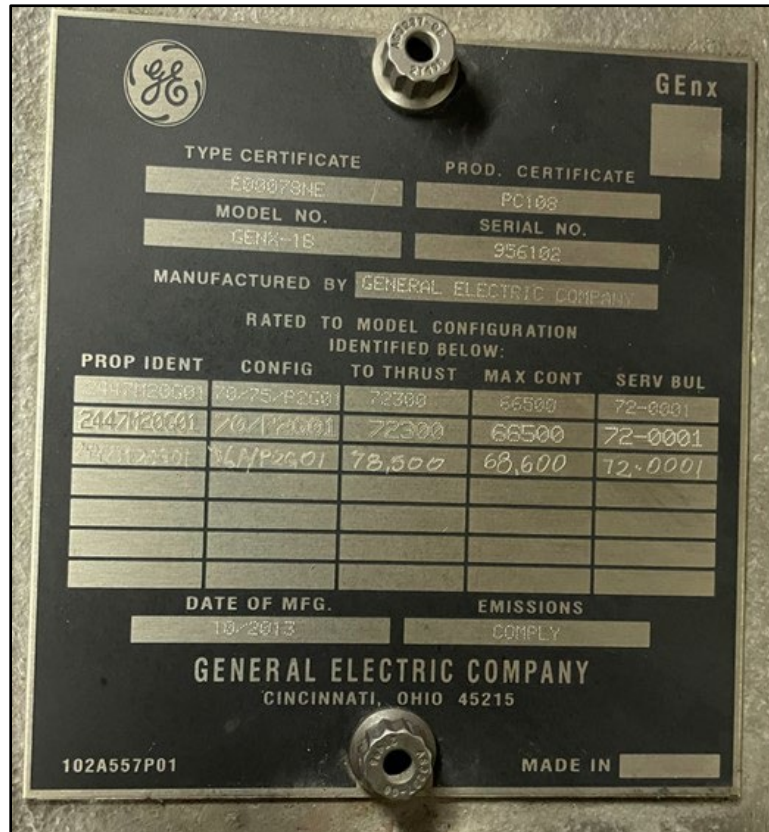


Figure 27. Engine Propulsor Data Plate

Upon the investigation team's arrival, all the hardware removed prior to our arrival was placed in boxes and staged in the secure disassembly area (Figure 28).



Figure 28. Engine Hardware Removed

Nine boxes with various engine components were shipped to GE's facility, which included engine mounting hardware, fan module attachment hardware, two fan blades, the FADEC, the EMU, the bifurcation panel, and loose hardware found at the event site. The components within the boxes were inventoried by GE personnel and stored in a secure trailer until their release.

Portions of the HPT stage 1 blade aft retainer were found within a box containing various pieces of hardware collected during the on-scene investigation, these pieces were separated out for further examination by GE's material laboratory (Figure 29).



Figure 29. BOXED HARDWARE, PORTIONS OF HPT STAGE 1 AFT BLADE RETAINER

The engine propulsor was installed in a vertical stand to facilitate disassembly and removal of the HPT hardware (Figure 30).



Figure 30. Engine Propulsor Upon Arrival

All the LPT stage 1 blades exhibited hard body impact damage, one airfoil was fractured at approximately midspan and missing the outboard material (Figure 31). The LPT shaft was intact and appeared undamaged. When looking at the rear of the engine, the LPT stage 7 blades were all present and appeared undamaged.

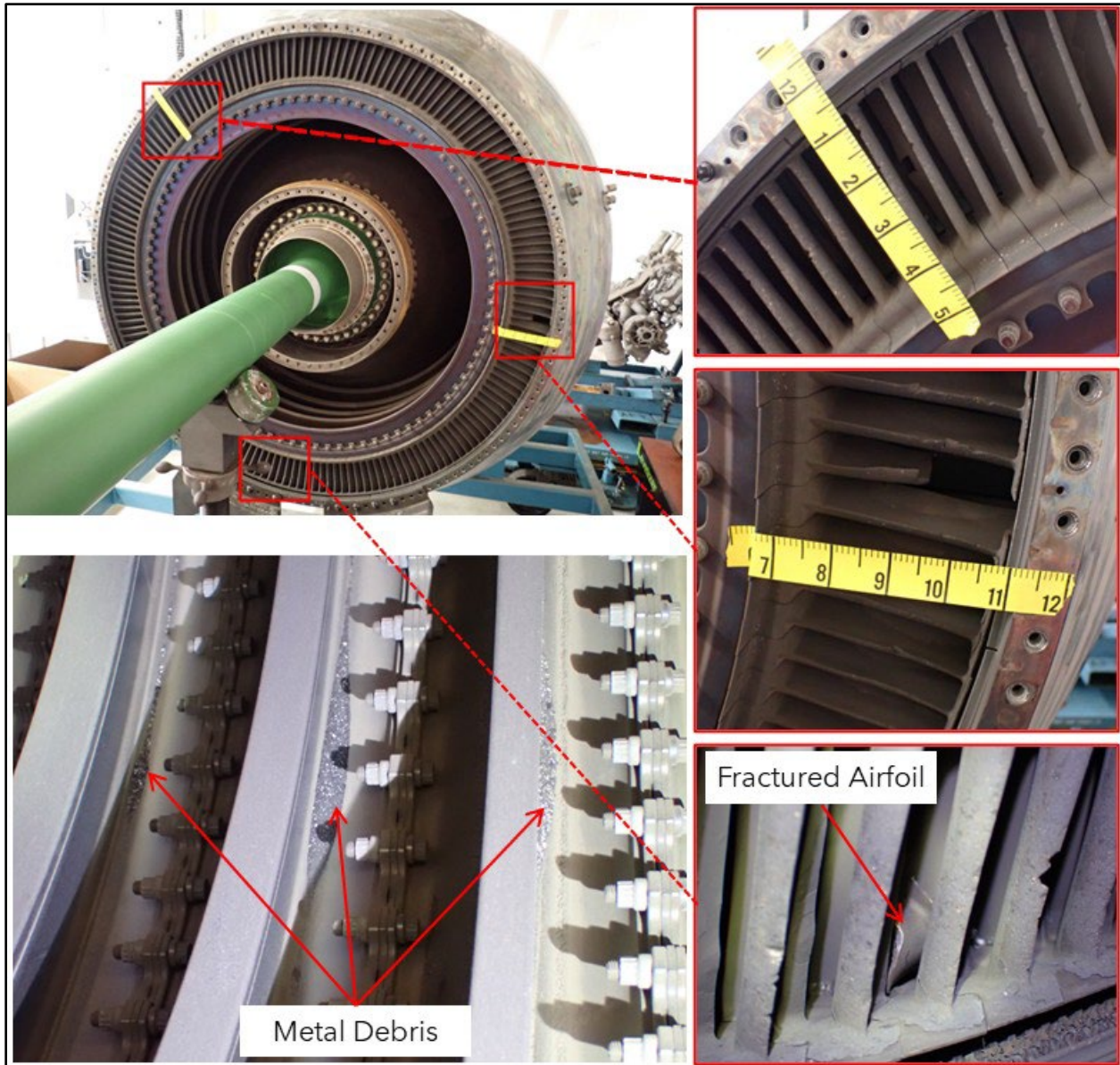


Figure 31. LPT - IMPACT DAMAGE AND LOSS OF MATERIAL

Located on the tables in the disassembly area were several damaged components that were removed prior to the team's arrival. This include one HPC stage 7 cooling air tubes, supplying cooling air to the 8 supply ports on the HPT case, was completely severed near the exit hole, which corresponded to the HPT case supply port that was missing, located within the exit hole (Figure 32). Other items on the table included an ignitor cable, a speed sensor/cable, and multiple pieces of the HPTACC ducts/manifolds, and the HPC stage 4 cooling manifold which contained multiple puncture holes and crushed/tearing damage (Figure 32).

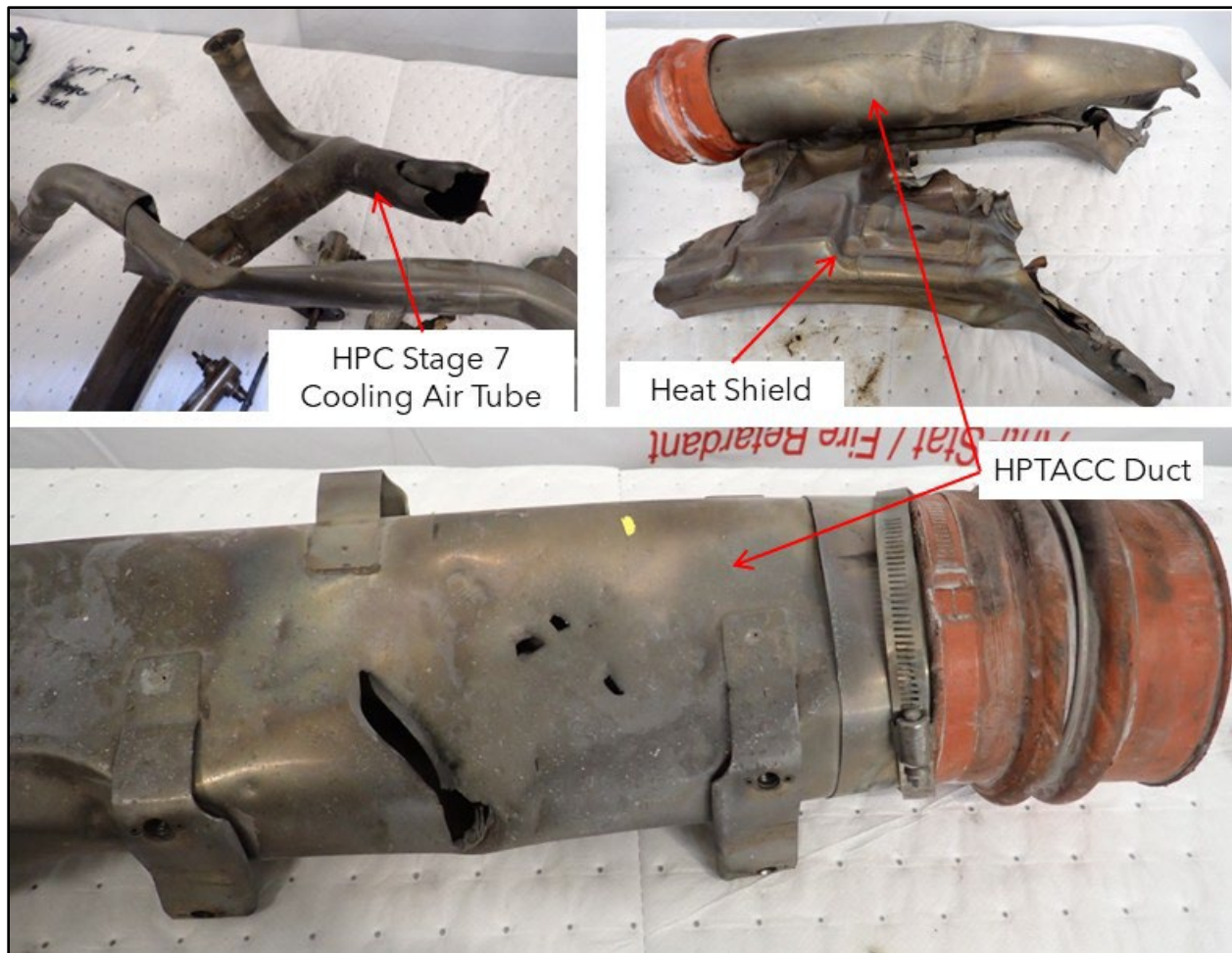


Figure 32. Damaged HPTACC Ducts and HPC Stage 4 Cooling Manifold

5.5.2 Removal of the TCF

In preparation for removal of the TCF from the HPT turbine case, the breakaway torques of the attachment bolts were measured and found to be within the normal experience during disassembly according to GE maintenance personnel. The nominal torque range according to the engine manual is 362-432 inch pounds. Every fifth attachment bolt was left in place and secured for removal of the TCF once the investigation team arrived (Figure 33).

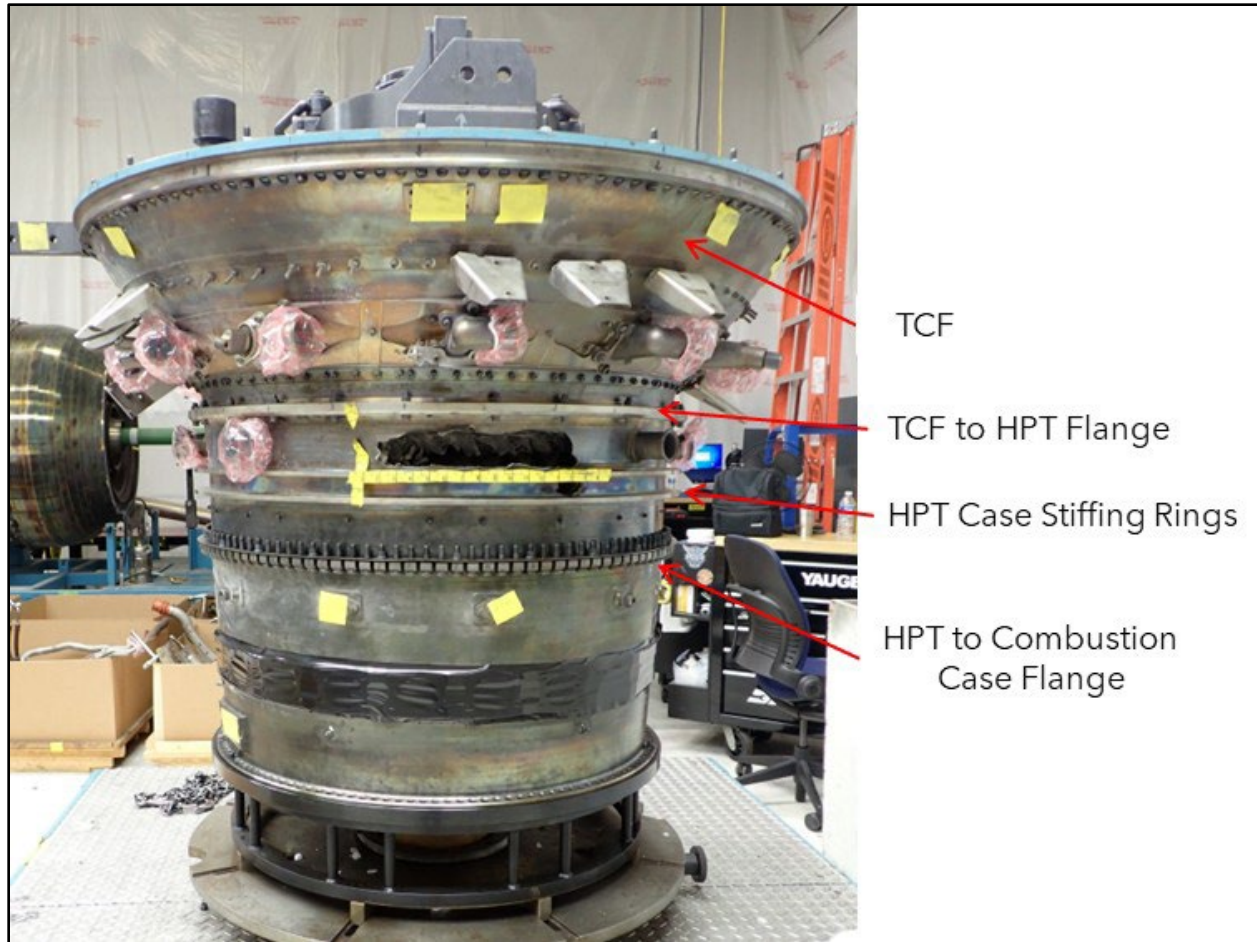


Figure 33. FLANGE LOCATIONS

With the TCF removed, multiple TCF flow path liners and struts, located on the forward side of the TCF, sustained hard body impact damage and had missing material. Multiple struts exhibited puncture holes and imbedded debris (Figure 34). The honeycomb seals located on the inner bore of the TCF appeared to have consistent trenching around the whole circumference.

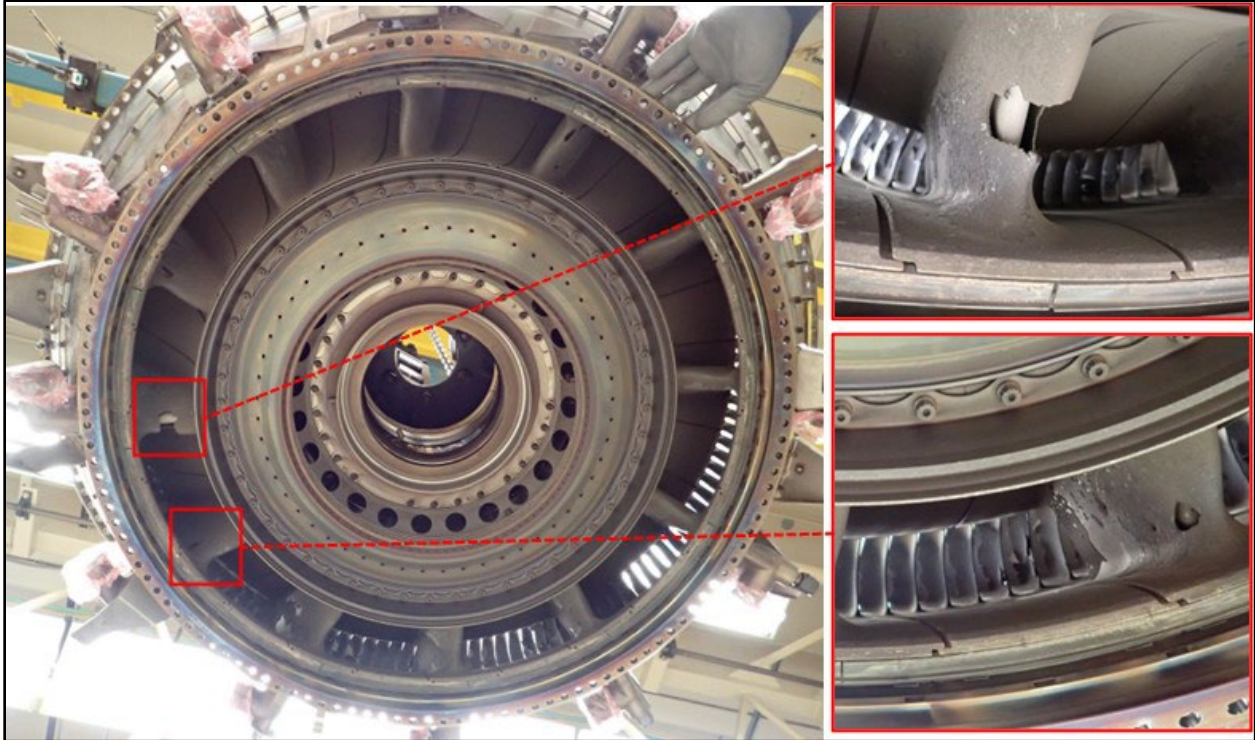


Figure 34. TCF Removed - Strut Impact Damage

All the LPT stage 1 nozzles segments (4 nozzle airfoils per nozzle segment), located on the aft side of the TCF, were present and engaged. The stage 1 nozzle airfoils exhibited hard body impact damage (Figure 35).



Figure 35. AFT SIDE OF TCF

5.5.3 HPT Case

With the HPT case removed, an unobstructed view of the uncontainment location showed the HPT case wall to be distorted outward and the edges appeared flared, jagged and exhibited mechanical damage (Figure 36). Every HPT stage 2 nozzle sustained damage to the airfoil leading edges, significant thermal distress and hard body impact damage.

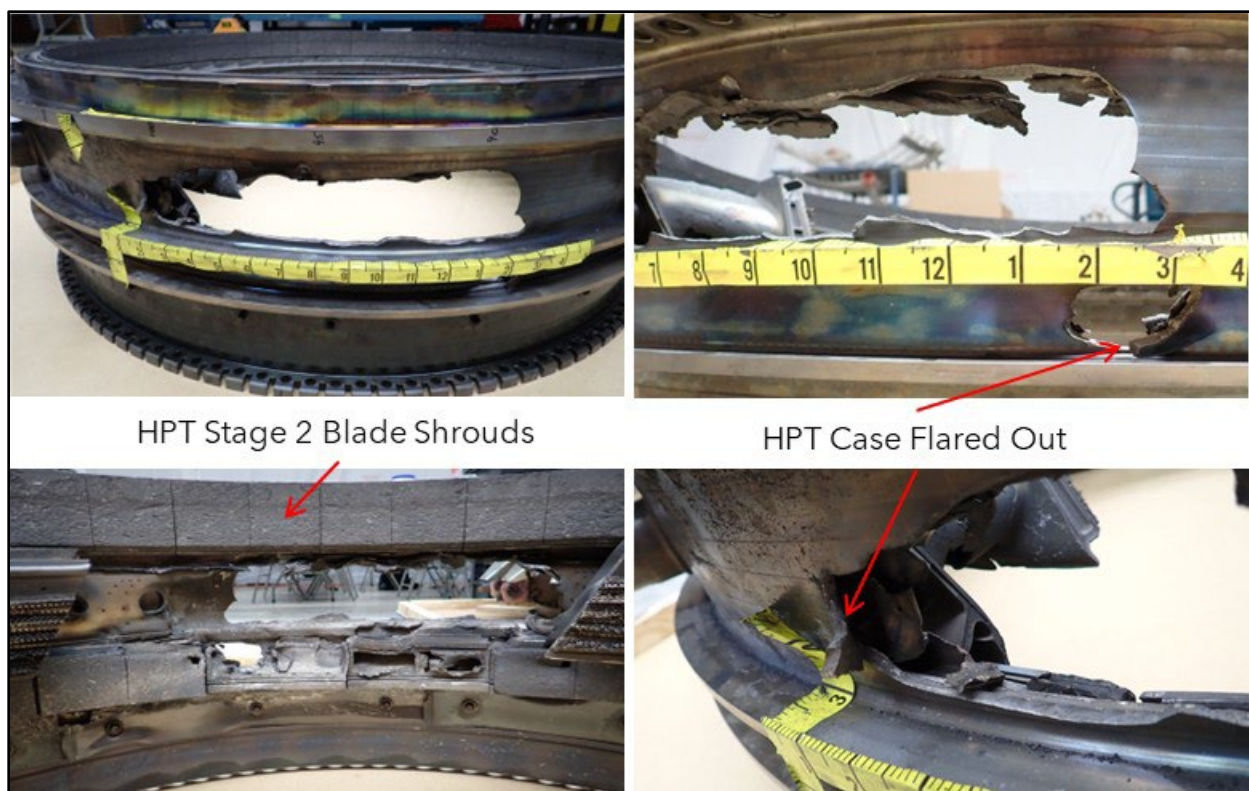


Figure 36. HPT Case Removed

The HPT turbine case was not disassembled any further for the purpose of potential future evaluation related to HPT containment capability.

5.5.4 Combustion Chamber/Case

The combustion chamber, forward of the HPT stage 1 nozzle segments, appeared typical of normal engine operation, there were no signs of fuel nozzle streaking, hot spots, or burn throughs (Figure 37). The HPT stage 1 nozzle consists of 20 nozzle segments installed into the combustion case; each nozzle segment contains two vanes that are brazed together to create one nozzle segment. All nozzle segments were accounted for, a quantity of 15 nozzle segments were found still engaged to the combustion case and a quantity of 5 nozzle segments adjacent to the exit hole were disengaged (Figure 37). The vane trailing edges sustained significant hard body impact damage and missing material.

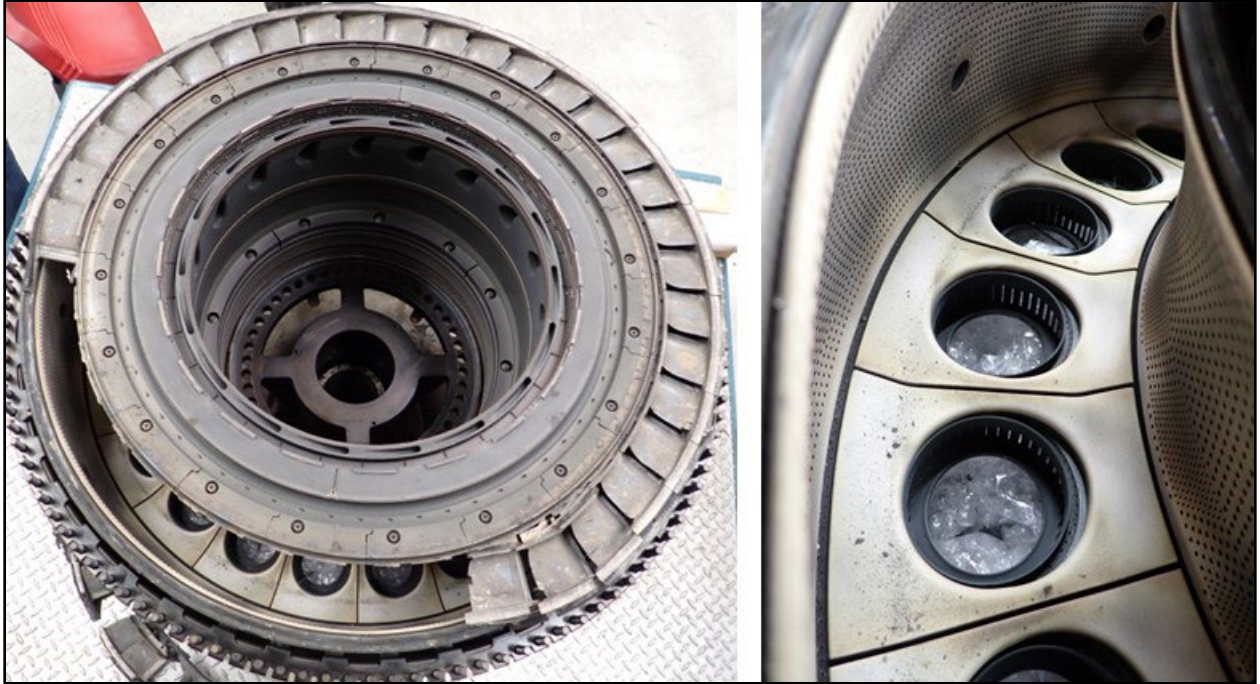


Figure 37. HPT Stage 1 Nozzle Segments and Combustion Chamber

5.5.5 HPT Rotor Assembly

The No. 4 roller bearing assembly appeared oil wetted, undamaged and with discoloration consistent with that of engine run hardware (Figure 38). The HPT aft seal, secured to the aft side of the HPT stage 2 disk by 15 bolts, was intact but loss of coating and peeling coating on the seal teeth was observed.

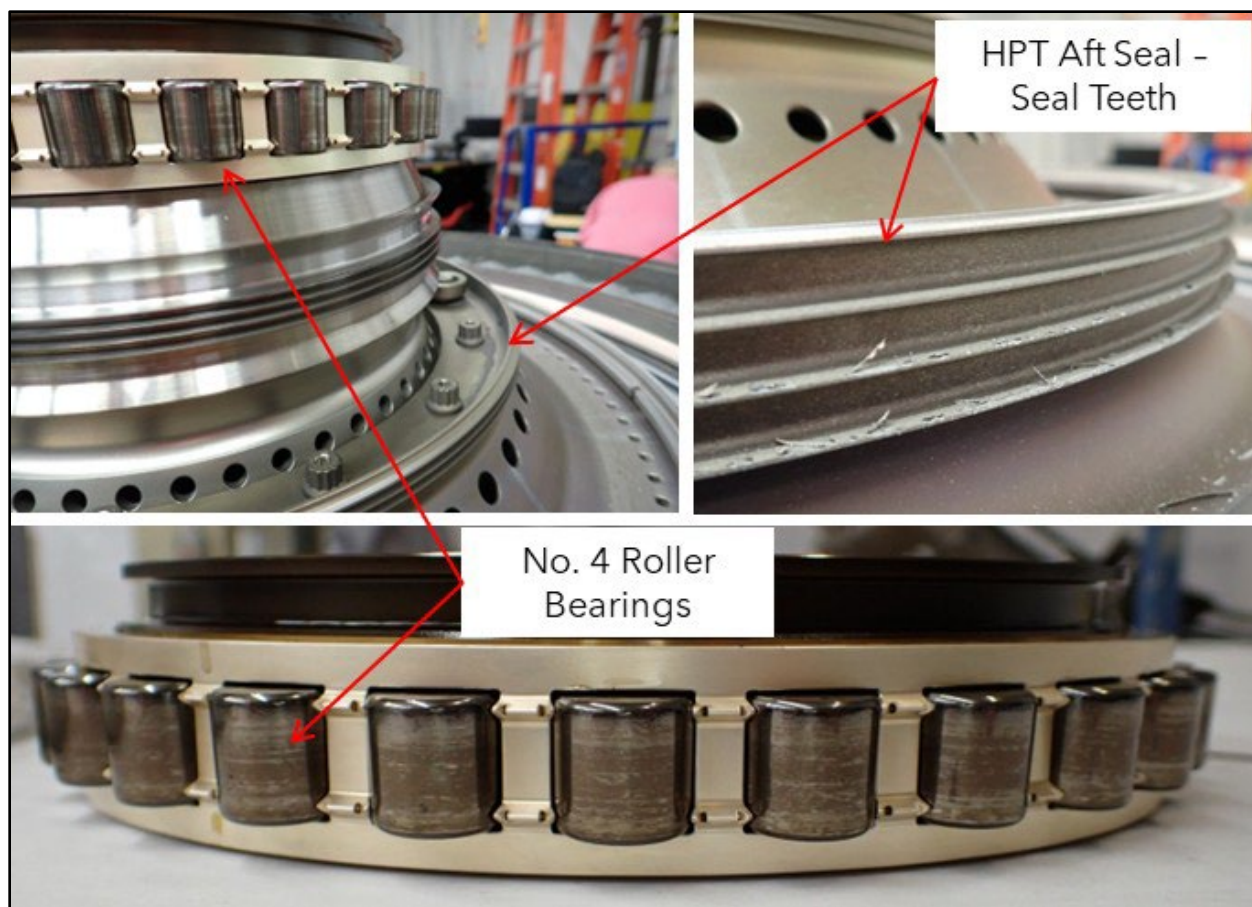


Figure 38. NO. 4 ROLLER BEARINGS AND HPT AFT SEAL

With the HPT case removed, the HPT rotor assembly was exposed (Figure 39).

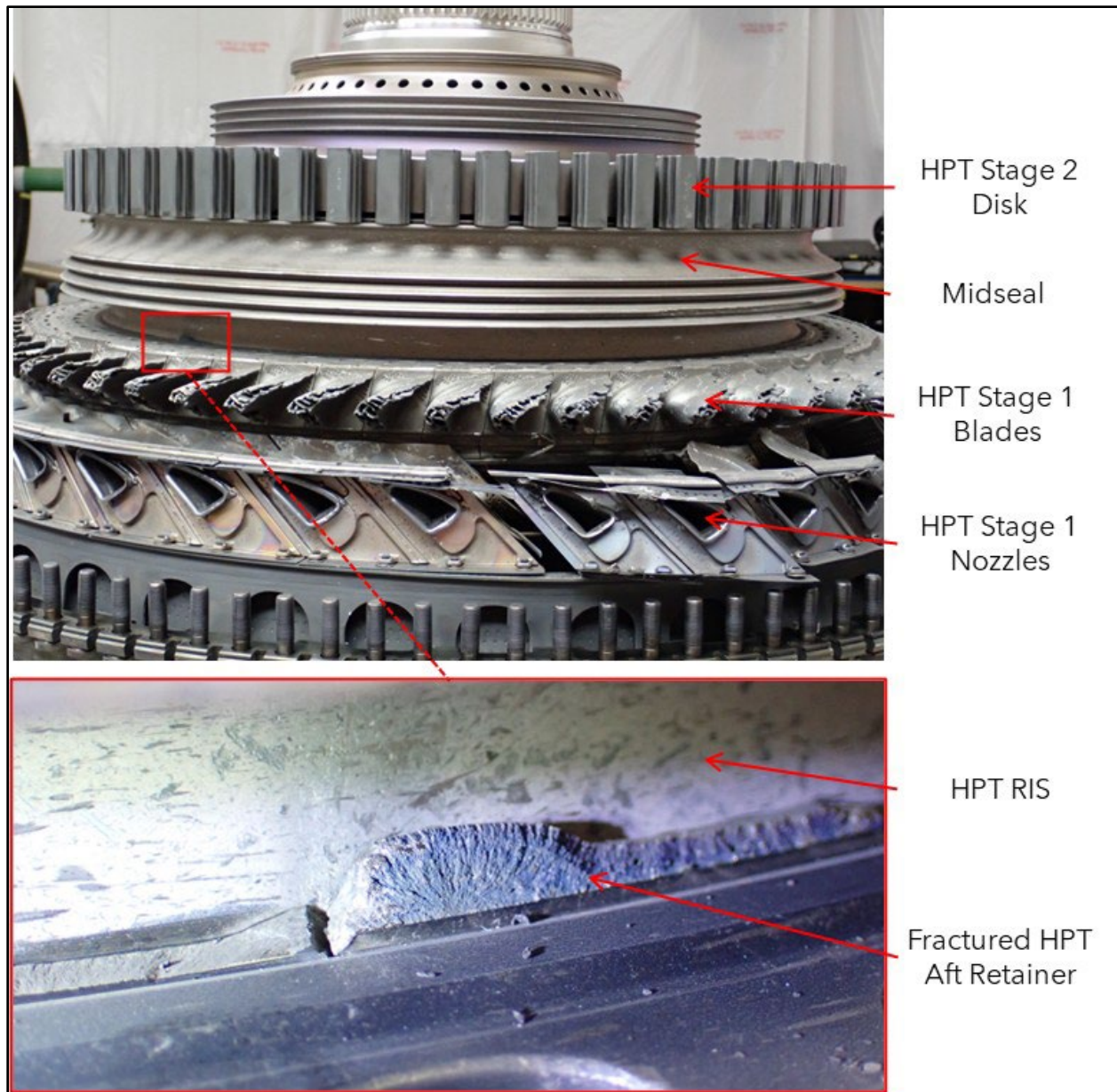


Figure 39. HPT Rotor Assembly - HPT Case Removed

All the HPT stage 2 blades were present and fractured transversely across all the airfoils between approximately 85%-95% blade span above the platform. The airfoils sustained leading and trailing edge hard body impact damage, missing material, tears, and material transfer (Figure 40).



Figure 40. HPT STAGE 2 BLADE DAMAGE

The HPT stage 1 blades were fractured transversely at varying span lengths with approximately half being of a similar length, the other half of the blades had an increasing span length (Figure 41). All the HPT stage 1 blades sustained significant thermal distress and hard body impact damage. The blade platform has a single seal on the forward face and a double seal on the rear face; these seals are referred to as angel wings seals and they regulate airflow around the blade and in the HPT stage 1 cavity. The aft side angel wings of all HPT stage 1 blades were damaged and sustained significant loss of material to all angel wings, the forward blade angel wings were all present and intact with only localized rubbing contact (Figure 42).

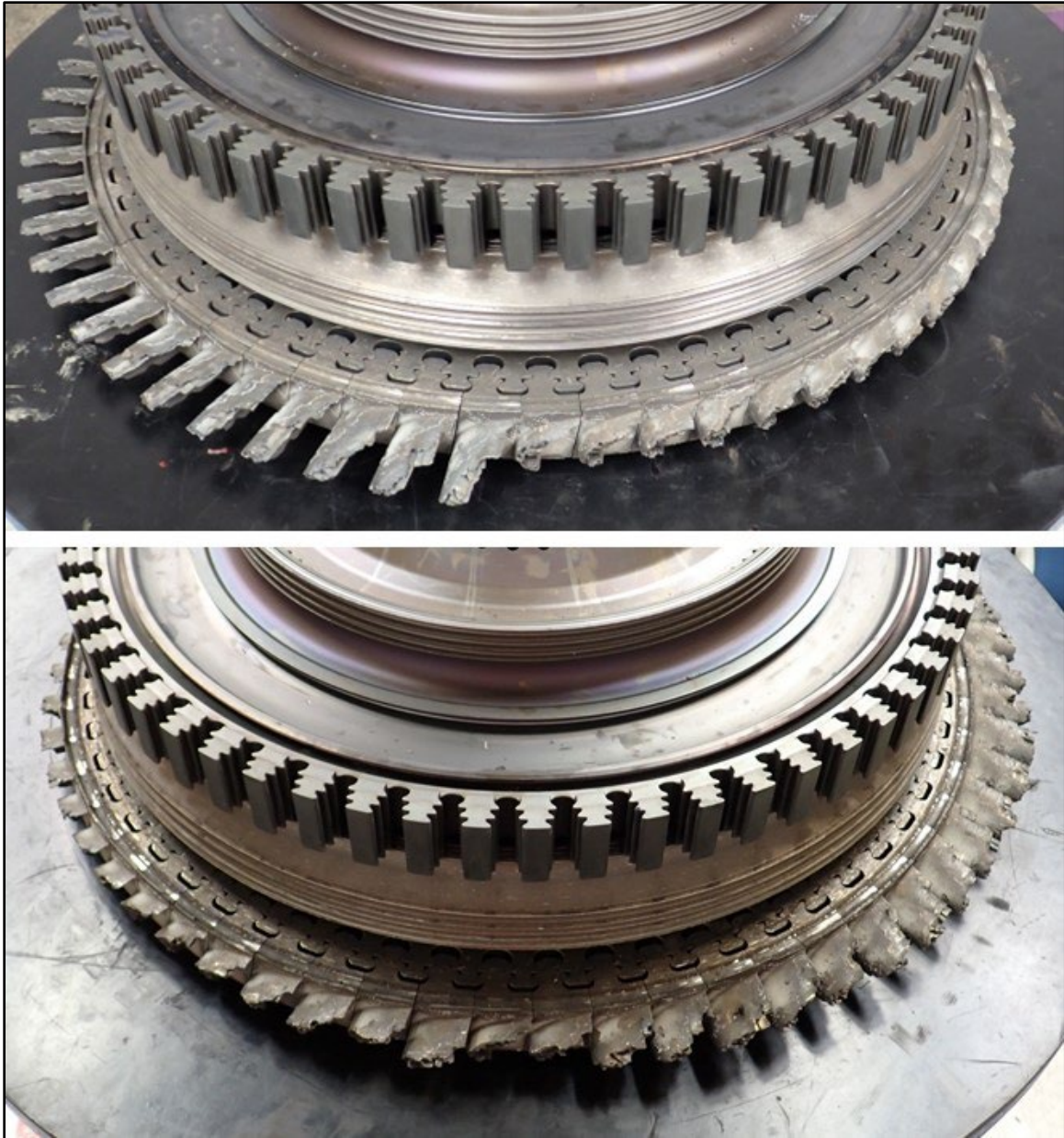


Figure 41. HPT Stage 1 Blades - Installed



Figure 42. HPT STAGE 1 BLADES REMOVED – SHORTEST AND LONGEST

The HPT rotor assembly was dimensionally checked on a precitech measuring machine to collect runout data for dimensions for potential later evaluation. Drop measurements were performed for dimension A and dimension B. See attachment 1.

The HPT rotor assembly was installed vertically on the stand to facilitate further disassembly and examination.

The HPT forward seal serves both as a blade retainer for the forward side of HPT stage 1 blades and also directs/modulates cooling air to the HPT stage 1 blades. The HPT forward seal appeared intact and visually in good condition. The forward side seal teeth were also visually in good condition with no rollovers, but coating chips were observed, which is consistent with rubs into the stationary honeycomb (Figure 43).

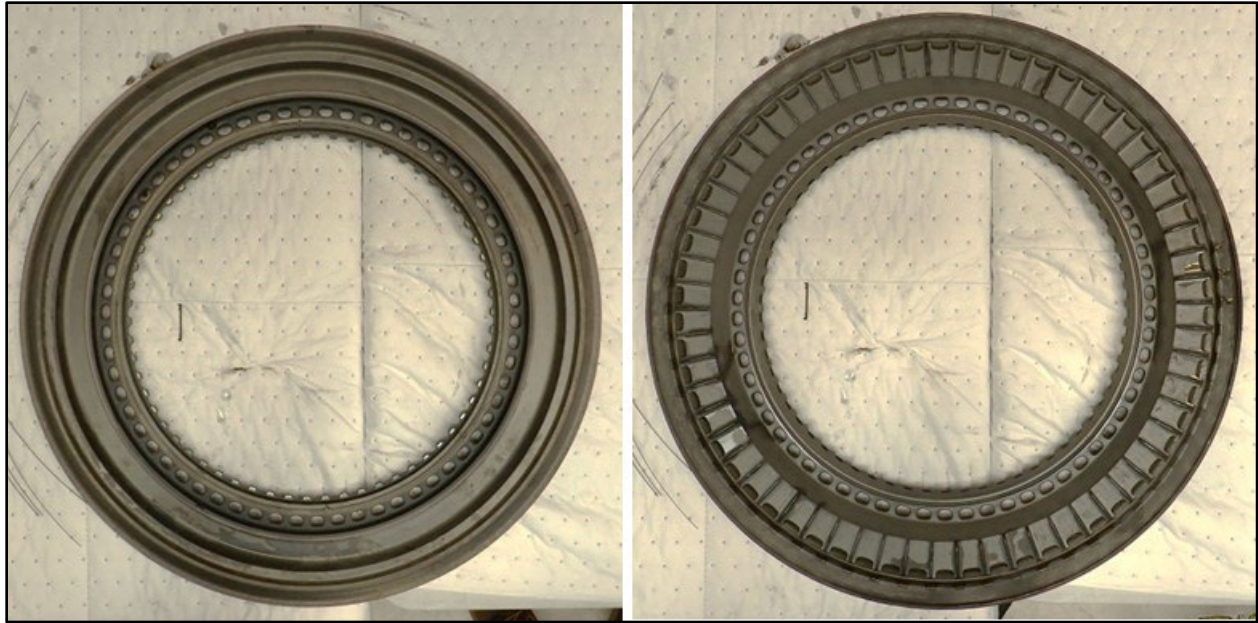


Figure 43. HPT FORWARD SEAL

During the removal of the HPT forward seal, the tooling made contact with the part and left a circumferential rub/cuff mark on the lower web of the forward side (Figure 44).

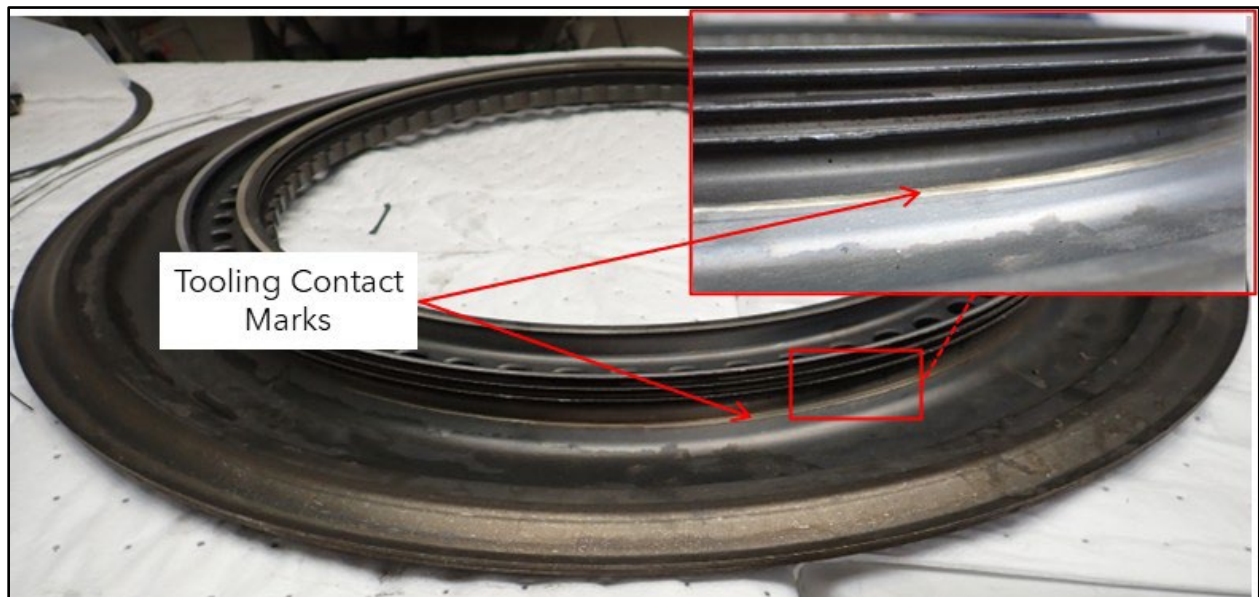


Figure 44. HPT FORWARD SEAL – DISASSEMBLY DAMAGE

A BSI of the joint between the HPT stage 1 disk and the HPT RIS was performed to evaluate the position of the HPT stage 1 aft blade retainer prior to removal of the coupling nut. The BSI showed the HPT stage 1 aft blade retainer was in the correct and seated position. The HPT coupling nut was removed from the HPT stage 1 aft shaft, the

threads and the splines of the coupling nut were visually in good condition and undamaged. The HPT stage 2 disk posts appeared intact and visually in good condition, the dovetails and splines on the inner diameter of the HPT stage 2 disk were visually in good condition and undamaged (Figure 45).



Figure 45. HPT STAGE 2 DISK

Drop check measurements were taken between the HPT stage 1 disk dovetail plane and that of the HPT RIS flange for later evaluation. Additional drop measurements were performed for dimension C. See attachment 1.

The HPT RIS is secured to the HPT shaft (which is integral with the HPT stage 1 disk) by a keyed/slotted locking ring on the HPT shaft, 4 anti-rotation locking keys/pins, and split ring. The HPT shaft keyed/slotted locking ring slides into a groove in HPT RIS aft flange to secure the HPT RIS axially. Four anti-rotation locking keys/pins, equally spaced 90° around, are used to prevent the HPT misdeal from rotating on the HPT shaft and the split ring it used to keep the anti-rotation locking keys/pins in place. The anti-rotation keys/pins were found in the typical condition (loose, not lodged or frozen). In order to remove the anti-rotation locking keys/pins, the lock ring, located within the HPT RIS aft flange groove, must first be removed.

The HPT RIS remained with the HPT stage 1 disk when the HPT stage 2 disk was removed. The HPT RIS sustained damage to all four of the seal teeth, circumferential scoring and contact marks were present on the seal teeth and the surfaces located forward and aft of the seal teeth. Once the assembly load between the HPT stage 1 disk and the HPT RIS was relieved, the HPT stage 1 aft blade retainer became visible on the forward side of the HPT RIS. As the HPT RIS was lifted, the HPT stage 1 aft blade retainer was observed to be partially attached to the HPT stage 1 disk.

In order to protect the HPT stage 1 aft blade retainer primary fracture surface from damage during its removal, metal blocks were placed between the HPT stage 1 disk and the HPT RIS around the outer diameter and on either side of the primary fracture surface of interest to prevent the HPT RIS and attached tooling from relaxing back onto the HPT stage 1 aft retainer after it disengaged from the HPT stage 1 disk (Figure 46). The hydraulic load was then released to allow tooling to be removed the disassembly process to continue.

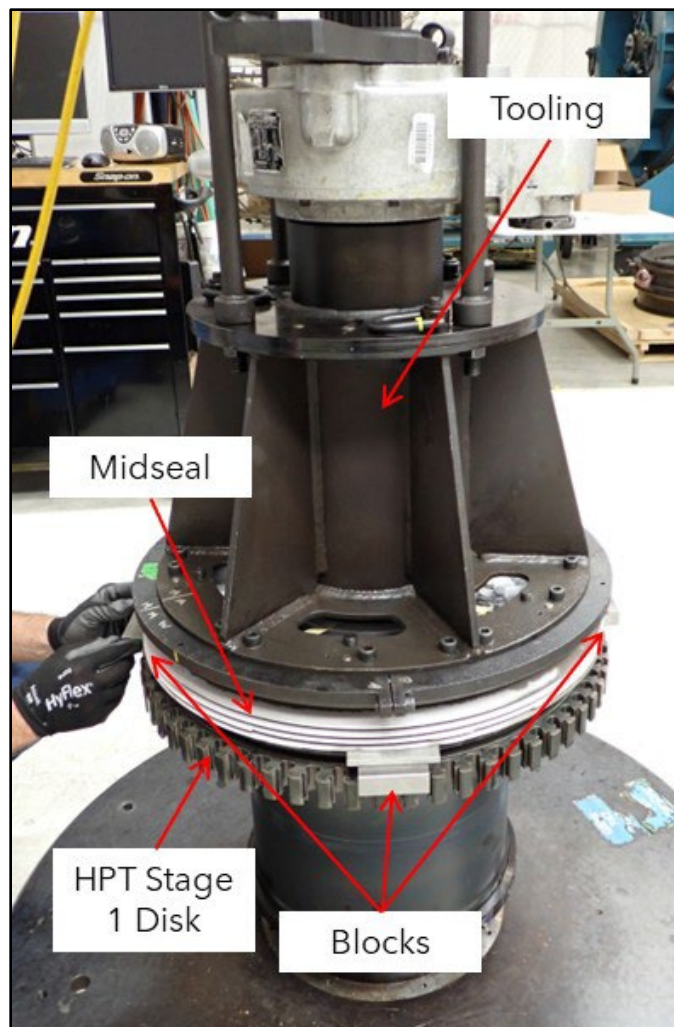


Figure 46. BLOCKED HPT STAGE 1 DISK AND HPT RIS

The HPT RIS and HPT stage 1 aft blade retainer assembly was transported to GE's material systems laboratory for initial photo documentation, and with approval from the NTSB, the metallurgist separated the HPT stage 1 aft blade retainer from the HPT RIS. Both parts were photo documented and an initial evaluation of the HPT stage 1 aft blade retainer fracture surface was performed (Figure 47 and Figure 48).

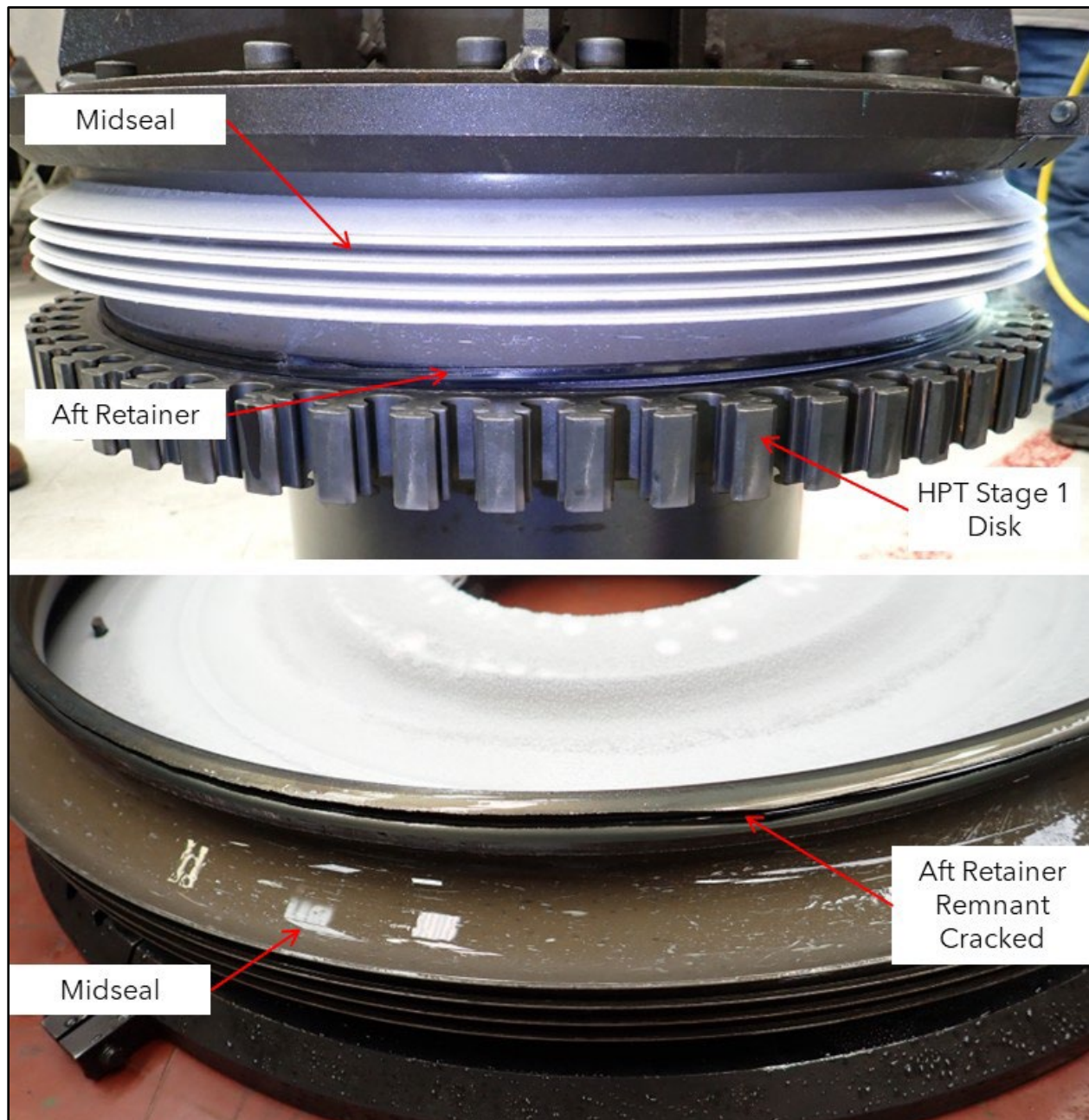


Figure 47. HPT RIS AND HPT Stage 1 AFT BLADE RETAINER

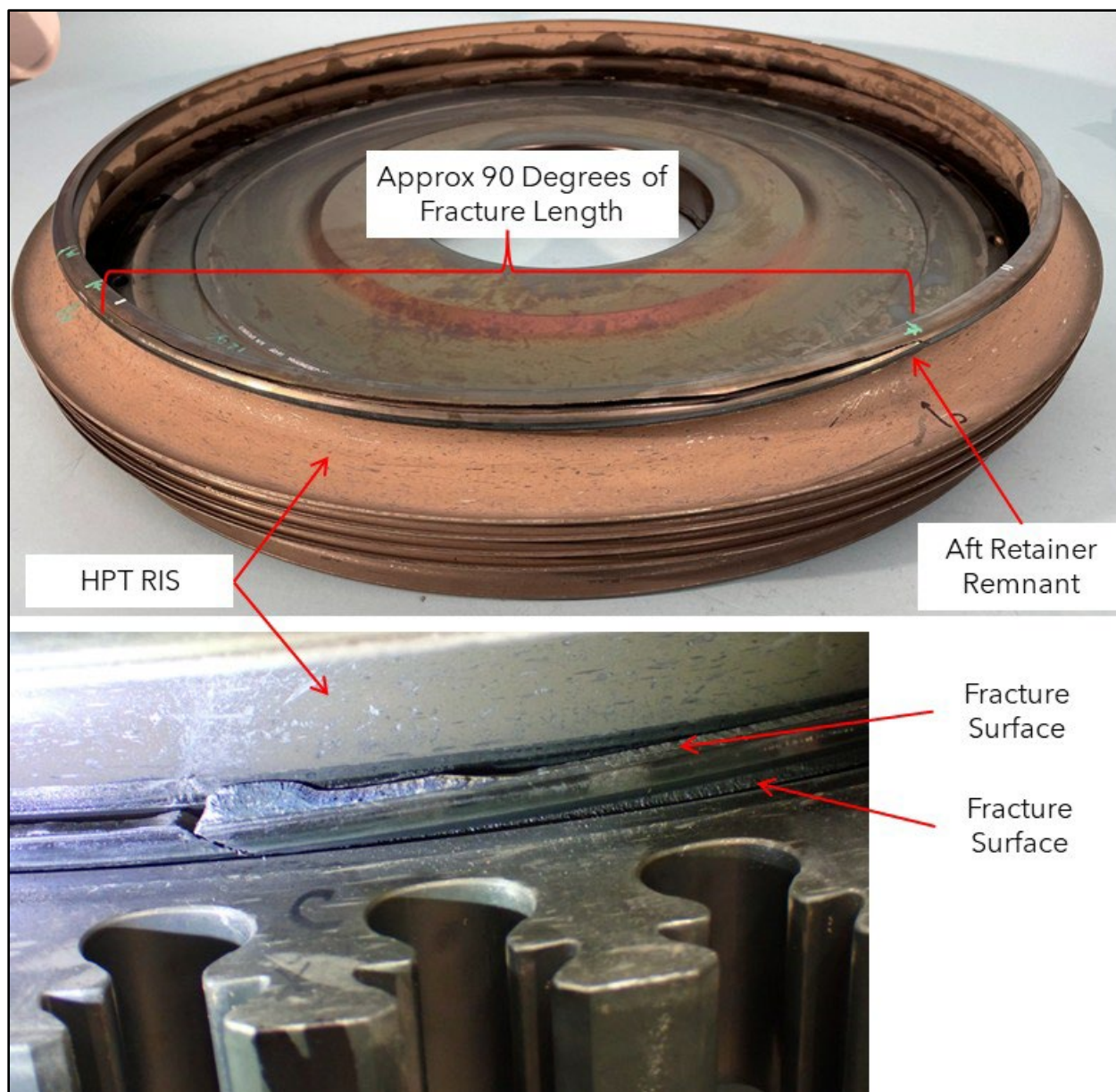


Figure 48. HPT RIS AND HPT Stage 1 AFT BLADE RETAINER

Once removed, the HPT stage 1 disk appeared intact and visually in good condition, the aft side of the disk posts exhibited impact/contact damage and rubbing contact was observed on approximately 90-degrees of the rim corner circumference, inboard of the dovetail slots (Figure 49).

All engine hardware was inventoried, repacked and stored in a secure trailer at GE's facility in Evendale, Ohio.



Figure 49. Stage 1 Disk

6.0 GE Aerospace Fire Assessment

A fire assessment of ESN 956102 external hardware from was performed by a GE Aerospace subject matter expert, which revealed no evidence of an under cowl fire. Localized mild heat distress was observed on some of the electrical wiring harnesses peek braid with no damage observed to the underlying wiring. Some of the fire loop grommets/isolators and some of the HPTACC and LPTACC rubber bellows showed evidence of localized mild heat distress and localized soot was observed on the nacelle bulb seal. All other remaining assessed hardware did not present evidence of heat or fire distress.

7.0 Metallurgical Analysis

This section provides a summary of more detailed information that is available in GE's Metallurgical Investigation Report that is attached to this report as Attachment 1, which will also be available in the investigation docket.

Various components were retained and transported to GE's material systems laboratory for further examination and dimensional verification, this included the remains of the HPT stage 1 aft blade retainer, the turbine case, the HPT RIS, HPT stage 1 turbine blades, and the HPT stage 1 turbine disk. All inspections (ex. visual, blue light, dimensional), testing, and analysis work performed on hardware other than the HPT stage 1 aft blade retainer found all dimensional and material characteristics to be conforming to GE's drawings and material specifications.

The remaining inboard portion of the HPT stage 1 aft blade retainer was dislodged from the HPT stage 1 disk and visually examined, which presented a 360-degree circumferential fracture surface located adjacent to the rabbet undercut fillet consistent with overstress (Figure 50). Additionally, a through-wall circumferential crack was observed at the inboard collet radius with a circumferential length spanning a sector of approximately 80-degrees consistent with primary fatigue (Figure 51). The two fracture surfaces connected at one end and at that transition region a parallel approximate 1.25 inch region consistent with secondary cyclic damage was observed in the undercut radius

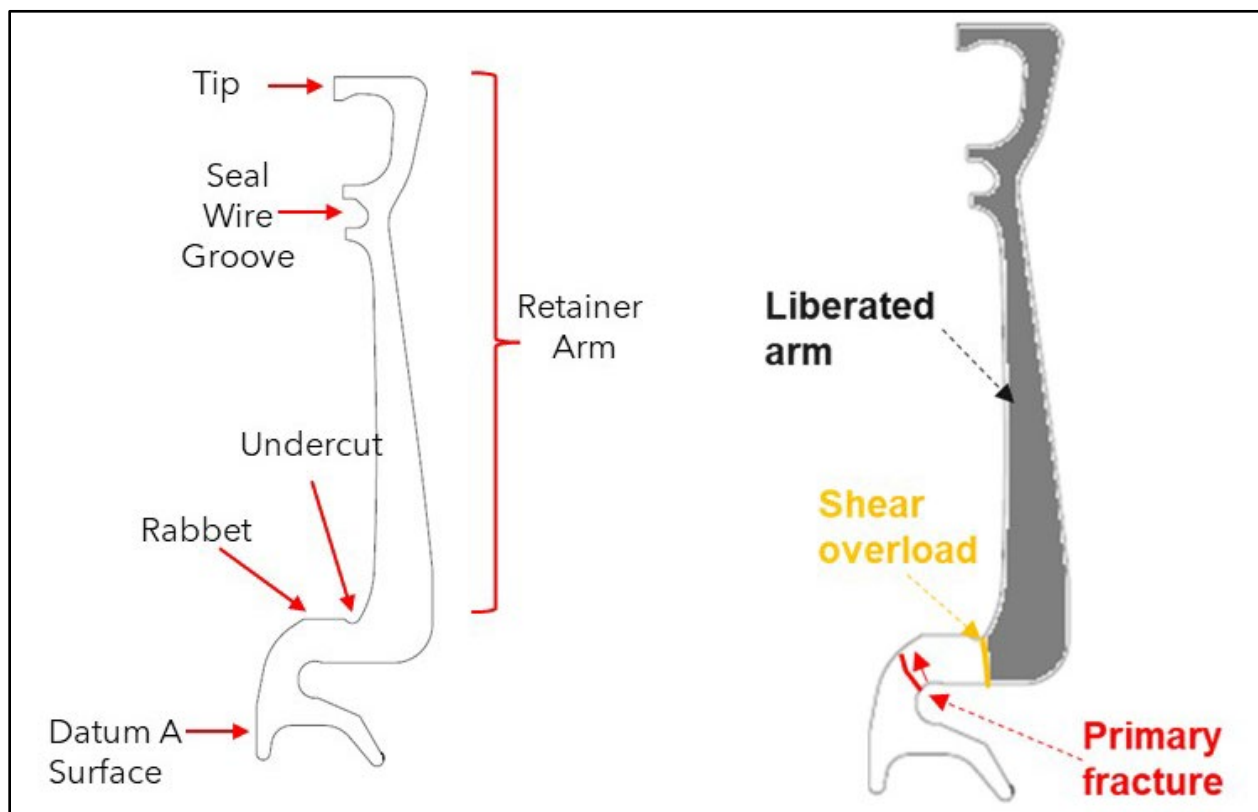


Figure 50. HPT Stage 1 Aft Blade Retainer - Cross Sectional View

Figure Courtesy of GE, Modified by NTSB

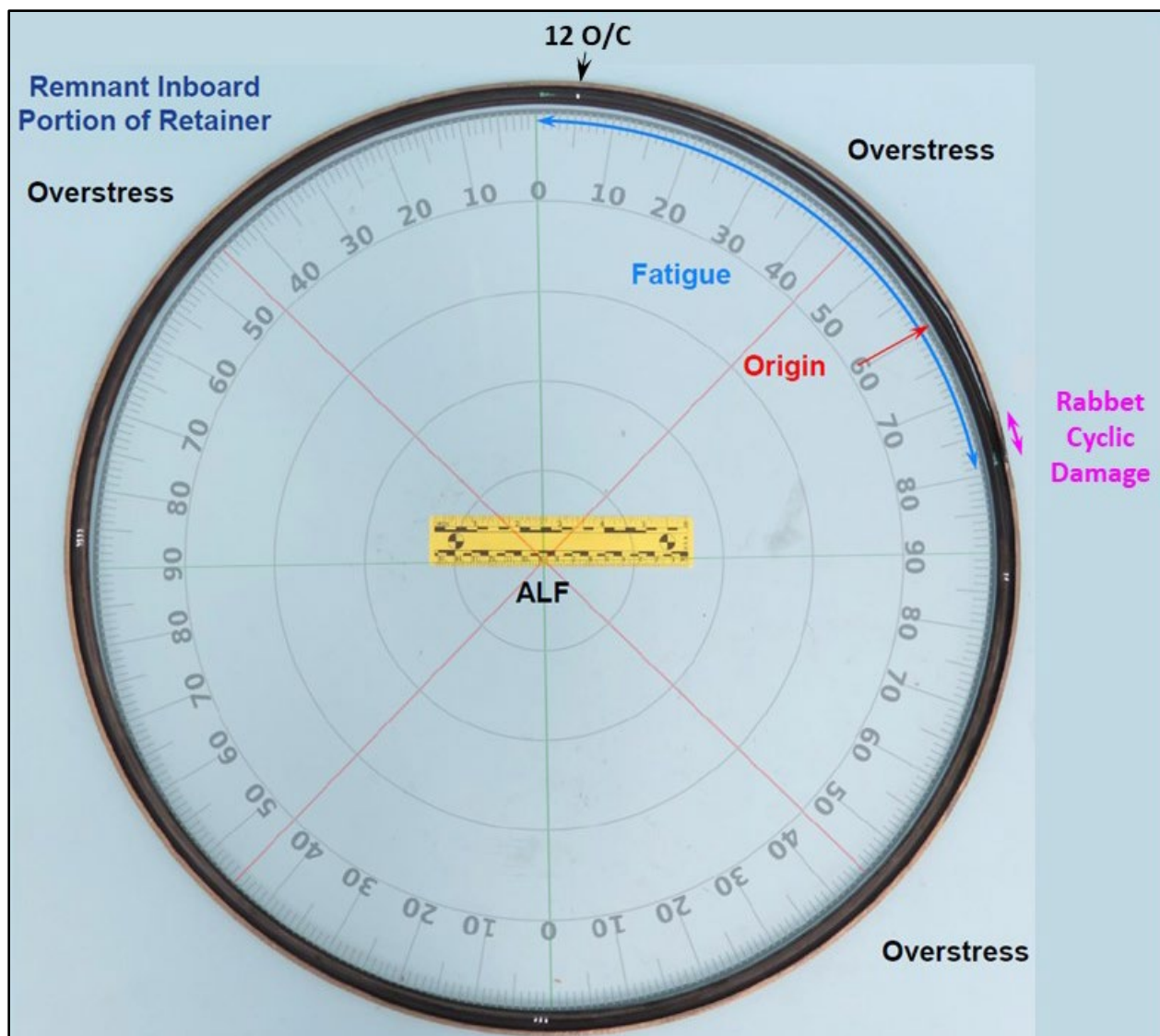


Figure 51. HPT Stage 1 Aft Blade Retainer

Figure Courtesy of GE, Modified by NTSB

Fractography revealed that the primary fatigue crack initiated in the upper collet radius at a single point of origin, with the origin plane approximately 0.048 inches aft of rabbet (Figure 52). The crack origin contained a surface connected anomaly that measured 0.005 inches by 0.015 inches by 0.0005 inches. Approximately 0.0017 inches of smeared material was observed at the origin which was added to the observed length of the inclusion to make it an estimated 0.015 inches long. The surface connected inclusion was analyzed with Energy-dispersive X-ray spectroscopy (EDS) and scanning electron microscopy (SEM) which determined the anomaly to be a cluster of particles enriched with Titanium, Aluminum, Zirconium, Magnesium, and Oxygen, which is consistent with OCNC⁸ particles (Figure 53 and Figure 54). This OCNC particle

⁸ The term Oxide/Carbonitride/Nitride/Carbide (OCNC) is a JENQC (Jet Engine Nickel Quality

cluster was also visible on the outboard side of the fracture but was significantly more smeared.

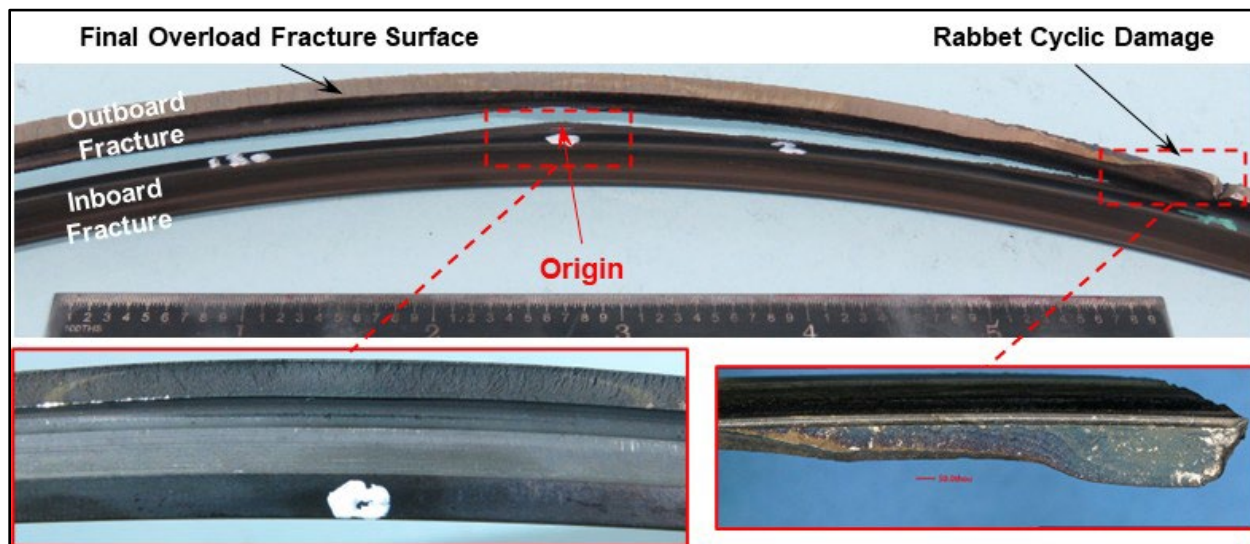


Figure 52. HPT Aft Blade Retainer Fracture Surfaces

Figure Courtesy of GE, Modified by NTSB

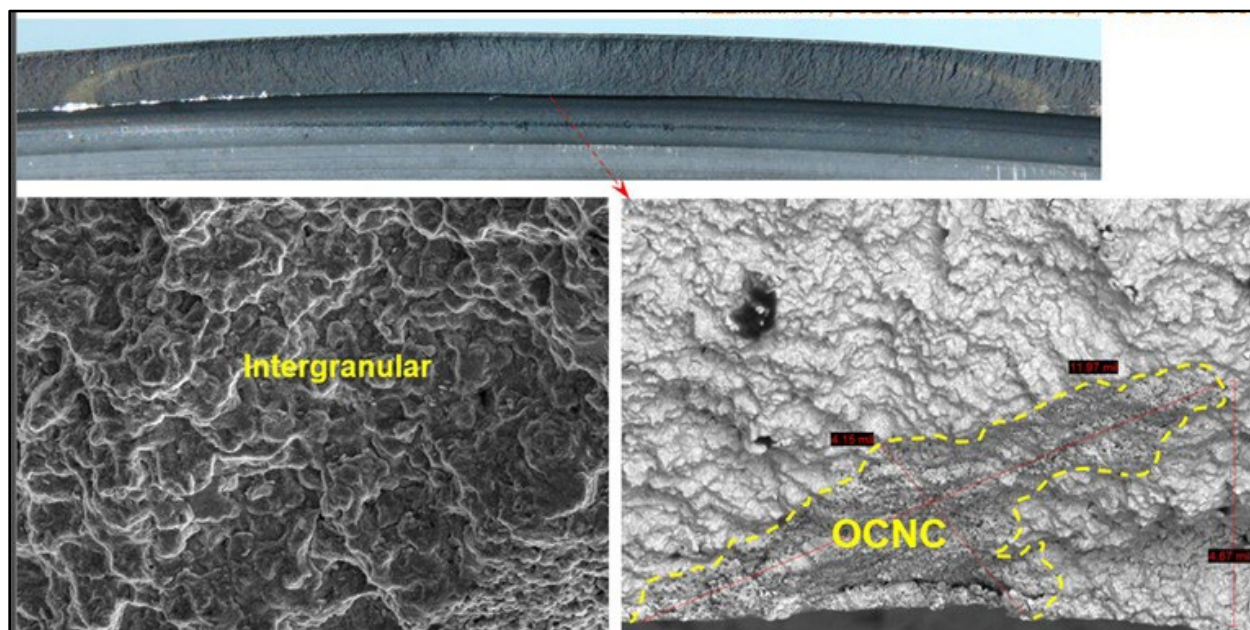


Figure 53. Inboard Side - Intergranular Fracture (L), OCNC Particle Cluster (R)

Figure Courtesy of GE, Modified by NTSB

Committee) defined standardized term to define specific melt-related anomalies (the group is formed by OEMs, the FAA, and melters). Similar terms are non-metallic inclusion (NMI) or to some degree stringer depending on size/shape and are a known nickel anomaly.

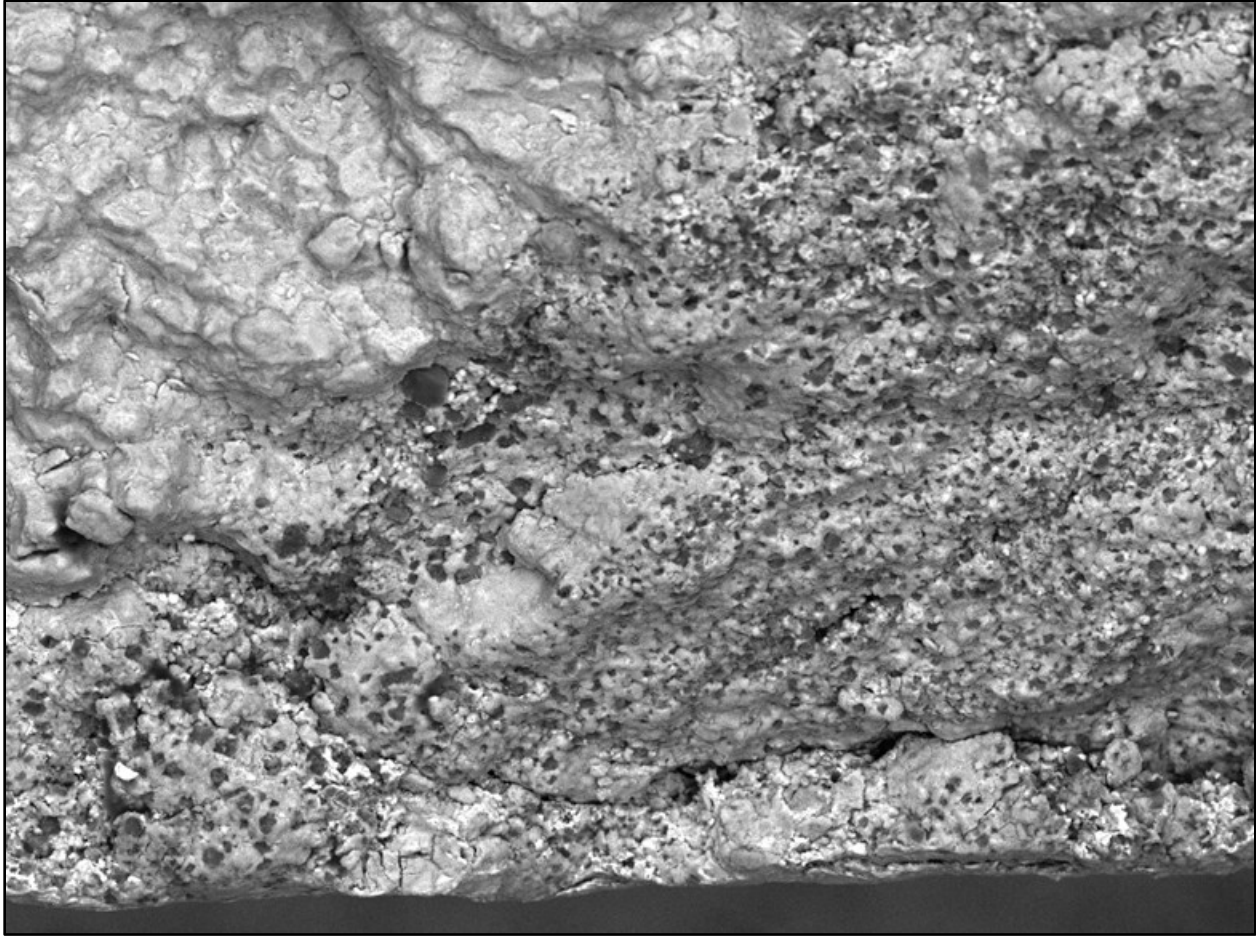


Figure 54. Close Up of OCNC Particle Cluster

Figure Courtesy of GE

Fractography of the collet crack, the 80-degree sector identified in the figures below as the inboard fracture, revealed the fracture surface surrounding the OCNC had intergranular morphology consistent with hold-time fatigue growth, also known as sustained peak low cycle fatigue (SPLCF). There were no visible striations; however, beach marks consistent with fatigue were observed on both sides of the origin. The fracture origin area was excised and mounted for further analysis.

An area of cyclic damage (secondary damage), approximately 1.25 inches long, was located coming out of the rabbet undercut and aligned with one end of the collet crack, the remaining circumference of the rabbet undercut crack plane had features consistent with overload (Figure 55). Fractography of the cyclic damaged area showed a fracture surface with intergranular features with some dimples which is consistent with a rapid hold-time fatigue/cyclic overload.

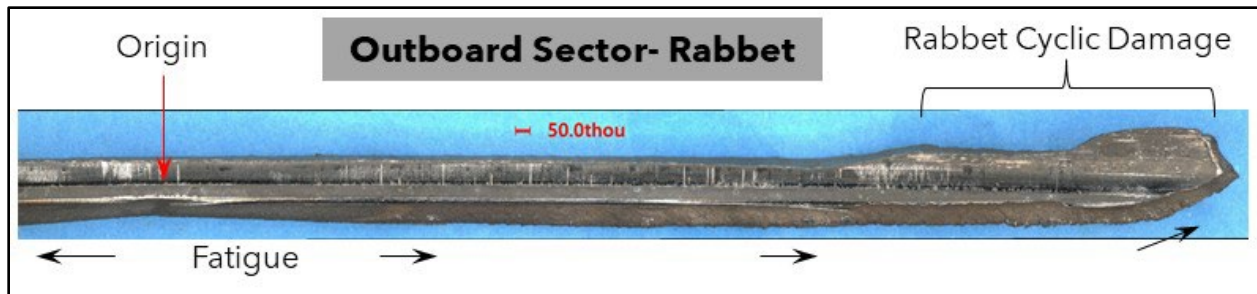


Figure 55. Outboard Side - Crack Origin (L), Rabbet Cyclic Damage (R)

Figure Courtesy of GE, Modified by NTSB

Additional cross sections of the HPT stage1 aft blade retainer (3:00, 6:00, 9:00 and 11:50 clock positions) were excised and mounted for additional analysis in order to compare dimensional and material characteristics with the origin area. The analysis results showed that all dimensions and material characteristics were found consistent and conforming except adjacent to the final overload fracture where the material was locally deformed.

No other surface anomalies were observed in the collet area and the collet surface condition was consistent with having been shot-peened and wet media blasted as required.

The material hardness was tested (micro-hardness using Vickers, 500g load) and found conforming to the material specification. The bulk material microstructure, average grain size and chemistry were found to all be consistent with properly processed Rene 65 material.

7.1 Detailed Analysis & Material Testing

This section provides a summary of more detailed information that is available in GE's Metallurgical Investigation Report that is attached to this report and will also be available in the investigation docket.

The HPT stage 1 aft blade retainer is made of a Rene 65 ring-rolled short transverse minimum LCF (with SPLCF) based material; this part does not have a chapter 5 life limit since the calculated life is sufficiently high and is not designated as a life-limited part, similar to other blade retainers for similar architecture engines. The collet feature has a calculated SPLCF life of greater than 200,000 cycles and meets the standard design practice of minimum-maximum-minimum 6,000 cycle requirement for damage tolerance. The collet feature experiences high first principal stress magnitudes (200+ kilopounds per square inch (ksi)) with a high mean stress, high R-ratio⁹ (0.5-0.7), and hold-time effects, which influences the crack initiation life.

⁹ R-ratio is the ratio of minimum stress divided by the maximum stress.

An initial fault tree was established to consider various aspects of the component design, specifically related to the collet feature, to determine and understand the root cause and whether there were any contributing factors with regard to crack initiation and failure of the HPT stage 1 aft blade retainer component (Figure 56).

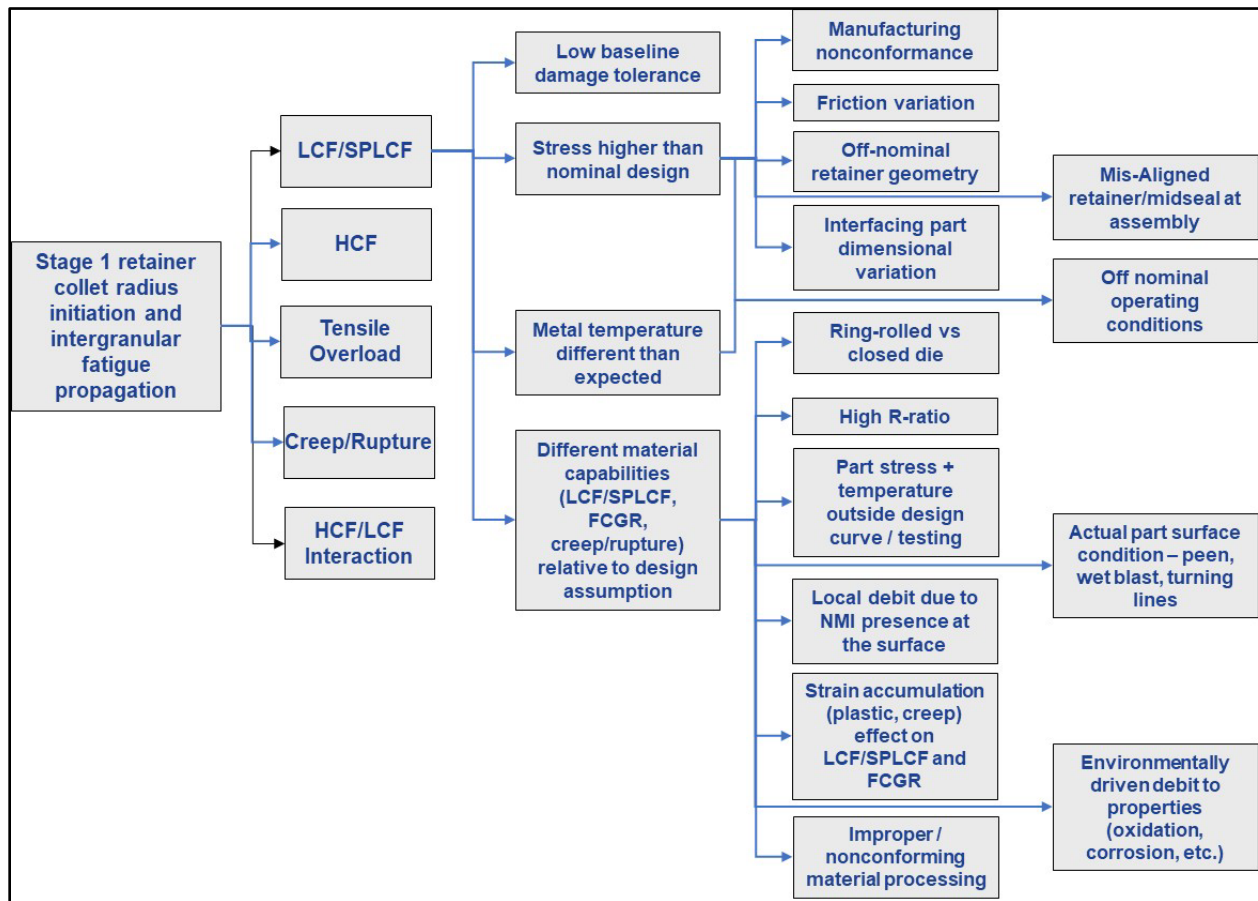


Figure 56. Initial Fault Tree Diagram

Figure Courtesy of GE

Analysis of the available test data suggests that at high stress conditions, surface connected OCNCs have a limited crack incubation/nucleation life and that surface connected OCNCs do not behave according to traditional crack growth threshold behavior. OCNCs are inherent inclusions to the triple melt manufacturing process for Rene 65 material and are present in the test bars used to establish the material capability. It is not uncommon for crack initiation to occur at OCNCs in test bars at high alternating stress and high temperature which suggests an incomplete understanding of the stress conditions that drive OCNC behavior at the unique mean and peak stress state experienced by the collet feature.

The existing stress models were used to assess the upper collet radius initiation life capability for ambient day, cold/warm start, thrust rating, and takeoff derate thrust

sensitivity. Retainer upper collet radius was calculated to have a high initiation life across all thrust ratings, hot ambient day, warm start, and derate thrust. A higher stress and/or debited material properties relative to the design curves was needed to close the gap between the calculated life and the observed life.

Due to the difference between the calculated life capability (e.g. crack initiation) and the quantity of cycles accumulated on each of the failed HPT stage 1 aft blade retainers, material sensitivity studies were identified to assess the sensitivity of the collet feature to various parameters. Material test plans were initiated to explore the Rene 65 material behavior and close the material lifing data gaps, with specific focus on LCF and SPLCF capabilities relative to engine operating conditions (high R-ratio and hold-times) to identify whether there's any influence on material properties, and if there are any gaps in the characterization of bulk material capability at the nonstandard retainer collet operating conditions. GE determined that the bulk material properties and model predictions were within expectations, which supports the local nature of the OCNC behavior in the root cause statement.

Feature testing was performed with test coupons excised from parts that were representative of the collet feature to assess material behavior and capability. It was found that activating surface connected OCNCs can reduce the fatigue life of the test specimen but not every OCNC would initiate a crack which suggests a bimodal behavior.

8.0 Related Events - HPT Stage 1 Aft Blade Retainer (Current Configuration)

In addition to the UAL (N27957) uncontained event, there was one additional uncontained failure event and one opportunistic finding (the part was found to be intact but cracked) during a shop visit. Both are currently being investigated (Table 2). The additional uncontained event occurred in November of 2023 on a foreign operated Boeing 787-9 powered by two GEnx™-1B76A/P2 turbofan engines. The opportunistic shop visit finding occurred during a scheduled removal/replacement of the HPT stage 1 turbine blades on a GEnx-1B74/75/P2 engine at the GE Caledonian shop in Prestwick, Scotland.

Table 2. History of Failed HPT Stage 1 Aft Blade Retainers

	Event Date	Retainer Cycles	Retainer Melt Manuf. & Date	OEM Ship Date	Retainer Installed	SV #1	SV#2
Operational Event #1 GEnx-1B76A/P2	Nov. 7, 2023	2046	April 8, 2020	Aug. 11, 2021	Oct. 2021	NA	NA
			Melt N0B14-1T2		MRO		
			23 inch flat		PRSV		
Shop Finding GEnx-1B74/75/P2	Dec. 2023	3110	May 10, 2016	Nov. 6, 2017	Jan. 2018	Apr. 2020	Dec. 2023
			Melt M6B62-5B2		Durham	Celma	Caledonia
			20 inch two-step		New	GVI	
UAL (N27957) GEnx-1B76A/P2 ESN 956102	May 14, 2024	1750	May 20, 2015	Jun. 20, 2016	Nov. 2016	May 2019	Dec. 2019
			Melt M5B70-3B2		Celma	EGAT	Caledonia
			20 inch two-step		PIP Upgrade	GVI	GVI

8.1 Summary of Operational Event No. 1

In November of 2023, a Boeing 787-9 powered by two GE Aerospace GEnx™-1B76A/P2 turbofan engines experienced an uncontained failure of the engine HPT and uncommanded inflight shutdown (IFSD) of the No. 1 (left) engine when it was reaching their assigned flight altitude of FL390 after departure. The flight crew reported vibration from the No. 1 engine and an EICAS (Engine Indicating and Crew Alerting System) message for “ENG Fail L” and a left engine Fire Warning were annunciated to the flight crew. The flight crew secured the engine, discharged both fire bottles, and performed an inflight air turnback where an uneventful single engine landing was made. Upon landing, ground inspection of the No.1 engine revealed metal in the tailpipe, damage and exit penetrations in the right side thrust reverser cowl, exit penetration hole in the HPT case, and missing and damaged HPT rotor parts.

At the time of this uncontained failure, the engine had an operating time of 6,939 CSN and contained HPT stage 1 aft blade retainer, P/N 2383M99P02 (Rene 65 material), S/N TMT1JEP9, with a time of 2,046 CSN. This HPT stage 1 aft blade retainer was produced from the material melt lot S/N N0B14-1T2 and manufactured April 8, 2020, and subsequently installed into the event engine in October of 2021 along with other used hardware during a performance restoration shop visit (PRSV).

An engine examination/teardown was conducted, the remaining portion of the HPT stage 1 aft blade retainer was removed and transported to GE’s material systems laboratory for examination. Examination found a 115-degree sector of the retainer contained a fracture surface in the collet area with the remaining circumference showing overload failure at the rabbet fillet, the fracture emanated from a single point of origin with intergranular crack propagation that is consistent with hold-time fatigue crack growth. The origin area on the inboard side of the fracture was smeared and uninterpretable due to secondary damage from mid seal contact, the outboard side of the fracture was not found. Metallography of the microstructure and chemical analysis showed the material was consistent with properly processed Rene 65 material.

8.2 Summary of Opportunistic Find at GE Caledonian

An engine was inducted into the GE Caledonian engine shop facility in December of 2023 for scheduled maintenance due to HPT stage 1 blade distress. The HPT module and HPT rotor assembly was fully disassembled to the piece part level. During the disassembly process, the technicians observed that the HPT stage 1 aft blade retainer released from the HPT stage 1 disk without the usual “pop” sound that is typically heard. The HPT stage 1 aft blade retainer, P/N 2383M99P02 (Rene 65) and S/N TMT11CNF, was removed from the rotor assembly and a through thickness crack was

found between the collet feature and the forward side of the retainer near the rabbet, spanning approximately 1/3 of the part circumference. At the time of the finding, the HPT stage 1 aft blade retainer had a time of 3,110 CSN.

This HPT stage 1 aft blade retainer was produced from the material melt lot S/N M6B62-5B2 and manufactured on May 10, 2016, shipped to the GE Durham engine shop facility in North Carolina and was subsequently installed into the event engine in January of 2018. The engine was later inducted twice for shop visits, the first shop visit was in February of 2020 (1,670 CSN) for HPT stage 1 blade damage and the second shop visit in April of 2020 for hot section distress and HPT stage 2 rotor replacement (the remaining rotor assembly hardware, including the aft retainer, was not de-stacked).

The HPT stage 1 aft blade retainer was transported to GE's material systems laboratory for examination, which found a 120-degree sector of the retainer with a through thickness fracture in the collet area. Fractography of the outboard side of the fracture revealed crack initiation from a single point of origin coinciding with a surface connected OCNC that measure 0.003 inches by 0.0015 inches by 0.0005 inches and intergranular crack propagation that was consistent with hold-time fatigue crack growth. The origin area on the inboard side of the fracture was smeared and uninterpretable due to secondary damage. Metallography of the microstructure and chemical analysis showed the material was consistent with properly processed Rene65 material.

8.3 Summary of UAL Failed HPT Stage 1 Aft Blade Retainer Information

As stated in Section 2 of this report, ESN 956102 from the UAL event contained HPT stage 1 aft blade retainer S/N TMT4TK46 with an operational time of 1,750 CSN, was produced from the material melt lot S/N M5B70-3B2 that was manufactured May 20, 2015, then shipped to the GE Celma engine shop facility on June 20, 2016 and subsequently installed into ESN 956102, the event engine.

GE's material laboratory analyzed all three HPT stage 1 aft blade retainers and found the fracture surface characteristics (initiation, propagation), were all consistent with each other. Two of the three HPT stage 1 aft blade retainers were confirmed to have a surface connected OCNC at the point of crack initiation.

9.0 HPT Stage 1 Aft Blade Retainer (Previous Configuration)

9.1 HPT Stage 1 Aft Blade Retainer Related Events

Prior to the events and shop finding mentioned above, there were 2 additional HPT stage 1 aft blade retainer failure events in 2016 and 2017 that led to engine IFSDs

but not uncontained turbine case failures. These failure events were investigated by GE Aerospace and they determined that the failures were due to a nonconformance in the perpendicularity of the rabbet feature relative to datum A. This nonconformance was found to increase stress in the undercut radius adjacent to the rabbet, causing early crack initiation and propagation to the point of retainer arm separation.

The HPT stage 1 aft blade retainers currently under investigation contain cracks that initiated from the collet feature whereas the retainers from the prior 2016/2017 events contained cracks that initiated at the rabbet undercut (Figure 57). The HPT stage 1 aft blade retainers currently under investigation do not have the same rabbet perpendicularity nonconformance as the prior 2016/2017 events.

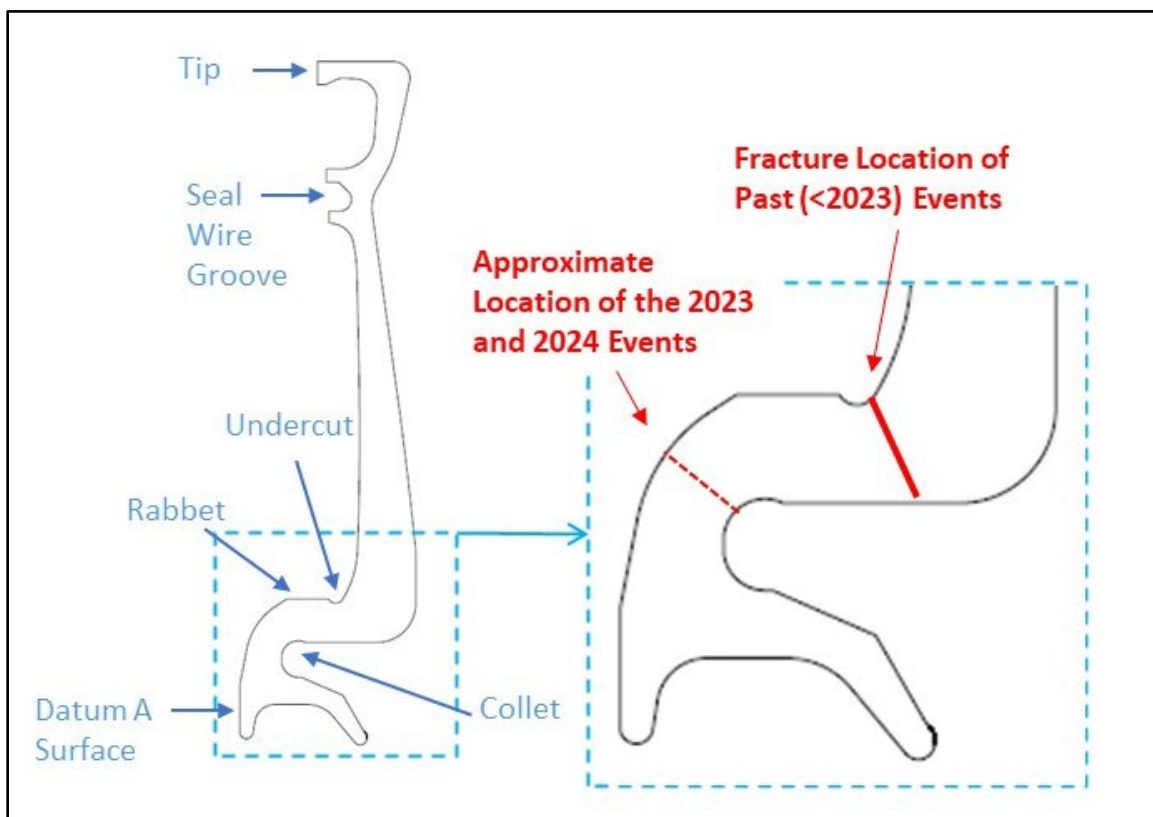


Figure 57. HPT Stage 1 Aft Blade Retainer – Current & Past Fracture Locations

Figure Courtesy of GE, Modified by NTSB

GE Aerospace pursued containment actions following the investigation of the 2016/17 events, identified all suspect parts and incorporated changes to the manufacturing quality assurance processes. GE also updated the manufacturing drawings to ensure the rabbet perpendicularity requirements were appropriately understood by the manufacturers. The drawing changes also included updates for the tip drop and seal wire groove depth to clarify design requirements and provide additional tolerances for the tip drop dimension to improve producibility. Additionally, a quality control inspection fixture was incorporated at the manufacturing level starting January

2016 for PN 2383M99P02 (Rene 65 material). The three HPT stage 1 aft blade retainers currently under investigation were all shipped from the manufacturer after January 2016 when the use of the inspection fixture was implemented.

9.2 Design/Configuration History

The original baseline configuration of the GENx HPT rotor assembly is known as Configuration 1 whereas the currently fielded HPT rotor assembly is known as configuration 2 (Figure 58). The configuration 2 rotor assembly was introduced with the GENx-1B/P2 and GENx-2B/P2 models, and its main differences relative to configuration 1 include: 1) a straight forward shaft on the HPT stage 1 disk, 2) a redesigned interface between the HPT stage 1 disk, HPT stage 1 aft blade retainer, and HPT RIS to introduce the collet design, and 3) Rene 65 material was introduced for the HT stage 1 aft blade retainer, HPT RIS, and HPT stage 2 disk. The configuration 1 rotor makes up approximately 25% of the current GENx fleet with the balance of the fleet containing the configuration 2 rotor. This investigation is focused on the configuration 2 rotor because the subject HPT stage 1 aft blade retainer configuration with the collet feature is applicable only to the configuration 2 rotor. There are a limited number of configuration 1 assemblies still in use, but this investigation is solely focused on the configuration 2 assembly.

There are two part numbers for the HPT stage 1 aft blade retainer in the configuration 2 HPT rotor assembly that are currently in use, P/N 2445M91P01 and P/N 2382M99P02, both part numbers are dimensionally identical. The original part number released in the configuration 2 HPT rotor assembly was P/N2445M91P01, made of Rene 88 material which is a powder metal¹⁰ based material. GE then released P/N 2382M99P02, made of a Rene 65 material which is a ring rolled forged¹¹ material, for the purpose of increasing manufacturing capacity. Production of the new part, P/N 2382M99P02, began on February 11, 2011, and GE discontinued production of the original part, P/N N2445M91P01, on December 13, 2014. As both part numbers being dimensionally identical, there are no differences in the assembly process or special conditions/requirements when installing P/N 2382M99P02 into an HPT rotor assembly.

¹⁰ Powder metal based parts are produced by using the powder metallurgy processes from metallic or nonmetallic particulate to create a discrete, or near net, shape.

¹¹ Ring rolled forging is a forging process involving extreme pressure and heat to form seamless circular rings of a material.

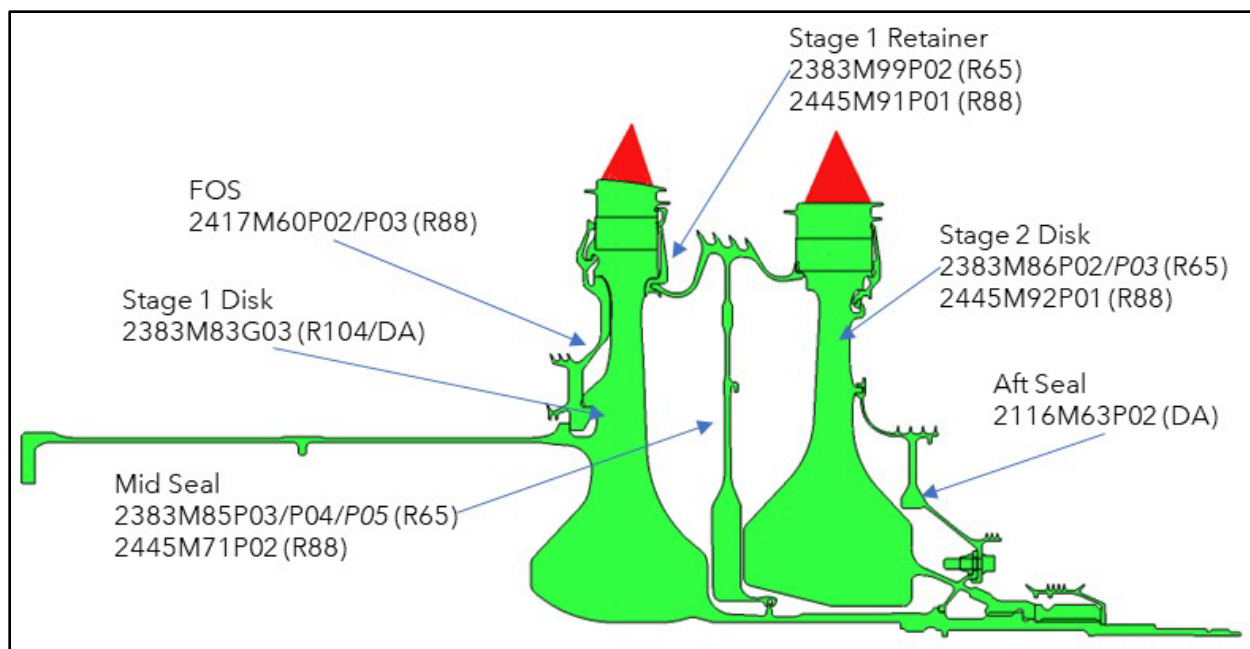


Figure 58. HPT Rotor Assembly (Configuration 2)

Figure Courtesy of GE, Modified by NTSB

At the time of the 2016/17 IFSD events mentioned above, both part numbers (P/N 2445M91P01 and P/N 2383M99P02) were installed in GEnx engines and in use by GEnx owners/operators. As mentioned above, GE updated the manufacturing drawings, for both part numbers, to clarify design and manufacturing requirements related to rabbit perpendicularity, tip and seal wire groove.

10.0 Containment Actions by GE Aerospace

Multiple containment actions were undertaken by GE to identify suspect HPT stage 1 aft blade retainers and to continuously monitor the fleet for changes to core vibration trends. The containment actions include the release of service bulletins instructing the nondestructive inspections and the use of a data analytic tool by GE personnel.

GE updated the life management strategy for the aft retainer with instructions to remove, scrap, and replace all HPT stage 1 aft blade retainer, P/N 2383M99P02, with a new zero time part.

10.1 GE Fleet Analytic Tool

GE developed a rules-based data analytic tool to evaluate and monitor engine core vibration trends (core vibes-rear accelerometer (ZVB2R)) for all engines within the GEnx-1B and GEnx-2B engine fleets. The data analytic tool was based on the engine

performance/vibration data from the three engines that contained the HPT stage 1 aft blade retainers failures/cracks, and rules established such that the core vibration shift of these engines would have issued an alert for their removal.

In the event the data analytic tool identifies a specific engine with a core vibration alert, a more thorough assessment will be performed by GE's engineering team on a case-by-case basis and in accordance with a decision tree process that accounts for other variables such as operational factors and time since overhaul. Not every vibration alert will result in the issuance of a customer notification report (CNR) but if all decision tree criteria are satisfied, GE would issue a Class A2 CNR to remove the engine from operation within 10 cycles .

GE's core vibration data analytic tool has been running continuously on a daily basis since July 1, 2024, and to date there have only been two engines that alerted for a core vibration shift that resulted in the issuance of a Class A2 CNR to the operator (the FAA is notified Title 14 CFR 21.3 failure, malfunctions or defects report). One engine was found to have HPT stage 1 blade platform distress that resulted in the core vibration shift. The second engine, operated by a foreign carrier, was alerted by the analytic tool in February 2025 and subsequently removed the engine from service for induction into a maintenance facility for further assessment. An ultrasonic inspection resulted in response signals that were indicative of a defect in the collet, disassembly of the engine/module to the piece part level allowed visual inspection of the whole HPT stage 1 aft blade retainer, specifically the collet feature, which confirmed the presence of a crack in the collet. This HPT stage 1 aft blade retainer was shipped to GE's material systems laboratory for further examination.

10.1 Service Bulletins GEnx-1B-72-0546 and GEnx-2B-72-0475

On June 5, 2024, GE Aerospace released two Category 5 (Impact A) service bulletins, service GEnx-1B 72-0546 (Revision R00) and SB GEnx-2B 72-0475 (Revision R00). SB GEnx-1B 72-0546 is titled "ENGINE - P2 HPT ROTOR ASSEMBLY (72-53-00) - RECOMMENDATION TO REPLACE THE HPT STAGE 1 BLADE RETAINER P/N 2383M99P02 AT PIECE-PART EXPOSURE". SB GEnx-2B 72-0475 is titled "ENGINE - P1 HPT ROTOR ASSEMBLY (72-53-00) - RECOMMENDATION TO REPLACE THE HPT STAGE 1 BLADE RETAINER P/N 2383M99P02 AT PIECE-PART EXPOSURE".

GE's intent with these SBs was to provide the recommendation and instructions for the removal of all affected HPT stage 1 aft blade retainer, P/N 2383M99P02, when exposed to the piece-part level and replacement with a new HPT stage 1 aft blade retainer after HPT rotor piece-part disassembly. Additionally, these SBs provide the instruction to clean and fluorescent penetrant inspect (FPI) the removed HPT stage 1 aft blade retainers and report the component's identification information and FPI results back to GE Aerospace for documentation.

At the time this report was being completed, there were at least 411 HPT stage 1 aft blade retainers that have been removed/inspected and there has been zero crack findings to date.

10.2 Service Bulletins GEnx-1B 72-0551 and GEnx-2B 72-0478

On August 26, 2024, GE Aerospace released two Category 4 (Impact A), SB GEnx-1B 72-0551 (Revision R00) and SB GEnx-2B 72-0478 (Revision R00), both are titled "ENGINE - HPT ROTOR ASSEMBLY (72-53-00) - HPT STAGE 1 BLADE RETAINER ULTRASONIC INSPECTION AT MODULE LEVEL".

GE's intent with these SBs was to provide the instructions and procedure for the ultrasonic inspection of the HPT stage 1 aft blade retainer, P/N 2383M99P02, when exposed at the HPT rotor assembly level. At the module level, the HPT stage 1 aft blade retainer is not disassembled to the piece part level and thus is not subjected to SBs GEnx-1B-72-0546 and GEnx-2B-72-0475 for removal/replacement. Any ultrasonic indication that exceeds the limit as defined in the applicable SB shall be cause for the HPT stage 1 aft blade retainer to be removed from service.

At the time this report was being completed, there were at least 51 modules that have been subject to the ultrasonic inspection and there have been zero crack indications to date.

10.3 Long Term Containment Plan

GE has begun taking steps toward re-establishing the manufacturing process for the previously approved retainer configuration, a Rene 88 alloy with P/N 2445M91P01, of the HPT stage 1 aft blade retainer with the long-term plan of making this Rene 88 retainer the primary configuration. The Rene 65 retainer, P/N 2383M99P02, will remain in production until production of the Rene 88 retainer has ramped up sufficiently to meet customer demand, at which point the production of the Rene 65 retainer will be terminated. Once manufacturing/production fully switches to the Rene 88 retainer, a change in design (CID) will be written to remove the Rene 65 retainer from the production rotor assembly parts list and update part drawings. The Rene 65 retainer, P/N 2383M99P02, will remain certified and listed in the illustrated parts catalog (IPC) but it will no longer be available as a spare part, further diminishing the population of the Rene 65 retainers through attrition.

The overall schedule for this transition is currently being developed by GE and will take into consideration various factors such as customer overhaul schedules, raw material availability and the forging house capacity.

G. ATTACHMENTS

Attachment 1. GE Aerospace - Metallurgical Investigation Report (DRB AvEng-558171.A)

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