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

June 28, 2016

POWERPLANT GROUP CHAIRMAN'S FACTUAL REPORT

NTSB No: DCA15FA185

A. ACCIDENT

Location: McCarran International Airport

Date: September 8, 2015

Time: 1613 Pacific daylight time

Aircraft: British Airways, Boeing 777-236ER, registration number G-VIIO, flight number 2276

B. POWERPLANT GROUP

Safety Board Powerplant Group Chairman: Jean-Pierre Scarfo
Powerplant Lead Engineer
Washington D.C.

General Electric Members: Les McVey
Flight Safety Investigator - Consulting Engineer
Cincinnati, Ohio

David Zigan
Flight Safety Investigator - Principal Engineer
Cincinnati, Ohio

Boeing Member: Michael Germani
Propulsion Safety
Seattle, Washington

Federal Aviation Administration Member: John Frost
Aerospace Engineer
Burlington, Massachusetts

United Kingdom Accredited Representative Air Accident Investigation Branch:	Justin Doxey Inspector of Air Accidents (Engineering) Hampshire, United Kingdom
United Kingdom Technical Advisor British Airways Member:	Trevor Bertrand Powerplant Engine and Fuel Technical Manager Heathrow Airport, Hounslow Middlesex, United Kingdom

C. SUMMARY

On September 8, 2015, at about 1613 Pacific daylight time (PDT), a British Airways flight 2276, a Boeing 777-236ER, registration number G-VIIO, powered by two General Electric GE90-85BG11 turbofan engines experienced a No. 1 engine (left) uncontained failure and subsequent fire during the takeoff ground roll on runway 07L at McCarran International Airport (LAS), Las Vegas, Nevada. The flightcrew aborted the takeoff, stopped the aircraft on runway 07L, and evacuated the airplane. The No. 1 engine, the inboard left wing, and a portion of the left fuselage experienced fire damage. The fire was extinguished by airport rescue and firefighting (ARFF) after the evacuation started. The 157 passengers, including 1 lap child, and 13 crew members evacuated via emergency slides on the runway. Initial reports indicated there were 5 minor injuries as a result of the evacuation (mostly abrasions). The airplane was substantially damaged. The flight was operating under the provisions of 14 Code of Federal Regulations (CFR) Part 129 flight from LAS to London-Gatwick International Airport (LGW) Horley, England.

On scene examination of the airplane and engine by the Powerplant group from September 9 – 15, 2015 revealed the following noteworthy damage to the No. 1 engine: 1) the right side (aft looking forward) of the engine cowls exhibited extensive thermal/fire damage and missing material throughout with the thrust reverser exhibiting the most distress, 2) the right thrust reverser translating sleeve exhibited 2 penetration holes located around the 1:30-2:00 o'clock position, 3) the right thrust reverser was missing much of the inner wall exposing the core of the engine, 4) the right side of the No. 1 engine strut exhibited some minor thermal distress, but no burn through holes or strut hole penetrations were noted, 5) the right engine thrust link, located at about the 1:00 o'clock position was severed in-plane with the high pressure compressor aft extension case, 6) main fuel supply line was intact; however, it was detached from the engine main fuel pump inlet, 7) the electrical leads for the hydromechanical unit high pressure shutoff valve were severed, 8) the high pressure compressor aft extension case was breached circumferentially from about the 11:30 – 5:00 o'clock position (58 inches circumferentially) exposing the internal compressor parts, 9) the high pressure compressor stage 8-10 spool was separated aft of the middle seal teeth between stages 8 and 9 and the stage 8 disk separated below the circumferential blade slot, and 10) several loose pieces were collected from the runway and surrounding area that were consistent with high pressure compressor stage 8-10 spool's stage 8 disk outer rim.

Disassembly of the engine was performed at the GE facility in Evendale Ohio from September 21-24, 2015 in the presence of the Powerplant Group. The high pressure compressor aft extension case was cut to gain access to the high pressure compressor (HPC) stage 8-10 spool. The HPC stage 8 disk was fully detached from the stage 7 and 9 disks and was fractured circumferentially about 200° around, just at the transition radius of the web to the blade circumferential dovetail slot rim. About 37 inches of the dovetail slot rim remained attached to the stage 8 disk. The impeller tube assembly was fractured and detached from the front side of the stage 8 disk and there was a mix of impeller tubes fractured, fairly intact, and some splayed open. Much but not all of the stage 8 disk forward rabbeted flange had fractured from the stage 8 disk and remained attached to the stage 7 disk. The rear drive arm of the high pressure compressor stage 8-10 spool was fractured and separated 360° in the general vicinity of the compressor discharge pressure seal land.

The pieces of the high pressure compressor recovered on site were first sent to the NTSB material laboratory for initial metallurgical examination. The received pieces accounted for less than 50% of the stage 8 outer rim. One of the recovered pieces with the dovetail slot and forward seal teeth still attached exhibited a hemi-elliptical shaped flat-fracture region that initiated in fatigue on the aft face of the web and transitioned circumferentially in both directions. A field emission scanning electron microscope was used to examine the surface in greater detail and the fracture region revealed an intergranular appearance near the initiation site and striations in the transgranular region of the fracture where striation density measurements could be taken to estimate the number of flight cycles from initiation/detection to failure. The NTSB materials laboratory estimated the number of flight cycles from detection to separation to be approximately 5,400 cycles. The last inspection of the event high pressure compressor spool was conducted by GE Wales 3,943 cycles prior to the failure.

After the initial examination of the high pressure compressor stage 8 pieces by the NTSB materials laboratory, they were sent to the GE Aviation Material Laboratory for further evaluation along with the stage 8 disk that was removed during the engine disassembly. Over several weeks, persons from the NTSB and FAA participated and oversaw much of the additional examination and testing. With all the recovered parts of the high pressure compressor stage 8 disk put together, it appeared that the entire disk rim and web was accounted for. GE's analysis of the stage 8 disk concluded: 1) the crack initiation propagated with intergranular features with local variations consistent with hold-time, high-alternating stress, low cycle fatigue (hold-time low cycle/sustained-peak, low cycle) consistent with the NTSB finding, 2) no microstructural anomalies or detrimental species in the grain boundaries were found near the fracture origin, 3) the material composition, hardness, and grain structure were as specified, 4) multiple secondary cracks were found, but only within 0.016 inches radially of the fatigue region, 5) just like the primary fracture, no microstructural anomalies were found at the secondary crack locations, and 6) the shot peening appearance on the forward web face had more pronounced peening dimples than the web aft face. GE also performed their own striation density calculation and they estimated the number of flight cycles from detection to separation to be between approximately 5,000 – 5,700 cycles, consistent with the NTSB finding.

In order to better understand how the crack could have initiated in the web of the stage 8 disk, GE performed a series of additional hardware testing (for example residual stress and strain distortion), computational analysis (reevaluated the LCF lifing based on the actual event hardware, including all the material review board accepted allowable deviations and under the worst material property conditions) and operation mission profile review (verify stresses during taxing time, takeoff thrust rating, shutdown, core speeds and temperatures, ambient takeoff temperature, etc.). All the predictive calculations that GE performed could not close on the event crack location at the number of cycles it was thought to have initiated the crack; however, the striation density curves that were developed for the event spool match well with the analytical predictive crack propagation rate.

Based on this event, GE developed and incorporated into the engine maintenance manual a set of unique non-destructive inspections for the HPC stage 8-10 spool focusing on the event crack location. These inspections can be performed at the piece part, rotor and module levels, as well as on-wing. Along with the addition of the engine manual non-destructive inspections, GE released three separate service bulletins (SB 72-1145, SB 72-1146, and SB 72-1151) to inspect all the GE90 HPC stage 8-10 spools with part number 1694M80G04 (failure event spool part number) and a selected number of spools part numbers 1844M90G01 & G02. The Federal Aviation Administration mandated those inspections by issuing airworthiness directives AD 2015-27-01 and AD 2016-13-05.

Based on the shot peening irregularity found on the event part (and review of other similar vintage spools found a similar reduced coverage on the aft side of stage 8 disk), GE initiated two 2 corrective actions. First, GE issued SB 72-1149 to provide a one-time improved shot peen repair for all HPC stage 8-10 spools (all base model GE90 part numbers). Since this is a one-time shot peen repair, this repair will <u>not</u> be incorporated into the engine manual. Secondly, GE modified the print drawing for the current in-production HPC stage 8-10 spool to include a check the shot peen intensity in the outer web areas.

TABLE OF CONTENTS

TABLE OF PHOTOS

TABLE OF FIGURES

TABLE OF TABLES

TABLE OF ACRONYMS

AAIB	Air Accident Investigation Branch	INSP	Inspection
AC	Advisory Circular	LAS	McCarran International Airport
ACC	Active Clearance Control	LGW	London-Gatwick International Airport
AD	Airworthiness Directive	LPC	Low Pressure Compressor
AGB	Accessory Gearbox	LPT	Low Pressure Turbine
ALF	Aft Looking Forward	MFP	Main Fuel Pump
ARFF	Airport Rescue and Fire Fighting	MRB	Material Review Board
ASCS	Air Supply Control System	N1	Fan/Low Rotor Speed in Percent RPM
ATA	Air Transport Association of America	N2	High Rotor Speed in Percent RPM
BA	British Airways	NA	Not Applicable
BSI	Borescope Inspection	NC	North Carolina
CDP	Compressor Discharge Pressure	NCW	Not Complied With
CFR	Code of Federal Regulations	NDI	Non-Destructive Inspection
CSLSV	Cycles Since Last Shop Visit	NTSB	National Transportation Safety Board
CSN	Cycles Since New	PDT	Pacific Daylight Time
CW	Complied With	PRSOV	Pressure Regulating and Shutoff Valve
DA	Direct Aging	PSIA	Pressure – Pounds Per Square Inch Absolute
DR	Departure Record	PSIG	Pressure – Pounds Per Square Inch Gage
ECI	Eddy Current Inspection	QAR	Quick Access Recorder
ECS	Environmental Control Systems	RDR	Runway Distance Remaining
EDS	Energy Dispersive Spectroscopy	RPM	Revolution Per Minute
EEC	Electronic Engine Control	SB	Service Bulletin
EM	Engine Manual	SEM	Scanning Electron Microscope
ESN	Engine Serial Number	SL	Service Letter
FAA	Federal Aviation Administration	SP	Special Procedure
FADEC	Full Authority Digital Engine Control	SPLCF	Sustained-Peak Low Cycle Fatigue
FDR	Flight Data Recorder	SPM	Standard Practices Manual
FHF	Fan Hub Frame	TCDS	Type Certificate Data Sheet
FPI	Fluorescent Penetrant Inspection	TCF	Turbine Center Frame
GE	General Electric	TEM	Transmission Electron Microscopy
GPS	Global Positioning Satellite	TR	Thrust Reverser
HMU	Hydromechanical Unit	TRF	Turbine Rear Frame
HPC	High Pressure Compressor	TSLSV	Time Since Last Shop Visit
HPSOV	High Pressure Shutoff Valve	TSN	Time Since New
HPT	High Pressure Turbine	UTI	Ultrasonic Inspection
HRC	Rockwell Hardness "C" Scale	VBV	Variable Bleed Valve
ID	Inner Diameter	VSCF	Variable Speed Constant Frequency
IDG	Integrated Drive Generator	VSV	Variable Stator Vane

D. DETAILS OF THE INVESTIGATION

1.0 ENGINE AND AIRPLANE INFORMATION

1.1 ENGINE DESCRIPTION

The accident airplane was powered by two GE GE90-85BG11 turbofan engines (**FIGURE 1**). The GE90-85BG11 is a dual-spool rotor, axial flow, high bypass turbofan that features a single stage fan and a 3-stage low pressure compressor (LPC) driven by a 6-stage low pressure turbine (LPT), an annular combustor, and a 10-stage high pressure compressor (HPC) driven by a 2-stage high pressure turbine (HPT). The engine is designed so that it is easy to remove and install modules during maintenance procedures without complete disassembly and assembly of the engine. The primary modules of the engine are the fan hub, core, LPT, accessory gearbox (AGB), and fan case. According to the engine's FAA Type Certificate Data Sheet (TCDS) E00049EN, Revision 18, dated February 14, 2014, the engine has a maximum takeoff thrust rating of 88,870 pounds and a maximum continuous thrust rating of 81,230 pounds, both flat-rated[1] to 86°F (30°C).

FIGURE 1: GENERIC GE90 TURBOFAN ENGINE

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

[1] 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.

1.2 Engine History

The No. 1 engine installed on the accident airplane was a General Electric (GE) GE90-85B-G11 turbofan engine, engine serial number (ESN) 900-294 (**Photo 1**). At the time of the fire event (accident), ESN 900-294 had accumulated 66,801 hours time since new (TSN), 9,992 cycles since new (CSN), 3,645 hours time since last shop visit (TSLSV), and 503 cycles since last shop visit (CSLSV). The last engine shop visit occurred in June/July 2014 at GE Aviation Wales, United Kingdom, (subsequently referred to as GE Wales). According to British Airways (BA), GE Wales performs all engine maintenance requiring shops visits.

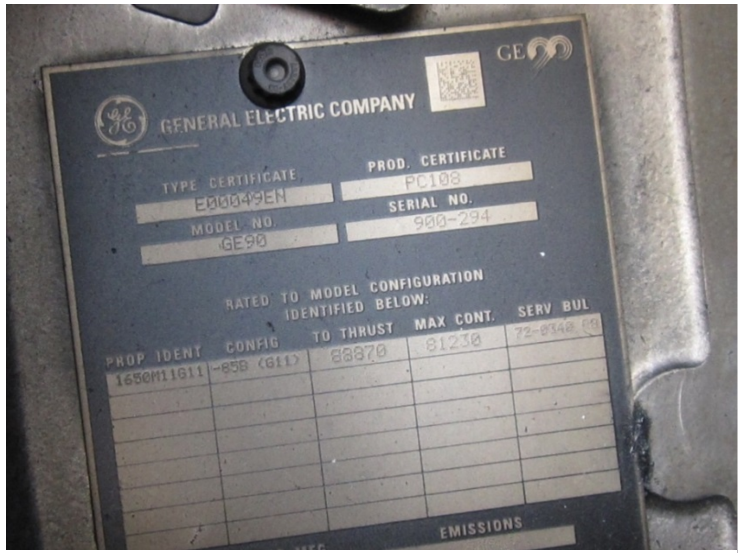

Photo 1: No. 1 Engine Data Plate

1.3 High Pressure Compressor Stage 8-10 Spool History

According to BA and GE maintenance records, the HPC stage 8-10 spool (subsequently referred to as the 'spool' for the remainder of this section) installed on the accident engine is PN 1694M80G04 and SN GWNHA236. The PN and SN are dot peened on the outer surface of the rear drive arm near the rear flange. Disassembly of the engine confirmed the SN of the HPC stage 8-10 spool; however, the PN could not be found due to damage to the rear drive arm in the vicinity of where the PN was anticipated to be (**Photo 2**). The spool was delivered on ESN 900-121 in April 1997 and first installed on airplane G-ZZZD in June 1997 where it remained until November 1999.[2]

Photo 2: Part Marking on the HPC Stage 8-10 Spool Located on the Rear Drive Arm

In November 1999, ESN 900-121 was removed due to HPT distress beyond manual limits and shipped to GE Wales. During the shop visit, the HPC module was not disassembled to piece part level but the HPC forward cases (top and bottom) and the aft extension case were removed and the

[2] The event HPC stage 8-10 spool (and event engine ESN 900-121) had accumulated 30 hours and 37 cycles when delivered to BA. ESN 900-121 had first been delivered to Boeing new in 1995 and during its first pre-delivery flight test of the customer airplane the engine stalled. Post-flight inspection revealed evidence of heavy fan rubs. Fan hardware including the blades and forward case were changed. Flight testing of the engine continued at Boeing on two different airplanes until the engine was removed in January 1996 and shipped to the GE Peebles facility for a series of engineering ground tests. While at Boeing the engine and the HPC stage 8-10 spool had accumulated 16 hours and 8 flight cycles. While at GE Peebles, the engine and the HPC stage 8-10 spool accumulated an additional 14 hours and 29 cycles.

spool seal teeth were eddy current inspected (ECI); no findings were noted.[3] At the time of the November 1999 engine removal, the spool had accumulated 10,097 hours TSN and 2,279 CSN. The spool was reinstalled on ESN 900-121 and the engine was installed on airplane G-RAES in February 2000 where it remained until September 2001.

In September 2001, ESN 900-121 was removed for customer convenience and was installed on airplane G-ZZZB that same month where it remained until January 2002. ESN 900-121 was removed for HPC toggle seal damage and sent to GE Wales. At the shop visit, the HPC forward stators were removed and the HPC stage 8-10 was not exposed as the HPC aft extension case was not removed. At the time of the January 2002 engine removal, the spool had accumulated 17,210 hours TSN and 3,489 CSN. ESN 900-121 was installed on airplane G-VIID in February 2002 where it remained until January 2004.

In January 2004, ESN 900-121 was removed for HPT distress beyond manual limits and sent to GE Wales. At that shop visit, the HPC spool was visually inspected and an eddy current inspection of the spool seal teeth was performed; no findings.[4] At the time of the January 2004 engine removal, the spool had accumulated 25,021 hours TSN and 4,711 CSN. The spool was reinstalled in ESN 900-121 and the engine was installed on G-VIIC in May 2004 where it remained until September 2008.

In September 2008, ESN 900-121 was removed for combustor distress beyond manual limits and shipped to GE Wales. The HPC module was removed and shipped to Snecma Service in Saint-Quentin-en-Yvelines France (subsequently referred to as Snecma) for module disassembly and the spool was shipped back to GE Wales for repair. At GE Wales, a full spool FPI and an ECI of the inertia weld was performed along with a strip, seal tooth blend repair, FPI and recoat of seal teeth per the engine manual (EM) and several unique repairs/approvals that were all approved on departure record (DR) 17-08-0706.[5] None of the items on the DR related to the stage 8 disk web. The FPI and ECI satisfied the intent of AD 2002-04-11; therefore the AD was complied with. At the time of the September 2008 engine removal, the spool had accumulated 47,499 hours TSN and 7,516 CSN. The repaired spool was returned to Snecma where it was installed in a serviceable HPC module and shipped back to GE Wales for installation in ESN 900-294 (accident engine). ESN 900-294 was installed on airplane G-VIIK in July 2009 where it remained until June 2014 when it was removed for HPT distress beyond manual limits.

In June 2014, ESN 900-294 was removed and shipped to GE Wales and during the shop visit it was discovered that the HPC forward case and forward stators were contaminated with hydraulic fluid (Skydrol®); the forward case and forward stators were removed and cleaned but the HPC stage 8-10 spool was not exposed. Since the HPC stage 8-10 spool was not exposed, AD 2002-04-11 compliance was not required because the HPC module was not at the piece part level. At the time of the June 2014 engine removal, the spool had accumulated 70,525 hours TSN and 10,956 CSN. The spool was reinstalled in ESN 900-294 and the engine was installed on airplane G-VIIO in January 2015 (accident airplane) where it remained until the accident. At the time of the accident, the HPC stage 8-10 spool, SN GWNHA236 had accumulated 74,170 hours TSN; 11,459 CSN; 26,671 hours since piece part exposure and 3,943 cycles since piece part exposure (**Figure 2**).

[3] ECI was not a requirement at that time but was performed to gather engineering data.
[4] ECI was not a requirement at that time but was performed to gather engineering data.
[5] At shop inspections, during the FPI, mirrors were used to view the webs.

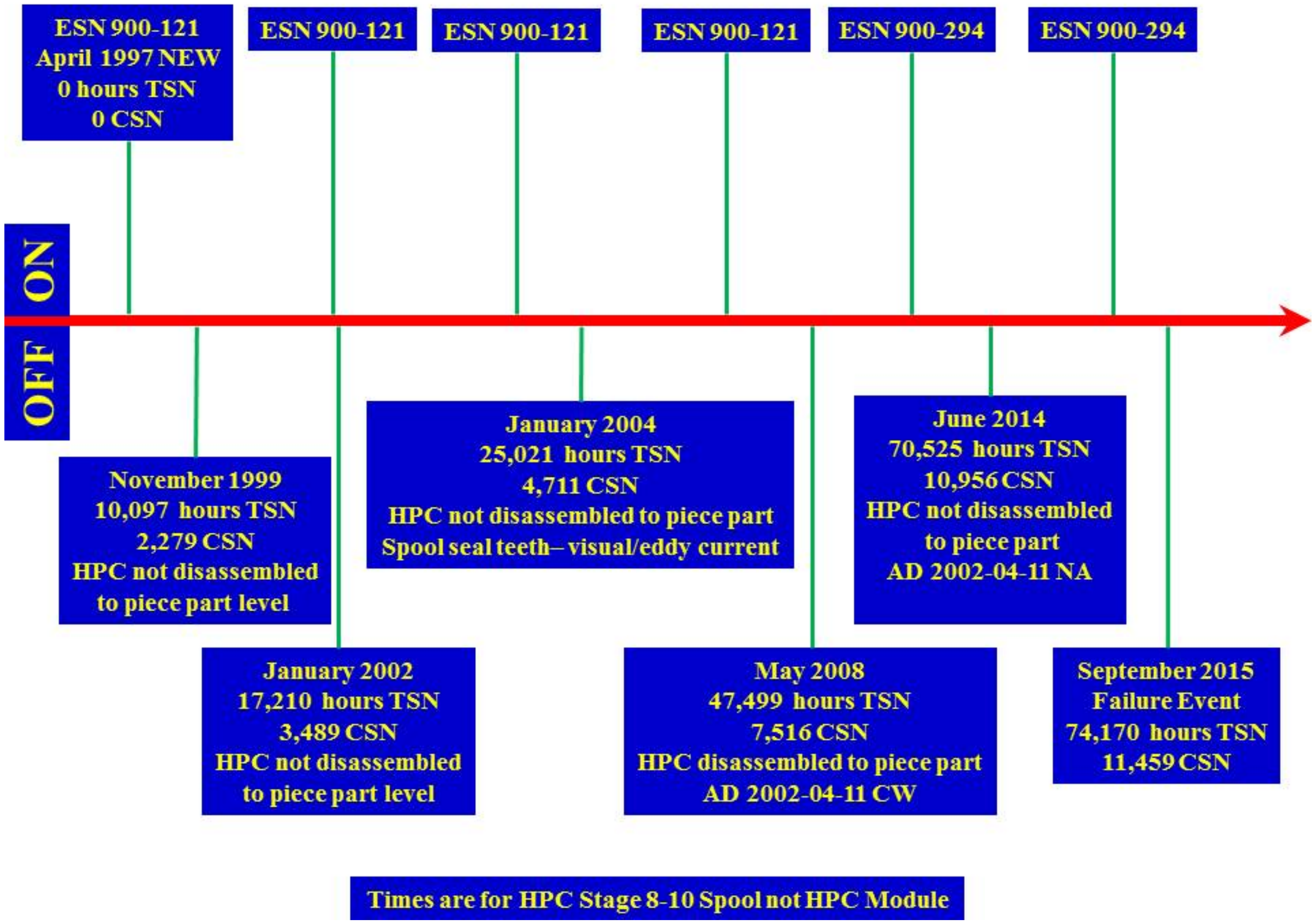

Figure 2: Timeline of HPC Stage 8-10 Spool, PN 1694M80G04 and SN GWNHA236

1.4 High Pressure Compressor Stage 8-10 Spool Life Limits and Special Inspections

According to the Chapter 05-11-02 *Compressor Rotor – Life Limits* in the GE90 engine manual (EM), the life limit (also referred to as the low cycle fatigue (LCF) life) for the accident HPC stage 8-10 spool, PN1694M80G04, is 16,600 cycles (See Section 6 *HPC Stage 8-10 Spool Life Limit Calculations* for additional details on LCF life).

AD 2002-04-11, which was referenced in the shop visit maintenance records, had an effective date of April 8, 2002 and required a mandatory inspection of selected parts at the piece part level and was intended to prevent critical life-limited rotating engine part failure, which could result in an uncontained engine failure and damage to the airplane. The AD, as it related to the HPC stage 8-10 spool, requires a fluorescent penetrant inspection (FPI) of the entire spool and an ECI of the inertia weld.

According to Chapter 05-21-00 *Engine Piece Part – Mandatory Inspection* in the EM, all HPC stage 8-10 spools regardless of the PN require an inspection in accordance with Task 72-31-08-200-001-001 that calls out an FPI per subtask 72-31-08-230-006 and an ECI per subtask 72-31-08-250-

021.[6] The EM inspection requirements reflect the AD requirements. See Section 7.0 *HPC Stage 8-10 Spool Engine Manual Inspections – Current At The Time of The Event* for detailed inspection instructions.

1.5 DESCRIPTION OF ENGINE NACELLE

The engine inlet cowl (also known as the inlet duct), fan cowl, thrust reverser (TR) and, turbine exhaust, consisting of the primary exhaust nozzle, forward and aft centerbodies, comprise the boundaries of the engine nacelle and consist of fixed and hinged components (**FIGURE 3**). The fixed components include the inlet cowl, primary nozzle, and the forward and aft centerbodies; while the hinged components include the fan cowl and TR. The inlet cowl is bolted to the engine fan case, the primary nozzle to the outer portion of the turbine rear frame (TRF), and the aft centerbody is bolted to the forward centerbody which is attached to the inner portion of the TRF. The left TR outer barrel incorporates a ventilation air outlet panel at the 6:00 o'clock position used for vent flow of the nacelle and is also used in the event of an over-pressurization event. The fan cowl and the TR are each in two halves, hinged at either side of the strut, and joined by latch hooks on the bottom centerline - fan cowl incorporates four latches while the TR incorporates 13 latches. All of the nacelle hardware, including the TR, is provided by Boeing and the engine is supplied by GE.

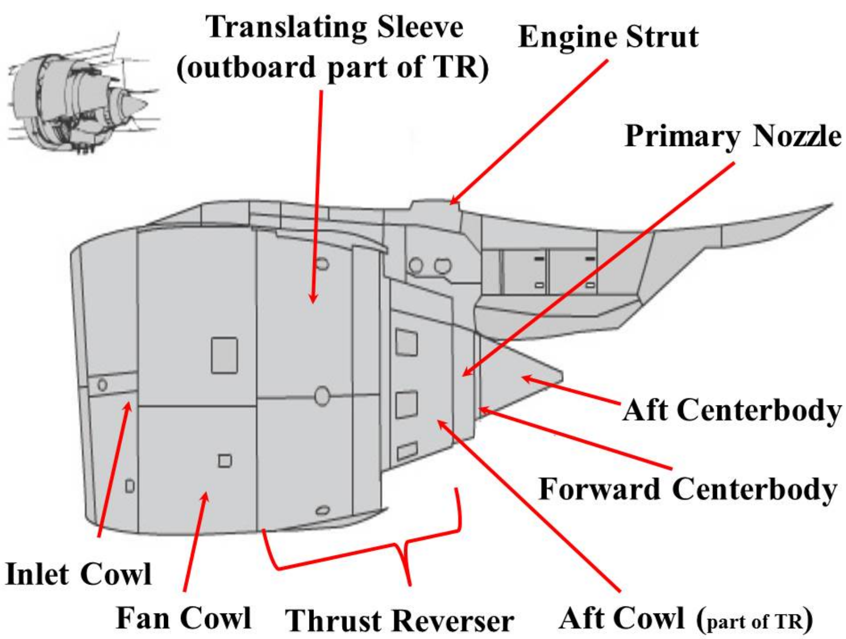

FIGURE 3: BOEING 777 NACELLE NOMENCLATURE AND LOCATION

[6] The Air Transport Association of America (ATA) 3-element (6-digit) numbering system is used to identify the engine or systems chapter (for example 72-31-08, element one "**72**" for the basic engine), engine section or major assemblies (element two "**31**" major components of the compressor section) and engine subjects, subassemblies or detail piece parts (element three "**08**" the HPC stage 8-10 spool). The EM covers two levels of maintenance - Engine Level Maintenance (condition) and Off-Engine Maintenance (overhaul). Maintenance coverage is supplied by the use of a numbering procedure that changes the second and third element of the section/subject number; for example, 72-31-00 and 72-00-31 as follows: Task 72-31-00 specifies the compressor rotor assembly and piece-parts used in a complete overhaul, (i.e. complete disassembly to piece part level). Task 72-00-31 specifies the compressor rotor assembly to be inspected and repaired without complete disassembly. Inspection started at this level may show damage or wear that makes it necessary to perform complete disassembly and overhaul procedures; that is, follow 72-31-00.

2.0 ON-SITE EXAMINATION OF NO. 1 ENGINE STILL INSTALLED ON AIRPLANE

The Powerplant Group, comprised of members from GE, Boeing, BA, Federal Aviation Administration (FAA), Air Accident Investigation Branch (AAIB, United Kingdom), and the National Transportation Safety Board (NTSB) convened at LAS on September 9, 2015 to perform a detailed examination of the accident engine and airplane and completed its work on September 15, 2015.

According to witnesses statements by airport personnel, video tape evidence, and flight data recorder (FDR) data, the accident flight started its takeoff roll on runway 07L at the intersection of taxiway A8 and runway 07L. The airplane then experienced a No. 1 engine failure and subsequent fire about 1,030 feet after the takeoff roll start, and came to a full stop about 1,927 feet from the start of takeoff roll (**FIGURE 4**)[7]. After the event and prior to the arrival of the investigative team, airport personnel walked runway 07L and the grassy area about 50 feet on either side of the runway to recover any engine/airplane debris. Several boxes/bags of debris were collected. According to airport personal cleaning runway 07L, the first metallic part was found on the south right runway edge line at about the 1,000 foot threshold marker and no other metallic debris was found until about the 3,000 foot mark. At the 3,000 foot mark, the vast majority of the metallic debris was collected (**PHOTO 3**). Between the 3,000 foot mark and where the airplane was stopped (about 4,500 foot mark), only soft material was recovered consistent with honeycomb and cowling type material. The debris was examined by the investigative team and pieces of cowling, HPC stator vanes segments, HPC blade dovetails and platforms, and several pieces of the HPC stage 8-10 spool along (**PHOTOS 4** and **5**, and **FIGURE 5**) with various pieces of case and manifold (plumbing) were recovered.

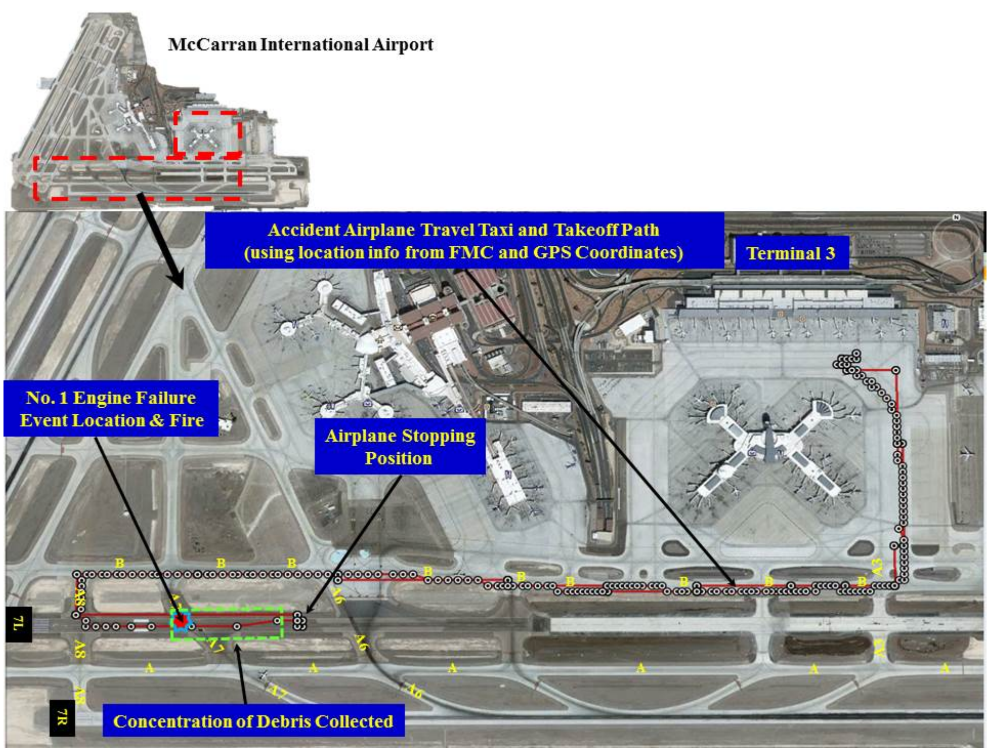

FIGURE 4: AIRPORT DIAGRAM WITH AIRPLANE PATH AND DEBRIS LOCATIONS

[7] The distances are average estimates based on global positioning satellite location data and engine/airplane parameters recorded on the flight data recorder. Since the times for the various parameters are offset; meaning they are recorded at different times, the average of the data before and after a specific event was used to bound the event time. See *Section 9.0 Flight Data Recorder and Event Time Line* for more details.

Photo 3: Metallic Debris Collected from Runway 07L

Photo 4: HPC Stage 8-10 Spool Fragments Collected from Runway 07L

Photo 5: HPC Stage 8-10 Spool Stage 8 Blade Circumferential Dovetail Slot and Forward Seal Teeth

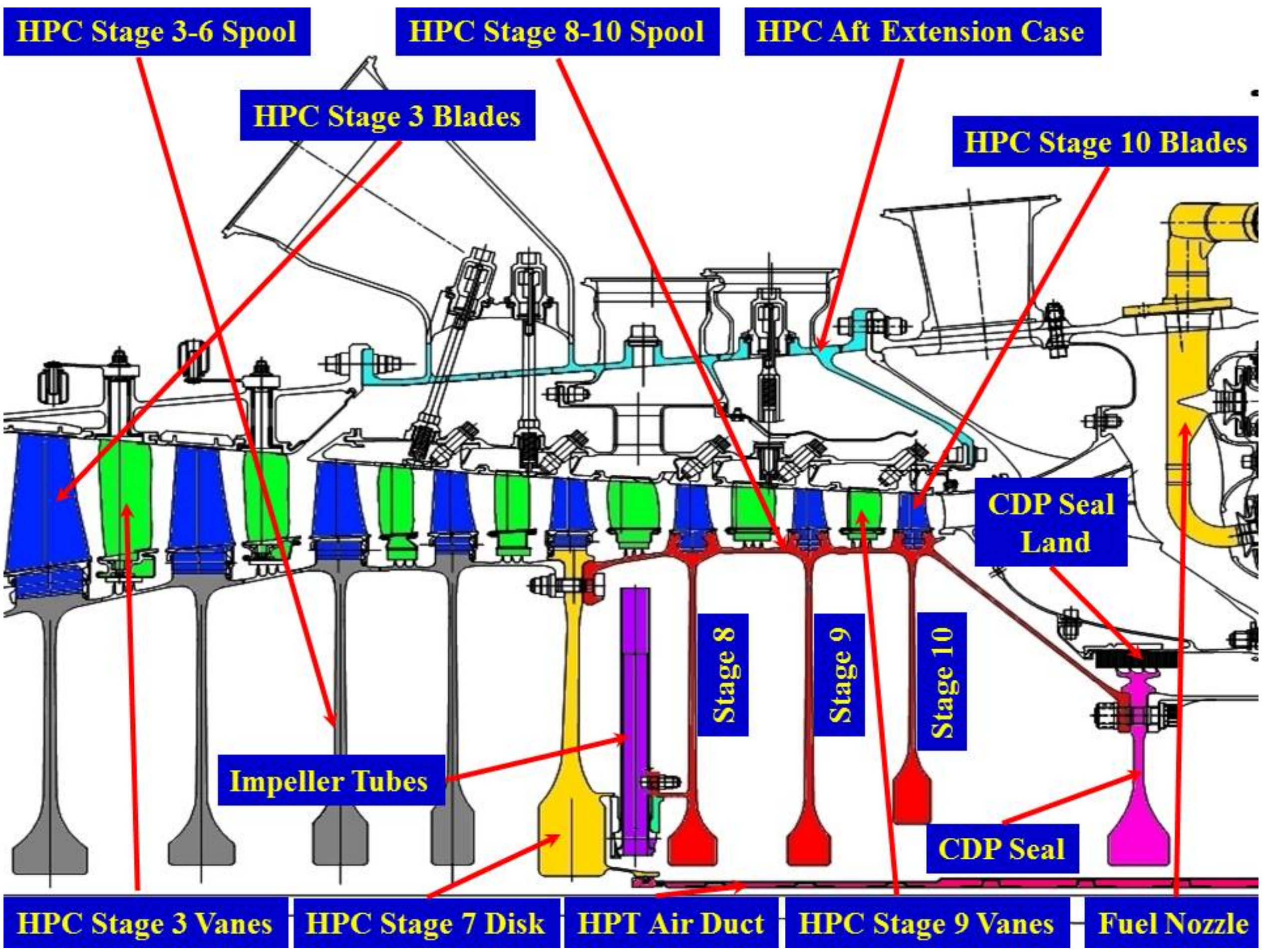

Figure 5: Cross-Section of High Pressure Compressor Focused on HPC Stage 8-10 Spool[8]

[8] This engine cross-section is for information purposes. The HPC stage 8-10 spool depicted is not the configuration of the spool installed in the accident engine. **Figure 5** depicts a HPC stage 8-10 spool with an inertia weld between stage 8 and 9 and between 9 and 10. The accident spool, PN 1694M80G04, only has an inertia weld between stage 8 and 9. See **Figure 6** for correct configuration.

2.1 Detailed Examination of the No. 1 Thrust Reverser and Engine Cowlings

In general, the right side of the engine exhibited far more thermal/fire distress than the left side (**Photos 6** and **7**).

Photo 6: Right Side of Engine

Photo 7: Left Side of Engine

Photo 8: Inlet Cowl w/ Outer Barrel Breach

The inlet cowl was still attached to the fan case; however, it exhibited thermal distress from the 12:00 to 8:00 o'clock position and was missing part of the outer skin of the lip from 4:00 to 5:30 (64 inches circumferentially x 5 inches axially) o'clock position but the inner barrel remained intact (**Photo 8**).

The fan cowls were latched and locked, of the 4 latches, only the most forward was not fully CLOSED (locked) but the hook (hooks are located on the right barrel and the latch keepers are located on the left barrel) appeared to be still engaged with the latch keeper. The fan cowls exhibited thermal distress from about the 12:00 to 8:00 o'clock position with the right cowling exhibiting more severe thermal damage; however, no breaches, holes, or burn-thrus were observed. The right cowl was charred, had significant loss of the epoxy binder, loose composite sheets, and when removed from the engine had lost its structural integrity and did not keep its barrel shape. The left fan cowl exhibited sooting, thermal distress and blistered paint and when removed it held its structural integrity and kept its barrel shape. The pressure relief door located on the left fan cowl was found in the OPEN position while the oil feed access door was found in the CLOSED and locked position.

The TR exhibited the most thermal distress of any of the nacelle components. The right TR translating sleeve was heavily thermally damaged/burned (charred and loose composite sheets with almost all of the epoxy binder missing) and two confirmed penetration holes: 1) located at 1:30 o'clock positon and 49 inches forward of the rear flange measuring 3 x 4 inches, 2) located at 2:00 o'clock positon and 44 inches forward of the rear flange measuring 5.5 x 3 inches. The distance between the centers of the penetration holes was about 14 inches. Broomstick handles were placed in the penetration holes to provide a general fragment exit direction (**Photos 9-12**). The trailing edge of the translating sleeve from about the 3:00 - 5:30 o'clock position (70 inches circumferentially) was missing 16 inches of axial material. In **Photo 12**, impact mark "A" was identified as a non-through hole where a piece of HPC stage 7 bleed air duct port and case (**Photo 13**) was found imbedded in the inner wall.

The TR inner wall from the 1:00 – 4:00 o'clock position was missing (about a 5 foot x 5 foot section) exposing the engine core – normally the engine core would not be able to be seen with the TRs closed (**Photo 14**).

Photo 9: Right TR Thru-Holes – Side View – Exit Angle from Hole 1

Photo 10: Right TR Thru-Holes – Forward Looking Aft - Exit Angle from Hole 2

Photo 11: Right TR Thru-Hole – Aft Looking Forward -Exit Angle from Hole 2

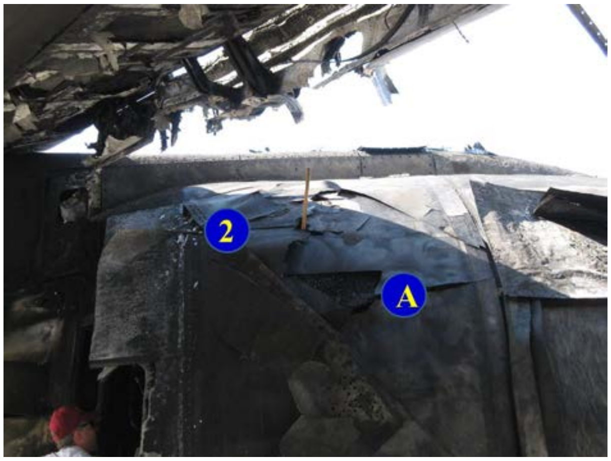

Photo 12: Right TR Thru-Hole – Side View - Exit Angle from Hole 2

PHOTO 13: HPC STAGE 7 BLEED AIR DUCT AND CASE PIECE FOUND WEDGED IN RIGHT TR TRANSLATING SLEEVE

PHOTO 14: MISSING TRANSLATING SLEEVE & INNER WALL MATERIAL ON RIGHT TR

The left TR was intact with no observed structural issues and exhibited only minor thermal distress from the 6:00 – 8:00 o'clock position – primarily sooting and blistered paint.

The aft cowl, which is part of the TR, was intact and exhibited soot around the pressure relief doors. The aft cowl has 6 pressure relief panels and all were found OPEN. All the visible TR latches - 5 located on the aft cowl and 1 on the translating sleeve - were fully latched and locked with all the latches flush with the barrel skin (PHOTO 15).[9] The primary nozzle and the forward and aft centerbodies remained attached to their respective hardware and appeared undamaged.

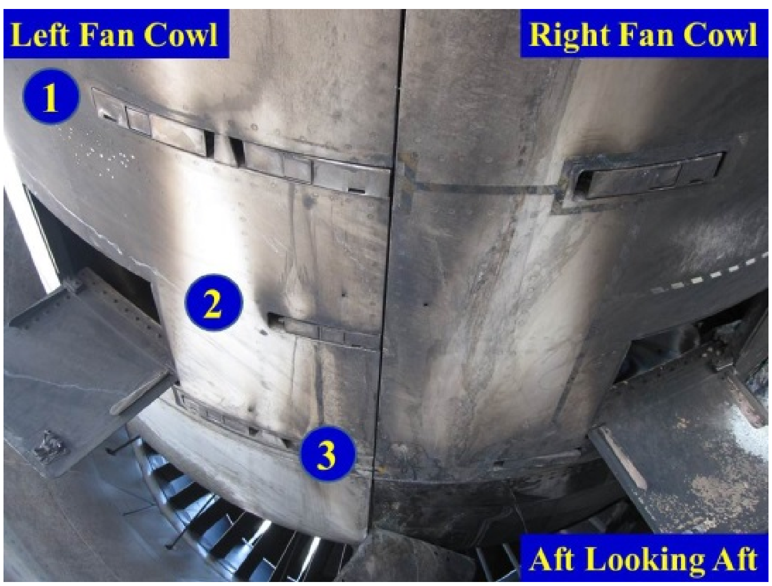

PHOTO 15: AFT COWL LATCHES

[9] There are 13 TR latches in total – some are not visible from the outside of the TR without opening panels.

2.2 Fire Suppression and Fuel Cutoff Documentation

Examination of the cockpit revealed that both thrust levers (throttles) were against the aft stops, both fuel control[10] cutoffs knobs were in the down and CUTOFF position (**Photo 16**) and the No. 1 engine fire handle was pulled and twisted to the right in the direction of the No. 2 bottle discharge position (**Photos 17** and **18**). The No. 2 engine fire handle was found not pulled and straight.

Toggling of the fuel control knob from RUN to CUTOFF closes the airplane spar valve and closes the engine-mounted fuel control high pressure shutoff valve (HPSOV) and pulling of the fire suppression handle performs several functions, such as closing the fuel control HPSOV if not already closed and closing the spar valve if not already closed, same as the fuel control cutoff switches on the pedestal.[11]

Photo 16: Thrust Lever and Fuel Control Run/Cutoff Position for Both Engines

Photo 17: Engine Fire Suppression Handle Positions for Both Engines

Photo 18: No. 1 Engine Fire Suppression Handle Pull and Twisted to the Right

There are two fire suppression bottles for the engines – bottle 1 and bottle 2 – and the bottles are shared by both engines. Each fire suppression bottle has a dedicated discharge port for each engine. The engine fire suppression bottles are located in the forward right cargo bay (**Photo 19**). A continuity check was conducted to determine if the fire suppression bottles were discharged by checking the pressure switch for each of the two fire suppression bottles. The electric connector was disconnected from the pressure switch and continuity was confirmed from pins 1 and 3 on each of the fire suppression bottles; this is consistent with low or no pressure in the bottles. When the fire suppression bottles are under normal pressure, the pressure switch is in the OPEN position; when low or no pressure, the switch is in the CLOSED position. After the continuity check was completed, both engine fire suppression bottles were removed to examine the detonation devices to see if they had been discharged. Visual examination of the discharge ports of each bottle revealed that the detonation device in one of the discharge ports in each bottle was missing the percussion cap which is consistent with the bottle being discharged (**Photos 20** and **21**). The part number for both engine fire suppression bottles are Boeing PN 34600042-1, manufacturer (Pacific Scientific) PN 5303W001-1 with the forward most bottle SN 14986F1 and the rearmost SN 20644F1. According to the manufacturer's specification, the empty

[10] Although the nomenclature on the pedestal calls out 'fuel control', the GE term for this accessory is a hydromechanical unit (HMU) and it will be referenced as such during the engine exam documentation.

[11] Along with closing the fuel control shutoff valve and the spar valve, pulling the fire suppression handle also performs the following actions: closes the environmental control system (ECS) pressure regulating and shutoff valve (PRSOV), closes the ECS high pressure shut off valve, closes the hydraulic pump supply/shut-off valve, de-pressurizes the hydraulic pump, de-energize the integrated drive generator (IDG) and variable speed constant frequency (VSCF) generator, arms the fire extinguishing system, and silences the aural warning.

weight of the fire suppression bottle is 16.4 pounds. The No. 1 fire suppression bottle weighed about 15 pounds and the No. 2 weighed about 16 pounds with the scale reading in whole pound increments. The left wing (No. 1 engine) spar valve actuator was accessed through the bottom of the left wing and visually the actuator was in the CLOSED position (**PHOTO 22**).

PHOTO 19: ENGINE FIRE SUPPRESSION BOTTLES INSTALLED IN THE CARGO BAY

PHOTO 20: DETONATION DEVICE WITH THE PERCUSSION CAP MISSING

PHOTO 21: DETONATION DEVICE WITH THE PERCUSSION CAP PRESENT

PHOTO 22: LEFT WING (NO. 1 ENGINE) REAR SPAR ACTUATOR IN CLOSED POSITION

2.3 ENGINE STRUT AND ENGINE MOUNT EXAMINATION

The engine is mounted to the strut at two anchor points; the front engine mount is located on the fan case and the rear mount is located on the TRF. Examination of the engine mounts while the engine was still installed on the airplane and again when the engine was removed did not reveal any apparent damage or distortion.

Thermal distress/fire damage was only noted on the right side of the engine strut - the left side was in good condition and no soot patterns were noted (See **PHOTOS 6** and **7**). The right side of the engine strut was burned, charred and on some of the panels the epoxy binder was missing and the composite plies were loose (**PHOTOS 23** and **24**). No burn through holes were noted. All the access panels were CLOSED and latched except for the pressure/relief and fire detector access panel which was partially OPEN, melted, and the latch missing (**PHOTO 24**).

PHOTO 23: ENGINE STRUT THERMAL/FIRE DAMAGE – LOOSE COMPOSITE PLIES

PHOTO 24: MELTED & OPEN PRESSURE/RELIEF AND FIRE DETECTOR ACCESS PANEL

2.4 NO. 1 ENGINE (LEFT) – ESN 900-294

Visual examination of the engine through the inlet cowl revealed that all the fan blades were present and exhibited minor thermal distress but were intact, and with the inlet cowl removed, the fan could be reached and it rotated by hand. Looking at the LPT stage 6 turbine blades through the primary nozzle revealed that all the blades were present and undamaged, there was some fluid that had pooled at the bottom TRF, and the LPT rotor rotated along with the fan with no binding or any grinding sounds.

With the fan cowls removed, visually the fan case appeared in good condition and was free of soot and the full authority digital engine control (FADEC) appeared in good condition. The left TR was removed first, followed by the right TR. Debris was collected from within the TR (**PHOTO 25**), from the ground beneath the engine as the TRs were removed, as loose pieces became dislodged after removing hardware from the engine to facilitate shipping, and as the engine was removed from the airplane. All loose metallic debris was examined and pieces consistent with the HPC stage 8-10 spool, such as pieces of the forward rabbeted flange, seal teeth sections, and circumferential blade slot sections were shipped to the NTSB metallurgical laboratory in Washington DC for failure and material analysis.

PHOTO 25: UNDERCOWL ENGINE DEBRIS

A partial initial borescope inspection (BSI) was performed on the HPC and HPT sections. The HP rotor could not be rotated through the gearbox, so the BSI was limited to the length of the scope through the respective borescope ports or HPC aft extension case breach. BSI of the engine while it was still installed on the airplane through the borescope ports on the left side of the engine starting on HPC stage 2 revealed minor HPC stage 2 blade airfoil damage and significant airfoil and vane damage starting at HPC stage 4 progressing downstream. Severe distress of the entire HPC stage 8-9 was observed and the spool (stage 8-10) was separated aft of the middle seal teeth between stages 8 and 9 and the stage 8 disk radially separated below the circumferential blade slot; this was consistent with hardware recovered from the runway (**Figure 6**). The stage 9 disk appeared intact with blade roots and platform present and still installed. The stage 10 disk was not visible.

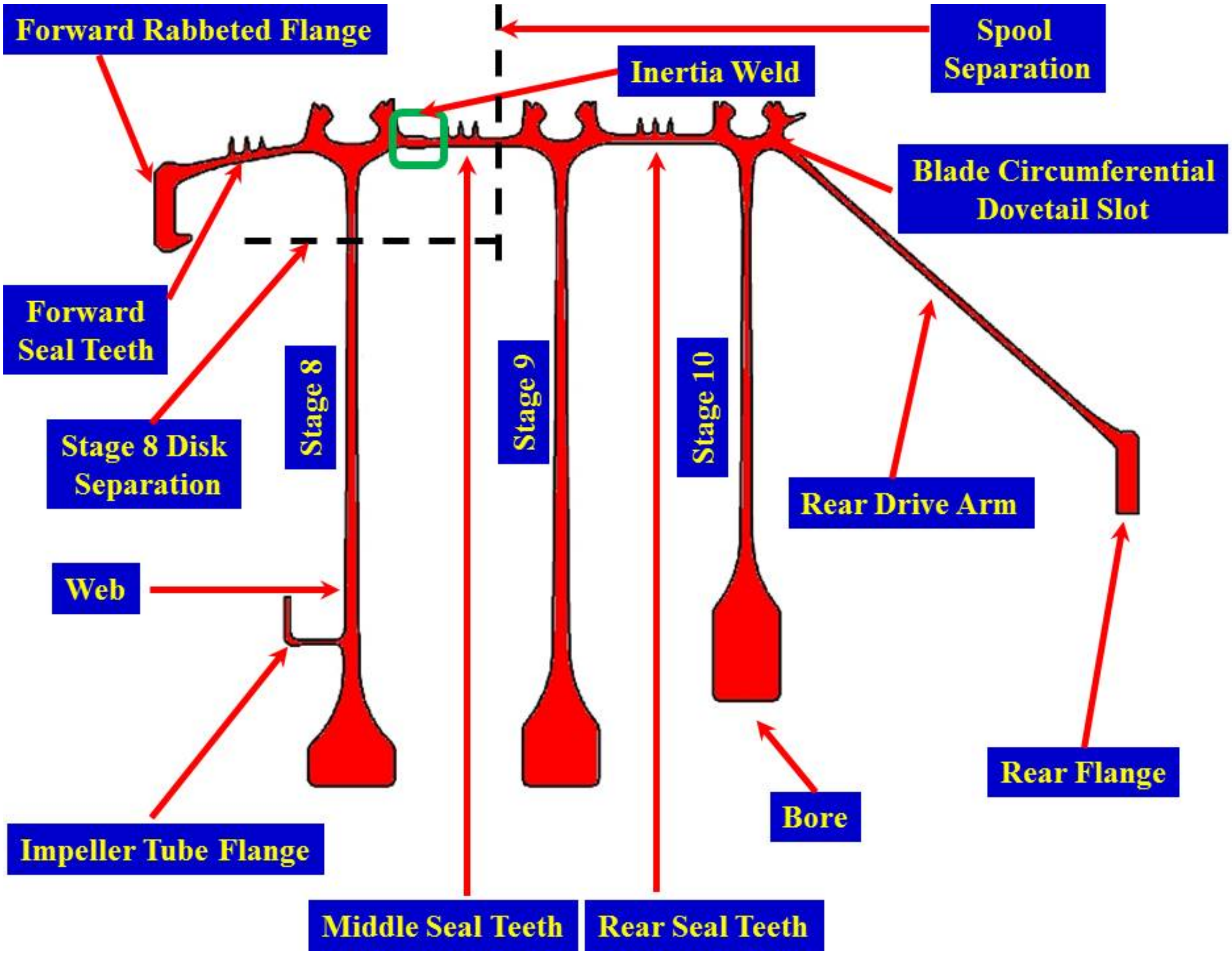

Figure 6: Borescope Finding of HPC Stage 8-10 Spool Separation
(Locations are relative and not exact)

The right engine thrust link, located at about the 1:00 o'clock position was severed in-plane with the HPC aft extension case (**Photo 26**). Several bleed ducts from the 2:00 – 3:00 o'clock position were severed in-plane with the HPC aft extension case (**Photo 27**). The HPC stage 10 bleed duct was disconnected from the HPC aft extension case – the tube and case were intact, all the attachment bolts were fractured (**Photo 28**). The main fuel supply line was intact; however, the inlet elbow of the main fuel pump (MFP) housing was fractured; the fuel supply line was completely separated from the pump and pieces of the elbow remained attached to the main fuel supply line (**Photo 29**).

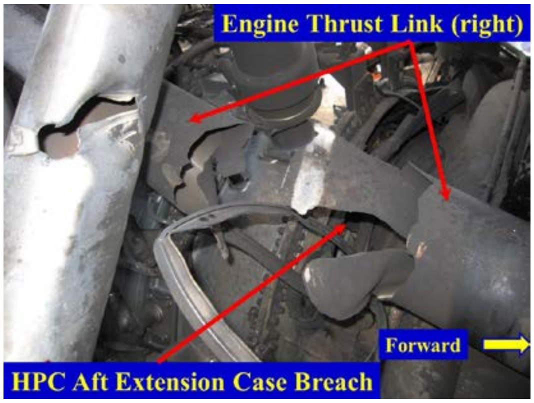

PHOTO 26: SEVERED RIGHT THRUST LINK

PHOTO 27: EXAMPLES OF SEVERED DUCTING

PHOTO 28: HPC STAGE 10 BLEED DUCT SEPARATION

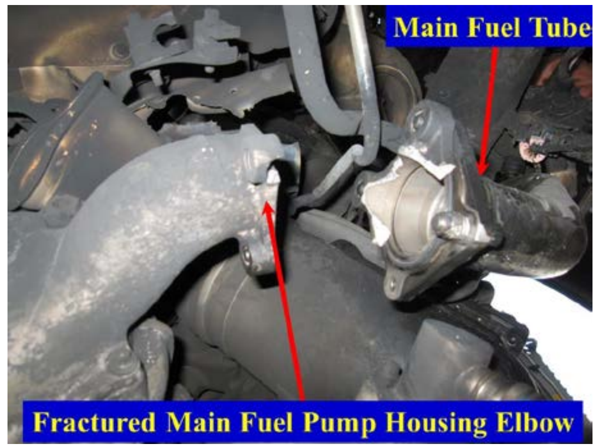

PHOTO 29: MAIN FUEL SUPPLY LINE SEPARATED FROM THE MAIN FUEL PUMP

The AGB housing exhibited a crack between the starter mount pad and the hydromechanical unit (HMU) mount pad that extended from the back of the gearbox (PHOTO 30), over the top, and stopped on the front. No visible oil was dripping from the crack. The lube and scavenge pump attachment flange was fractured around the entire circumferential and parts of the flange remained attached to the gearbox (PHOTO 31).

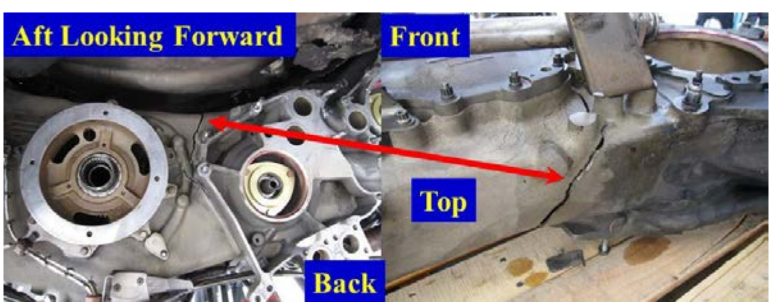

PHOTO 30: FRACTURED REAR HOUSING OF GEARBOX

PHOTO 31: FRACTURED LUBE AND SCAVENGE PUMP ATTACHMENT FLANGE

The engine heatshield exhibited numerous exit penetration holes centered about the 5:00 o'clock position in-line with the breach in the HPC aft extension case (**Photo 32**). The engine heatshield aft skirt was pierced by a section of what appears to be part of the HPC aft extension case flange (**Photos 33** and **34**).

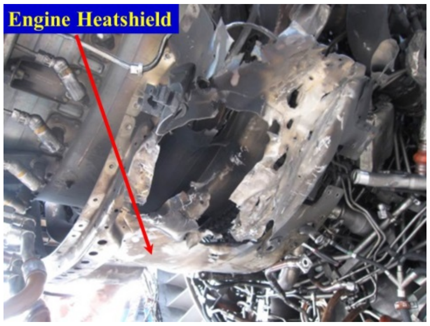

Photo 32: Penetration Holes in the Engine Heatshield

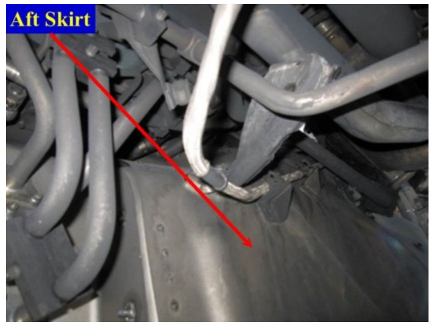

Photo 33: Engine Heatshield Aft Skirt – Front Side

Photo 34: Engine Heatshield Aft Skirt – Aft Side

The HMU electrical leads for the HPSOV were sooted, thermally damaged, and severed near the locations where the main fuel line was severed (**Photo 35**). A piece of what appears to be HPC aft extension case flange with an attachment bolt still installed pierced through the forward side of the HMU housing (**Photo 36**) with the HMU removed, the input shaft of the MFP was bound and could not be rotated by hand.

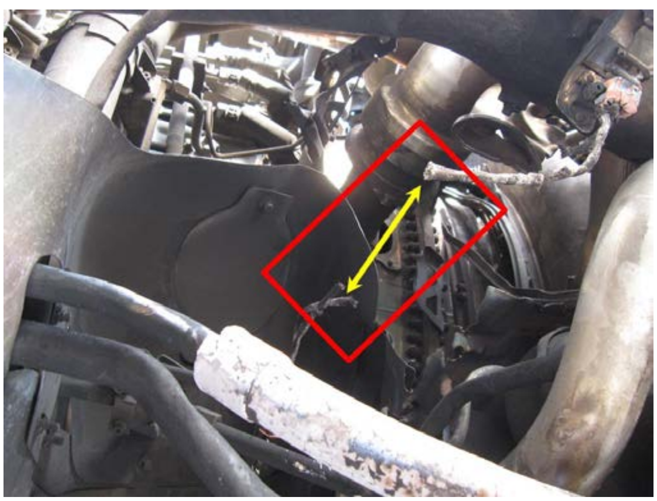

Photo 35: Severed HMU Electrical Leads

Photo 36: HPC Aft Extension Case Lodged in the HMU

The HPC aft extension case was breached circumferentially from about the 11:30 – 5:00 o'clock position (58 inches circumferentially) (**Photos 37** and **39**) and axially from the stage 4 bleed manifold to the rear flange. The case skin was missing but in those areas where it remained, the case skin was torn and pushed outwards. A breach of the HPC aft extension case was also located at about the 9:30 o'clock position, coincident with a HPC stage 7 bleed port (**Photo 40**), and the case skin was torn, pushed outwards, and pieces of compressor debris were wedged in the tear. The tear measured about 10 inches circumferentially x 2.5 inches axially.

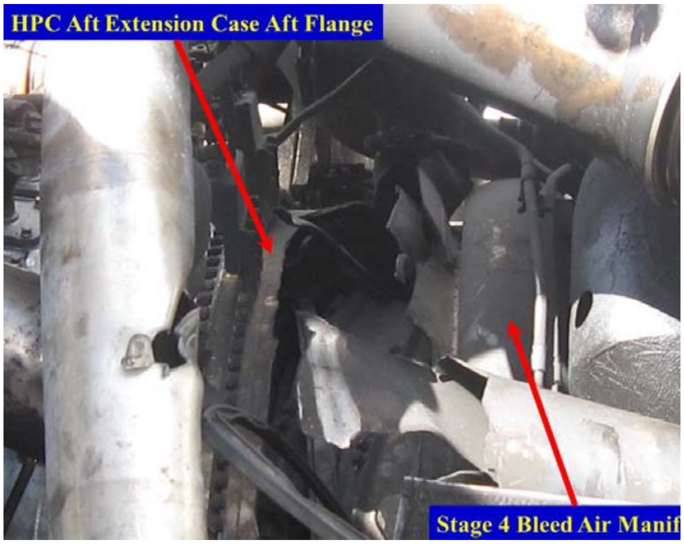

PHOTO 37: HPC AFT EXTENSION CASE BREACH AT TOP OF ENGINE

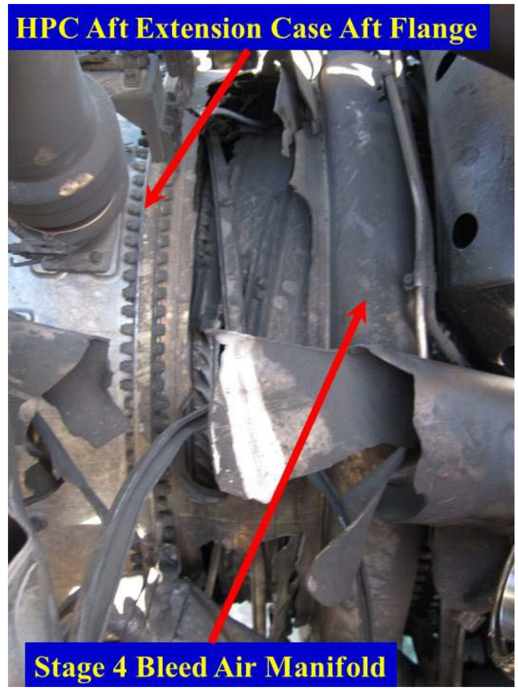

PHOTO 38: HPC AFT EXTENSION CASE BREACH AT 3:00 O'CLOCK POSITION

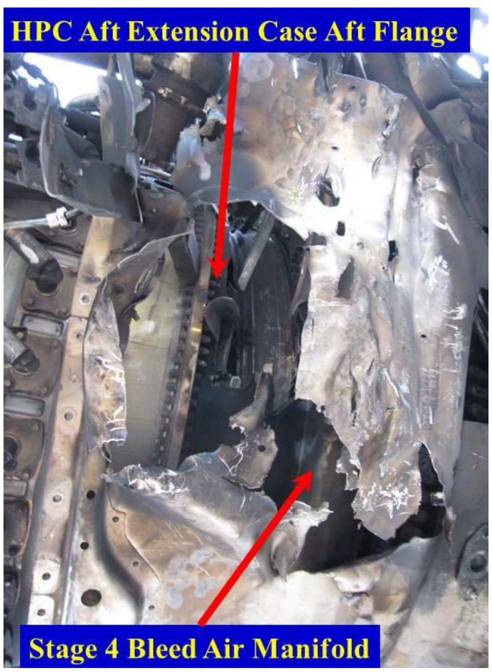

PHOTO 39: HPC AFT EXTENSION CASE BREACH AT BOTTOM OF ENGINE

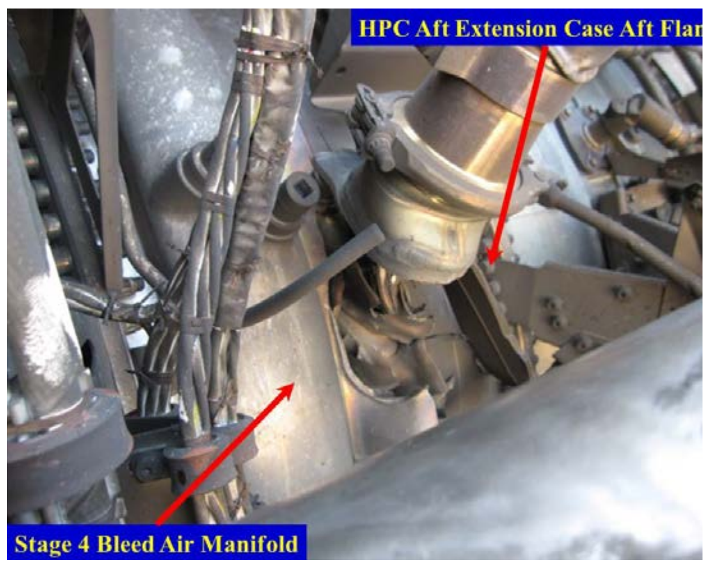

PHOTO 40: HPC AFT EXTENSION CASE BREACH AT 9:30 O'CLOCK POSITION

The aft left, aft right, aft lower fire loops appeared to be intact; however, the aft right loop was distorted and its support tube fractured (**PHOTO 41**). All the fire loop isolators were present, most were sooted, and some exhibited minor thermal distress (**PHOTO 42**).

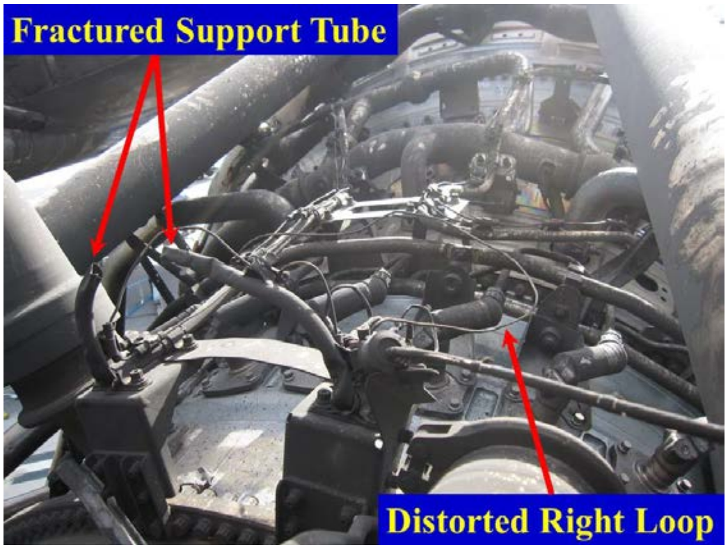

PHOTO 41: FIRE LOOPS ON TOP OF ENGINE – RIGHT AFT DISTORTED

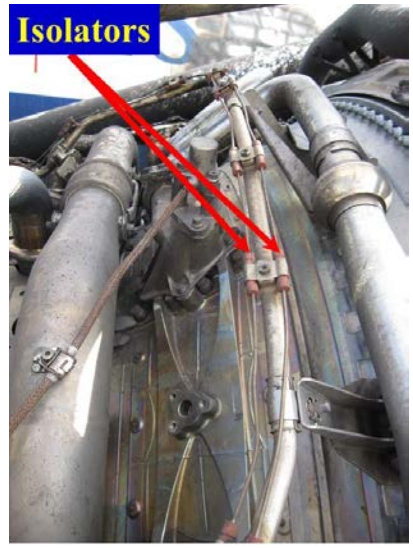

PHOTO 42: FIRE LOOP ISOLATORS IN GOOD CONDITION

2.5 HPC STAGE 8-10 SPOOL FRAGMENT SEARCH

A preliminary trajectory analysis was performed by GE and Boeing using FDR data, video of the event, and witness statements from airport personnel to determine a search area for engine debris that departed the engine (**FIGURE 7**). On September 15, 2015, a group comprised of persons from GE, Boeing, BA, and the NTSB walked from the intersection of taxiway A8 to taxiway A6 along the south edge of runway 07L and from taxiway A7 to the airport perimeter south fence. Two pieces consistent with the HPC stage 8-10 spool with seal teeth were recovered on the south east corner of the intersection of taxiway A7 and runway 07L (**PHOTOS 43** and **44**). A HPC stage 8 blade platform was also recovered in the center of taxiway A7, just north of runway 07R.

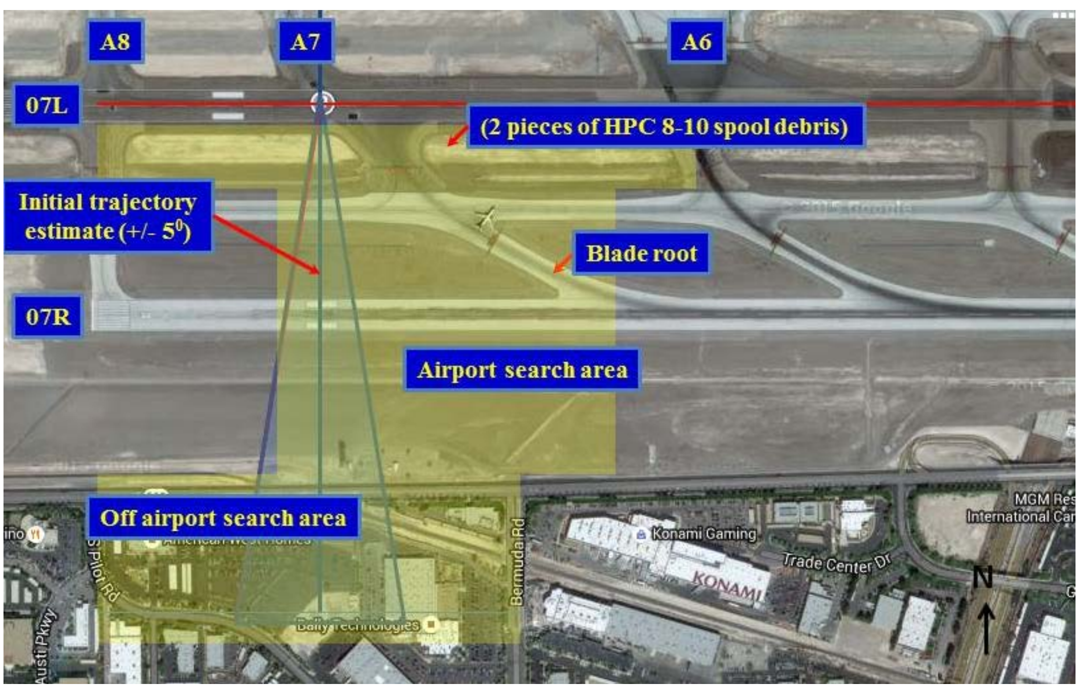

FIGURE 7: PRELIMINARY TRAJECTORY PATH, SEARCH AREA, AND PARTS RECOVERED LOCATION

PHOTO 43: RECOVERED HPC STAGE 8-10 SPOOL FRAGMENT FROM INTERSECTION OF TAXIWAY A7 AND RUNWAY 07L

PHOTO 44: RECOVERED HPC STAGE 8-10 SPOOL FRAGMENT FROM INTERSECTION OF TAXIWAY A7 AND RUNWAY 07L

A search was also performed off the airport property; no parts were recovered (See **FIGURES 7** and **8** for search and recovery areas).

FIGURE 8: AREA OUTSIDE OF AIRPORT PERIMETER FENCE WHERE PARTS SEARCH WAS CONDUCTED

3.0 ENGINE DISASSEMBLY

The Powerplant Group, comprised of members from GE, Boeing, BA, FAA, AAIB, and the NTSB convened at the GE facility in Evendale Ohio on September 21, 2015 to perform a detailed disassembly of the accident engine and completed its work on September 24, 2015. Prior to the arrival of the Powerplant Group, the NTSB authorized GE to remove items listed in **TABLE 1** to facilitate and expedite the engine disassembly.

TABLE 1: LIST OF APPROVED PARTS REMOVED TO FACILITATE ENGINE EXAM
LPT Active Clearance Control (ACC) system – Cooling manifolds around LPT forward to fan hub frame (FHF), all ducts, manifolds, clamps, and brackets
HPT ACC system – Ducting from FHF to HPT cooling manifolds, clamps and brackets
Stage 4 and 7 air piping – from each connection (HPCS to HPT case), and all associated clamps and brackets
Variable Stator Vane (VSV) and Variable Bleed Valve (VBV) muscle lines
All oil pressure and scavenge lines
All fuel lines that connect to the fuel manifolds
All wiring harnesses
VSV actuators and VBV actuators
Engine Mounts
LPT case-to-Turbine Center Frame (TCF) flange joint bolts - kept 20 bolts at the 12:00 and 6:00 o'clock positions and 10 bolts at the 3:00 and 9:00 o'clock positions – many of the bolts were already sheared

Removal of the LPT rotor revealed that the LPT shaft had circumferential score mark spanning 38 inches axially and starting 2.25 inches forward of the change in LPT shaft diameter (**PHOTO 45**). The LPT stage 1 nozzle vanes were all present, intact and in good condition as were the LPT stage 1 blades. The LPT module was not disassembled any further. The knife edges of the "B" sump aft inner air seals were damaged and worn down almost to the backing strip.

PHOTO 45: LPT SHAFT CIRCUMFERENTIAL SCORE MARKS

The fan hub module, comprised of the fan[12] and booster (LPC stages 1-3) sections, was removed exposing the No. 3 bearing area and the inlet gearbox. The bevel gear of the radial drive shaft and the horizontal drive bevel gear were both severely damaged (**PHOTO 46**). The radial drive shaft bevel gear teeth were smeared almost entirely flat (no appreciable tooth height left), melted, and exhibited a bluish heat tint. The horizontal drive bevel gear teeth were smeared; over the outboard 80% of the teeth length, about 50% of the tooth height was missing. The inboard 20% of the tooth length exhibited essentially the entire tooth height. The horizontal drive bevel gear did not exhibit the melting that the radial drive bevel gear did.

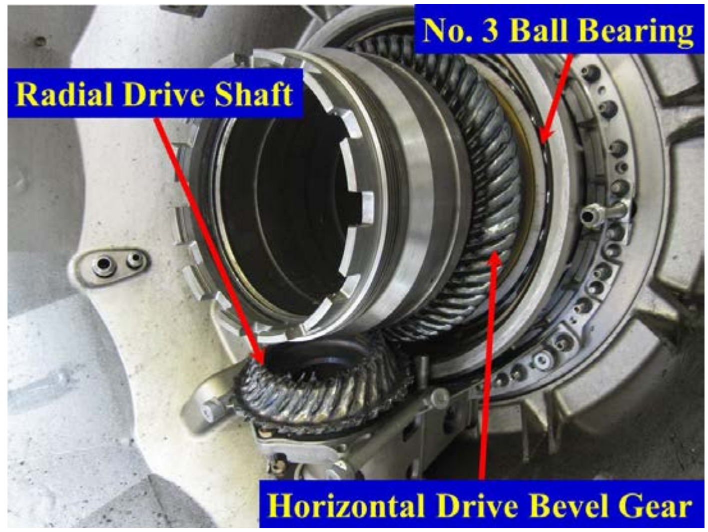

PHOTO 46: INLET GEARBOX - DAMAGE TO BEVEL GEARS

The No. 1 ball bearing (**PHOTO 47**), the No. 2 roller bearing (**PHOTO 48**), and the No. 3 ball (See **PHOTO 46**) and radial bearings all appeared intact, in good condition, and the roller elements were shiny.

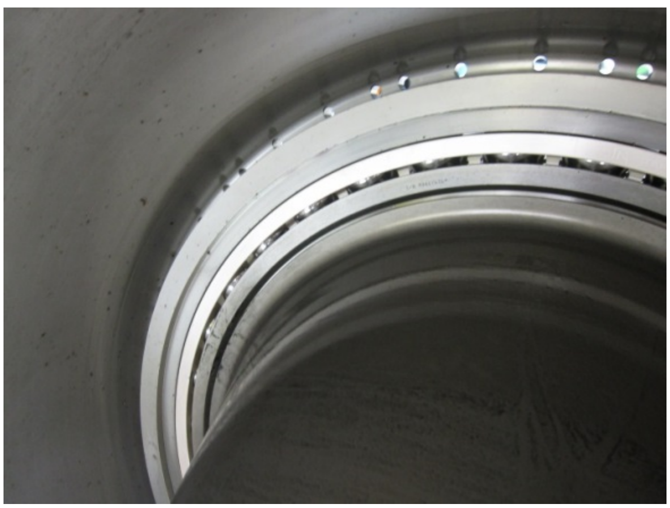

PHOTO 47: NO. 1 BALL BEARING

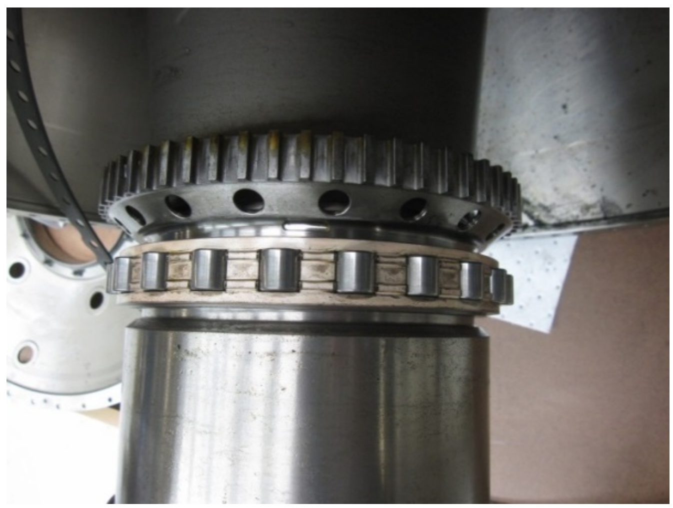

PHOTO 48: NO. 2 ROLLER BEARING

With the fan module removed, the engine was rotated vertically on its nose to facilitate the cutting of the HPC aft extension case. A hand held grinder was used to cut through the HPC aft extension case that remained intact, essentially cutting from the 5:00 to 11:30 o'clock position (**PHOTO 49**).

[12] The fan case and the fan blades were removed at LAS to facilitate transportation of the engine.

PHOTO 49: CUTTING OF THE HPC AFT EXTENSION CASE TO REMOVE HPC STAGE 8-10 SPOOL PIECES

After cutting through the HPC aft extension case, the back end of the engine was lifted off and the HPC stage 9 and 10 disks remained with the back portion (**PHOTO 50**). The HPC stage 8 disk remained with the forward end along with the HPT air duct. The HPC stage 8 disk was fully detached from the stage 7 disk, and the HPT air duct was loose within the bore (**PHOTO 51**). The stage 8 disk was fractured circumferentially about 200° around just at the transition radius of the web to the blade circumferential dovetail slot rim. About 37 inches of the dovetail slot rim remained attached to the disk (**PHOTO 52**).[13]

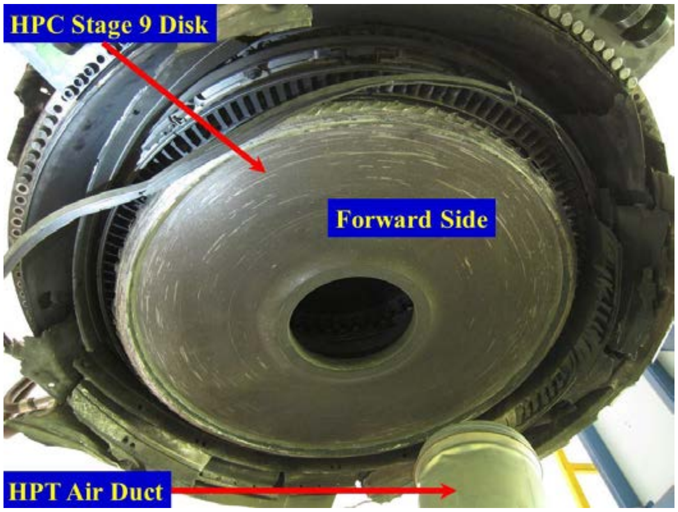

PHOTO 50: HPC STAGE 9 DISK STILL ATTACHED TO BACK HALF OF THE COMPRESSOR

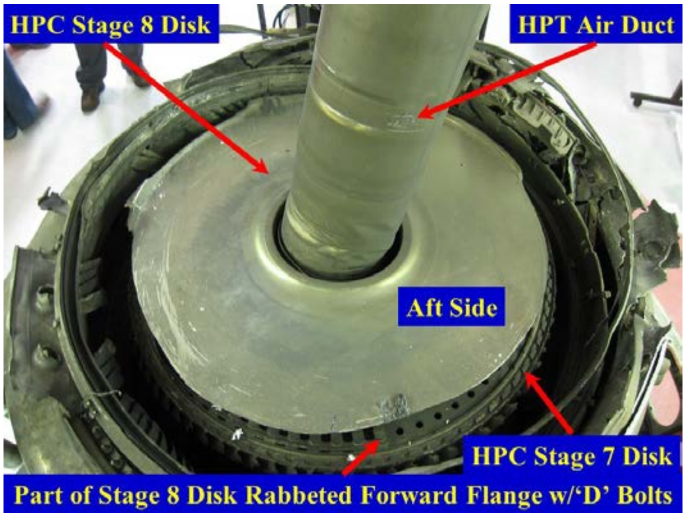

PHOTO 51: HPC STAGE 8 DISK LOOSE & SITTING ON THE IMPELLER TUBE ASSEMBLY & THE HPT AIR DUCT LOOSE WITHIN THE DISK BORES

[13] The HPC stage 8 disk has about a 27 inch diameter which translates into a rim circumference of about 85 inches or approximately 7 feet.

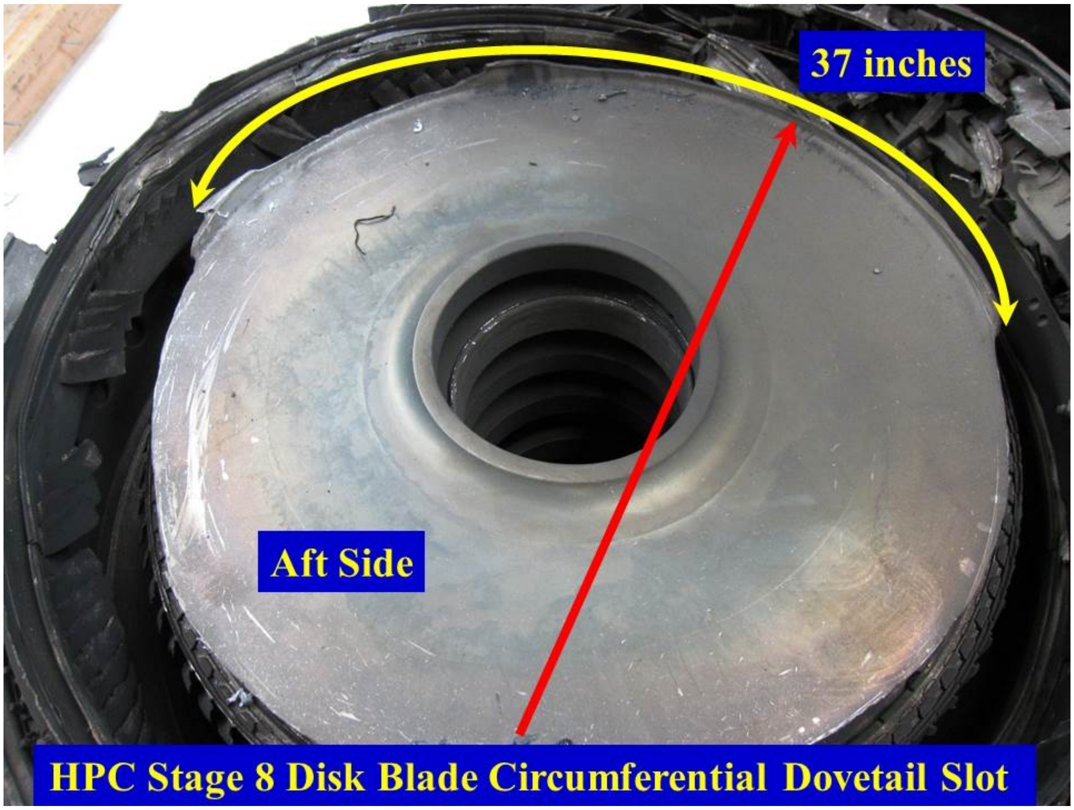

Photo 52: HPC Stage 8 Disk with 160° of Dovetail Slot Rim Still Attached

With the HPC stage 8 disk removed, the impeller tube assembly was exposed. The impeller tube assembly was fractured and detached from the front side of the stage 8 disk, and the carrier was ovalized; no longer round. The attachment flange on the impeller tube assembly carrier was partially present with all the bolt hole scallops pulled through and fractured. Some of the impeller tubes were fractured at the carrier, some were fairly intact, and some were splayed open (**Photo 53**). The mating attachment flange on the stage 8 disk forward side was completely fractured and missing (**Photo 54**).

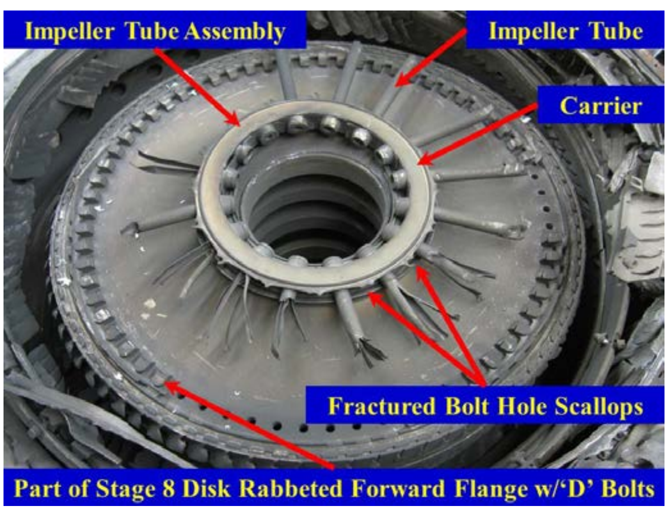

Photo 53: Damaged Impeller Tube Assembly

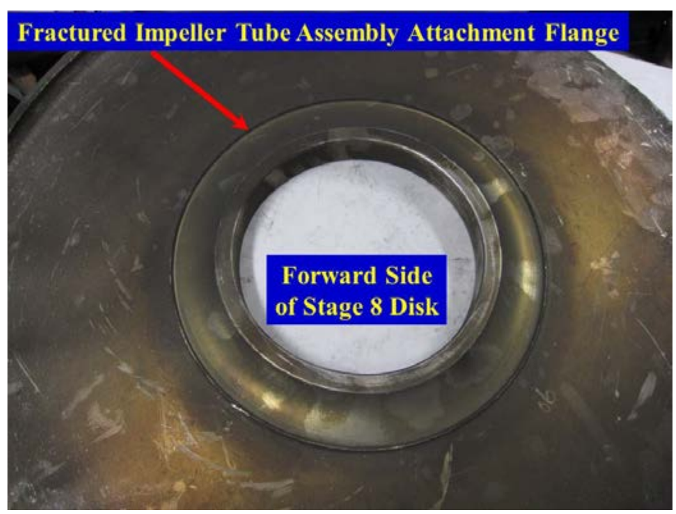

Photo 54: Missing Impeller Tube Assembly Attachment Flange

With the impeller tube assembly removed, the HPC stage 7 disk aft face was exposed. The impeller tube assembly support arm, which is integral to the back side of the stage 7 disk, was present and about a 240° continuous section was pushed outward and was torn near the transition radius of the support arm to the disk web. The HPT air duct forward seal land, integral to the back of the stage 7 disk at the bore, was completely fractured and missing. Two sections of the forward rabbeted flange remained attached to the aft face of the stage 7 disk; two sections of the flange, including the bolts, totaling 13 attachment bolt locations were also missing (**Photo 55**). These missing forward flange pieces were recovered at LAS and during the engine disassembly.

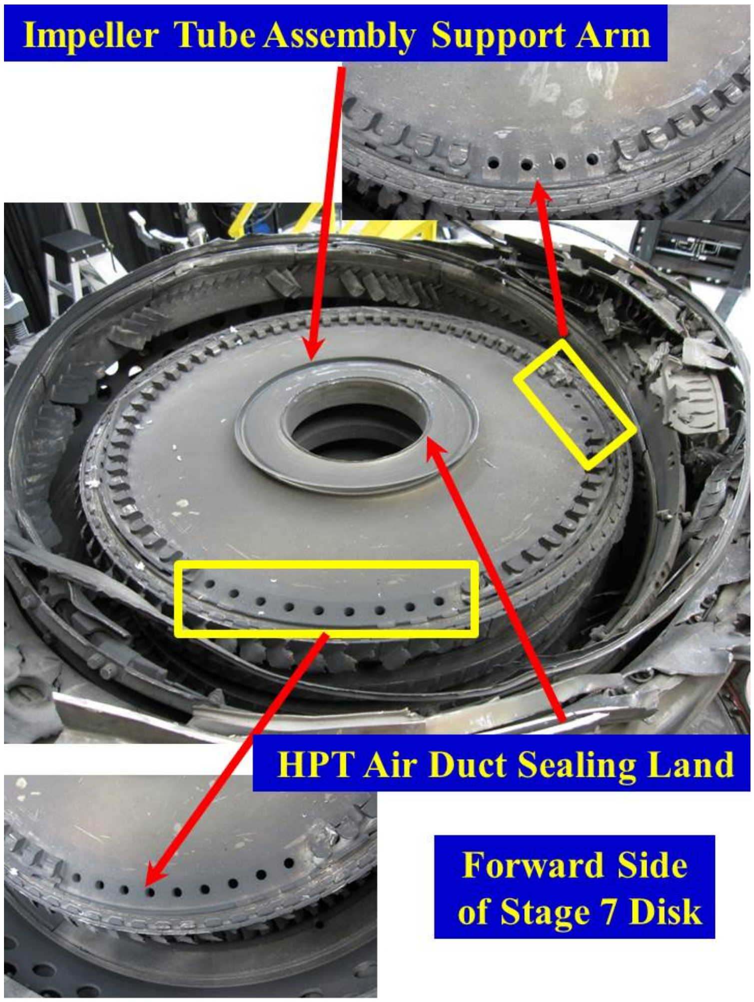

Photo 55: HPC Stage 7 Disk with Impeller Tube Assembly Support Arm Damage and Pieces of the HPC Stage 8-10 spool Forward Flange Still Attached

The rear drive arm of the HPC stage 8-10 spool was fractured and separated 360° about 1.375 inches forward of the rear flange (**Photo 56** and **Figure 9**) in the general vicinity of the compressor discharge pressure (CDP) seal land. The portion of the rear drive arm that remained with stage 8 and 9 disks exhibited a 0.8 inch wide 360° circumferential scoring mark and bluish heat tinting. The aft portion of the drive arm remained attached to the CDP seal and all the bolts were present.

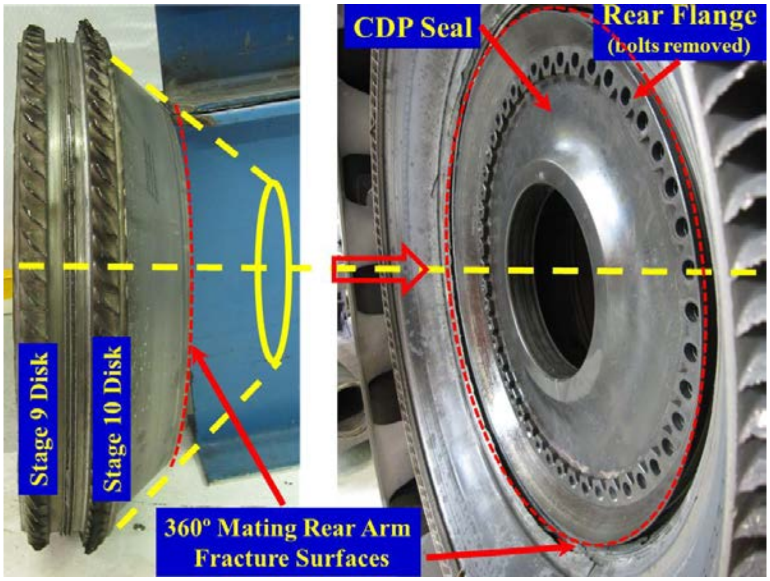

PHOTO 56: FRACTURED REAR DRIVE ARM

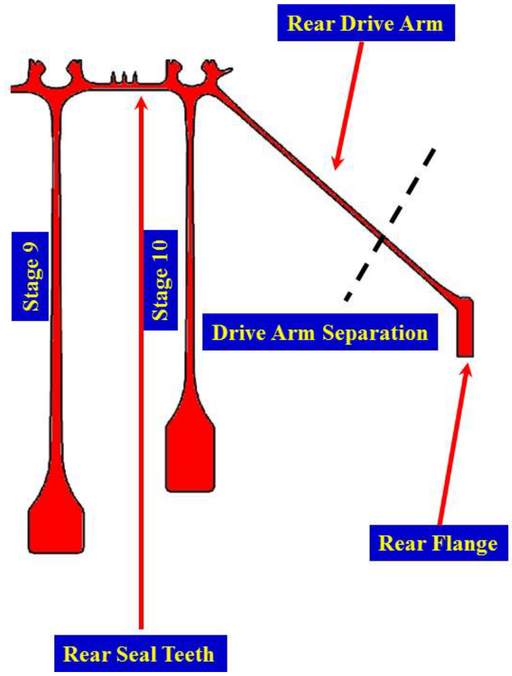

FIGURE 9: FRACTURED REAR DRIVE ARM LOCATION
(Location is relative and not exact)

The HPT air duct was twisted and distorted (consistent with torsional buckling) at both ends of the duct. The forward portion of the air duct exhibited three separate, about 180° circumferential, rub marks/contact marks and the locations of those marks were consistent with contact with the stage 8 (❶), 9 (❷) and the CDP seal (❸) bores (PHOTO 57). The forward seal land that mates with the seal on the stage 7 disk was damaged and exhibited circumferential rubs.

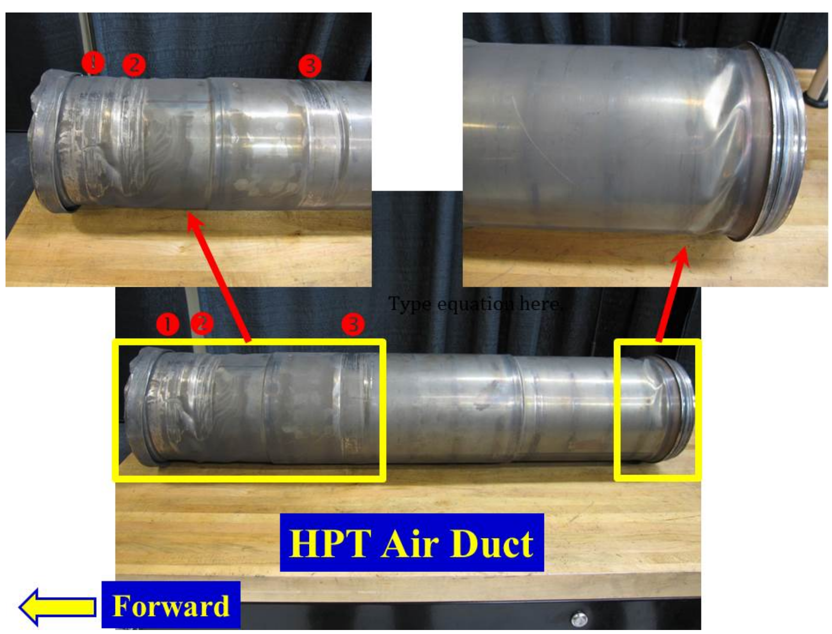

PHOTO 57: HPT AIR DUCT DAMAGE – TWISTING AND PARTIAL CIRCUMFERENTIAL SCORING

4.0 METALLURGICAL EXAMINATION OF HPC STAGE 8-10 SPOOL FRAGMENTS

4.1 METALLURGICAL EXAMINATION CONDUCTED BY THE NTSB

All the HPC stage 8-10 spool pieces that were recovered on-site were shipped to the NTSB Material Laboratory in Washington DC for further examination. A group comprised of members from GE, FAA, and the NTSB convened at the NTSB Material Laboratory on September 14, 2015 to commence examination of the hardware and completed their work on September 15, 2015. For complete details of the metallurgical findings, see Materials Laboratory Factual Report No. 15-131, dated January 15, 2016. After aligning the recovered pieces of the stage 8 disk, it was determined that just less than 50% percent of the stage 8 outer rim had been accounted for (**PHOTO 58**).

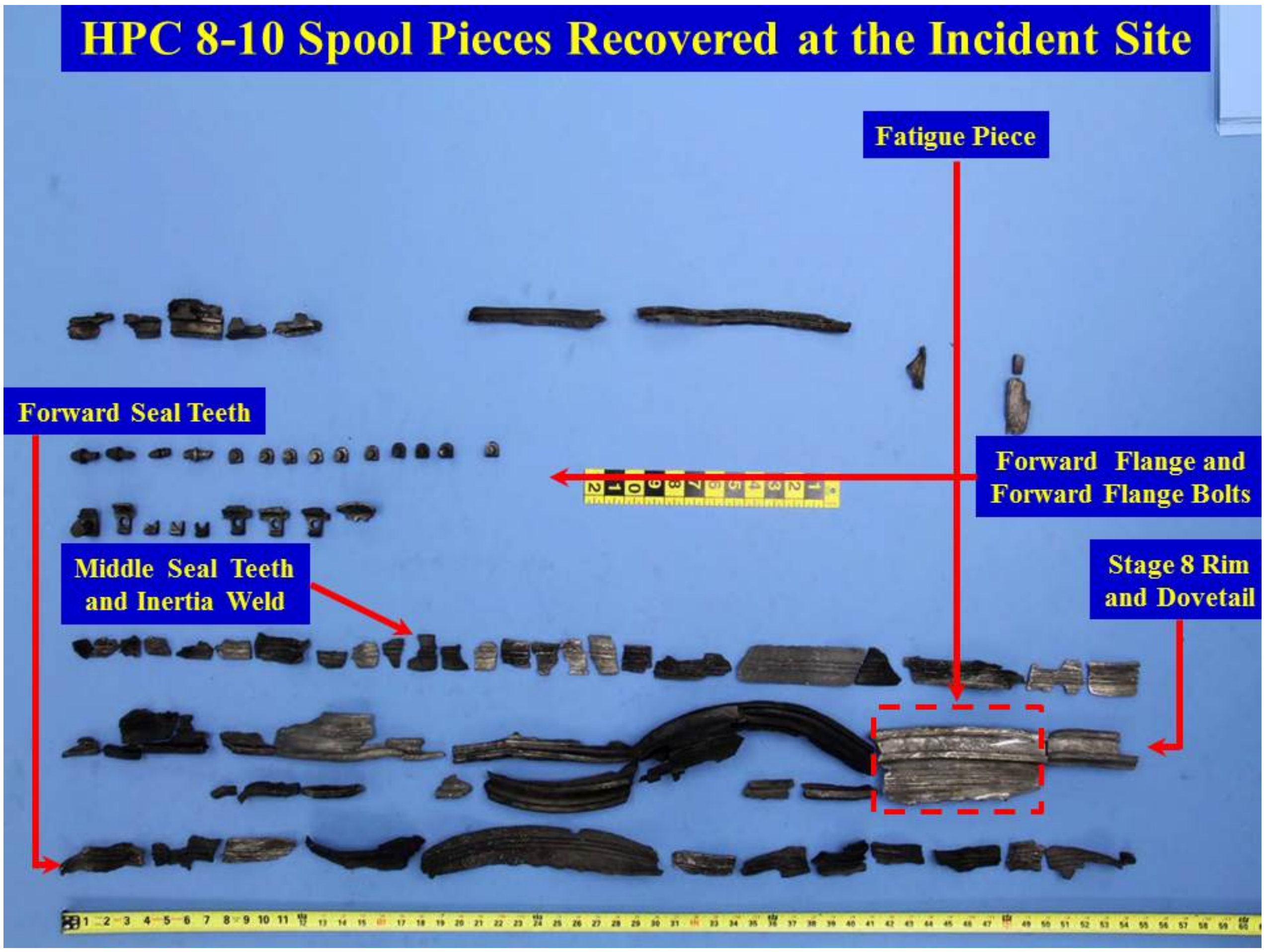

PHOTO 58: HPC STAGE 8-10 SPOOL PIECES RECOVERED ON-SITE. PIECE WITH FATIGUE HIGHLIGHTED

Examination of all the stage 8 fragments revealed a piece (See **PHOTO 58** and **PHOTO 4 -** leftmost piece) with a tinted/discolored flat-fracture region oriented in the circumferential plane located approximately 0.9 inch inboard of the blade slot bottom at the thinnest part of the web (web thickness in this area conformed to print dimensions of 0.118 inches) (**PHOTOS 59** and **60**). Detailed examination of the flat fracture region indicated that the fracture initiated along the aft face of the disk and initially progressed in an intergranular mode before gradually transitioning to a transgranular mode consistent with low-cycle fatigue (LCF) (**PHOTO 61**).[14] As the crack continued to progress in LCF, the flat-fracture region transitioned to a tinted/discolored slanted-fracture region (see the region labelled cyclic tensile in **PHOTO 60**). Taken together, the tinted/discolored regions extended through the web in the forward/aft direction (approximately 0.118 inch) and approximately 1.7 inch in the circumferential direction. Outside of this region, the appearance of the fracture surface was consistent with a tensile overstress fracture.

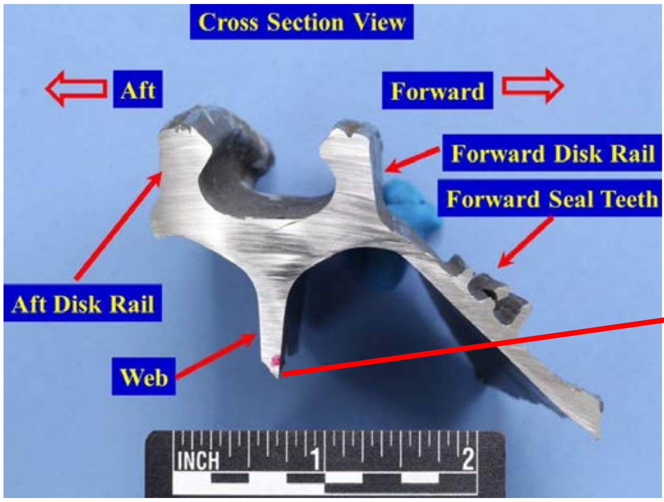

PHOTO 59: CROSS-SECTION - GENERAL LOCATION OF CRACK

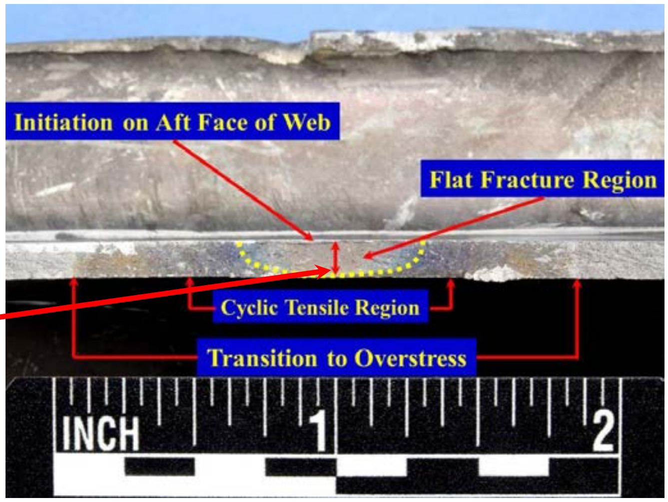

PHOTO 60: FRACTURE ORIGIN AT HIGHER MAGNIFICATION

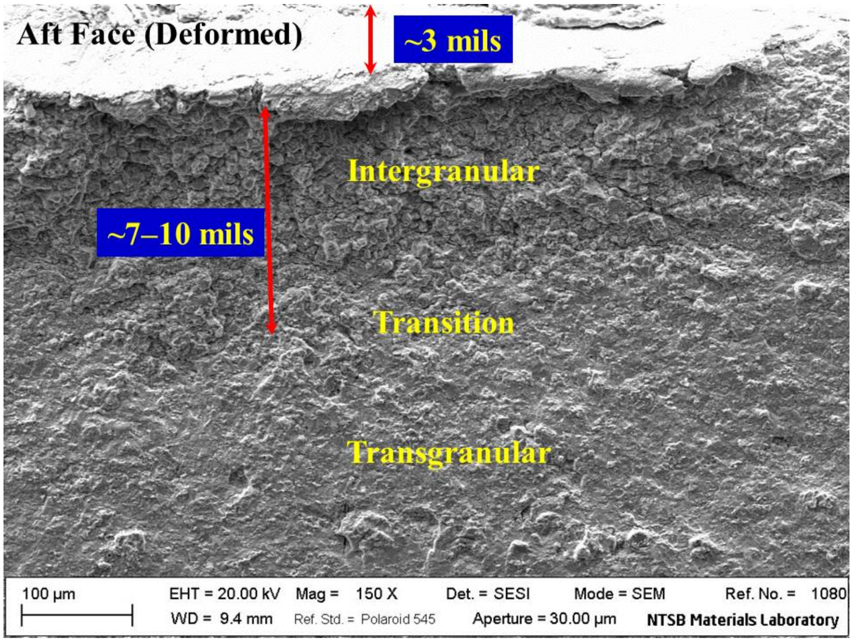

PHOTO 61: SCANNING ELECTRON MICROSCOPE IMAGE OF FRACTURE SURFACE

[14] The terms intergranular and transgranular describe how the crack advances through the metal, which is composed of three-dimensional crystals called grains. When the crack advances along the boundary between grains it advances in an intergranular mode. When the crack advances across the interior of a grain it advances in a transgranular mode.

A series of field emission scanning electron microscope (SEM)[15] images was taken in the transgranular LCF regime and striation density measurements were made on those images. [16] The purpose of striation density measurements was to estimate the number of flight cycles that elapsed between the point of detectable crack initiation (0.015 inches as defined by FAA Advisory Circular (AC) 33.70-1) and the end of the flat-fracture LCF region. It was assumed that one striation correlated with one flight cycle and that the forces during takeoff, when the stresses on the web are at their highest, were driving the advancement of the crack. The striation calculation was performed in the transgranular region where the striations were well defined and it was estimated that the fatigue crack progressed by LCF approximately 5,400 cycles beyond the point of detectable crack initiation. See Materials Laboratory Factual Report No. 15-131S, dated January 15, 2016 for details on the striation count calculation method. The last piece part inspection of the event spool was 3,943 cycles prior to the failure.

After the metallurgical examination was completed at the NTSB Laboratory, the parts were shipped to the GE Aviation Materials Laboratory in Evendale, Ohio where the rest of the HPC stage 8-10 spool parts were quarantined and additional examination of all the pieces was conducted. The intergranular region was examined in greater detail during subsequent examinations performed at GE.

4.2 Metallurgical Examination Conducted by GE

Further evaluation of the HPC stage 8-10 spools pieces was conducted at the GE Aviation Material Laboratory over several weeks with persons from the NTSB and FAA participating and overseeing much of the additional examination and testing. For complete details of the metallurgical findings, see GE Metallurgical Investigation Report Log No. FAL2015-17359. With all the recovered parts of the HPC stage 8-10 spool put together, it appeared that the entire HPC stage 8 disk rim and web was accounted for and the fracture origin found on the piece recovered on-site matched with the facture origin on the stage 8 disk that was removed from the engine during the engine exam (**Photos 62** and **63**). According to GE, the crack initiation propagated with intergranular features with local variations consistent with hold-time, high-alternating stress, LCF (hold-time LCF/sustained-peak, low cycle fatigue (SPLCF)). The crack then transitioned to transgranular, striated areas that decreased in density progressing through the web thickness, consistent with more cyclic-driven LCF. Deformation was noted

[15] In an optical microscope, light from a source is focused on the sample and the image is formed when the sample reflects and absorbs different wavelengths of the light and your eye detects the difference to form an image. An electron microscope works in a similar manner but instead of using light, it uses electrons to bombard the sample in order to create images and to understand its composition. A scanning electron microscope (SEM) is one of the types of an electron microscope that offer high resolution and high magnification. The SEM focuses on the surface and composition of sample by scanning the surface of a sample with an incident electron beam. Electrons from the sample scatter creating secondary electrons typically of low energy value or the electrons from the incident beam bounce upon impact with the sample creating backscattered electrons typically of higher energy values; both of which are collected to create three-dimensional images that are black and white. In the secondary image mode, the differences in surface topography are represented by variations in gray scale intensity while in backscatter mode, the image is mapping changes in material density. Some SEMs have an energy dispersive x-ray spectroscopy (EDS) detector that captures x-rays emitted from the sample during the creation of secondary electrons. When creating secondary electrons, x-rays are emitted as electrons from the high energy outer shells fill the void left by the ejection of lower energy shells electrons in order to stabilize the atoms state. The x-rays emitted are characteristic in energy and wavelength of the element that emitted them, so the composition of the sample can be determined. Elements that have high atomic number will have several x-ray elemental peaks while elements that have low atomic number have few x-ray elemental peaks. The various elemental peaks represent the shell that the electrons were ejected from and the shell from which the electrons were filled.

[16] Striations are linear features on a fatigue fracture surface that indicate how far the crack advances with each stress cycle.

at the fracture origin on both fracture halves at the aft surface (fracture origin) and according to GE was consistent with post-fracture, event-related impact damage.

PHOTO 62: STAGE 8 DISK SHOWING FATIGUE REGION AND SECONDARY CRACK LOCATIONS

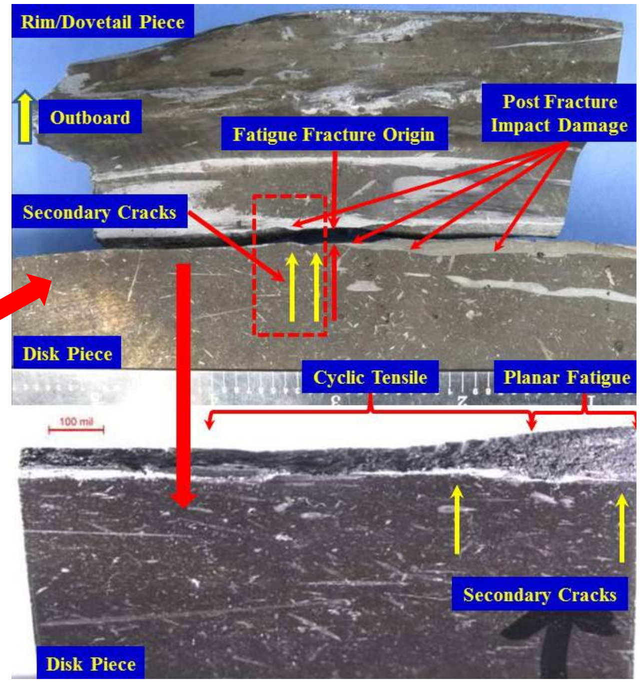

PHOTO 63: STAGE 8 DISK AND MATCHING RECOVERED RIM/DOVETAIL SECTION SHOWING FATIGUE REGION AND SECONDARY CRACK LOCATIONS

PHOTOS COURTESY OF GE

Metallographic mounts adjacent to the origin area (excluding the post-fracture surface damage) showed no microstructural anomalies. A SEM energy dispersive spectroscopy (EDS) of the fracture surface revealed only elements consistent with the specified material - Inconel® 718DA[17] (**PHOTO 64**). Etching confirmed that the grain structure was as specified and also that no heat affected zones were present (**PHOTO 65**). Micro-hardness traverses on the forward and aft sides of the disk revealed that the material had the proper hardness on both sides, between 46-48 on the Rockwell Hardness "C" Scale (HRC) (**PHOTO 66**). Further analysis of the intergranular region using techniques such as high high-resolution SEM and transmission electron microscopy (TEM)[18] did not reveal any detrimental species at the grain boundaries, only base metal oxides.

[17] Inconel® 718 is a wrought precipitation hardenable nickel base alloy. The 'DA' stands for Direct Aging heat treat process that improves fatigue durability by obtaining at specific microstructure. This material is made up of Chromium (Cr - 19%) Iron (Fe - 18%), and Niobium (Nb - 5.3%); traces of Carbon (C), Titanium (Ti), Aluminum (Al), and Molybdenum (Mo) all less than 1%; and the remainder is Nickel (Ni).

[18] TEM, like the SEM, is another type of electron microscope that offers high resolution and high magnification. Unlike SEM, TEM focuses on the inside of the sample and produces a two-dimensional fluorescent images by directing the incident beam through the sample; the sample must be thin enough for the electrons to pass.

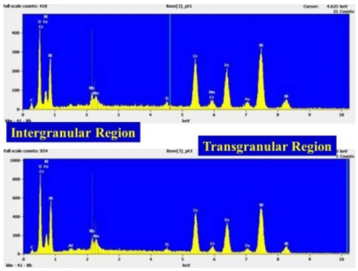

Photo 64: Material Composition

Photo 65: Etched and Polished Fracture Showing Proper Microstructure

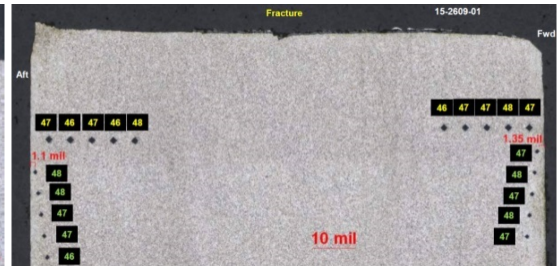

Photo 66: Micro-Hardness Measurements at Various Locations

Photos Courtesy of GE

Multiple secondary circumferential cracks were found on the aft face of the web within 0.016 inches radially of and circumferentially adjacent to the fatigue region. The secondary cracks measured up to approximately 0.003 inches circumferentially by 0.0005 inches deep and initiated in a combination of intergranular/transgranular modes at the base of oxidized/etched carbides with no surface/microstructural anomalies (**Photo 67**). According to GE, based on the level of oxidation and circumferential orientation, the cracks were consistent with pre-existing fatigue cracks and were not event-related. Furthermore, since the secondary cracks were only localized in the area near the fracture origin, this indicates a locally high stress area. No damage or microstructural anomalies were found at the secondary cracks; similar to what was found on the primary fracture. No secondary cracks were found on the diametrically opposed side of the aft web in the same radial plane or on the forward face (with one exception at/near the slant fracture).

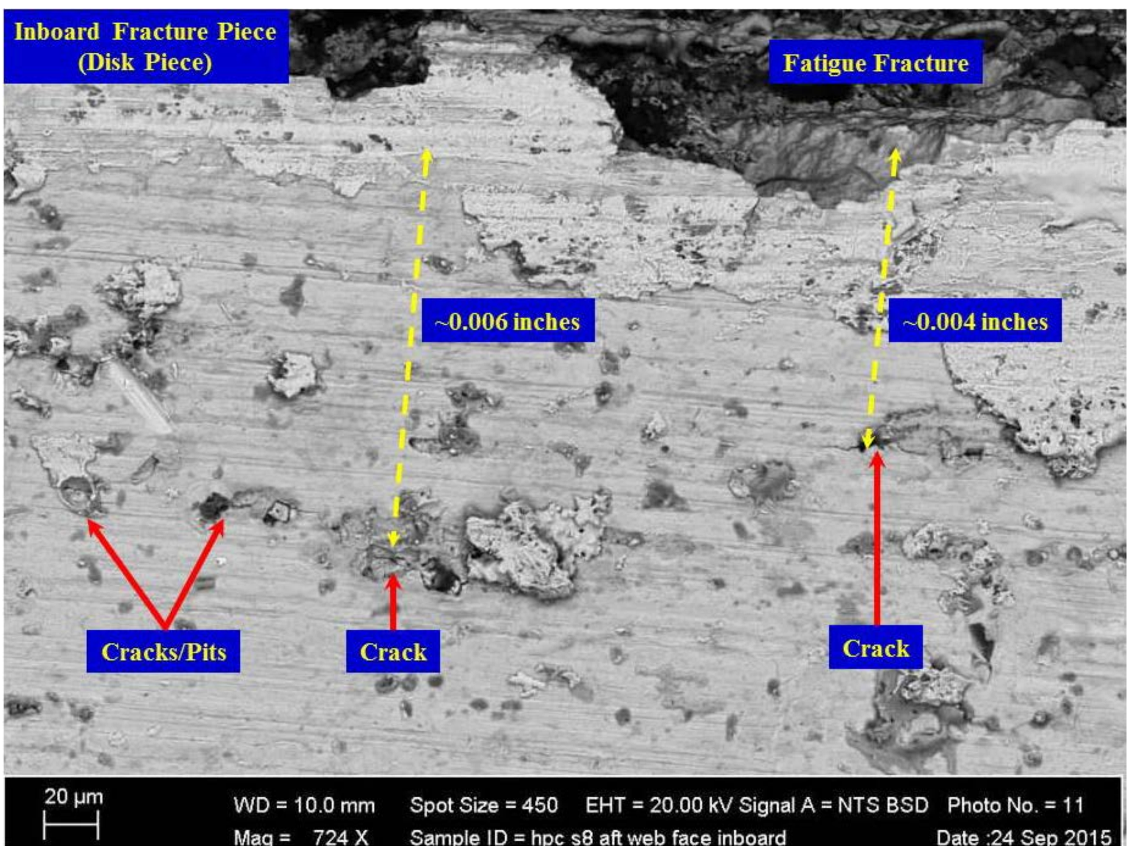

Photo 67: Secondary Cracking Found in Stage 8 Disk Piece

Photo Courtesy of GE

Examination of both the forward and aft face of the stage 8 web near the fracture radius found a visible difference in the shot peening appearance, with the forward web face having more pronounced peening dimples than the web aft face (**PHOTO 68**). Since no anomalies were noted at the fracture initial site, residual stress measurements were taken to determine the effect that the variation of the peening may have on the part. The measurements were taken on the event disk on both the forward and aft faces of the web at the fracture origin and at the opposite side where the dove tail slot was still in place. Measurements were also taken on two out-of-service HPC stage 8-10 spools of similar vintage (sister spools) for comparison. According to GE, the results showed that on the forward web, the sister spools' compressive profile conformed to expectations for a well peened surface. However, the forward web on the event disk was less compressive near the fracture surface but approached the expected profile further inboard radially (**FIGURE 10**). For the outer aft web, all the disks – the event and sister spools – showed compressive profiles that did not conform to expectations with a well peened surface but according to GE the profiles and surface variation were consistent for as-turned surfaces with little or no peening (See **APPENDIX C** in GE laboratory report for details) (**FIGURE 11**). Along with residual stress measurements, GE performed two additional measurements/tests; LCF testing and strain (grain distortion) measurements. LCF testing was conducted on specimens of the event stage 8 disk and comparison disks. All but one of the fatigue samples conformed to the GE average life curve for the specified material (See **APPENDIX D** in GE laboratory report for details). One sample from the event disk fell between the average life curve and the minimum life design curve. Strain measurements of the event web material near the fracture origin or the secondary cracks were taken to determine if there was any additional work that may have been pre-existing beyond the normal machining and peening inducted stresses. According to GE, the uniform level of strain on the aft face was less than on the forward side but was consistent with the difference in the peening observed on each side of the web (See **APPENDIX D** in GE laboratory report for details).

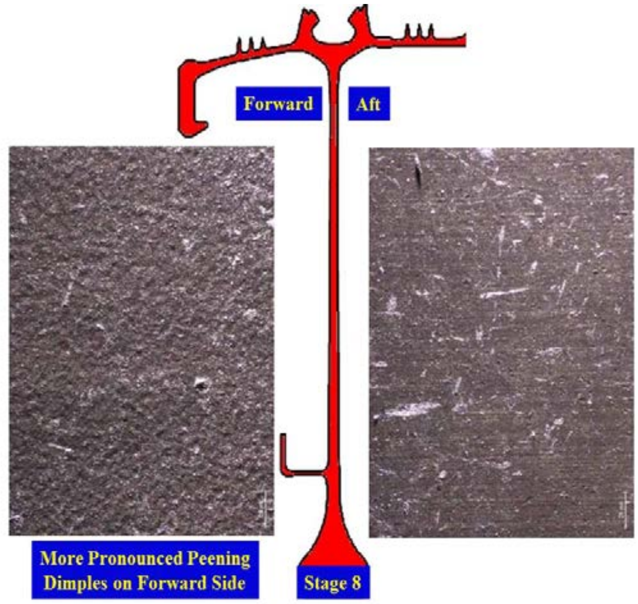

PHOTO 68: VARIATIONS IN SHOT PEENING APPEARANCE

PHOTO COURTESY OF GE

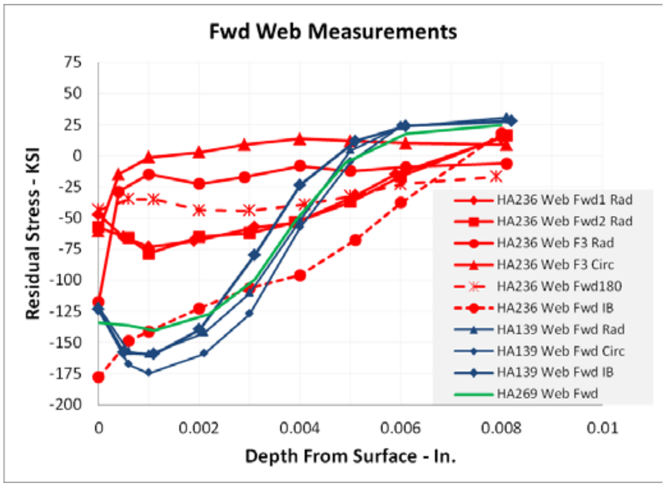

FIGURE 10: FORWARD WEB RESIDUAL STRESS

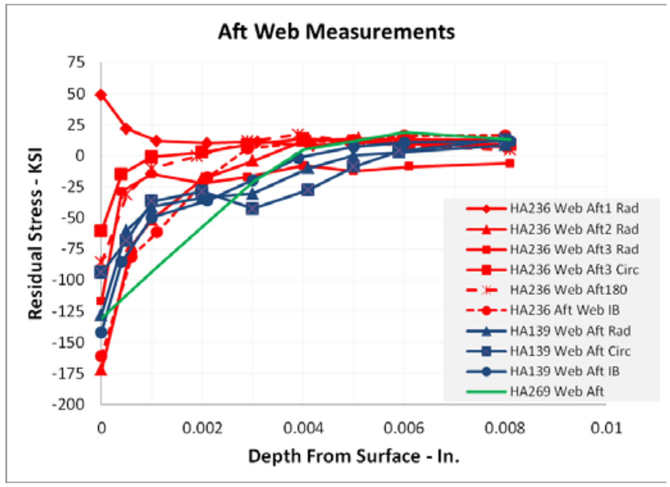

FIGURE 11: AFT WEB RESIDUAL STRESS

FIGURES COURTESY OF GE

GE also conducted independent striation density measurements though the LCF region, which did not include the initial intergranular region, and came up with cycle counts between approximately 5,000 – 5,700 cycles, similar in magnitude to the NTSB estimate.

5.0 HPC STAGE 8-10 SPOOL MANUFACTURING HISTORY & LIFE LIMIT CALCULATIONS

5.1 HPC STAGE 8-10 SPOOL LINEAGE

In the GE90 EM, Chapter 05-11-02 *Compressor Rotor – Life Limits* lists 4 approved PNs for the HPC stage 8-10 spool and their life limit (**TABLE 2**). For simplicity PN 1694M80G04[19] (the event spool) will be referred to as 'M80' and PNs 1844M90G01 and G02[20] will be referred to as 'M90'. The event spool was a 'M80'; a total of 163 were manufactured and shipped between May 1994 and end of production in November 1999. In 1996, a process improvement was introduced that changed the weld geometry to improve inertia weld upset stability. This process improvement did not require a PN change. Seventy-four (74) 'M80' spools, including the event spool, were manufactured and shipped prior to the inertia weld process improvement.

The only current in-production HPC stage 8-10 spool is PN 1844M90G02. The major differences between the 'M80' and the 'M90' configurations are that: 1) the 'M80' is a two-piece forging spool with an inertia weld between stage 8 and 9 (See **FIGURE 6**) whereas the 'M90' configuration is a three piece forging spool with an inertia weld between stage 8 and 9 and between 9 and 10 disks (See **FIGURE 5**), 2) the 'M90' has an increased thickness in the rim and web for all three stages, and 3) the 'M90' has an increase in the stage 8 bore size. Both 'M80' and 'M90' configurations are constructed of the same material.

TABLE 2: GE 90 HPC STAGE 8-10 SPOOL LIFE LIMITS		
Part Number	Life Limit	Comments
1687M40G26	3,000	Out-of-Production, Limited Use on Specific Subpopulation of Engine, In-Service
1694M80G04	16,600	Out-of-Production, In-Service, Event PN
1844M90G01	16,700	Out-of-Production, In-Service
1844M90G02	16,700	In Production, In-Service

5.2 MANUFACTURING HISTORY OF EVENT SPOOL – PN 1694M80G04/SN GWNHA236

The event HPC stage 8-10 spool was composed of two separate forgings that are inertia welded together. The forward section was made of Inconel® 718DA and was comprised of the forward rabbeted flange, the forward seal teeth, and the stage 8 disk. The rear section was made of Rene® 88DT and it comprised the middle and aft seal teeth, stages 9 and 10 disks, and the rear drive arm. The forward and rear sections were inertia welded just aft of the stage 8 blade circumferential dovetail slot and forward the middle seal teeth (See **FIGURE 6**).

Both forgings (forward PN 4013382-523P02 and rear PN 4013382-708P01) for spool SN GWNHA236 were manufactured by Wyman Gordon in Massachusetts in 1994 and delivered to the GE Aviation facility in Wilmington North Carolina (NC) for machining and inertia welding. The event forward section (stage 8 disk) was one of 16 forgings manufactured by Wyman Gordon in Grafton Massachusetts in April 1994 from a 10 inch round billet. The forward forging was then shipped to GE Aviation Wilmington in November 1994 where it was rough turned, semi-finished cut, and the forward and aft sides of the stage 8 disk web were finished cut in preparation for inertia welding. The machining operations created PN 4013382-558P06. The event spool rear section (stage 9 and 10 disk) was one of

[19] 'M80G01', 'M80G02', and 'M80G03' configurations were never certificated.
[20] Difference between 'M90G01' and 'M90G02' is a rear flange bolt hole geometry change.

18 forging manufactured by Wyman Gordon in Worchester Massachusetts in June 1994 from a 12¼ inch round billet. Before the completed rear forging was shipped to GE Aviation Wilmington for processing, it was first sent to GE Cincinnati for rough turning in the area between stage 9 and 10 disk webs and then sent back to Wyman Gordon for solution and age heat treat before it was sent to GE Aviation Wilmington in December 1994 for final processing. At GE Aviation Wilmington, the aft forging was rough turned, semi-finished cut, and the forward and aft sides of the stage 9 and 10 disk webs were finished cut in preparation for inertia welding. The machining operations created PN 4013382-794P02. PNs 4013382-558P06 (forward section) and 4013382-794P02 (rear section) were inertia welded to create PN 1694M80G04 – the event spool PN.

Inertia welding is a process by which the two pieces are welded together by mechanical friction due to the relative motion of one piece to the other under high loads that causes the material at the weld joint to plastically displace and fuse together. After inertia welding, the one-piece spool is heat treated, the flash (material upset that is created during the inertia welding process) is machined, weld sonic inspected, the spool is rough turned, finished cut, blade slots machined, bolt holes drilled, and the knife edge seals machined before the part is FPIed and ECIed. After the spool passed the various inspections, it was sent to Wilmington National Peening in Wilmington NC for shot peening. The seal teeth are coated at this point. After peening, the spool was returned to GE Aviation Wilmington for final dimensional inspection and cleaning before shipping. In June 1995, the event spool SN GWNHA236 was shipped from GE Aviation Wilmington to Snecma for assembly into HPC module PN 1650M53G01, SN WM022022 and that module was delivered to the GE facility in Durham NC for engine assembly.

5.3 SHOT PEENING OPERATION –'M80' AND 'M90' SPOOLS

During the metallurgical examination of the event HPC stage 8-10 spool, visual examination of the forward and aft surfaces of the stage 8 disk web revealed a difference in the shot peen appearance, especially in the outer radial portions. The shot peen on the forward side was more pronounced than on the aft side. Based on this visual difference, the shot peening operation sheets for the 'M80' and 'M90' spools were reviewed including a visit to Wilmington National Peening to view the process. It should be noted that the 'M80' spool has been out of production since November 1999, and the lance used to perform the shot peening had been discarded and the print of the lance was not retained. Therefore, only a review of the shot peening records for the event 'M80' spool could be performed, but for the 'M90' a review of the tooling and process was also performed. Essentially the process used to shot peen the event 'M80' spool is very similar to the current 'M90' spool process. When comparing the two shot peening process, any significant differences will be highlighted.

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

For the event 'M80' spool, surface coverage (% saturation), peening intensity, peening locations, and required intensity verification locations are all called out on the part drawing. The intensity is specified as 0.005-0.008 inches using the 'A' strip (**Photo 69**) and the intensity must be verified at the beginning and end of each run of a lot of parts and every eight hours of continuous running on a given set-up. Almen test strips are placed at designated locations on a scrapped part to verify intensity of the shot peening at those key areas. Review of the Almen test strip locations for the 'M80' part showed that the test strips were located on the forward (No. 21) and aft (No. 31) surfaces of the stage 8 disk web as well as on the ID surface underneath the middle seal teeth (No. 40B). No Almen test strips were located in the area of the fatigue fracture on the event 'M80' spool which was in the stage 8 disk web about 1 inch below the stage 8 blade slot. A review of the 'M90' Almen test strip locations revealed: 1) there was a significantly greater number of test strip locations than what was on the event 'M80' and 2) no test strips locations for 'M90' spool are in the same vicinity of the fatigue fracture found on the event 'M80' spool. Currently GE has added the Almen test strip for the 'M90' spool in the vicinity of the event 'M80' fracture location.

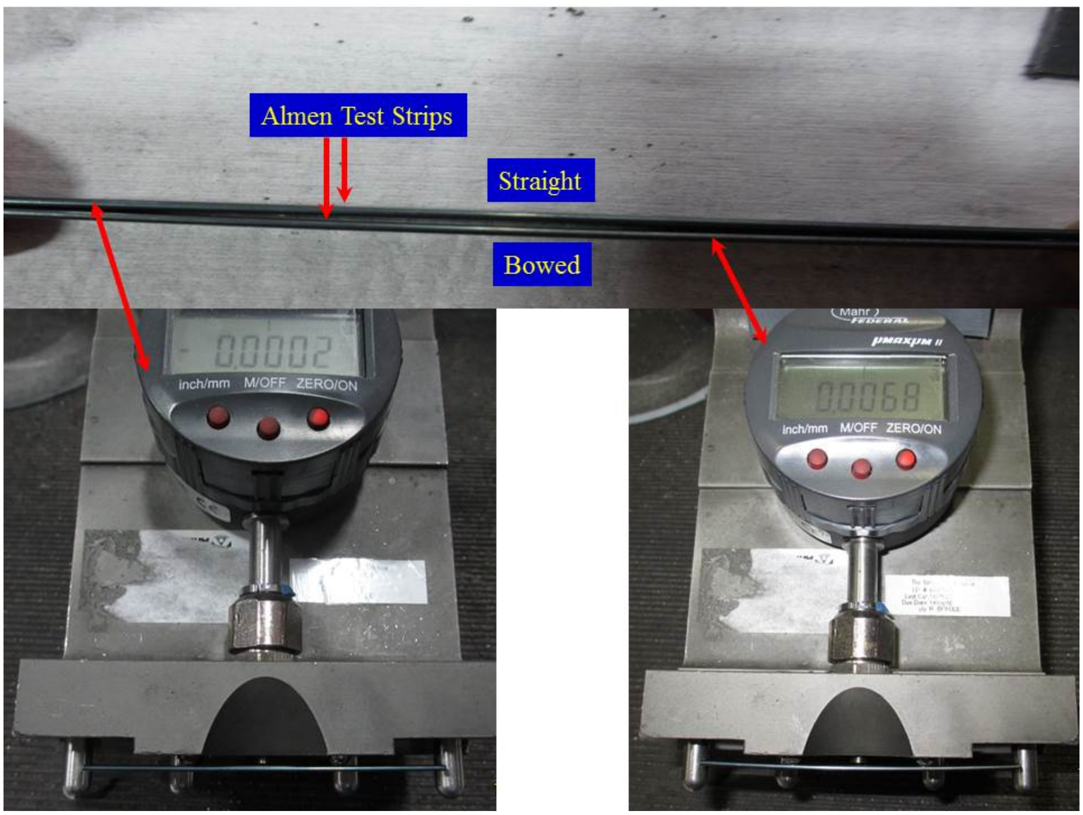

Photo 69: Exemplar Almen Strip Test for the 'M90' Spool

In 2006, Wilmington National Peening started using a new automated peening booth and new individually pressurized lance on the 'M90' spools (recall 'M80' were out of production in 1999 so the 2006 peening process changes only applied to 'M90' spools). GE reviewed 'M80' and 'M90' spools using the pre-2006 peening process change and compared the intensity and coverage in the upper aft web in the location of the event spool crack initiation site with 'M90' spools peened using the post-2006 peening process change. According to GE, inadequate intensity and coverage were found on the pre-2006 peened process changed spools whereas the post-2006 peened process changed spools showed adequate coverage.

5.4 MATERIAL REVIEW BOARD HISTORY – SN GWNHA236

When a part nonconformance is found, it is annotated on a deficiency record sheet. If the nonconformance or deviation is not specifically addressed by the manufacturing drawing and is deemed significant, a material review board (MRB), which is made up of different manufacturing and design engineers, reviews that deficiency and provides a disposition which is also recorded on that same sheet. Any nonconformance listed on the deficiency record, whether or not a MRB review is warranted, must be closed out before the part is acceptable.

Review of the manufacturing routing and deficiency record sheets for the forward (PN 4013382-558P06) and rear (PN 4013382-794P02) sections revealed that there were no nonconformances that warranted a MRB review and that all the nonconformances were cleared. Review of the manufacturing routing and deficiency record sheets for the spool (PN 1694M80G04) revealed a total of 15 nonconformances that were subject to a MRB disposition. Of those 15, GE deemed five had a potential impact on the stress distribution – favorably or unfavorably – on the entire spool. Of those five that impacted the stress distribution, only one related to the stage 8 disk web and it related to the radial position of the middle spacer arm and axial positioning of the disk web in relation to the dovetail blade slot; the combination of which can produce both tensile and compressive stresses during different phases of the flight/mission profile creating a condition referred to as "oil canning". Review of all the other nonconformances listed on the deficiency record revealed no defects or under minimum material thickness conditions relating to the stage 8 disk and all the nonconformances were cleared.

6.0 HPC STAGE 8-10 SPOOL LIFE LIMIT CALCULATIONS

The LCF life for both the 'M80' and 'M90' spools are base-lined against the GE-90B thrust, thermal, and stress levels; according to GE the event engine, a GE90-85B has lower stress than the GE-90B. The published life limit for the 'M80' is 16,600 cycles with the limiting feature being the stage 8 disk-to-forward rabbeted flange transition radius.

GE performed stress and lifing analysis on the event spool using standard LCF curves for minimum/average material thickness and properties at the GE-90-85B rating; this included all the MRB dimensions ('oil canning'), BA specific operational mission (taxing time, takeoff thrust rating, shutdown, core speeds and temperatures, ambient takeoff temperature, etc.), and shutdown stresses, etc. Based on the analysis, the fracture location (stage 8 web) on the event spool has a calculated LCF life about 1.5 times (1.5x) (29,800/20,800 = 1.4) greater than the event spool limiting feature (stage 8 disk-to-forward rabbeted flange transition radius) at the minimum material properties and about 9x (187,400/20,800 = 9) at the average material properties. Based on the striation count analysis that the NTSB and GE performed, the crack initiated at roughly 6,000 CSN which is 5x (29,800/6,000=5) less than the predicted value for the failure location under the worst conditions (highest stresses and temperatures and minimum material properties); the GE analysis could not close the gap between the predicted LCF crack initiation and the estimated actual event crack initiation. It should be noted that all the predictive calculations that GE performed could not close on the event crack location at the number of cycles that was thought to have initiated the crack; however, the striation density curves that were developed for the event spool match well with the analytical predictive crack propagation rate (**FIGURE 12**).

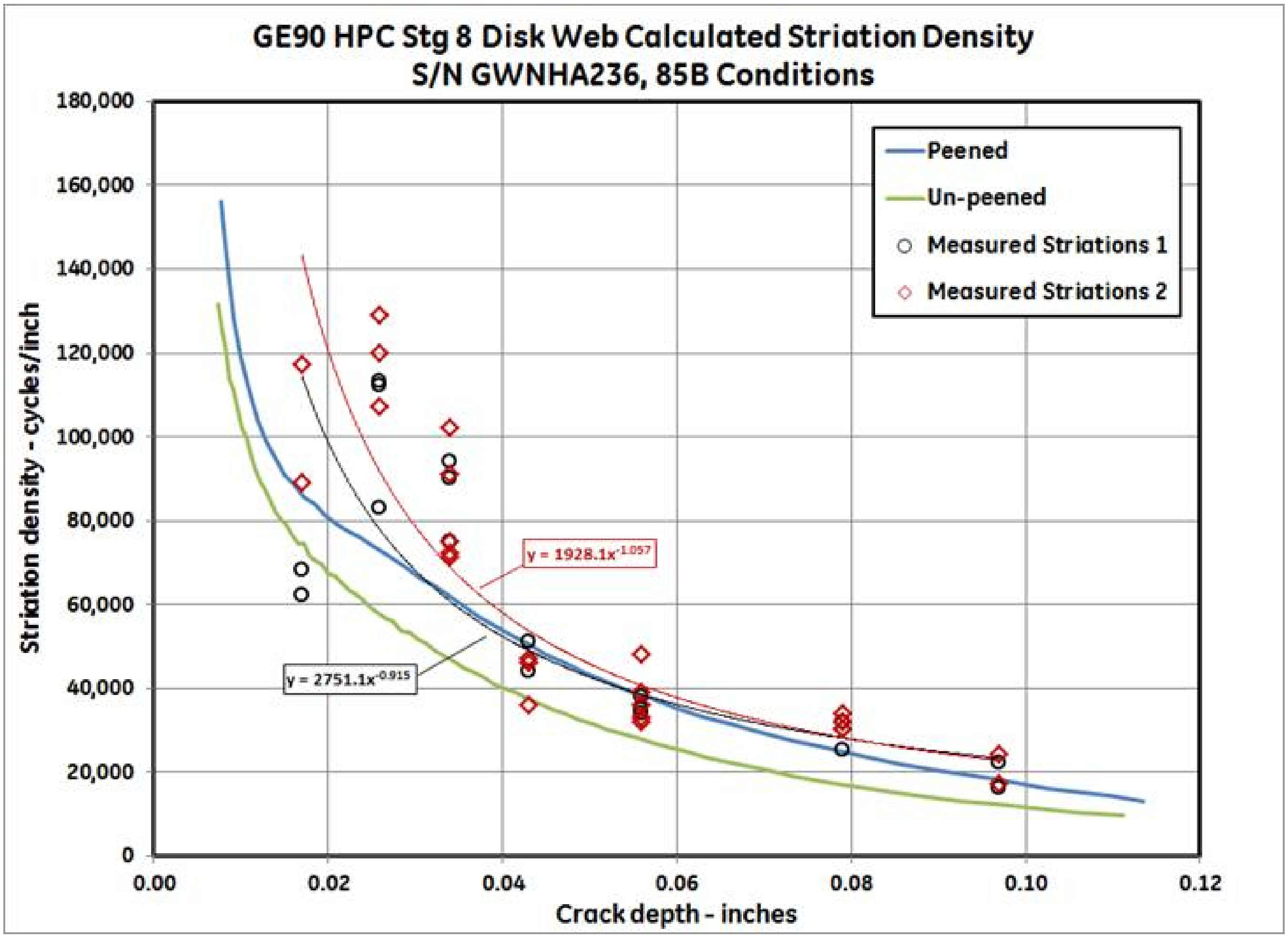

FIGURE 12: STRIATION DENSITY CURVE

FIGURE COURTESY OF GE

7.0 HPC STAGE 8-10 SPOOL ENGINE MANUAL INSPECTIONS – CURRENT AT THE TIME OF THE EVENT

7.1 HPC STAGE 8-10 SPOOL PIECE PART INSPECTION

According to EM Task 72-31-08 Inspection 001, the HPC stage 8-10 spool is cleaned using four step alkaline cleaning and acidic descaling (without inhibited phosphoric acid) before a class 'G' FPI of the entire spool and an ECI of the inertia weld (Task 72-31-08 Special Procedure 001) is conducted. The cleaning procedures includes the following CAUTION at the end of the procedure

> CAUTION: PARTS MUST BE THOROUGHLY DRY PRIOR TO FPI INSPECTION. WATER ON THE PARTS MAY DEGRADE THE SENSITIVITY OF THE PROCESS, ESPECIALLY FOR THE WATER WASHABLE PENETRANT PROCESS

A class 'G' FPI is performed in accordance with standard practices manual (SPM) Task 70-32-02-230-01 and critical areas such as disk bores, bolt holes, dovetail slots, locking lug slots, flanges, fillets, inertia welds, and interstage seal teeth. FPI processes are categorized according to type of process by class (Class 'A', 'B', 'C', 'D', and 'G') and degree of sensitivity by level (Level 1, 2, 3, and 4). Parts such as the HPC stage 8-10 spool are critical rotating parts so more sensitive FPI material and processes are used. The class of the FPI refers to method of penetrant application and indicates penetrant dwell times, emulsification, and developer requirements. Class ‘G’ is an ultrahigh (highest penetrant class and highest sensitivity) and requires the use of a post-emulsifiable penetrant with a hydrophilic remover. The method of application of the penetrant (spray, brush, immersion, or flow) and the method of inspection (visual, ultraviolet borescope, or specialized video/camera) depends on the size and geometry of the part inspected. According to the SPM inspection, hub bores and faces can be inspected using mirrors, an ultraviolet borescope can be used to aid inspection of the recessed cavities or inside diameters (ID) of shafts, and a charge-coupled device (CCD) camera to see remote internal parts are all permissible FPI visible inspection techniques. According to GE, GE Wales (the facility that inspected the event HPC stage 8-10 spool) was and currently is the only facility inspecting the spool at the piece part level and they were employing the mirror inspection technique only.

The EM, not the SPM, specifies the visual inspection criteria and the serviceable and repairable limits. According to the EM, indications less than 0.015 are not interpretable. There are no specific inspections called out for the web area. The inspection criteria for the web is covered in the general inspection subtask 72-31-08-220-053, -054, -055 that relate to all surfaces except the interstage seal teeth. Subtask 72-31-08-220-053 is a crack inspection; no cracks are allowed (except on the interstage seal teeth) and if cracks were found there is no repair and the part must be replaced. Subtask XX-054 and -055 relate to silver corrosion pitting and red deposits or oxidation without pitting respectively. Silver corrosion pitting is not serviceable nor repairable whereas red deposits or oxidation without pits is serviceable.

7.2 HPC Rotor/Module (Module) [21] Level Inspection

According to EM Task 72-00-31 Inspection 001, the stage 8-10 spool has a general visual inspection for cracks; no cracks are allowed except for in the interstage seal teeth. This was the inspection requirement and criteria at the last HPC module removal in June 2014. On September 3, 2015, GE issued an incremental change to the EM to include Task 72-0031 Special Procedure 004 for an ECI of the stage 9-10 inertia weld. An incremental change was also issued at that same time to Task 72-00-31 Inspection 001 to call out Special Procedure 004. The rotor/module incremental change was immediately available after its release electronically on the GE Customer Web Center.

[21] The core module is made up of ten axial flow high pressure compressor (HPC) rotor and stator stages. The No. 3 roller bearing supports the forward end of the HPC rotor. The No. 3 ball bearing which is also located in the front of the HPC rotor takes the thrust load.

8.0 CORRECTION ACTIONS

8.1 PROACTIVE SPOOL INSPECTIONS

GE performed proactive spool inspections to better understand the root cause of the failure event. GE inspected 59 HPC stage 8-10 spools (48 'M80' and 11 'M90') at the piece part level from engines that had already been removed from service. All 59 spool were subjected to an ECI of the stage 8 web and tube flange inspection and 31 of those were fully inspected over all three stages. As previously mentioned in Section 5.1 *HPC Stage 8-10 Spool Lineage*, in 1996 a process improvement for the inertia weld was implemented that did not require a PN change. The subpopulations of 48 'M80' spools included those with and without the inertia weld process improvements. No crack-like indications were found on any of the spools. The entire population of 'M80' still in-service, that is to say either currently in flight operation, installed in an engine that can be installed on an airplane for service, or in a maintenance shop for inspection, is 56.

GE selected three engines with 'M80' spools installed for removal and inspection based on the spool manufacturing date, CSN, and manufactured geometry (similar approved/allowed deviations as event spool). The intent was to remove the engines and then conduct the spool inspections at the engine, module, rotor, and piece part level implementing the techniques developed due to this event (See Section 8.2 for specifics on the various level inspections).

8.2 ENGINE MANUAL CHANGES - INSPECTIONS

As mentioned in Section 7.0, at the piece part level, the entire HPC stage 8-10 spool is subjected to an FPI inspection and only the inertia weld is subjected to an additional or special inspection (Task 72-31-08 Special Procedure (SP) 001 – inertia weld special ECI). This was the requirement at the time of the failure event and when the event HPC was last removed in June 2014; there were no other specific spool inspections at any of the higher assembly/module/engine levels. Subsequently a HPC stage 9-10 inertia weld ECI was included but no special inspections for any of the disk webs. Based on this failure event, GE developed and incorporated unique tooling and comprehensive inspections instructions for the HPC stage 8-10 spool web at the piece part level, rotor level (blades still installed), module level (core module separated from the fan hub module and the LPT module), and engine level.

8.2.1 Engine and HPC Assembly Inspections

On November 10, 2015, GE issued an incremental change to Task 72-00-00 Engine Assembly Inspection 001 calling out an ECI of the stage 8 aft disk web of the HPC stage 8-10 spool (72-00-31, Special Procedure 006). This inspection is only to be performed when the aft flange of the CDP seal is exposed (See Sections 8.2.2, 8.2.3, and 8.2.4 for specific details on the inspection). Special Procedure 006 and the conditions for the inspections (CDP seal exposure) were also called out in Task 72-00-30 HPC Assembly Inspection 001 and the entire SP 006 was included in Task 72-00-31 HPC Rotor Assembly. Along with SP 006, entire Task 72-00-31 SP 007, which calls for an UTI of the stage 8 web, was also incorporated in the HPC Rotor Assembly Inspection (See Section 8.3 for specific details on the inspection). It should be noted that the SPs 006 and 007 were released the day before (November 9, 2015) the incremental changes to the Engine and HPC Rotor Assembly Inspections (November 10, 2015) were released. All the incremental changes to the Engine and HPC Rotor Assembly Inspections were immediately available electronically on the GE Customer Web Center after issuance. **FIGURE 16** provides a list of all the new HPC stage 8-10 spool inspections.

8.2.2 HPC Stage 8-10 Spool Piece Part Inspections & Repair

At the piece part level, two additional inspections were added to the EM for the HPC stage 8-10 spool. On October 20, 2015, GE issued: 1) incremental change Task 72-31-08 SP 002 to ECI the stage 8 forward web, stage 9 disk forward and aft web, stage 10 disk forward and aft web, and the tube flange and 2) incremental change 72-31-08 SP 003 to ECI the stage 8 aft web (**FIGURE 13**). An incremental change was also issued at that same time to Task 72-31-08 Inspection 001 to call out SPs 002 and 003. On November 9, 2015, GE also issued Task 72-00-31 Special Procedure 007 to Ultrasonic Inspect (UTI) the stage 8 web. This inspection was designed and intended to be performed at the rotor level (See Section 8.2.3); however, this inspection can also be performed at the piece part level as well. All these piece part incremental changes were immediately available electronically on the GE Customer Web Center after issuance.

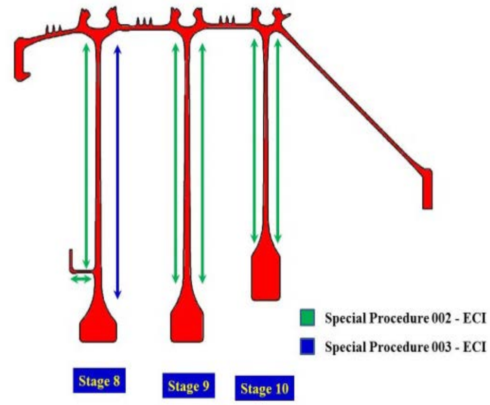

FIGURE 13: PIECE PART SPECIAL PROCEDURE ECI LOCATIONS

8.2.3 HPC Rotor Assembly Level Inspections

At the rotor assembly level, two additional inspections were added to the EM for the HPC stage 8-10 spool. On November 9, 2015, GE issued: 1) incremental change Task 72-00-31 SP 006 to ECI the stage 8 aft disk web and 2) incremental change 72-00-31 SP 007 to UTI the stage 8 web (**FIGURE 14**). <u>**Both**</u> the ECI and UTI are to be performed at the rotor assembly level; there is no option to perform one and not the other. Although SP 006 inspection can be performed at the rotor level, it can also be performed at the module level as well (See Section 8.2.4). Similar to the piece part SPs, the rotor level incremental changes were immediately available electronically on the GE Customer Web Center after issuance.

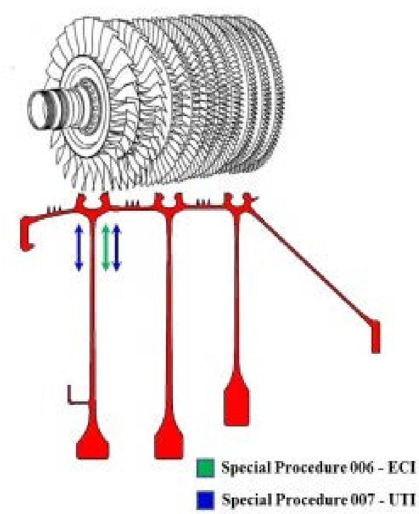

FIGURE 14: ROTOR LEVEL SPECIAL PROCEDURE ECI AND UTI LOCATIONS

8.2.4 HPC Module Level Inspection

At the module level, one additional inspection was added to the EM for the HPC stage 8-10 spool. On November 10, 2015, GE issued incremental change the stage 8 aft disk (FIGURE 15). The incremental changes were immediately available electronically on the GE Customer Web Center after issuance.

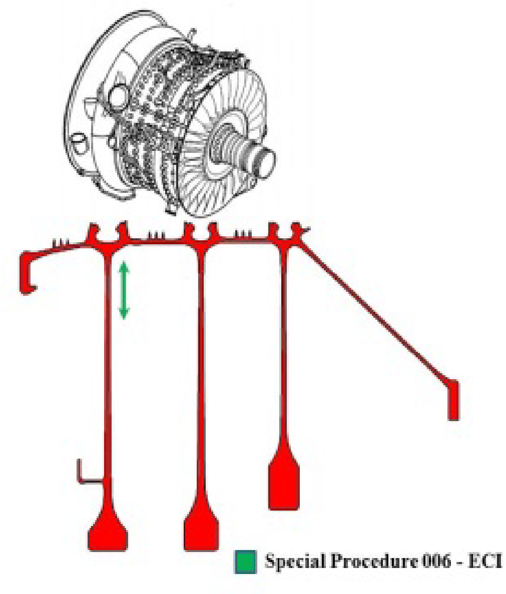

FIGURE 15: MODULE LEVEL SPECIAL PROCEDURE ECI LOCATION

Date Issued	ATA Chapter/Tasks	Description
October 20, 2015	72-31-08, SP 002	Piece Part – ECI (Stage 8 forw and tube flange, Stages 9 & 10 forward/aft web)
October 20, 2015	72-31-08, SP 003	Piece Part – ECI (Stage 8 aft w
October 20, 2015	72-31-08, INSP 001	Piece Part – Calls SP 002 & SF
November 9, 2015	72-00-31, SP 006	Rotor Level – ECI (Stage 8 aft
November 9, 2015	72-00-31, SP 007	Rotor Level – UTI (Stage 8 we
November 10, 2015	72-00-31, INSP 001	Rotor Level – Calls SP 006 &
November 10, 2015	72-00-00, INSP 001	Module Level – Calls 72-00-3 when CDP seal aft flange expo

FIGURE 16: OVERVIEW OF THE INSPECTIONS FOR THE HPC STAGE 8-10 SPOOL

8.3 SERVICE BULLETINS & LETTER, AIRWORTHINESS DIRECTIVES, AND DRAWING CHANGES

On November 24, 2015, GE issued a category 2[22] service bulletin (SB) 72-1145 titled *Engine – Compressor Rotor Assembly (72-31-00) – One-Time On-Wing Ultrasonic Inspection of the Stage 8 Web of the Stage 8-10 Spool*. The SB provided instruction to perform an on-wing UTI of the stage 8 web of affected 'M80' HPC stage 8-10 spools. Based on manufacturing history, cycles accumulated, and the best available data at the time the SB was issued, GE identified two groups of affected 'M80' HPC stage 8-10 spools that should be inspected as soon as possible. GE recommended that spools in Group 1 be inspected within 150 cycles and spools in Group 2 be inspected within 300 cycles from the date of issuance of the SB. Group 1 is comprised of 2 spools manufactured prior to the 1996 inertia weld process improvement with over 11,000 cycles accumulated and Group 2 is comprised of 4 spools manufactured prior to the 1996 inertia weld process improvement with over 10,000 cycles accumulated. Although the SB provides instructions for performing an on-wing UTI, compliance, should the operator choose to, can also be accomplished by in-shop ECI and UTI procedures discussed in Sections 8.2.2, 8.2.3, or 8.2.4. Similar to the incremental changes to the EM, this SB was immediately available electronically on the GE Customer Web Center after issuance. The FAA issued AD 2015-27-01 with an effective date of January 27, 2016, that required the inspection of the HPC 8-10 spools and in the time frame listed in SB 72-1145.

BA was the launch customer for the Boeing 777, so their engines, components, and parts on the GE 90 typically have the most operational time accumulated. Five of the six spools listed in SB 72-1145 and mandated by AD 2015-27-01 were installed in engines operated by BA. As of January 18, 2016, all five of the BA HPC stage 8-10 spools had been inspected using UTI procedures and no defects or anomalies were found. The one other HPC stage 8-10 spool was on a GE lease engine and it was inspected on November 14, 2015 using UTI procedures and no defects or anomalies were found as well.

On February 12, 2016, GE issued a category 2 SB 72-1146 titled *Engine – Compressor Rotor Assembly (72-31-00) – One-Time On-Wing Ultrasonic Inspection of the Stage 8 Web of the Stage 8-10 Spool P/N 1694M80G04*. The SB provided instructions to perform a one-time on-wing UTI of the stage 8 web of all the remaining 'M80' HPC stage 8-10 spools not covered by SB 72-1145. SB 72-1146 called out the inspection of 50 'M80' spools; SB 72-1145 called out 6 'M80' spool for a total of 56. GE recommended that the spools be inspected before exceeding 10,500 CSN or within 500 cycles from the date of issues of the SB, whichever occurs later. SB 72-1146 required that the person conducting the inspection have specialized non-destructive inspection (NDI) certification and must receive practical training in the use of the procedure and tooling and show proficiency. Based on the focused requirements for the inspector, SB 72-1146 inspections will be initially conducted by a GE customer support team and not the airplane operator. On February 4, 2016, before SB 72-1146 was issued, Boeing released service letter (SL) 777-SL-72-002 informing the 777 operators of the issuance of SB 72-1145, its accompanying AD 2015-27-01 and the soon to be published SB 72-1146 with the anticipation that the FAA will follow suit with an AD.

On June 10, 2016, GE issued a category 2 SB 72-1151 titled *Engine – Compressor Rotor Assembly (72-31-00) – On-Wing Ultrasonic Inspection of the Stage 8 Web of the Stage 8-10 Spool*. This

[22] The following is how GE defines a category 2 service bulletin: **Usual Statement**: Do as soon as possible without effect on revenue service or before XX hours, YY cycles, or a specific end date or specific interval. **Explanation**: Compliance is recommended based on GE technical evaluation and when an aircraft can stay at a line station or maintenance base with the capability to do the procedure. Justification for hour, cycle, and end date requirements will be based on technical considerations only (safety, risk analysis, etc.). This category may cause non-routine operator action and must be complied with on all engines presently in the shop

SB superseded SB 72-1146 and differed from it in several key aspects. Firstly, SB 72-1151 expanded the inspection population size to not only include all ‘M80’ spools but a limited subpopulation of the ‘M90G01’ and ‘M90G02’ (138 ‘M90’ spools) manufactured prior to an inertia weld process improvement. GE introduced an inertia weld process change on December 17, 2001 for the ‘M90’ spool (‘M80’ spool production had ended in November 1999) that added damping material to all the disk webs to reduce vibratory stress into the spool. As already mentioned, no anomalies were found during the physical or metallurgical exams to account for the crack initiation of the event spool. Even though there is no definitive evidence to suggest that elevated vibratory stress introduced during the inertia weld process caused or contributed to the failure of the event spool, GE decided to expand their field management plan to include all spools, ‘M80’ and ‘M90’ manufactured prior to the vibration damping process improvement. Secondly, SB 72-1151 reduced the inspection threshold. SB 72-1146 suggested an inspection threshold of 10,500 CSN or within 500 cycles from the date of issues of the SB. Additionally, to cover all GE90 model thrust ratings, SB 72-1151 recommended to reduce the inspection threshold from 10,500 CSN to 9,000 CSN and kept the requirement for the inspection to be performed with 500 cycles from issuance. Thirdly, SB 72-1151 recommended a repetitive inspection within every 500 cycles until the ECI was performed as opposed to SB 72-1146 which only recommended a one-time on-wing UTI inspection.

On June 24, 2016, the FAA released AD 2016-13-05 with an effective date of July 29, 2016 that mandated inspection of all ‘M80’ and a select number of ‘M90’ HPC stage 8-10 spools. The AD required an UTI or ECI of the stage 8 aft web upper face after reaching 8,000 CSN but before exceeding 9,000 CSN, or within 500 cycles in service after the effective date of the AD, whichever occurs later. Although AD 2016-13-05 did not mention SB 72-1146 or SB 72-1151 specifically, except for the repetitive inspection called out in SB 72-1151, the AD incorporated the information provided in both. Although SBs 72-1145 and 72-1146, and ADs 2015-27-01 and 2016-13-05 provide for a one time inspection of all ‘M80’ and limited population of ‘M90’ spools, all ‘M80’ and ‘M90’ spools will be subjected to repeated shop inspections at the module, rotor, or piece part level in accordance with the engine manual inspections called out in Section 8.2 *Engine Manual Changes* of this report.

During the metallurgical examination of the event HPC stage 8-10 spool, visual examination of the aft surface of the stage 8 disk outer web revealed lower than expected shot peen coverage. Examination of other ‘M80’ spools found the same condition. Although GE concluded that the lower than expected shotpeen on the aft surface of the stage 8 disk was not sufficient to be the root cause for the fatigue crack initiation and eventual fracture of the spool, GE issued SB 72-1149 titled *Engine - Compressor Rotor Assembly (72-31-00) - HPC Stage 8-10 Spool Web and Spacer Arm Shotpeen Repair* on March 29, 2016 to provide a one-time improved shot peen repair for serviceable ‘M80’ and ‘M90’ HPC stage 8-10 spools (all part numbers) to address possible shoot peening inconsistencies. According to GE, since this is a one-time shot peen repair, this repair will not be incorporated into the EM. In addition, GE modified the print drawing for the ‘M90’ spool to include a check for the intensity of the shot peen intensity in the outer web areas. This change took effect in February 2016.

9.0 FLIGHT DATA RECORDER AND EVENT TIME LINE

The flight data recorder (FDR), the Quick Access Recorder (QAR)[23] and the electronic engine control (EEC) were downloaded and tabular data along with various plots were created. A sequence of events timeline was created based on data from the FDR and QAR, and is provided in **Table 3**. It should be noted that the parameters are sampled at different rates based on the need for the fidelity of that data and are recorded at different times. For example, on the FDR the groundspeed, fan and low rotor speed (N1), engine fail warning, HPSOV position, fuel spar valve, TLA, and master warning are all sampled every second; engine fire warning, engine fuel lever cutoff, and high rotor speed (N2) are sampled every 2 seconds; the fire suppression bottle pressure (normal/low) and GPS data points is sampled every 4 seconds; longitudinal acceleration is sampled 4 times a second. Also, each second is partitioned into multiple fractions of a second so that even if two parameters are recorded in that same whole second, the order that the parameters were recorded may be chronologically different. Due to variations in sample rates and recording times, it was not possible to align all the parameters perfectly on a common time, instead the information is provided in chronological order and represents the best estimate of the occurrence.

Table 3: Incident Timelines	**NOTE -** After takeoff, all data references are for the No. 1 engine	
Estimated Elapsed Time (hour:minutes:seconds)	Event Description	Comments
T = -00:22	Both thrust levers advanced for takeoff (referred to as thrust lever angle (TLA))	Engine 1 (left) – TLA moves from 40.4° to 45.7° and continues to maximum value of 78.4° Engine 2 (right) – TLA moves from 40.1° to 47.8° and continues to maximum value of 78.4
T = -00:21	Engine 1 & 2 N2 starts to increase On QAR data, N2 is sampled 1 per second instead of every 2 seconds on FDR. Information to right is FDR data and N2 starts to increase at 97769.43 & 97770.43 (QAR)	Engine 1 – N2 moves from 82.88% to 85.63% and continues to climb Engine 2 – N2 moves from 82.25% to 86.5% and continues to climb
T = -00:21	Engine 1 & 2 N1 starts to increase On QAR data, N1 is sampled 1 per second as in the FDR. Information to right is FDR data and N1 starts to increase at 97770.53 & 97770.53 (QAR)	Engine 1 – N1 moves from 33.88% to 35.88% and continues to climb Engine 2 – N1 moves from 33.38% to 36.5% and continues to climb
T = -00:21	Airplane takeoff roll starts	Longitudinal acceleration starts to climb – moves from 0.006g to 0.012g and continues to accelerate then hold until the engine event
T = -00:20	Airplane takeoff groundspeed comes alive	Groundspeed starts to increase
T = -00:12	Engine 2 N2 reached value for takeoff	112%
T = -00:11	Engine 2 TLA reaches takeoff setting (maximum value)	78.4°

[23] Quick Access Recorder data is a copy of the FDR and thus records the same data. It allows a quick and easy recovery of the raw data recorded in FDR. Only raw data is used for the event analysis.

T = -00:11	Engine 1 N2 reached value for takeoff	112%
T = -00:11	Engine 1 TLA reaches takeoff setting (maximum value)	78.4°
T = -00:10	Engine 1 N1 reached value for takeoff	99%
T = -00:09	Engine 2 N1 reached value for takeoff	99%
T = -00:06	Engine 1 failure event	Longitudinal acceleration starts to decrease from 0.24 – 0.166
T = -00:05	Engine 1 fuel cutoff valve - **CLOSED**	Wires to the HMU were found severed This fuel cutoff valve is the high pressure shutoff valve located in the engine HMU
T = -00:05	Engine 1 warning - **ON**	About 70 knots N2 speed about 75%
T = -00:05	Burner pressure drops dramatically	Decreased from 460 psia to 181 psia
T = -00.05	Engine 1 TLA to idle	TLA moves from 78.4° to 33.4°
T = -00:04	Master warning - **ON**	
T = -00:04	Maximum groundspeed	75 knots
T = -00:04	Engine 2 TLA to idle	TLA moves from 78.4° to 33.8°
T = -00:04	Airplane starts to decelerate	Longitudinal acceleration moves from positive to negative indicating braking
T = -00.03	Brake pressure increases followed by groundspeed decrease	Left main - moves from 80 psig to 3008 psig Right main - moves from 48 psig to 2976 psig
T = 00:00	**Engine 1 fire warning - ON**	**N2 speed is about 28.75% and decreasing**
T = +00:03	Master warning - **OFF**	
T = +00:09	Engine 1 overheat caution - **ON**	
T = +00:09	Airplane stopped	Groundspeed to 0
T = +00:09	Parking brake set	
T = +00:19	Auto throttle - **DISENGAGED**	
T = +00:22	Engine 1 fuel cutoff lever – **IN CUTOFF**	This is on the pedestal in the cockpit - First action to address fire
T = +00:22	Engine 1 fuel spar valve - **CLOSED**	
T = +00:22	Engine 1 fail warning - **OFF**	
T = +00:25	Air supply control system (ASCS) HPSOV - **CLO**	This is bleed valve controlled closed by the fire handle
T = +00:30	Engine fire bottle 1 discharge pressure - **LOW**	
T = +00:42	Engine 1 fire warning - **OFF**	
T = +00:45	Engine fire bottle 2 discharge pressure - **LOW**	
T = +01:15	APU speed begins to increase	
T = +01:34	All entry doors – **NOT CLOSED, LATCHED AND LOCKED**	
T = +01:52	APU running discrete is set (APU RPM ~ 100%)	
T = +02:09	Forward Cargo Smoke Warning - **ON**	Related to cargo smoke
T = +02:08	Engine 2 fuel cutoff lever - **IN CUTOFF**	
T = +02:10	Engine 2 fuel spar valve - **CLOSED**	
T = +02:11	Engine 2 fuel cutoff valve - **CLOSED**	Engine HPSOV closing
T = +02:12	Forward cargo smoke warning - **OFF**	Related to cargo smoke
T = +02:19	Last APU speed signal (APU ~ 100%)	

13sec 22sec 30sec 130sec

Jean-Pierre Scarfo
Aerospace Engineer
Powerplant Lead