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Office of Aviation Safety

Washington, DC 20594



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POWERPLANTS

Group Chair's Factual Report

May 30, 2025

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TABLE OF ACRONYMS

ACMMB	Abbreviated Component Maintenance Manual Book
AGB	Accessory Gearbox
ALF	Aft Looking Forward
ATP	Acceptance Test Procedure
BEA	Bureau d'Enquêtes et d'Analyses pour la Sécurité de l'Aviation Civile
CAD	Cockpit Advisory Display
CDP	Compressor Discharge Pressure
CDS	Cockpit Display System
CMM	Component Maintenance Manual
CSS	Control System Software
CT	Computed Tomography
DECU	Digital Engine Control Unit
ECU	Engine Control Unit
EDS	Electron Dispersive Spectroscopy
EECU	Electronic Engine Control Unit
EEPROM	Electrically Erasable Programmable Read-Only Memory
ESHP	Equivalent Shaft Horsepower
ESN	Engine Serial Number
EV	Electro Valve
FAA	Federal Aviation Administration
FADEC	Full Authority Digital Engine Control
FCU	Fuel Control Unit
FP	Fuel Pump
FWD	Forward
HCF	High Cycle Fatigue
HMU	Hydromechanical Unit
HP	High Pressure
HPT	High Pressure Turbine
HSI	Hot Section Inspection
HV	Vickers Pyramid Number
ID	Inner Diameter
IGS	International Governor Services
LCF	Low Cycle Fatigue
LH	Left Hand
LLP	Life Limited Part
MCD	Magnetic Chip Detector
NTSB	National Transportation Safety Board
NGV	Nozzle Guide Vane
OEI	One Engine Inoperative

OLL	On Line Loader
OSS	Operating System Software
P&WC	Pratt & Whitney Canada
PG	Propeller Governor
PPC	Propeller Pitch Control
PT	Power Turbine
RGB	Reduction Gearbox
RH	Right Hand
SafranHE	Safran Helicopter Engines
SEM	Scanning Electron Microscope
SHP	Shaft Horsepower
SN	Serial Number
SRZ	Secondary Reaction Zone
TBO	Time Between Overhaul
TCDS	Type Certificate Data Sheet
TCP	Topologically Close Packed
TSI	Time Since Inspection
TSN	Time Since New
TSO	Time Since Overhaul

A ACCIDENT

Location: Pompano Beach, FL, USA
Date: August 28, 2023
Time: 0846 EST (Eastern Daylight Time)
Helicopter: Airbus EC135 T1, Registration N109BC

B POWERPLANTS GROUP

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C SUMMARY

On August 28, 2023, about 0846 eastern daylight time, an Airbus Helicopters Deutschland (Airbus) EC135 T1 helicopter, N109BC, crashed into a structure after an uncontrolled descent. Ground witness video showed the helicopter flying straight and level with a trail of grey and black colored smoke emanating from the airframe section aft of the No. 1 (left) engine compartment., and subsequently a partial separation of its tail boom, leading to its uncontrolled descent. The helicopter was operated as a *Title 14 Code of Federal Regulations Part 135* air medical flight.

The on-scene investigation was performed on August 28, 2023, by representatives from the National Transportation Safety Board (NTSB), the Federal Aviation Administration (FAA) and Broward County Sheriff's Department. The helicopter wreckage was recovered and transported to Florida Air Recovery's facility in Fort Pierce, Florida for the initial wreckage exam which was performed August 29 - September 4, 2023.

The initial wreckage exam found both engines were in their normally installed positions and exhibited thermal damage to all external surfaces due to the post-crash fire. The No. 1 engine contained multiple fractured power turbine blades which were visible when looking through the engine exhaust duct. The power turbine blades in the No. 2 engine were present and intact. Both engines were shipped to Safran Helicopter Engines USA in Grand Prairie, Texas for teardown and examination.

The electronic engine control units (EECU) were found within the avionics compartment, both exhibited thermal damage. The housing for the No. 1 EECU was intact but the housing for the No. 2 EECU was partially burned through. The EECUs were removed and shipped to the NTSB's laboratory in Washington D.C. for further examination. The No. 1 EECU was subsequently shipped to the Bureau d'Enquêtes et d'Analyses pour la Sécurité de l'Aviation Civile (BEA) in Paris France for further examination and data recovery. Due to the damage to the No. 2 EECU, the nonvolatile memory chip was removed, a data image was created which was then sent to the BEA for data recovery. The BEA recovered the stored data from both EECUs. The recovered data from the No. 1 EECU contained one snapshot of the engine parameters during the accident flight. The No. 2 EECU contained multiple snapshots of the engine parameters during the crash and post-crash fire sequence.

The accident engines were examined on September 25-29, 2023, at Safran Helicopter Engines USA in Grand Prairie, Texas. In attendance were group members from Safran Helicopters Engines (SafranHE), the BEA, Airbus Helicopters, Metro Aviation, Broward County Sheriff's Department, the FAA, and the NTSB.

The examination of the No. 1 engine found thermal damage to all exterior surfaces and wiring harnesses, all external tubes and tube connections appeared intact

and fully engaged, and all external components were found in their normally installed positions. Disassembly of the engine found five power turbine blades fractured below the blade platform, material loss and hard body impact damage was present throughout the gas generator turbine and power turbine sections of the engine and rotational damage was present throughout the engine.

The examination of the No. 2 engine found thermal damage to all exterior surfaces and wiring harnesses, all external tubes and tube connections appeared intact and fully engaged, and all external components were found in their normally installed positions. Disassembly of the engine found the compressor impeller; gas generator turbine, and power turbine components were all found intact and undamaged.

The hydromechanical fuel control units from the No. 1 and No. 2 engine were further examined at the Safran Helicopter facility in Tarnos, France on April 9-10, 2024. The measured fuel flow of the No. 1 hydromechanical unit, in the as received condition, was approximately 90% of the pump speed consistent with the fuel flow recorded in the No. 1 EECU snapshot data stop. Inspection of the electro-valve showed the valve was in an open position which is consistent with an engine in an operating condition. The measured fuel flow of the No. 2 hydromechanical unit, in the as received condition, was approximately 99% of the pump speed. Inspection of the electro-valve showed the valve was in an open position which is consistent with an engine in an operating condition.

Safran's material/failure analysis laboratory further examined hardware from both the No. 1 and No. 2 engines; evidence of thermal overheating was present on various components in both engines. The No. 1 engine hardware contained evidence of localized heat tinting of various components and localized microstructural transformation within the material of the high-pressure turbine blades and power turbine blades, both above and below the blade platforms.

The Safran laboratory performed fractography and metallography on the five fractured power turbine blades. All presented a similarly shaped area of crack initiation/propagation with characteristics consistent with fatigue, initiation occurred below platform in the blade shank at the corner of the leading-edge side and the pressure side, no material anomalies were found at the initiation sites.

D DETAILS OF THE INVESTIGATION

1.0 Engine Information

1.1 Engine Description

The Arrius 2B1 is a two-spool turboshaft engine comprised of a single stage centrifugal compressor, reverse flow annular combustor, single-stage gas generator

turbine (HPT), and a single-stage free turbine power turbine (PT) (Figure 1). The engine is controlled by an EECU¹/hydromechanical metering unit (HMU). The Arrius 2B1 engine is considered a modular engine, the module 1 (M01) module is comprised of the accessory and reduction gearbox (RGB) and the module 2 (M02) module is comprised of the gas generator and power turbine.

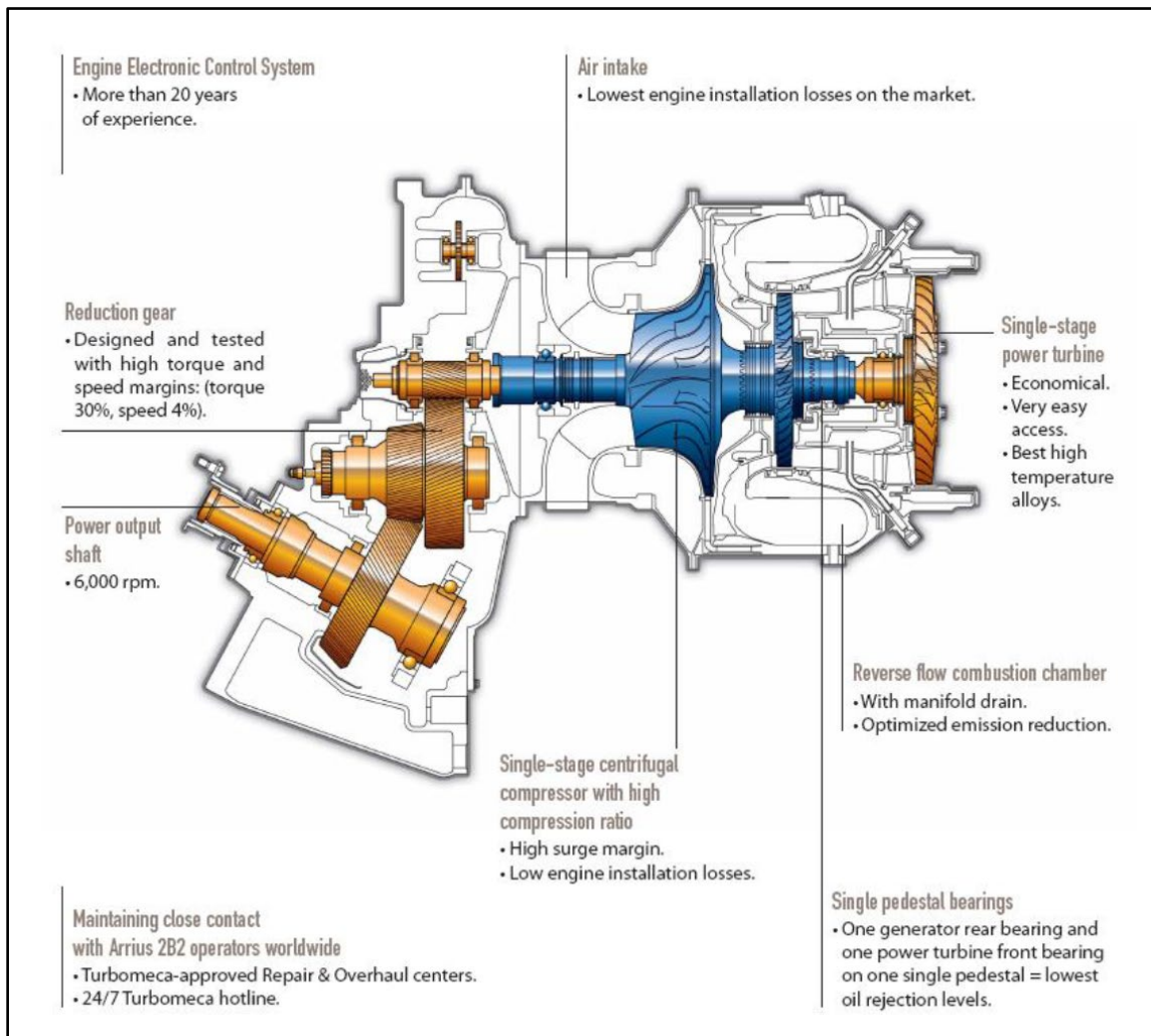


Figure 1. SafranHE Arrius 2B1 - Engine Cross-Section

(Figure Courtesy of SafranHE)

According to the FAA Type Certificate Data Sheet (TCDS) E34NE, revision 13, dated April 22, 2022, the Arrius 2B1 turboshaft engine has maximum continuous shaft horsepower power (SHP) of 547 SHP and a maximum SHP at takeoff (5 minutes) of 577 SHP. All directional references to front and rear; left and right; top and bottom; and clockwise and counterclockwise are made aft looking forward (ALF).

¹ The EECU may also be referred to as a digital engine control unit (DECU), an engine control unit (ECU) or a full authority digital engine control (FADEC).

1.2 Logbooks and Maintenance

According to the engine logbook and SafranHE records as of August 27, 2023 (the day prior to the accident), the No. 1 (left hand) engine serial number (ESN) 30175 and the No. 2 (right hand) ESN 30182 had a Time Since New (TSN²) of 5,403.2 hours and a calculated Time Since Overhaul (TSO) of 2,097.4 hours. The last 100 hour inspection of the helicopter recorded in the engine logbook was performed on May 23, 2023, on both engines, by Broward County Sheriff's Office (BSO), Florida. The last engine overhauls recorded in the engine logbooks were performed on June 15, 2016, for both engines, by Safran Helicopter Engines USA.

The No. 1 engine was last installed on the accident helicopter on September 16, 2020, following removal of the engine for the replacement of M01/reduction gearbox module and replacement of rear bearing. The No. 2 engine was installed the accident helicopter on June 15, 2016, following the scheduled engine overhaul.

A summary of maintenance activities recorded in the logbook for the No. 1 engine between January 2020 and August 27, 2023, is shown below in Table 1. A summary of maintenance activities recorded in the logbook for the No. 2 engine between January 2020 and August 27, 2023, is shown below in Table 2.

Table 1. Maintenance Records – No. 1 ESN 30175

Date	Engine Total Time	Work Performed
August 27, 2023	5,403.2	<i>Day before accident (based on the helicopter flight log)</i>
May 23, 2023	5,327.2	100 hour inspection, oil analysis, power assurance check
July 29, 2022	5,249.3	100, 200, 1000 hour inspection, oil analysis, R/R FCU and preference injector
March 17, 2022	5,249.3	Cycle count audit, corrected values: N1 5329.2, N2 8493.3
January 12, 2022		HMU (SN 234B) repaired/reinstalled by Safran USA
July 29, 2021	5,198.2	100 hour inspection, oil analysis, power assurance check
April 21, 2021	5,167.4	500 hour, borescope HP turbine, compressor, and diffuser
December 23, 2020		HMU overhauled by Safran Helicopter Engines USA, AD 2020-0033
November 21, 2020	5,117.1	R/R LH and RH fuel manifold
September 16, 2020	5,105.9	M01/Gearbox and PT/rear bearing replaced
July 27, 2020	5,103.6	400 hour inspection, preference injector valve inspection
July 25, 2020	5,102.9	100 hour inspection, oil analysis, power assurance check
July 20, 2020	5,095	200 hour inspection
July 12, 2020	5,089.1	AD 2018-12-03 inspected preferential injector
April 24, 2020	5,015.2	100 hour inspection, oil analysis, power assurance check

² TSN is synonymous with Engine Total Time (ETT)

Table 2. Maintenance Records – No. 2 ESN 30182

Date	Engine Total Time	Work Performed
August 27, 2023	5403.2	<i>Day before accident (based on the helicopter flight log)</i>
May 23, 2023	5327.2	100 hour inspection, oil analysis, power assurance check
July 29, 2022	5249.3	100, 200, 1000 hour inspection, oil analysis, R/R preferential injector
July 29, 2021	5198.2	100 hour inspection, oil analysis, power assurance check
April 22, 2021	5168.3	500 hour, borescope HP turbine, compressor, and diffuser
February 27, 2021	5132.3	Engine oil and oil filter change
July 27, 2020	5103.6	400 hour inspection, injector valve inspection
July 25, 2020	5102.9	100 hour inspection, oil analysis, power assurance check
20-Jul-20	5095.0	200 hour inspection
July 12, 2020	5089.1	AD 2018-12-03 inspected preferential injector
April 24, 2020	5015.2	100 hour inspection, oil analysis, power assurance check

2.0 On Scene & Initial Powerplant Wreckage Exam

The on-scene investigation and accident site documentation occurred August 28, 2023, with group members from Broward County Sheriff's Department, the FAA, and the NTSB.

The helicopter wreckage was recovered and transported to Florida Air Recovery in Fort Pierce, Florida for the initial wreckage exam, which occurred August 29 – September 4, 2023. The initial exam included group members from the SafranHE, Airbus Helicopters, Metro Aviation, Broward County Sheriff's Department, the FAA, and the NTSB.

The following engine related information was documented during the on-scene investigation and the initial wreckage exam by on-site NTSB personnel.

2.1 No. 1 Engine (S/N 30175)

The No. 1 engine was present in its normally installed, left hand position No. 1 location (Figure 2). The engine exterior exhibited thermal damage. The exhaust support (with struts) and exhaust pipe were deformed to the right and forward. The exhaust pipe had one puncture that was seen on the lower-outboard (6:00 to 9:00 o'clock) quadrant. The puncture measured about 1 inch long. The compressor inlet struts, and adjacent surfaces had a dark grey to black colored material adhered to it, primarily concentrated on the 12:00 o'clock strut. The compressor impeller was able to be manually rotated. Manual rotation of the compressor impeller resulted in

movement without evidence of binding, but a complete rotation was not attempted to prevent further damage to the gas generator.



Figure 2. No. 1 Engine

Four PT blades were missing from their root end and a fifth PT blade had fractured chordwise and its outboard half was missing (Figure 3). The remaining visible PT blades exhibited varying degrees of damage on the outboard end of the blades. The visible PT blades exhibited a brown-colored staining near their root. Two of the visible stators (immediately upstream of the PT) exhibited missing material near the outboard end. The air intake screens (referred to as guards) were found installed but damaged, the carbon fiber air intake lips that support the air intake screens were present but damaged, and the braze material on the air intake screens appeared melted (Figure 3).



Figure 3. No. 1 Engine - View of Power Turbine Blades (L) and Inlet Screen (R)

The external surfaces of the PT containment ring showed no evidence of breaches or punctures. However, the 3:00 o'clock strut immediately aft of the PT had a puncture through its upper surface. The fuel lines to the preferential injector and the left and right fuel injector manifold remained installed and did not appear fractured. The oil scavenge line for the rear bearing remained installed but the scavenge oil chip detector housing was not found and likely consumed by fire. The remainder of the scavenge oil lines were present.

The fuel lines were present and did not appear fractured. There was no evidence of the oil and fuel connections being loose or improperly installed.

The oil filler cap remained installed on the gearbox. The oil and fuel filters remained installed; their pre-blockage pop-up indicators were present but could not be seen through the glass covers. The glass cover for the oil filter pre-blockage pop-up indicator was removed and the pop-up indicator was adhered to the glass cover. The glass cover for the fuel filter pre-blockage poppet indicator was removed and the pop-up indicator was extended. The filters were not removed at this time.

The engine was found disconnected from the main transmission gearbox at the connection point between the power output shaft and the main gearbox. The power drive shaft was not able to be manually rotated. The power drive shaft splines did not exhibit any evidence of anomalous wear or fracture.

The manual mode throttle³ (at the engine) was found at approximately the 12:00 o'clock position, consistent with the neutral position. The emergency fuel shutoff valve, installed below the engine deck with the fuel lines still connected, was determined to be in the open position by inserting a flexible plastic rod through one of the fuel lines connected to the shutoff valve assembly and the plastic rod came out of the fuel line on the other end of the shutoff valve assembly. The fuel shutoff valve assembly was removed for future examination and inspection using computed tomography (CT).

2.2 No. 2 Engine (S/N 30182)

The No. 2 engine was present in its normally installed, right hand No. 2 location (Figure 4). The engine exterior exhibited thermal damage. The engine was found disconnected from the main transmission gearbox at the connection point between the power output shaft and the main gearbox.

The compressor inlet struts were covered in soot and had a dark grey to black colored material adhered to it, primarily concentrated on the 12:00 to 3:00 o'clock area. Manual rotation of the compressor impeller resulted in movement without evidence of binding, but a complete rotation was not attempted to prevent further damage to the gas generator. The oil filler cap remained installed. The oil and fuel filters remained installed. All fuel and oil lines were present and did not exhibit any evidence of fracture. There was no evidence of the oil and fuel connections being loose or improperly installed.

³ The manual mode throttle is an Airbus supplied component.



Figure 4. No. 2 Engine

The PT blades were visible through the tail pipe, all PT blades were present and appeared intact with no abnormal discoloration (Figure 5). The air intake screens were found installed but damaged, the carbon fiber air intake lips that support the air intake screens were present but damaged, and the braze material on the air intake screens appeared melted (Figure 5). The lower portion of the emergency fuel shutoff valve was not present in its normally installed location, but the fuel line remained connected to the upper fitting (on the engine deck) of the shutoff valve. The No. 2 engine manual mode throttle (at the engine) was also found at approximately the 12:00 o'clock position, consistent with the neutral position.



Figure 5. No. 2 Engine - View of Power Turbine Blades (L) and Inlet Screen (R)

2.3 Electronic Engine Control Units

The No. 1 engine EECU and the No. 2 engine EECUs were found within the avionics compartment of the helicopter, both exhibited thermal damage and were discolored with soot (Figure 6). The No. 2 EECU housing was not intact and partially consumed, making internal components visible with the edges of the hole in the housing bent inward. The data plate for the No. 1 EECU showed part number (P/N) 70EMF01290 and serial number (S/N) 8ALD0295CE, and the data plate for the No. 2 EECU showed P/N 70EMF01290 and S/N 8ALD0258CE.

The serial link connectors for the P201 and P202 wiring harnesses were still attached to both EECU housings and appeared undamaged. The canon plugs for the left hand (LH) No. 1 EECU were detached (unplugged) by hand, allowing the No. 1 EECU to be removed from the helicopter. The canon plugs for the right hand (RH) No. 2 EECU were not able to be removed by hand, so the wiring harnesses were cut and the No. 2 EECU was removed with the canon plugs still attached.

The EECUs were retained for further examination (Figure 7 and Figure 8).

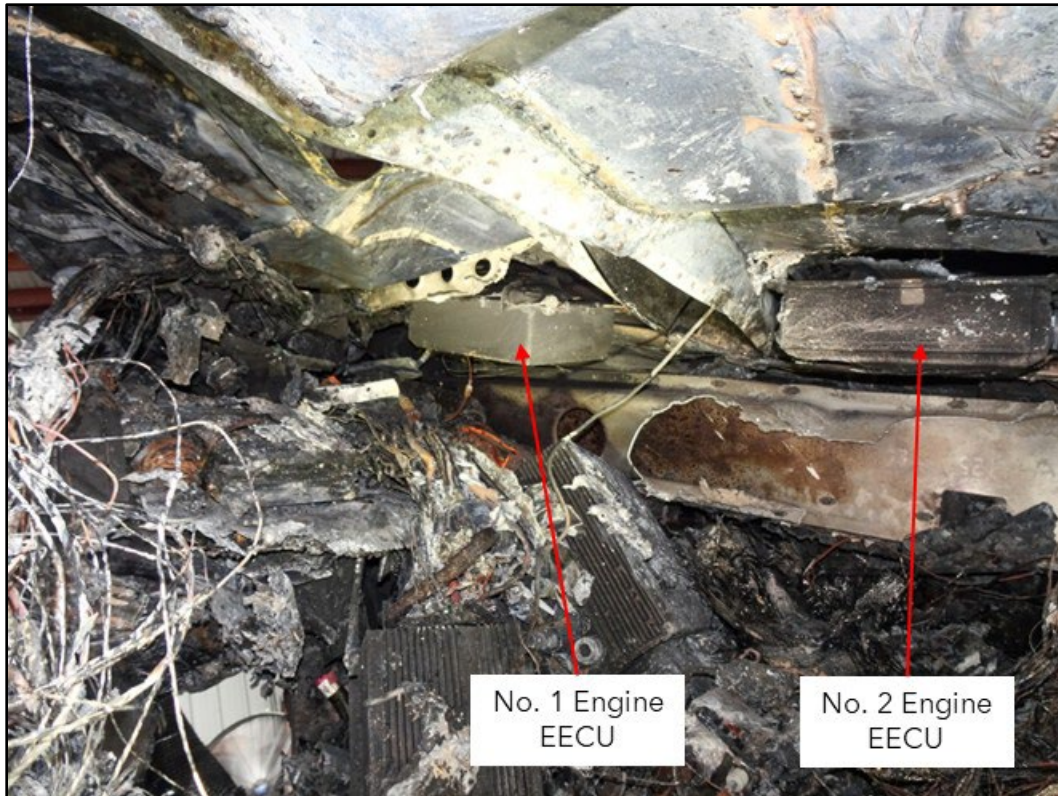


Figure 6. View of EECUs in Aft Compartment



Figure 7. No. 1 Engine EECU - Removed



Figure 8. No. 2 Engine EECU - Removed

3.0 Engine Examinations

3.1 ESN 30175 Examination

The engines were shipped to Safran Helicopter Engines USA in Grand Prairie, Texas for the examination of both engines. Group members from the SafranHE, the BEA, Airbus Helicopters, Metro Aviation, Broward County Sheriff's Department, the FAA, and the NTSB convened on September 25, 2023, and the examination was completed on September 29, 2023.

3.1.1 As Received

The No. 1 and No. 2 engines and engine logbooks were secured in a wood shipping crate in the Safran Helicopter Engines Training Center when the investigation team arrived (Figure 9). The No. 1 engine bottom mounting bracket was not shipped with the engine, so the No. 2 engine bottom bracket was removed and installed on the No. 1 engine to facilitate installation in a work stand (Figure 10). The ESN on the engine data plate was verified as 30175 (Figure 11).



Figure 9. Engine Shipping Crate, as Received

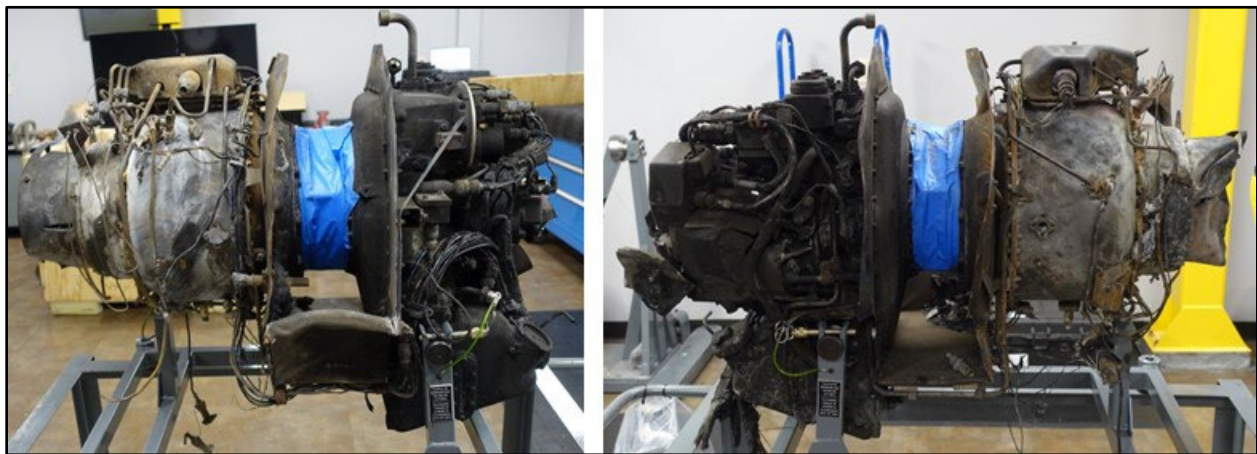


Figure 10. ESN 30175 As Received - Left Side (L) Right Side (R)



Figure 11. ESN 30175 Data Plate

All external engine surfaces were sooted/discolored. Multiple engine wire harnesses exhibited thermal damage and were charred and brittle. The engine inlet was coated with a dark residue that was bubbled from the 9:00 to 6:00 o'clock positions (upper right quadrant, ALF) (Figure 12). Samples of the dark residue on the inlets were collected for analysis at SafranHE and the inlet case was retained for shipment to the NTSB Materials Laboratory in Washington, DC for analysis. The aft intake firewall had 360-degree deformation. Resolidified metal slag was adhered to external lines/wires on the turbine case at the 9:00 o'clock position⁴.

The turbine exhaust case had crushing damage from the 5:00 to 12:00 o'clock positions. There was a penetration in the turbine exhaust case at the 3:00 o'clock strut aft of the PT rotor blades (Figure 13). The material on the outside of the turbine exhaust case penetration was deformed radially inward toward the engine.

The power turbine gas temperature (T4.5) confirmation box and rear turbine bearing magnetic chip detector housing, normally mounted on the aft flange of the turbine case near the 6:00 o'clock position, were thermally consumed and missing. The M02 data plate was separated from the turbine case and missing.

⁴ According to the airworthiness group, the No. 1 engine came to rest on its left side (9:00 o'clock) during the accident sequence.



Figure 12. Engine Inlet Residue

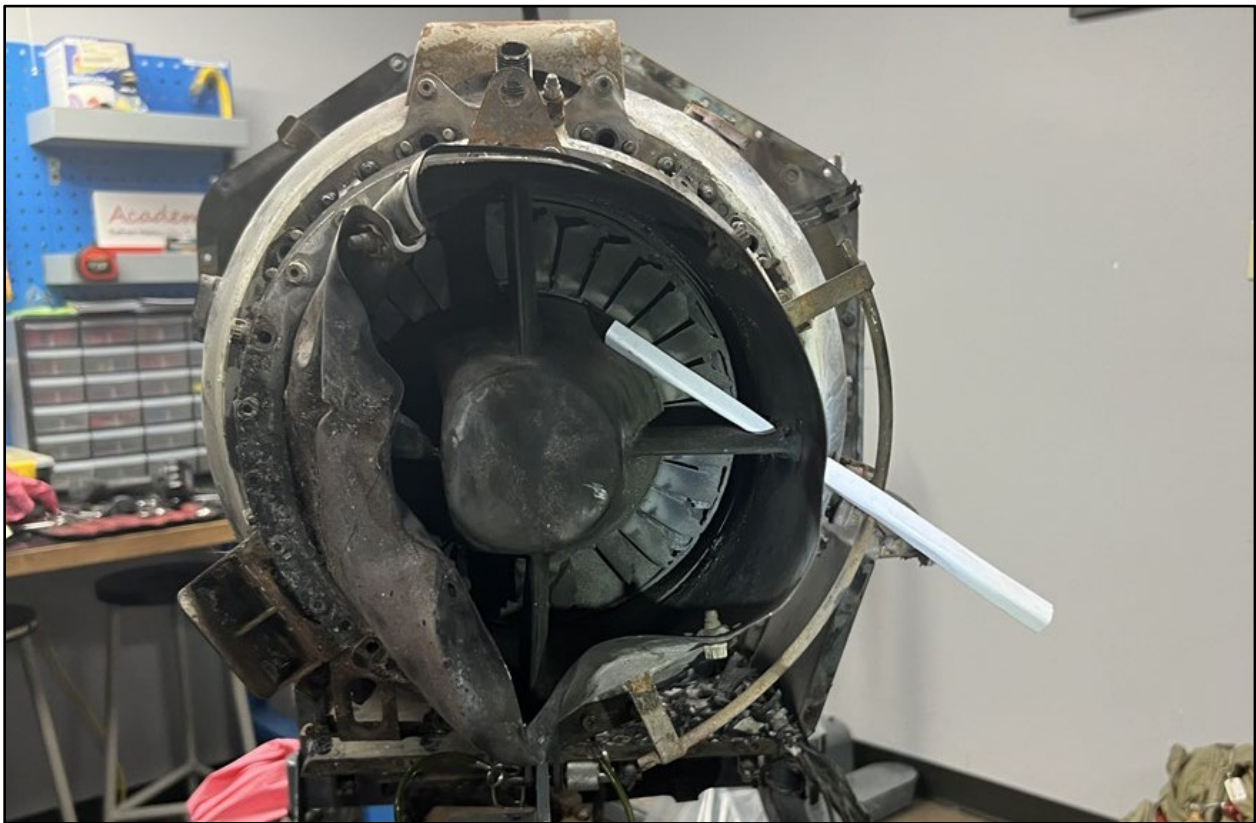


Figure 13. Turbine Exhaust Case Crushing Damage and Turbine Case Hole

The adjusted fuel valve assembly cover was secured on the turbine case (12:00 o'clock) and there was no evidence of fuel line chafing due to contact with the cover. The stirrup fitting that secures the preference injector pipe assembly and the right injector half manifold assembly was installed correctly as outlined in SafranHE Service Letter No. 3052/23/ARS2. After the adjusted fuel valve assembly cover was removed, dark discoloration was visible on all assembly external surfaces, the fire protection coating on the stop electro-valve was thermally damaged (Figure 14). All other fuel lines appeared intact and the joints appeared properly engaged.

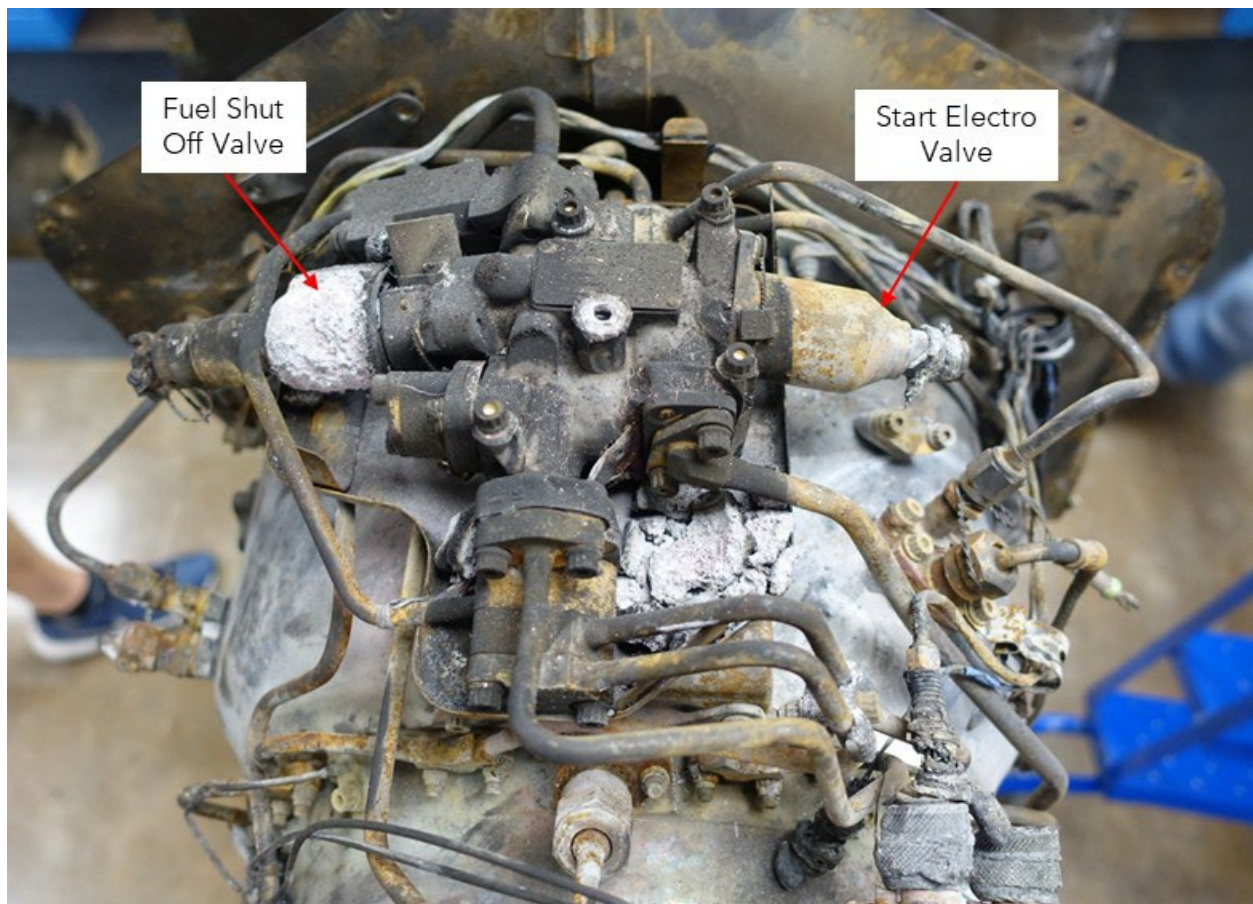


Figure 14. Fuel Adjusted Valve Assembly

Residue was adhered to the oil filter housing on the accessory gearbox (AGB), and a sample collected for analysis at SafranHE. The fuel filter was dry, and no debris/obstructions were visible between the filter pleats. There was residual oil in the oil filter and no debris/obstructions were visible (Figure 15). The M01 electrical magnetic chip detector (MCD) plug had fine magnetic debris accumulated on the magnet that appeared to bridge the gap⁵ (Figure 16). A sample of the chip detector

⁵ Debris that spans the MCD gap will result in a chip light in the cockpit when the system is functioning per design.

debris was collected for analysis at SafranHE. The oil reservoir⁶ was intact with no evidence of tank breaches; the remaining oil was drained during disassembly and found to be approximately 1.5 to 2 liters of oil. The oil did not smell acid/abnormal and appeared normal for service run engines. Oil samples were collected for analysis at SafranHE.



Figure 15. Fuel and Oil Filters **Fuel Filter Damage During Examination



Figure 16. AGB Magnetic Chip Detector Plug Debris

The PT rotor was rotated by hand and there was concurrent rotation of the power drive shaft, consistent with PT shaft continuity. The starter generator pad rotated with resistance and grinding noises were audible but there was concurrent rotation of the centrifugal compressor consistent with gas generator shaft continuity.

⁶ The oil reservoir for the Arrius 2B2 engine has a maximum capacity of 1.29 US gallons (4.9 liters).

All external air, fuel and oil lines were removed, visually examined, and photographed. There was no visible evidence of breaks, cracks, or obstructions on any of the external lines.

The N1 (gas generator rotational speed) and N2 (power turbine rotational speed) sensors were found intact, the connecting wires were thermally damaged but appeared still connected to the engine wiring harness and were otherwise unremarkable. The N1C speed sensor (lower sensor) had a fine metal sliver on the sensor magnet.

The front (cold zone) fire detector, located forward side of the M01 module at the 12:00 o'clock position, was found in the proper position but twisted and thermally damaged. The rear (hot zone) fire detector, located at the 6:00 o'clock position on the turbine case, was found connected in the proper position but thermally damaged. The torque sensor, located next to the output shaft flange, was found in the proper position but the connecting wires were found thermally damaged and broken.

3.1.2 Centrifugal Compressor and Diffuser Assembly

The centrifugal compressor impeller had dark discoloration on all blade surfaces. There were indications of contact rub and material accumulation adhered to the impeller blade leading edges that could be removed with a knife (Figure 17). There was corresponding, heavy, and uniform, 360-degree rotational scoring/rub and material loss on the impeller cover, consistent with axial impeller contact. The impeller cover also appeared discolored with visible dark areas where the impeller blades came to rest and white spots observed 360-degrees around the cover surface (Figure 17). The gas generator front thrust (ball) bearing was oil wetted and rotated smoothly.

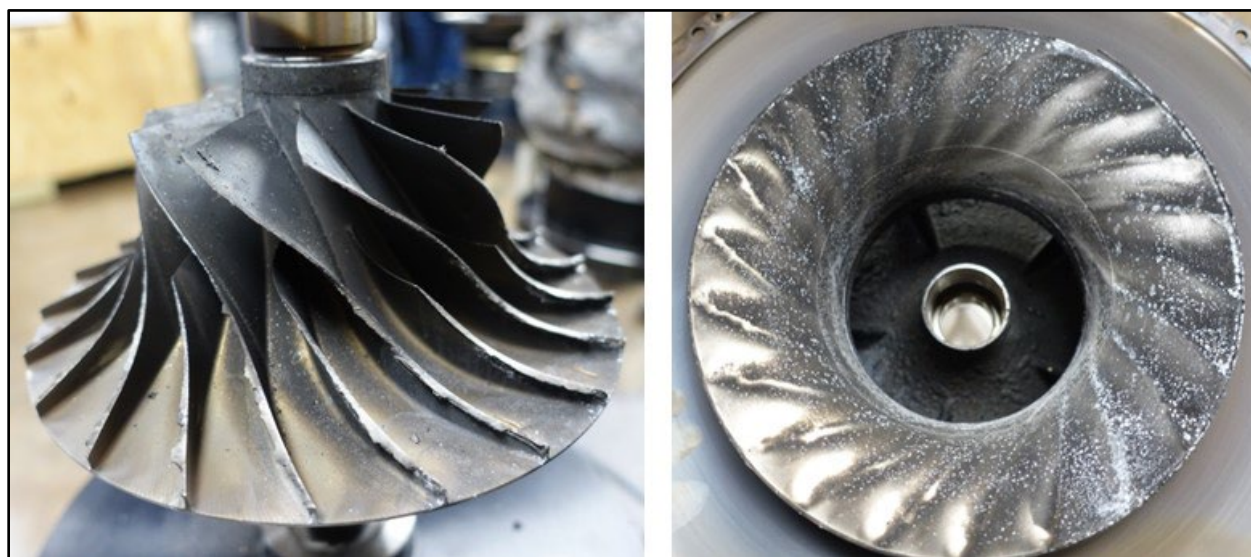


Figure 17. Centrifugal Impeller (l), Impeller Cover Material Accumulation (R)

The gas generator turbine spline nut was removed with hand tools and was well below normal torque⁷ that would be needed to stretch the gas generator shaft. The aft impeller curvic coupling of the impeller (impeller is made of titanium based alloy) exhibited fretting wear and material displacement on the curvic teeth contact surfaces. The mating curvic coupling teeth on the front of the gas generator turbine disk (the gas producer turbine disk is made of a nickel-based alloy) were intact (Figure 18).

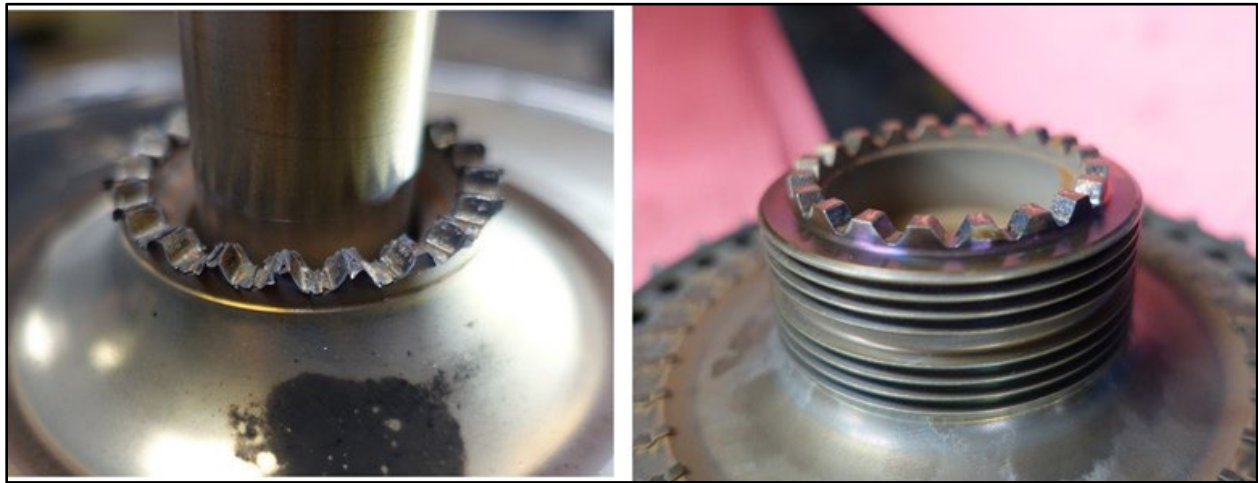


Figure 18. Impeller Curvic Coupling (L), Gas Generator Turbine Disk Forward Curvic Coupling(R)

The compressor diffuser vane leading edges appeared rough with non-uniform wear. According to SafranHE, vane erosion is not typical of similar service run engines. Discoloration and white spots were also observed around the entire 360-degree circumference of the diffuser vane inlet (Figure 19).

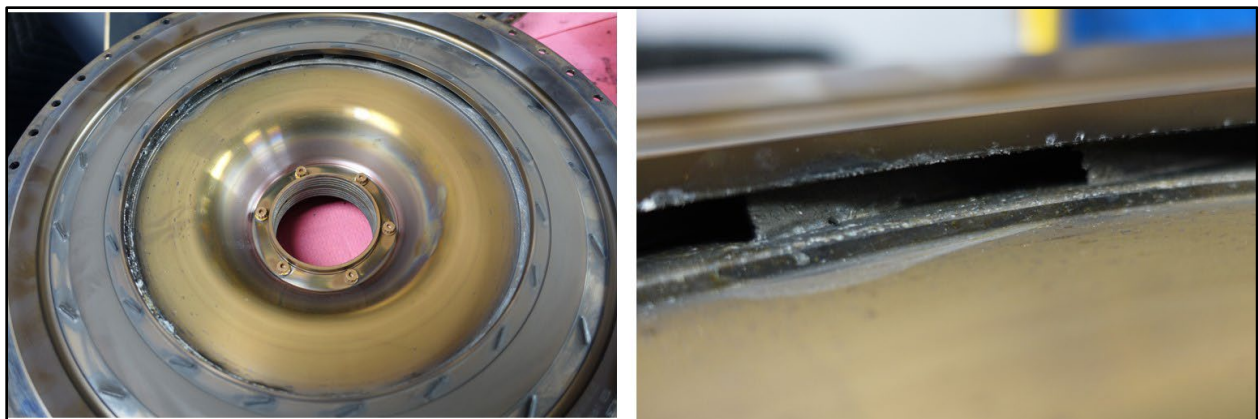


Figure 19. Diffuser (L), Close Up of Diffuser Vane Leading Edge Erosion (R)

⁷ During engine disassembly 14.6kPA is usually required to stretch the gas generator turbine shaft to facilitate removal of the spline nut.

3.1.3 Combustor

The combustion chamber was intact with no evidence of burn through or uneven wear, and the flame tubes and pre-vaporization tubes appeared undamaged (Figure 20). Black spots were observed under the fuel pre-vaporization tubes in the combustor that were typical for service run engines according to SafranHE. A small amount of fine ash like debris was present in the chamber. The left and right main fuel manifold injectors and the preference injector were intact with dark black discoloration on multiple injector tips (Figure 21).



Figure 20. Combustion Chamber

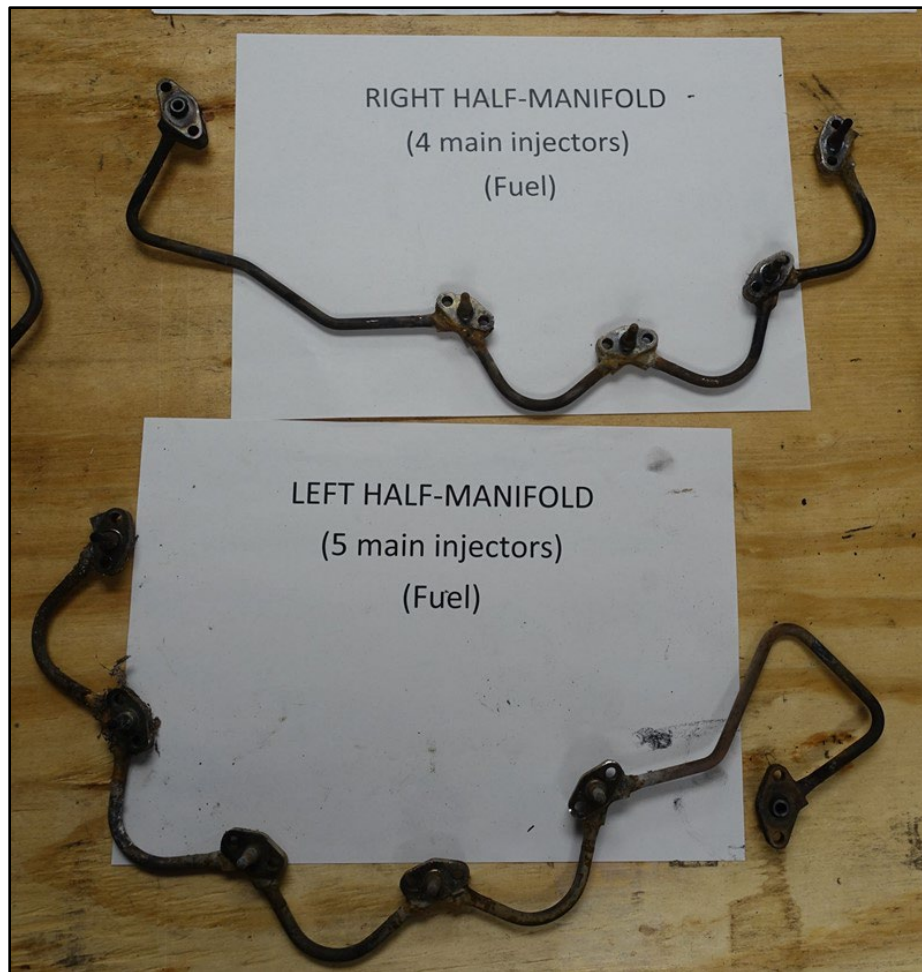


Figure 21. Fuel Injector Manifolds

3.1.4 Gas Generator Turbine

The gas generator turbine nozzle guide vanes (NGV) had metal spray on the pressure and suction vane surfaces. The gas generator turbine NGVs exhibited trailing edge damage and material loss from the 2:00 to 7:00 o'clock positions (Figure 22). The labyrinth seal in the center bore of the gas generator turbine NGV assembly had deep cuts in the abradable material where the gas generator turbine forward shaft teeth made contact.



Figure 22. Gas Generator Turbine NVGs, Trailing Edge Side

The gas generator turbine blades were all fractured transversely at about mid span. The remaining blade airfoil tip surfaces appeared rough/jagged and thermally distressed. The gas generator turbine blades had white discoloration on the leading-edge surfaces (Figure 23). All gas generator turbine disk posts and blade fir trees appeared undamaged and all blade locks were present.

The gas generator turbine shroud segments were all present and had 360-degree rub. One gas generator turbine shroud segment, located at the 5:00 o'clock position, exhibited missing material (Figure 24).



Figure 23. Gas Generator Turbine, Forward Side (L), Aft Side (R)



Figure 24. Gas Generator Turbine Shroud - Missing Material

The gas generator turbine rear bearing appeared undamaged but had no residual oil and appeared dry. The roller bearing was intact and rotated with minimal resistance (Figure 25).



Figure 25. Gas Generator Turbine Rear Bearing

3.1.5 Power Turbine

All PT disk blades were fractured transversely across the airfoil and various lengths; the fracture surfaces appeared rough/jagged and thermally distressed (Figure 26) with light metal spray observed adhered to the leading and trailing edges airfoils. Five PT disk blades were separated below the blade platform; all five fracture surfaces exhibited a similar thumbnail shaped area, located at the corner of the leading edge and pressure side of the blade shank, that had flat fracture features in comparison to the rough features on the remaining fracture surface (Figure 27).



Figure 26. PT Disk – Forward Side (L), Aft Side (R)

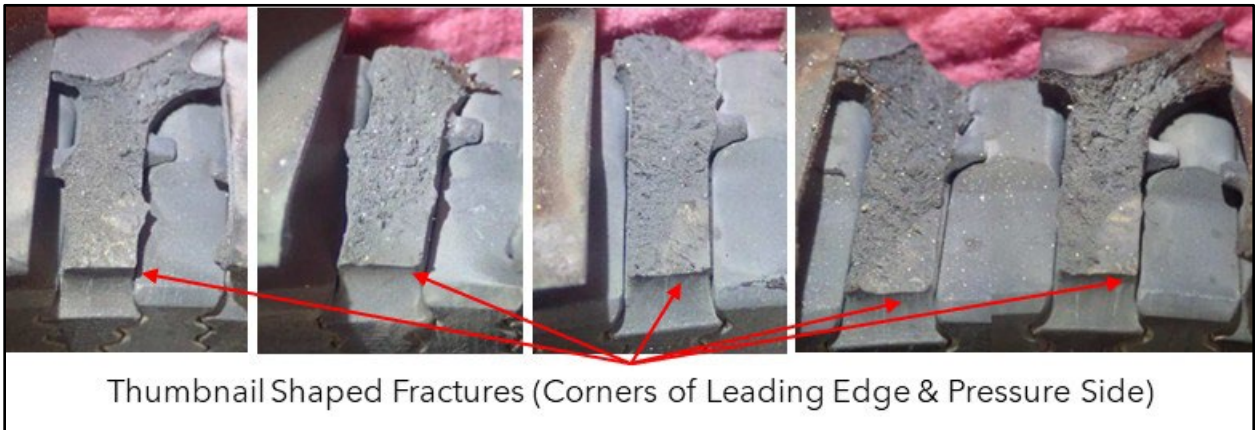


Figure 27. Fractured PT Disk Blades

All the PT NGVs had dark discoloration, with residue and metal spray adhered to all vane surfaces. A single vane was thermally consumed at the 6:00 o'clock position and there was a crack at the adjacent vane inner diameter platform. The trailing edges of all the NGVs exhibited impact and thermal damage 360-degrees around the nozzle assembly. The trailing edge damage and material loss was most severe between the 3:00 and 7:00 o'clock positions (Figure 28).

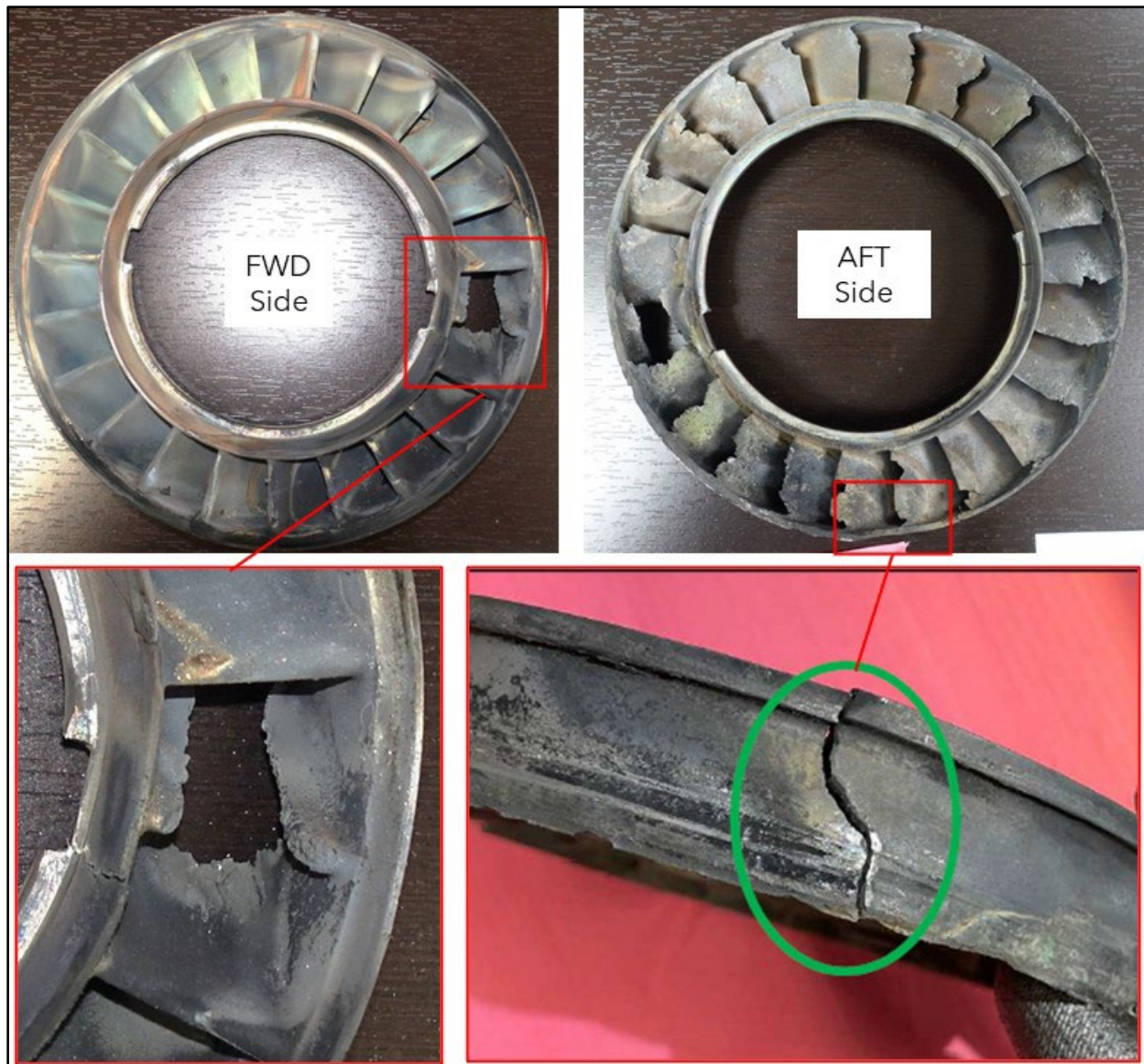


Figure 28. PT NGVs

The PT outer shroud exhibited two areas of impact and thermal damage which made removal during disassembly difficult, and the subsequent removal process induced additional damage/deformation to the outer diameter of the part (Figure 29). The PT containment shield, located outboard of the PT outer shroud, showed evidence of impact damage (Figure 30).

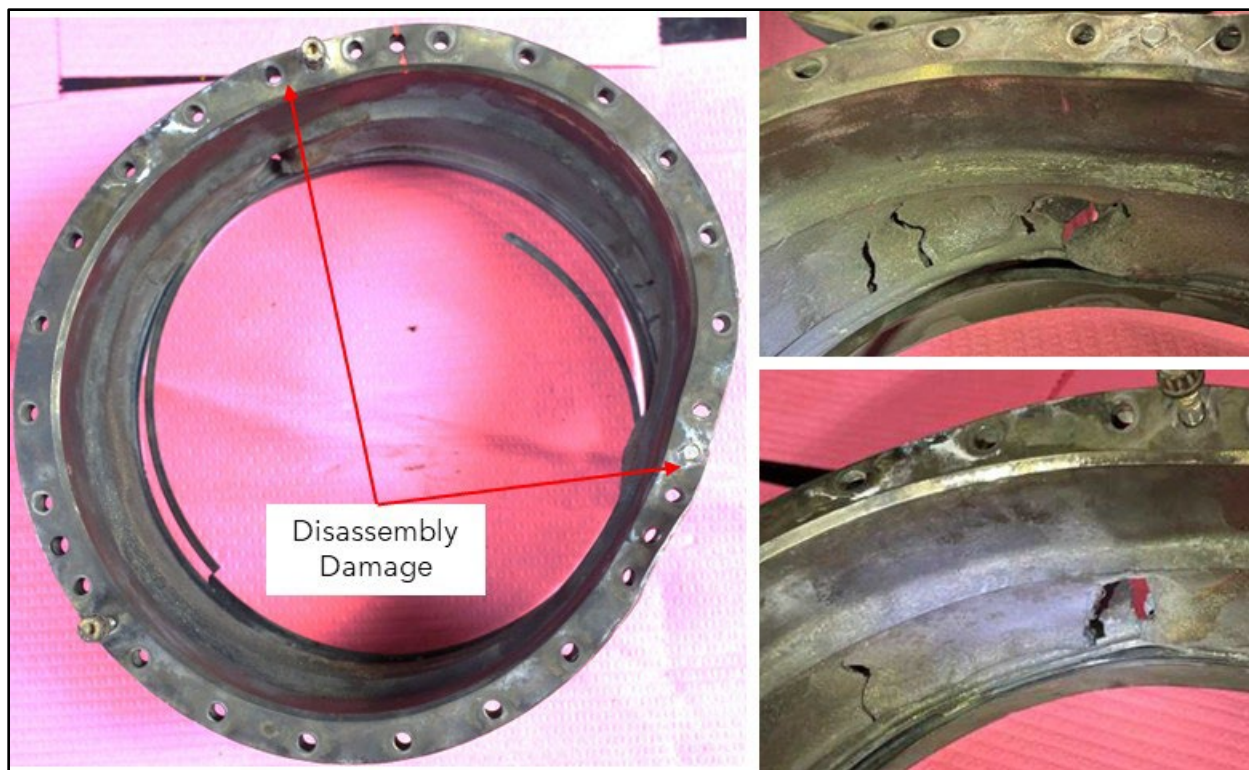


Figure 29. PT Shroud -Damaged During Removal



Figure 30. PT Containment Shield - Impact Mark

The PT thrust (ball) bearing assembly was destroyed. Multiple ball bearing elements were smeared into the bearing cage and additional loose bearing elements were collected in the bottom of the bearing housing (Figure 31). The bearing races also exhibited deformation and smearing. Dark powdered material was recovered from the turbine bearing housing when the ball bearings were collected.



Figure 31. PT Thrust Bearing

The inner support struts located at the 6:00 and 9:00 o'clock positions of the rear bearing support were fractured near the fillet between the strut and the inner ring (Figure 32). The oil supply tube and the oil scavenge tube, located within the 12:00 and 6:00 o'clock struts respectively, were visually undamaged. All four of the outer flange struts of the rear bearing support struts were found fractured (Figure 33) from the outlet diffuser case.

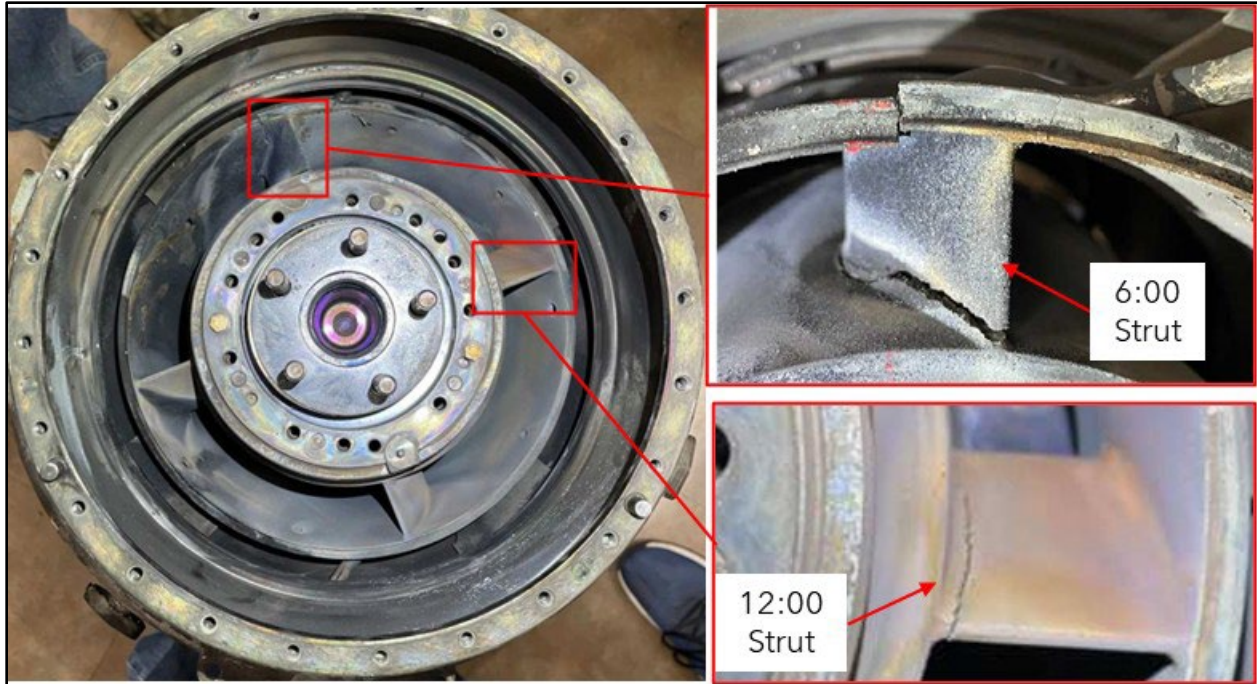


Figure 32. Rear Bearing Support - Forward Side

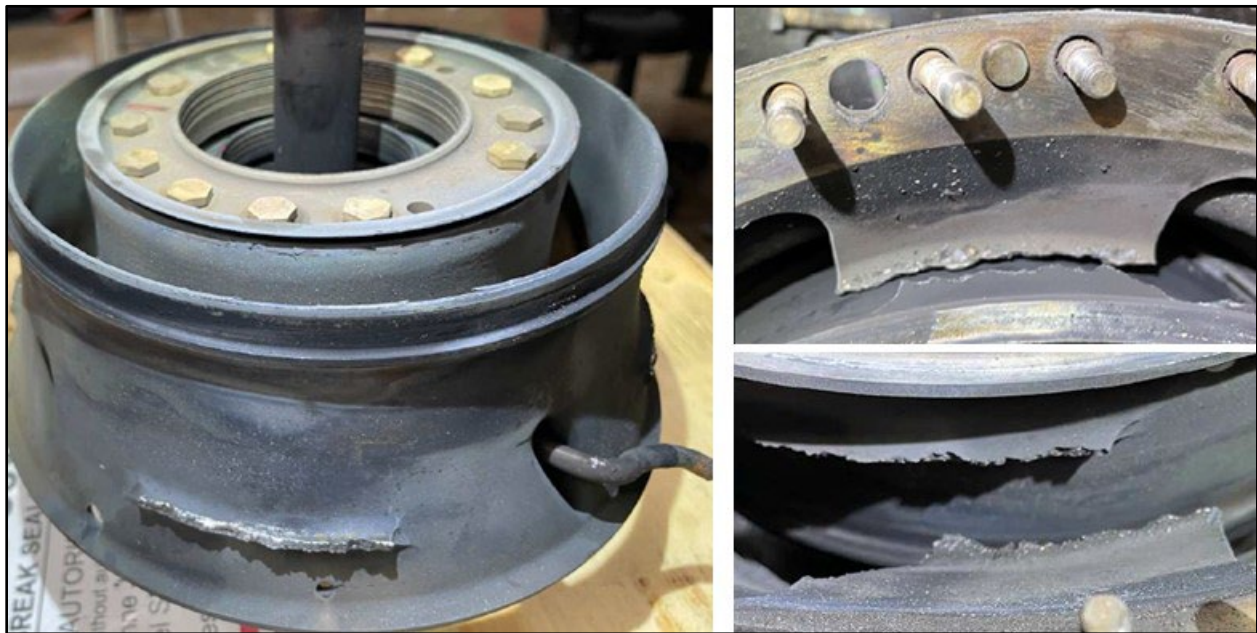


Figure 33. Rear Bearing Support - Aft Side

3.1.6 M01 (Reduction Gearbox) Module

The PT rotor, connected to the power drive shaft, spun freely and the bearings rotated smoothly with corresponding rotation of the M01 module.

The PT shaft, connecting the PT rotor to the M01 module was found undamaged but with a residual coating of soot on the rear of the shaft, the splines were in visually good condition and the remainder of the shaft was otherwise unremarkable. The M01 module was not disassembled due to a lack of evidence of internal damage.

3.1.7 Hydromechanical Metering Unit

The No. 1 engine HMU was darkly discolored and covered in soot residue but otherwise appeared intact (Figure 34). The input drive spline could be rotated smoothly by hand. The HMU was retained for future bench testing and examination. See Section E Paragraph 1 for further details.

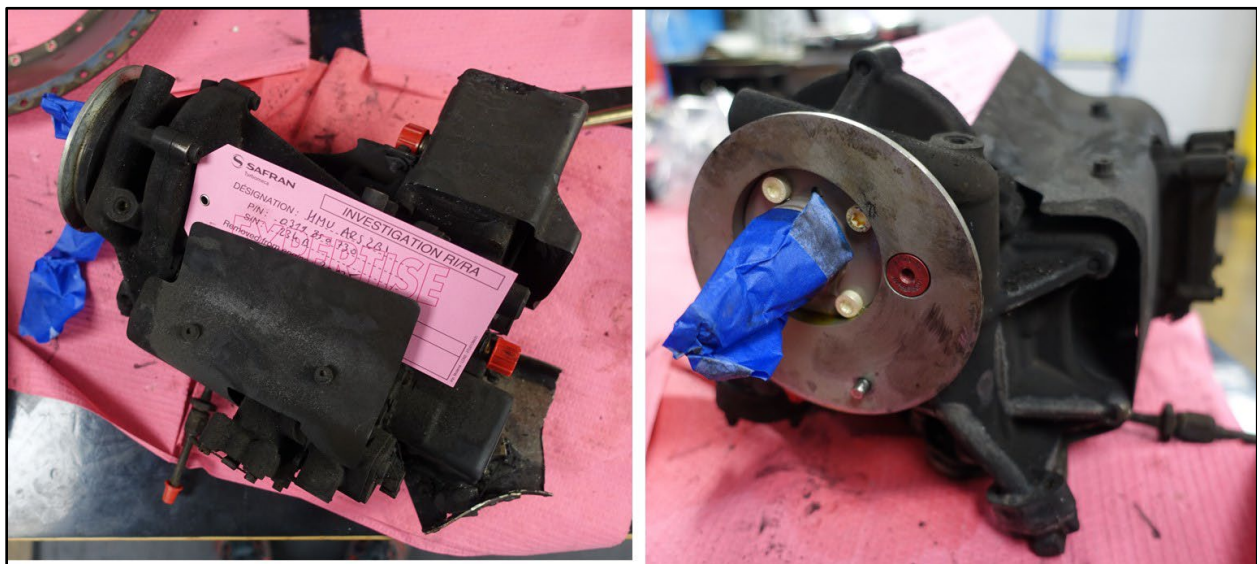


Figure 34. No. 1 Engine HMU

3.2 ESN 30182 Examination

3.2.1 As Received

All external engine surfaces were sooted/discolored (Figure 35). Multiple engine wire harnesses exhibited thermal damage and were charred and brittle (Figure 36). The ESN on the engine data plate was verified as 30182 (Figure 37).



Figure 35. ESN 30182 As Received - Left Side (L), Right Side (R)



Figure 36. ESN 30182 As Received - Front/Gearbox Side



Figure 37. ESN 30182 Data Plate

The engine inlet was coated with a dark residue that was bubbled from the 11:00 to 5:00 o'clock positions (upper right quadrant, ALF) (Figure 38). A residue sample was collected for analysis at SafranHE and the inlet case was retained for shipment to the NTSB Materials Laboratory in Washington, DC for analysis.



Figure 38. Engine Inlet Residue

The turbine exhaust case exhibited localized deformations at three of the four bolt locations connecting the exhaust case to the exhaust pipe, and the power turbine blades appeared unremarkable when looking through the exhaust pipe (Figure 39).



Figure 39. Exhaust Pipe Deformations (L), PT Blades (R)

The adjusted fuel valve assembly cover was secured on the turbine case and there was no evidence of fuel line chafing due to contact with the cover (Figure 40). The stirrup fitting that secures the preference injector pipe assembly and the right injector half manifold assembly was installed correctly as outlined in SafranHE Service Letter No. 3052/23/ARS2. After the adjusted fuel valve assembly cover was removed, the fire protection coating on the stop electro-valve was thermally damaged. All fuel lines appeared intact and flange connections/interfaces appeared properly engaged.

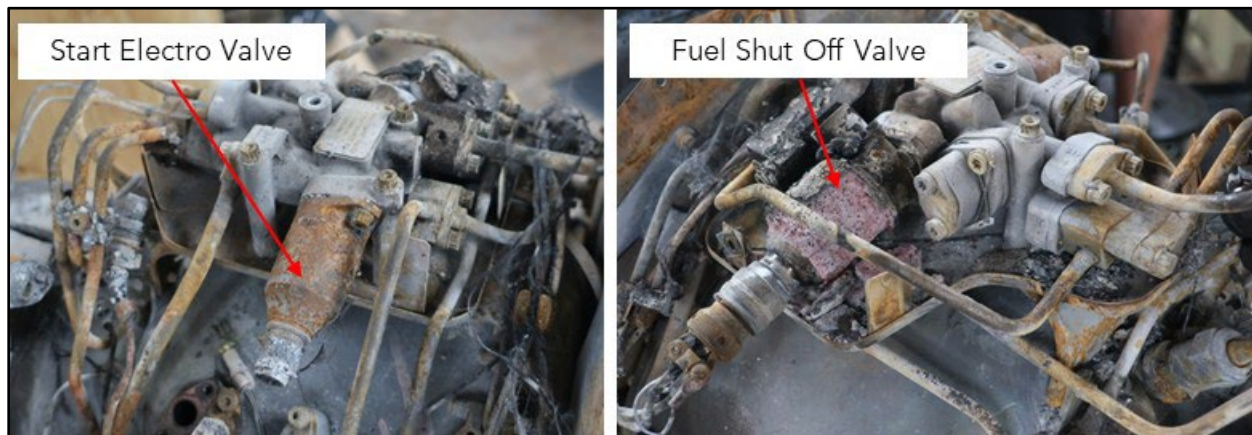


Figure 40. Fuel Adjusted Valve - Right Side (L), Left Side (R)

Burned residue was adhered to the exterior of the fuel and oil filter housings on the AGB. The fuel filter was fuel wetted when removed from the housing with no visible debris/obstructions between the filter pleats. The oil filter was oil wetted when removed from the housing with no visible debris/obstructions between the filter pleats. (Figure 41).



Figure 41. Fuel Filter (L), Oil Filter (R)

The electrical MCD for the M01 module did not have any ferrous debris attached to the magnet. The rear bearing electrical MCD (labeled M02) as well as the MCD housing sustained thermal damage with no evidence of ferrous debris on the magnet but a sample was collected for analysis at SafranHE (Figure 42).



Figure 42. MCDs - (M01) Reduction Gearbox (L), (M02) Rear Bearing (R)

The oil reservoir was intact with no evidence of tank breaches, the remaining oil was drained during disassembly and found to be approximately 2 liters of oil. The oil did not smell acrid/abnormal and appeared normal for service run engines; an oil sample was collected for analysis at SafranHE.

The PT rotor and the power drive shaft would not rotate by hand. The compressor impeller could be rotated by hand with concurrent rotation of the starter generator pad consistent with gas generator shaft continuity.

All external air, fuel and oil lines were removed, visually examined, and photographed. There was no visible evidence of breaks, cracks, or obstructions on any of the external lines. All N1 and N2 speed sensors appeared intact and unremarkable, the connecting wires were thermally damaged but still appeared connected to the engine wiring harness. Fine metallic particles were found attached to the N2A speed sensor magnet. All 4 temperature probes for the T4.5 harness appeared visually undamaged. The T4.5 conformation box and rear turbine bearing magnetic chip detector housing, mounted on the aft flange of the turbine case near the 6:00 o'clock position, was present but thermally damaged.

The front (cold zone) fire detector, located on the forward side of the M01 module at the 12:00 o'clock position, was found in the proper position but twisted and thermally damaged. The rear (hot zone) fire detector, located at the 6:00 o'clock position on the turbine case, was found connected in the proper position but thermally damaged. The torque sensor, located next to the output shaft flange, was found in the proper position but the connecting wires were found thermally damaged and broken.

3.2.2 Centrifugal Compressor & Diffuser Assembly

The centrifugal compressor impeller had dark discoloration (sooting) on all surfaces with no visual indications of contact or damage, but loose debris was found on the impeller and into the diffuser (Figure 43). The impeller cover was discolored with indications of contact rubbing on approximately a 10 degree section of the

impeller trailing edges, and the impeller cover interior surface had a dimpled appearance (Figure 43). The gas generator thrust bearing was oil wetted and rotated smoothly.

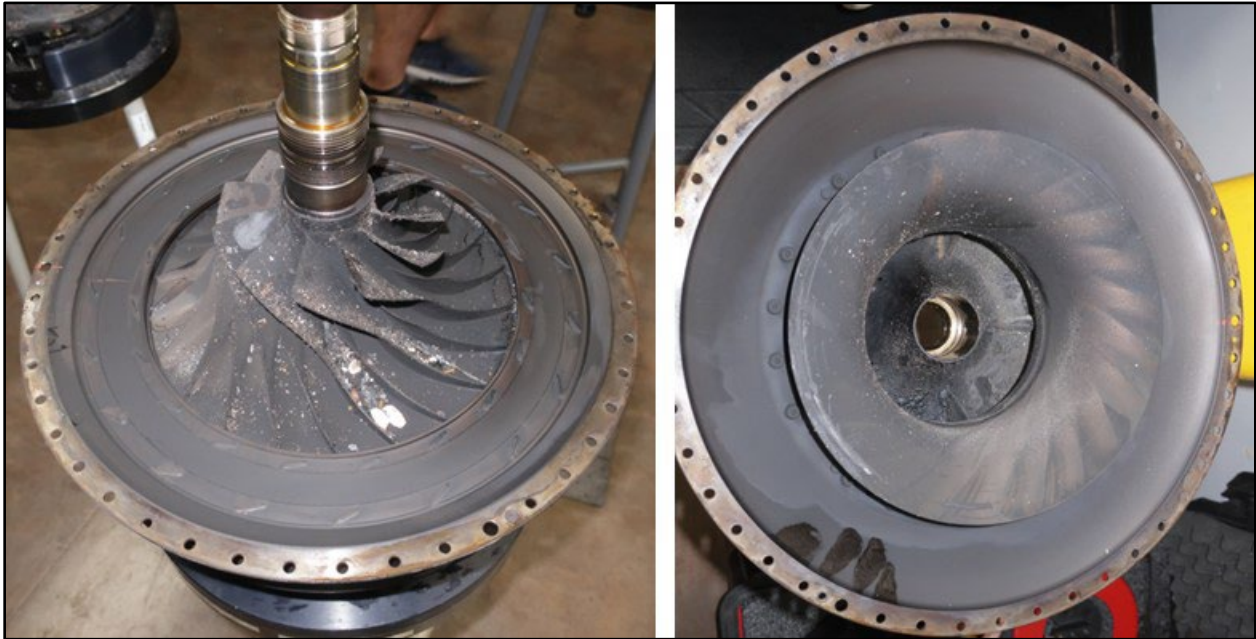


Figure 43. Centrifugal Compressor Impeller (L), Impeller Cover (R)

The aft impeller curvic coupling on the impeller and the mating curvic coupling on the front of the gas generator turbine disk showed no evidence of damage and appeared typical of service run engines (Figure 44).



Figure 44. Gas Generator Turbine Disk Curvic Coupling (L), Impeller Curvic Coupling (R)

3.2.3 Combustor

The combustion chamber was intact with no evidence of burn through or uneven wear, and the flame tubes and pre-vaporization tubes were undamaged (Figure 45). The chamber had dark discoloration on all surface and black spots were observed under the fuel pre-vaporization tubes in the combustor, typical for service run engines according to SafranHE.

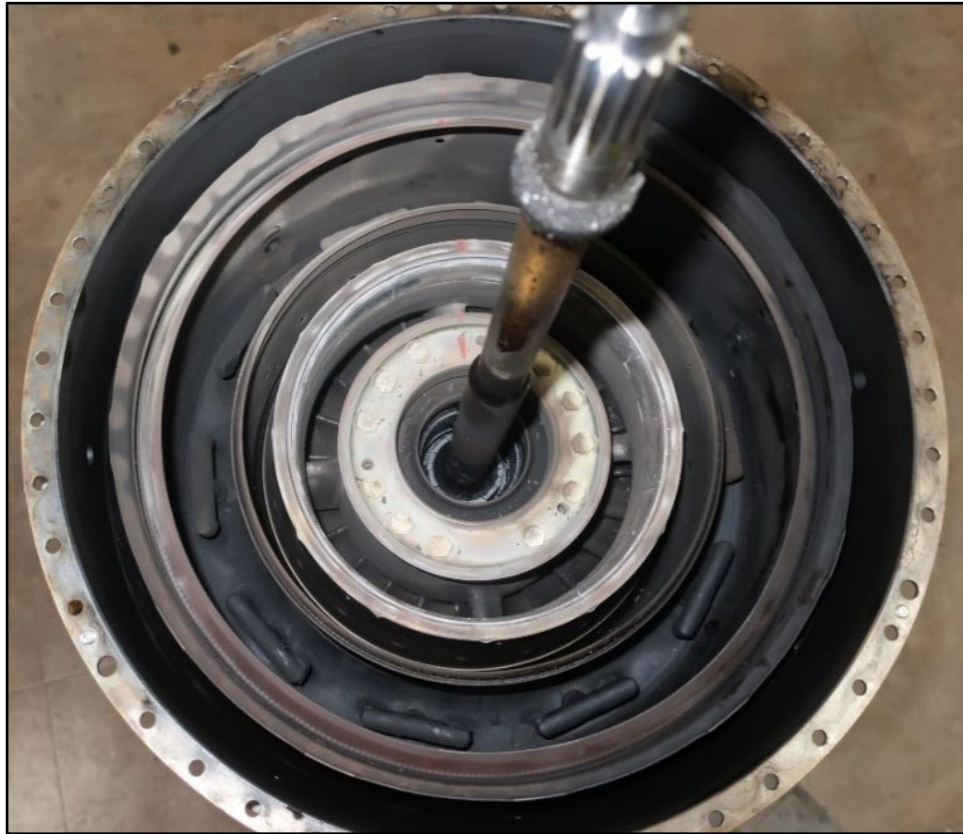


Figure 45. Combustion Chamber

The left and right main fuel manifold injectors and the preference injector were intact with dark black discoloration on multiple injector tips (Figure 46).

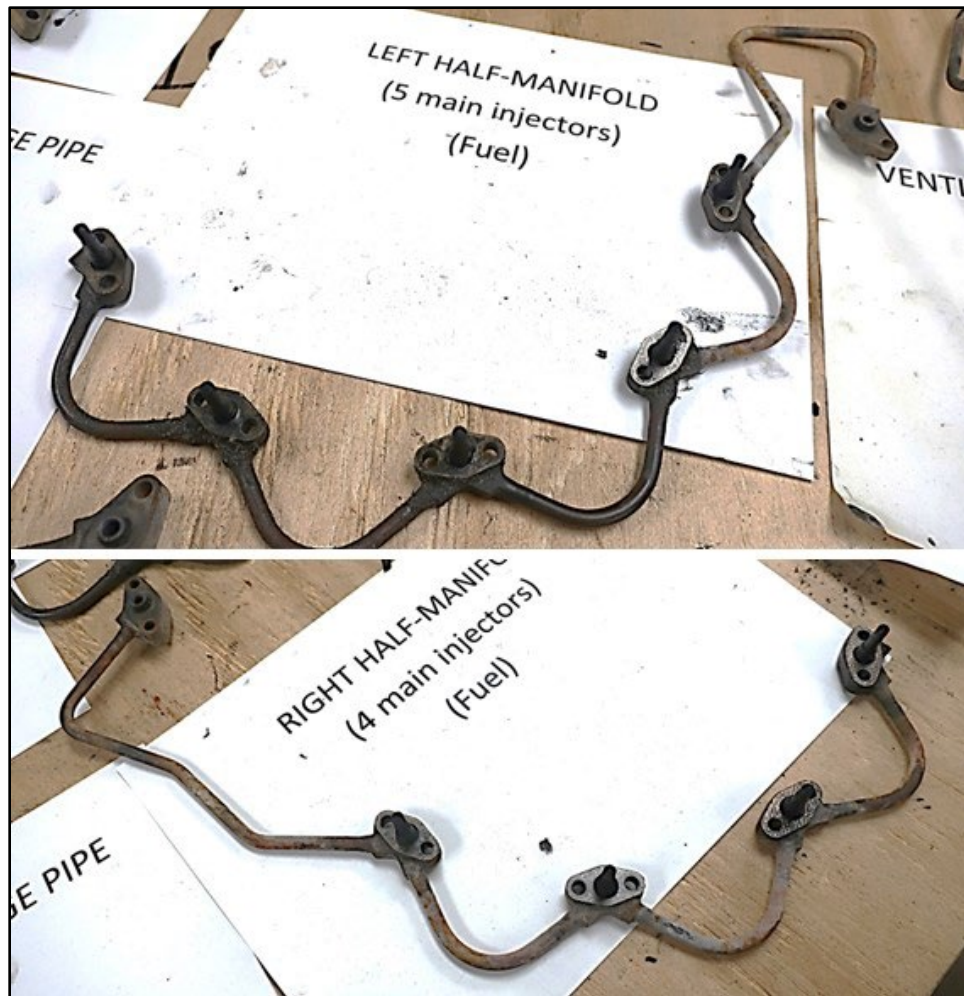


Figure 46. Fuel Injector Manifolds

3.2.4 Gas Generator Turbine

The gas generator turbine NGVs were all present and had a dark discoloration or sooted appearance but were otherwise unremarkable. The gas generator turbine disk blades appeared intact with no evidence of damage (Figure 47). Some gas generator turbine NGVs exhibited localized white discoloration and a flaking appearance, similar to the gas generator turbine disk blades.

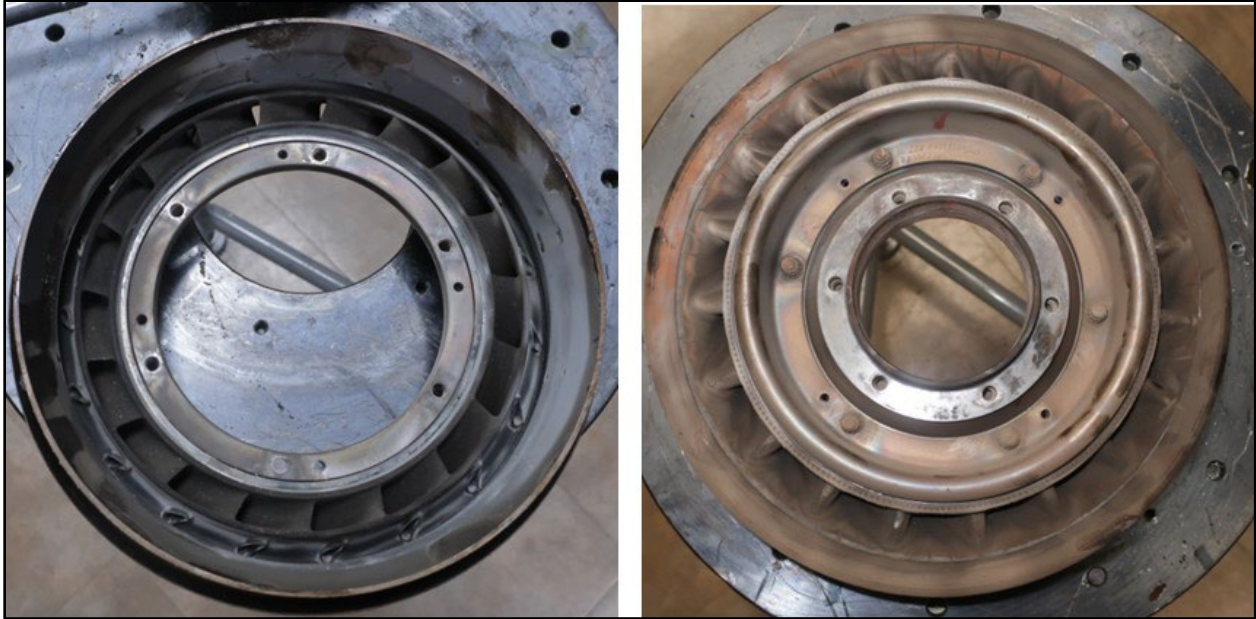


Figure 47. Gas Generator Turbine NGVs - Trailing Edge (L), Leading Edge (R)

The gas generator turbine disk blades appeared intact with no evidence of damage (Figure 48) and exhibited a white discoloration and a flaking appearance on the leading edge surfaces near the blade root. The airfoils of gas generator turbine disk blades also exhibited a white discoloration on the concave (pressure) side. All gas generator turbine disk posts and blade fir trees appeared undamaged and all the blade locks were present. The gas generator turbine shroud segments were all present and unremarkable.

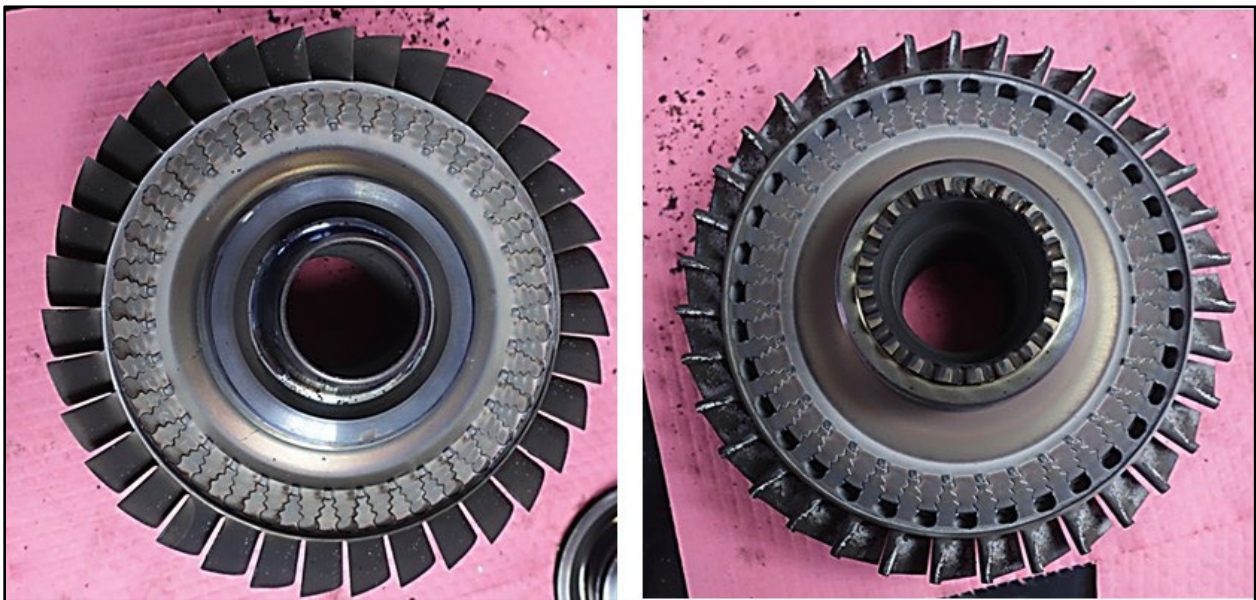


Figure 48. Gas Generator Turbine Disk/Blades - Trailing Edge (L), Leading Edge (R)

The gas generator turbine rear bearing had no residual oil and appeared dry. The roller bearing was intact and rotated with minimal resistance (Figure 49).



Figure 49. Gas Generator Turbine Rear Bearing

3.2.5 Power Turbine

All PT disk blades were intact and complete with no evidence of damage (Figure 50 and Figure 51). The PT inner shroud was removed, revealing static blade contact marks resulting from the radial sliding action and contact between the PT disk blade tips and the PT inner shroud (Figure 50). With the removal of the PT inner shroud, the PT disk freely rotated with corresponding rotation of the power drive shaft through the reduction gearbox.

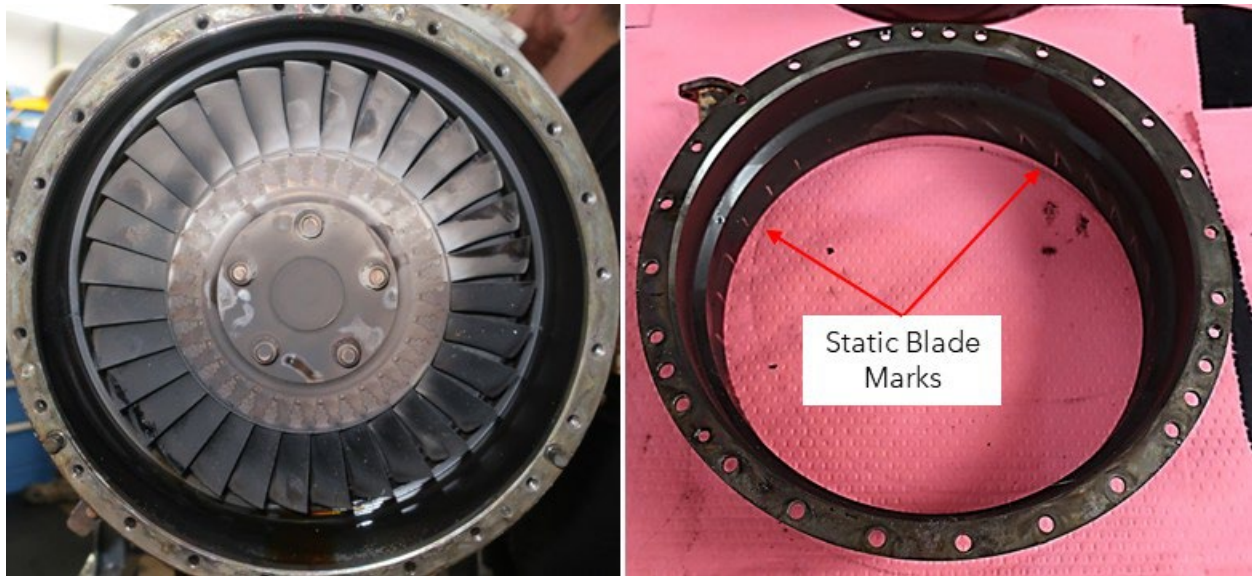


Figure 50. PT Disk/Blades Installed (L), PT Inner Shroud (R)

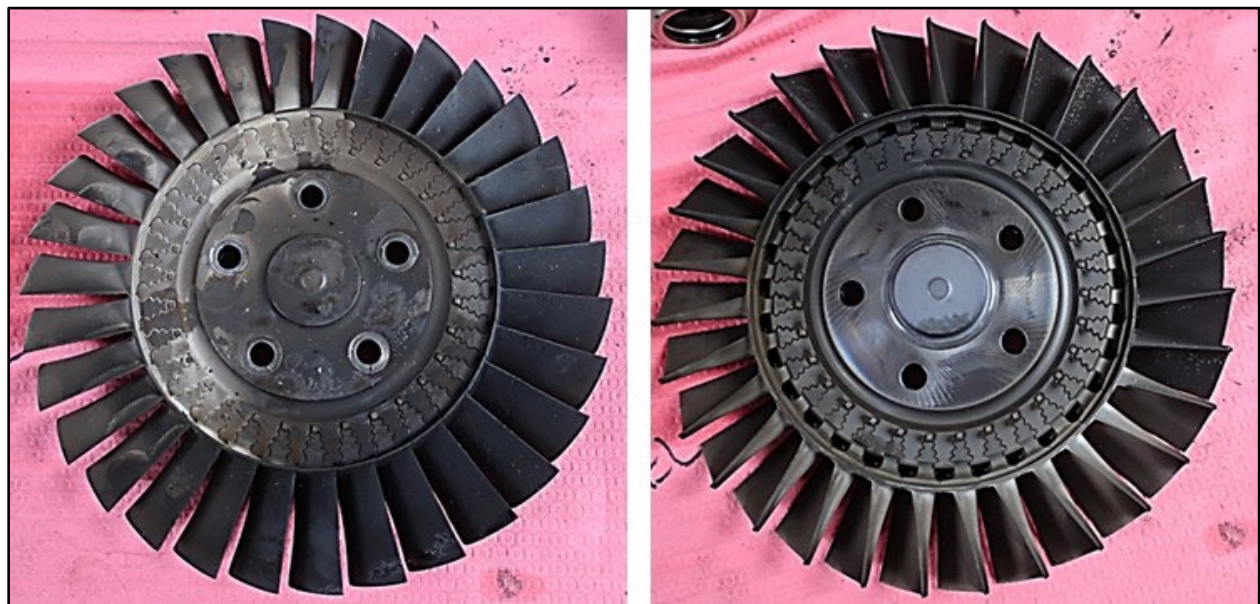


Figure 51. PT Disk/Blades - Trailing Edge Side (L), Leading Edge Side (R)

The PT NGVs had dark discoloration and residue adhered to all vanes but was otherwise unremarkable (Figure 52).

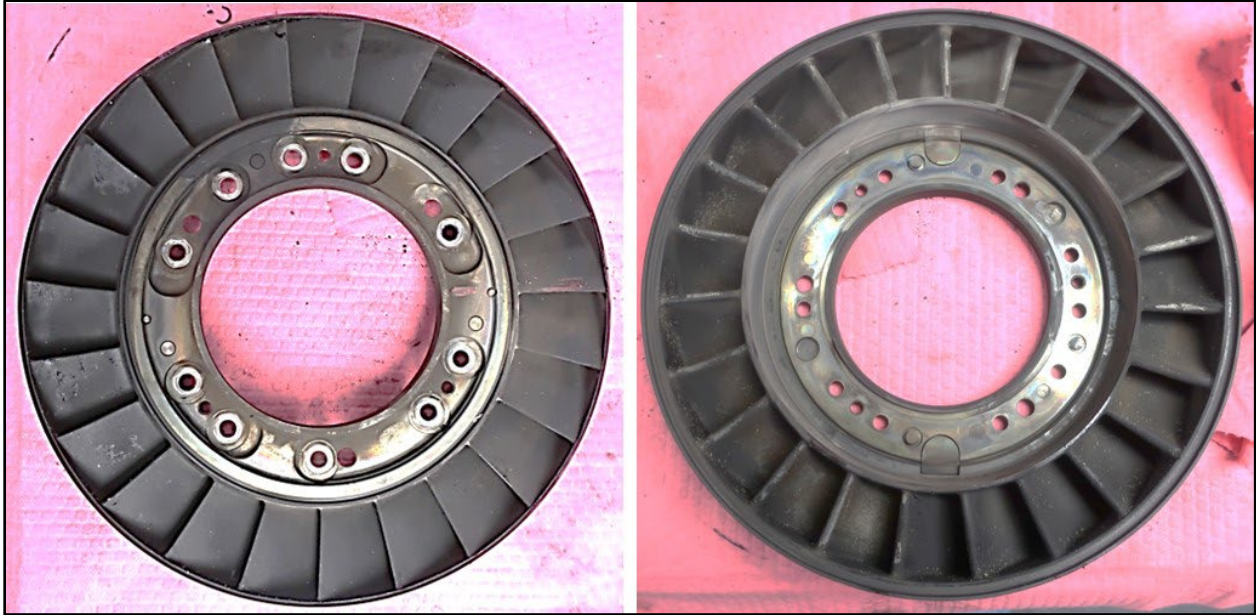


Figure 52. PT NGVs - Trailing Edge Side (L), Trailing Edge Side (R)

The PT thrust bearing appeared oil wetted and discolored with coking. The bearing elements came out of the bearing cage upon removal of the bearing, but otherwise there was no evidence of damage (Figure 53).



Figure 53. PT Thrust Bearing

3.2.6 M01 (Reduction Gearbox) Module

The PT rotor and power drive shaft did not spin freely until after the PT inner shroud was removed, after which the PT rotor rotated smoothly. The PT shaft, connecting the PT disk to the M01 module, was found undamaged, the splines were in visually good condition and were otherwise unremarkable. The M01 module was not disassembled due to a lack of evidence of internal damage.

3.2.7 Hydromechanical Metering Unit

The No. 2 engine HMU had dark discoloration and was covered in soot residue. The HMU case appeared to be intact (Figure 54). The input drive spline could be rotated smoothly by hand. The HMU was retained for further disassembly at SafranHE. See Section E Paragraph 1 for further details.



Figure 54. No. 2 HMU

E POST ENGINE EXAM EXAMINATIONS

1.0 Hydromechanical Metering Units (Fuel Control Units)

The HMUs were packaged and shipped to the SafranHE facility in Tarnos, France, a maintenance facility that repairs HMUs for further examination and bench testing. The preliminary workscope for both HMUs included visual examination, bench/rig testing, disassembly and internal component examination. Examination of the HMUs was performed April 29-30, 2024, with representatives of the BEA in attendance.

1.1 No. 1 Engine HMU P/N 0319850730, S/N 234B

The as-received visual examination found the No. 1 engine HMU S/N 234B to have black deposits, a sooted appearance on all external surfaces and on the fire protection covers (Figure 55). The manual mode control lever, located at approximately the 12:00 o'clock position of the HMU when installed on the engine, was found in a position consistent with the neutral position and was deformed/bent. The o-ring seals located on the fuel drain and the fuel pump input shaft appeared undamaged.

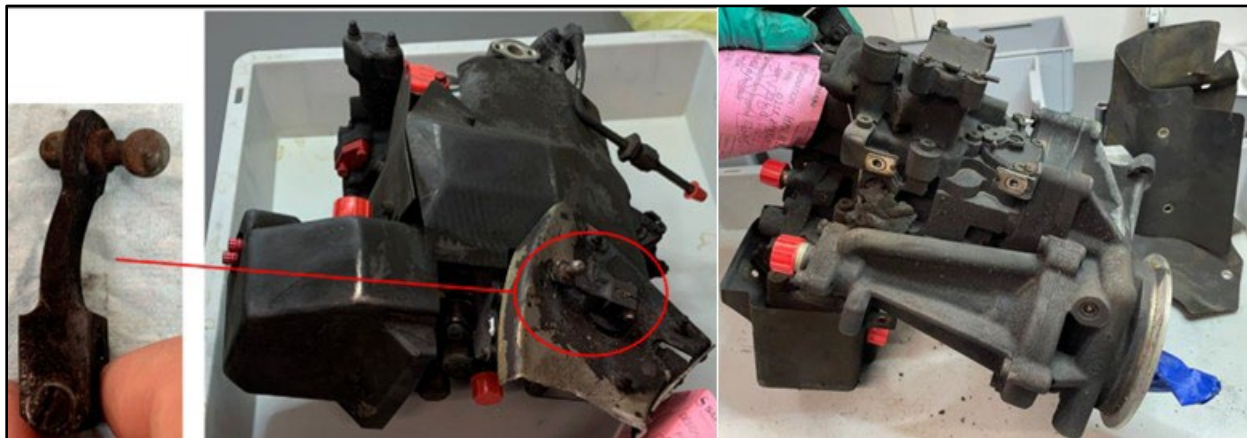


Figure 55. No. 1 Engine HMU - As Received (R), Bent Mode Control Lever (L)
(Figure Courtesy of SafranHE)

The HMU resolver is a position transmitter that provides a metering needle position signal to the EECU. The resolver provides the fuel metering valve feedback position through a rotary variable differential transformer, where an angular position will correspond to a fuel flow. A discrepancy of more than 7 degrees between the resolver angle (indicated by the resolver) and the expected metering unit angle (calculated by the EECU) will result in a failure indication in the cockpit.

The HMU's resolver angular position was tested in accordance with acceptance test procedure⁸ (ATP) 0319949570, which is used for new production HMUs, with the resultant resolver position measuring 45.28 degrees. According to ATP 0319949570, the measured position most closely corresponds to a fuel flow of 122 ±5 liters/hour (Figure 56).

90	10792	12C	300	45,5	+0,8 -0,8			2050	122	+5 -5
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Figure 56. HMU Resolver Position - ATP Table

The HMU was bench tested in the as-received condition, with the resolver angular position still at 45.24 degrees, to assess its functionality. The test bench was set up in accordance with ATP 0319949570 criteria for the fuel pump with two exceptions, the actuator and the compressor outlet pressure (P3) air were not connected to avoid altering the HMU state with the use of the manual mode operation. The bench test fuel flow was measured at 120.8 liters per hour and determined to be within limits for a resolver angular position of 45.24 degrees, and no discrepancies were observed during the bench test. An additional bench test was performed to assess the range of motion of the HMU actuator while the HMU was still on the test bench. The test revealed the HMU's resolver would not move.

The HMU was then disassembled for examination of the internal components. The manual mode control lever was located in the neutral position, the microswitch position was consistent with that of the manual mode control lever position, and the load limiter was also in the neutral position (Figure 57). The disk connected to the manual control lever was found slightly rotated but not enough for the control lever to be out of the neutral position, consistent with the deformed manual mode control lever and likely a result of the disassembly process to access the load limiter (Figure 58). The microswitch housing cover membrane appeared dry but otherwise undamaged.

⁸ SAFRAN refers to ATP as "cahier des charges techniques" (CCT) which translates to Acceptance Test Procedure (ATP) and is the technical specification and test procedure for new production HMUs.

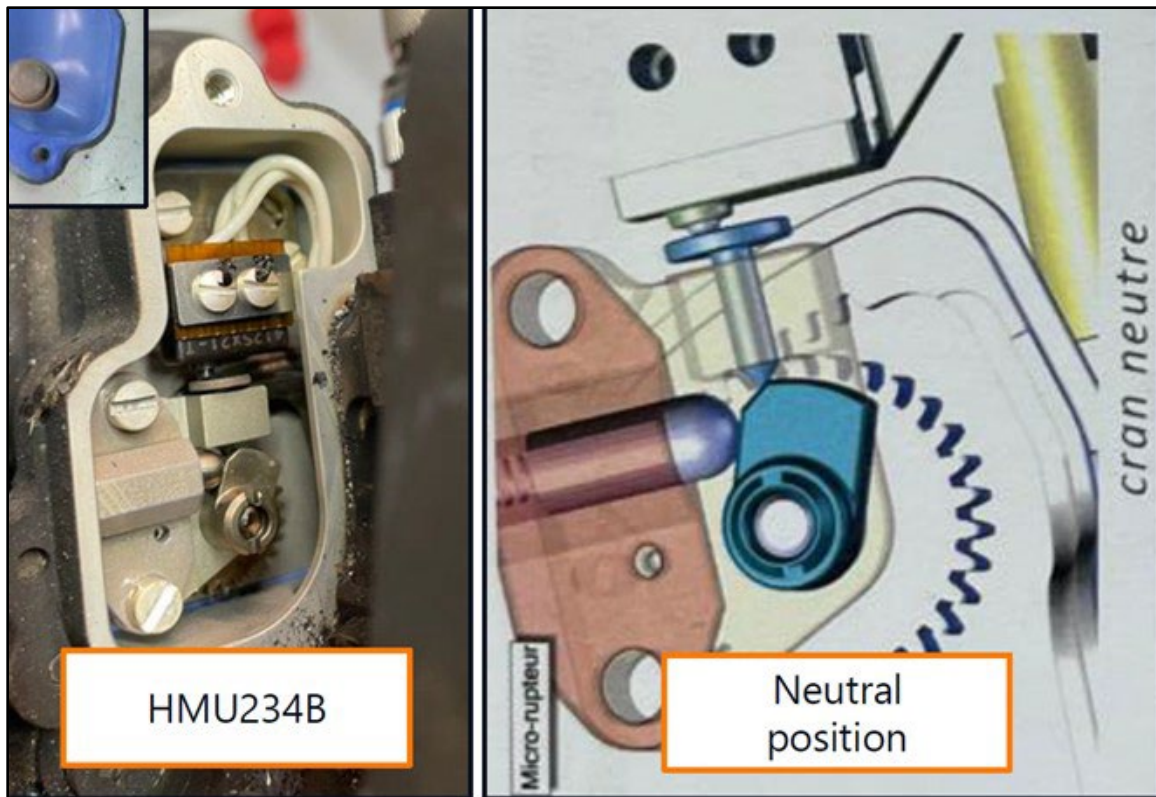


Figure 57. HMU Manual Mode Control Lever

(Figure Courtesy of SafranHE)

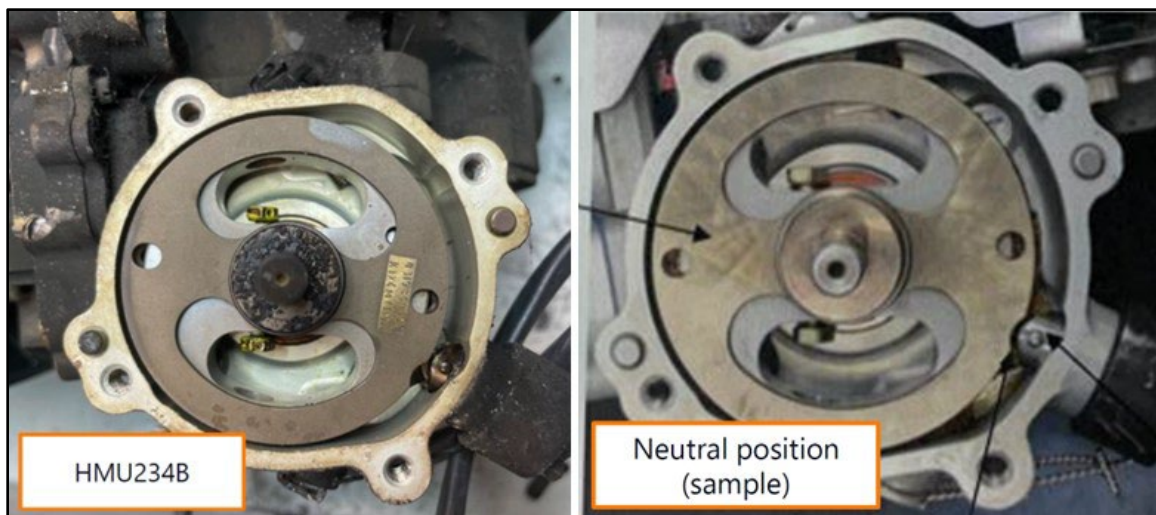


Figure 58. HMU Load Limiter

(Figure Courtesy of SafranHE)

The pressure differential (Δ -P) membrane was shiny but undamaged, consistent with there being remnant fuel within the housing. The P3 housing seal appeared undamaged, and the housing smelled of fuel (Figure 59). The resistive torque of the high-pressure pump was tested, and measured 0.078 deca-Newton meters, which was

within SafranHE's ATP specification 0319948750 of less than 0.15 deca-Newton meters.

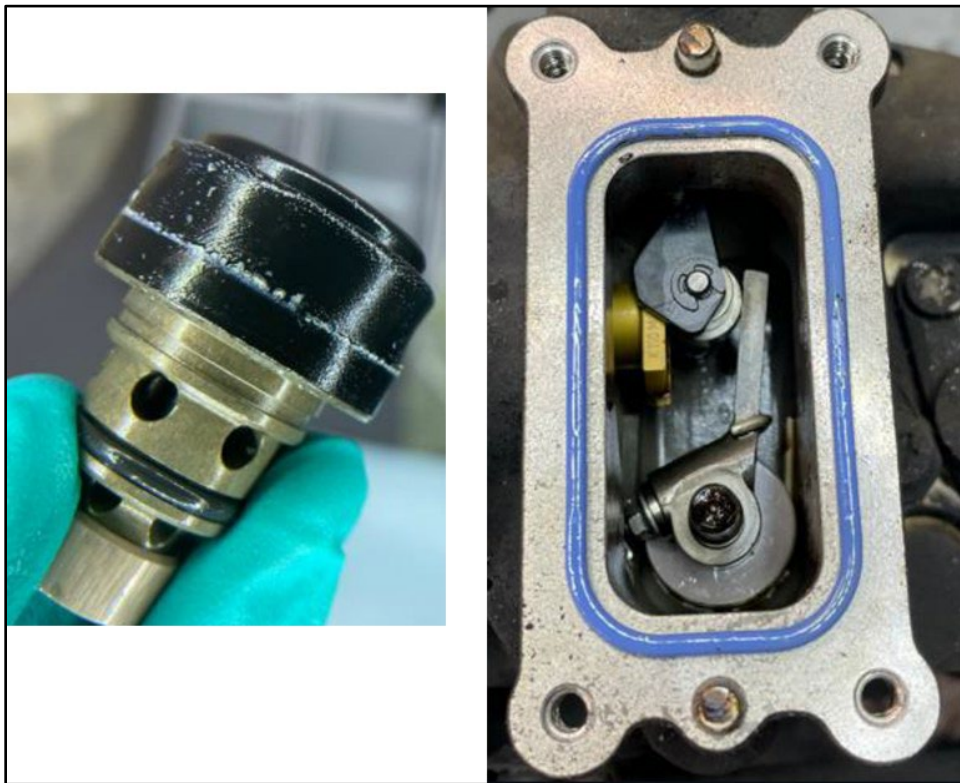


Figure 59. A-P membrane (L), P3 Housing Seal (R)

(Figure Courtesy of SafranHE)

Following the as-received bench test, the P3 membrane was removed from the HMU and appeared undamaged. The rubber membrane was found to be stiffer than a typical membrane but consistent with high temperature overheating.

The HMU resolver, P/N 9550171760, S/N1864, was electrically tested prior to removal from the HMU in accordance with Abbreviated Component Maintenance Manual Book (ACMMB) task 73-23-99-775-802-A01. The insulation resistance and the resistance of each winding was measured and was found to be within limits. The electrical continuity was found to be out of tolerance. The metallization resistance test for electrical continuity was measured at 7.69 milli-Ohms, which is not within the maximum requirement of 0.80 milli-Ohms. The electrically conductive paste that is used for the assembly of the resolver on the HMU body was found to be thermally damaged. The resolver was then removed from the HMU for further examination. The preliminary inspection of the resolver found black deposits on all external surfaces, the internal resolver shaft and flange surfaces were found clean and free of any foreign debris. The mechanical check of the shaft found no damage on the splines, but the shaft would not rotate. The grommets appeared thermally distressed, but no other damage was observed. The resolver was electrically tested in accordance with ACMMB

task 73-23-99-775-802-A01; all measurements were found within the specified limits, and no other discrepancies were observed. Further disassembly of the resolver found the resolver shaft was seized in place due to the glue/resin that is part of the assembly that had melted and resolidified around the rotor and stator, preventing rotor rotation.

The radiographic inspection (X-ray) inspection of the engine's stop electro-valve was performed and showed the valve was in an open position, with the valve not in contact with the valve seat, consistent with an engine in an operating condition. Electrical testing of the electro-valve was performed in accordance with task ACMMB 73-09-90-775-801-A01 and found there was no continuity so no further electrical testing was possible.

1.2 No. 2 Engine HMU, P/N 0319850730, S/N 287B

The as-received visual examination found the No. 2 engine HMU S/N 287B external surfaces to be covered in debris with an oily residue (Figure 60). The manual mode control lever was found in a position consistent with the neutral position. The o-ring seals, located on the fuel drain and the fuel pump input shaft, appeared undamaged.



Figure 60. No. 2 Engine HMU - As Received

(Figure Courtesy of SafranHE)

The HMU’s resolver angular position was tested in accordance with ATP 0319949570, with the resultant resolver position measuring 81.64 degrees. According to ATP 0319949570, the measured position most closely corresponds to a fuel flow of 204 ± 5 liters/hour (Figure 61).

99,6	11944	1F4	500	80,5	+0,8			3150	204	+5
					-0,8					-5

Figure 61. HMU Resolver Position - ATP Table

The HMU was not bench tested but was disassembled for examination of the internal components. The manual mode control lever was found in the neutral position, the microswitch position was consistent with that of the manual mode control lever position, and the load limiter was clutched in the neutral position (Figure 62). The microswitch housing cover membrane appeared dry but otherwise undamaged (Figure 63).

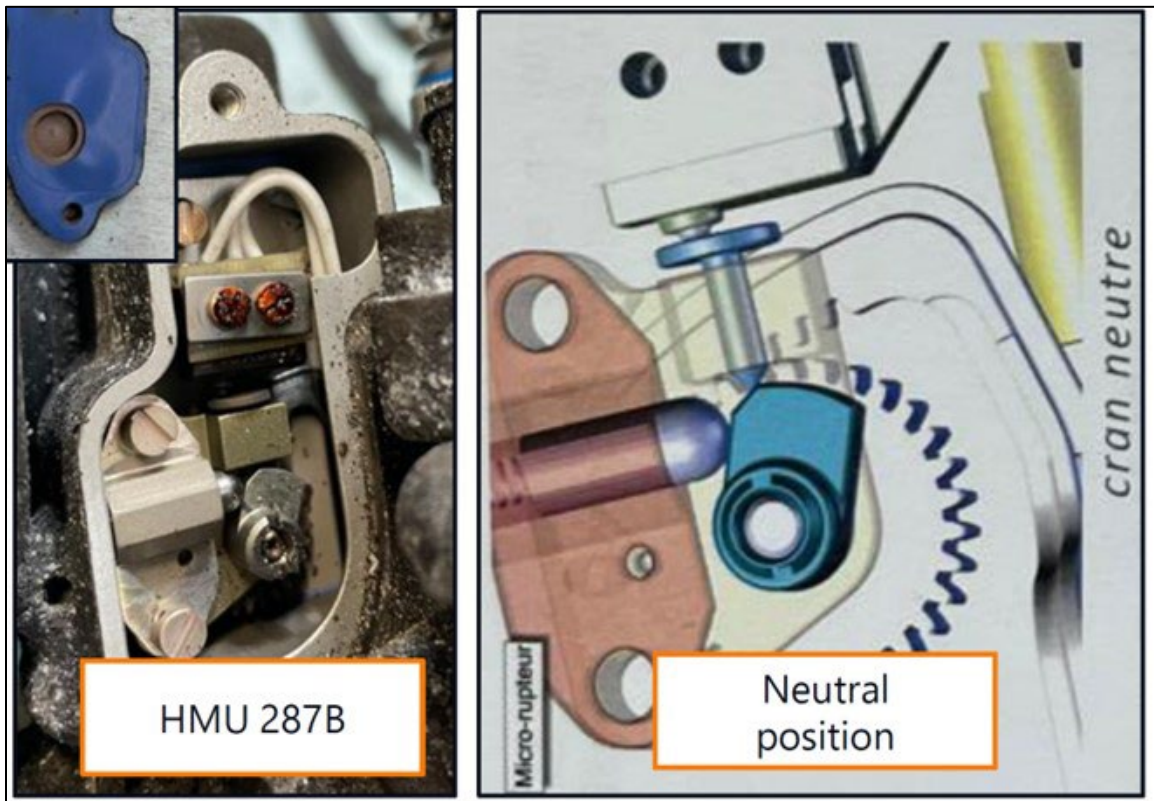


Figure 62. HMU Manual Mode Control Lever

(Figure Courtesy of SafranHE)

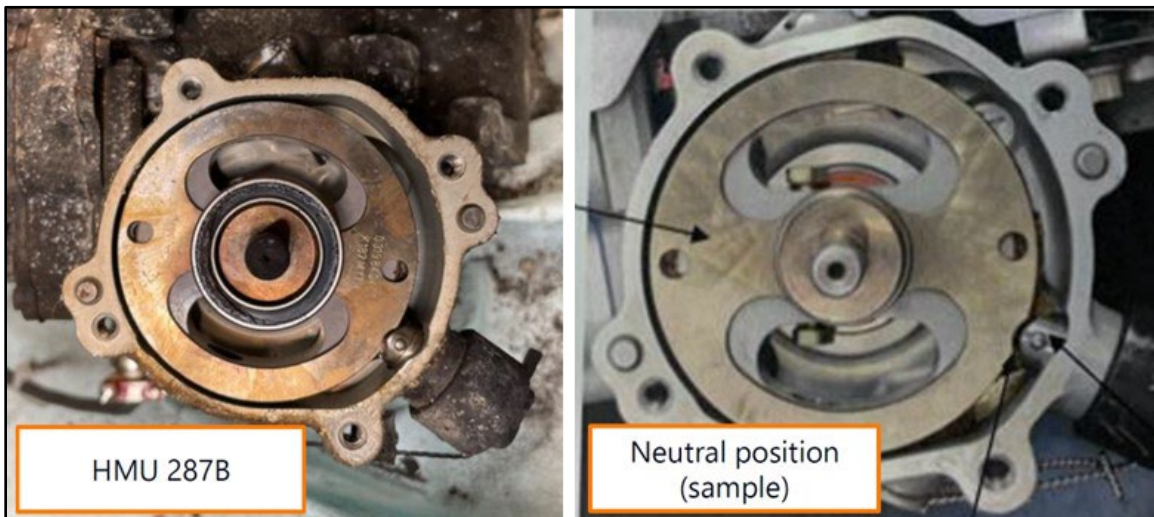


Figure 63. HMU Load Limiter

(Figure Courtesy of SafranHE)

The Δ -P membrane was shiny but undamaged, consistent with there being remnant fuel within the housing. The P3 housing was found to be dry and the seal appeared undamaged (Figure 64). The resistive torque of the high-pressure pump was tested, and measured at 0.059 deca-Newton meters, which was within SafranHE's ATP specification 0319949570 of less than 0.15 deca-Newton meters.



Figure 64. Δ -P membrane (L), P3 Housing Seal (R)

(Figure Courtesy of SafranHE)

The X-ray inspection of the engine's stop electro-valve was performed and showed that the valve was in an open position, with the valve not in contact with the valve seat, consistent an engine in an operating condition. Electrical testing of the electro-valve was performed in accordance with ACMMB 73-09-90-775-801-A01 and found there was no continuity so no further electrical testing was possible.

2.0 Fuel Injection Hardware

The main fuel injection system is made up of two half manifolds (comprising the primary injectors) and the preference injector. The preference injector is known as injector number 1, the right half manifold contains 4 injectors that are numbered 2

through 5, and the left side manifold contains 5 injectors that are numbered 6 through 10 (Figure 65). The two half manifolds are secured to the rear side of the combustion chamber casing and the preference injector supply tube is attached to the rear side of the combustion chamber at approximately the 1:00 o'clock position.

The half manifolds and preference injector supply, from both engines, were further examined by SafranHE to assess their post crash/fire condition.

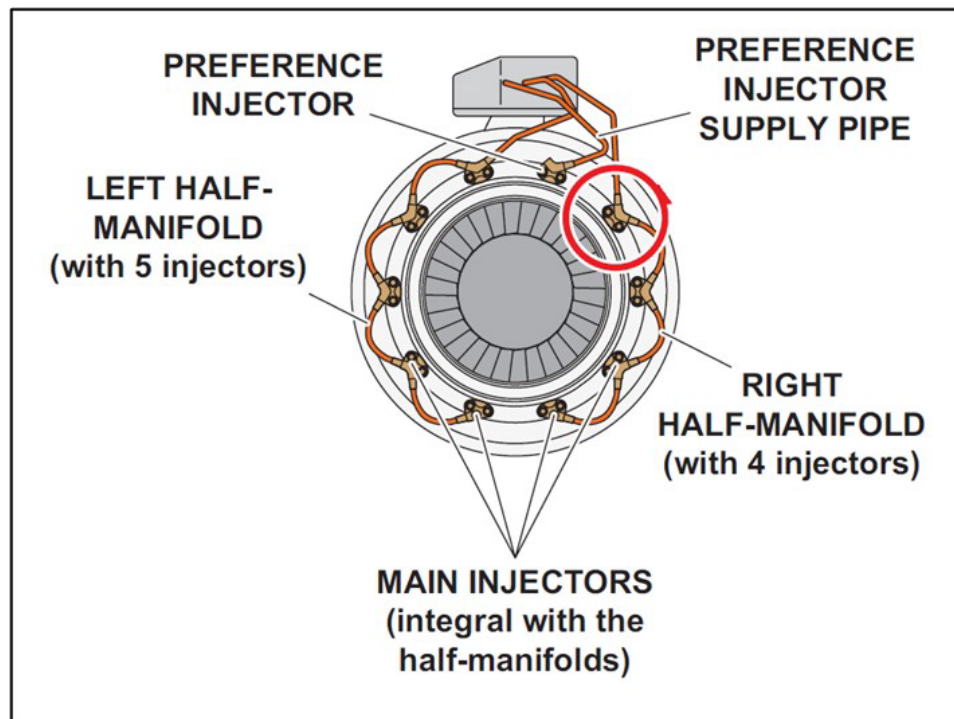


Figure 65. Fuel Injector Locations

(Figure Courtesy of SafranHE)

2.1 No. 1 Engine Fuel Injection Hardware

The visual inspection results of the No. 1 engine fuel injector output holes are provided in Table 3. Also included are the PNs and SNs of the hardware according to the inspections recorded in the engine logbooks.

Following the visual examination, the injectors were flow checked in accordance with ACMMB 73-15-01-950-001, where the acceptance criteria for the preference injector is 22.5 ± 0.7 liters per hour and the acceptance criteria for the manifold injectors is 21.2 ± 0.8 liters per hour. The injector flow check indicated that all injectors were blocked and no fuel flowed through the injectors.

Table 3. No. 1 Engine Fuel Injector Visual Inspection Results

Injector Type/Number	Part Number	Serial Number	Install Date
Preference Injector	0319630140	40009	July 2022
Injector #1	appeared one output hole was partially blocked		
Right Side Manifold	0319637170	10157	Nov. 20, 2020
Injector #2	appeared free of obstruction		
Injector #3	appeared one output hole was partially blocked		
Injector #4	appeared free of obstruction		
Injector #5	appeared one output hole was completely blocked		
Left Side Manifold	0319637180	10519	Nov. 20, 2020
Injector #6	appeared two output holes were completely blocked		
Injector #7	appeared free of obstruction		
Injector #8	appeared free of obstruction		
Injector #9	appeared free of obstruction		
Injector #10	appeared free of obstruction		

All fuel supply lines/tubes were non-destructively inspected with a fluorescent penetrant inspection method to check for cracks as potential sources of leakage. Penetrant inspection found minor anomalies, but no crack indications or fuel leakages were observed. The results are provided in Table 4.

Table 4. Penetrant Inspection of No. 1 Engine Fuel Tubes - Results

Part Name	Result
Automatic Bleed Pipe (Metering Unit Purge Valve)	No indication
Metering Valve/Start Valve Pipe (Fuel valve assy supply pipe)	Indication -blowhole along welding bead
Start injector Ventilation Pipe	No indication
Left Start Injector Supply Pipe	No indication
Right Start Injector Supply Pipe	No indication
Fuel Valve Assy Drain Pipe	No indication
Main Fuel Injector Drain Pipe (Stop Purge Valve Purge Pipe)	No indication
Adjusted Fuel Valve Assy Pipe	No indication
Pipe Bleed Assy/Metering Valve (Drain Pipe Assy-HPBP Pump Assy)	Indication of corrosion pitting within welding bead, within heat affected zone and along the pipe but without perforation
Pipe Metering Valve/Bleed (Metering Valve Air Bleed Valve)	No indication

2.2 No. 2 Engine Fuel Injection Hardware

The visual inspection results of the No. 2 engine fuel injector output holes is provided in Table 5. Also included are the part numbers and serial numbers of the hardware according to the engine logbooks.

Table 5. No. 2 Engine Fuel Injector Visual Inspection Results

Injector Type/Number	Part Number	Serial Number	Install Date
Preference Injector	0319630140	39144	July 2022
Injector #1	appeared one output hole was partially blocked, one was completely blocked		
Right Side Manifold	0319637170	11594	Nov. 20, 2020
Injector #2	appeared two output holes were partially blocked		
Injector #3	appeared two output holes were partially blocked		
Injector #4	appeared free of obstruction		
Injector #5	appeared one output hole was partially blocked		
Left Side Manifold	0319637180	10732	Nov. 20, 2020
Injector #6	appeared free of obstruction		
Injector #7	appeared one output hole was partially blocked, one was completely blocked		
Injector #8	appeared two output were holes partially blocked		
Injector #9	appeared two output were holes partially blocked		
Injector #10	appeared one output hole was partially blocked, one was completely blocked		

Following the visual examination, the injectors were flow three separate times for comparison. The injector flow check results are provided in Table 6.

Table 6. Fuel Injector Flow Test - Results

Injector Type/Number	Test #1	Test #2	Test #3
Preference Injector (22.5 ±0.7 Liters/hour)			
Injector #1	15.06	13.56	13.10
Right Side Manifold (21.2 ±0.8 Liters/hour)			
Injector #2	21.75	21.68	21.69
Injector #3	18.65	18.40	18.50
Injector #4	18.16	15.48	14.38
Injector #5	14.02	21.94	21.96
Left Side Manifold (21.2 ±0.8 Liters/hour)			
Injector #6	20.62	21.92	21.95
Injector #7	14.49	15.96	16.13
Injector #8	17.49	17.43	17.08
Injector #9	22.08	22.05	21.22
Injector #10	0.04	0.08	0.07
Cumulative Flow	147.28	157.87	153.12

All fuel supply lines/tubes were visually inspected for defects as potential sources of leakage. No indications were observed.

3.0 N1 and N2 Speed Sensors

The speed indicating system measures the rotational speeds of the gas generator and the power turbine using electro-magnetic sensors, phonic wheels and wiring harnesses connecting the sensors to the EECU and cockpit indicators. The gas generator speed is measured by two N1 speed sensors, identified as N1B and N1C, located on the front of the AGB around the fuel pump drive casing. The power turbine speed is measured by three N2 sensors, identified as N2A, N2B, and N2C, located on the front face of the RGB casing around the front of the intermediate gear. There is one additional speed sensor, identified as N1A and is located on the engine alternator, which functions as a redundancy signal to the N1B signal.

The speed sensors produce an electrical signal that is proportional to the rotational speed and the number of gear teeth on the phonic wheel. The output signals from the N1C and N2C sensors transmit to the respective cockpit speed indicators. The output signals from the N1A, N1B, N2A, and N2B sensors are transmitted to the EECU. (Figure 66).

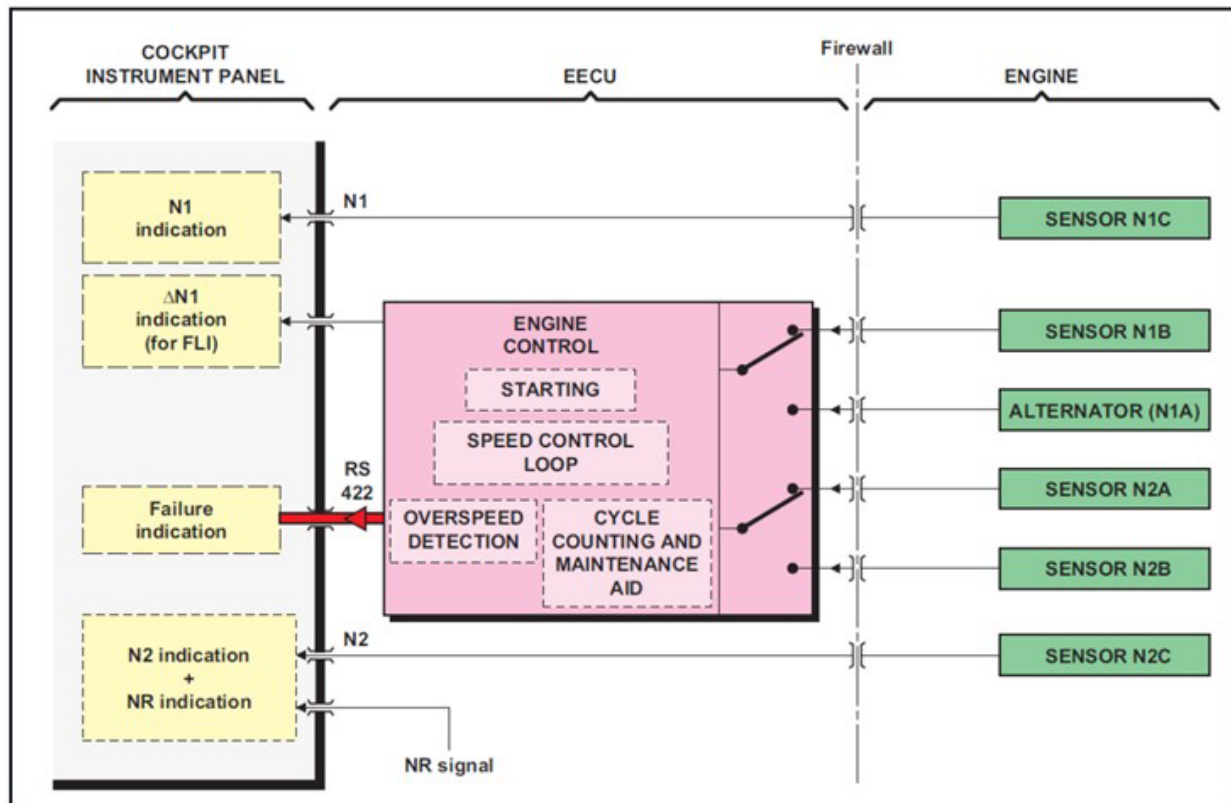


Figure 66. N1 and N2 Sensor Diagram

(Figure Courtesy of SafranHE)

All six sensors from the No. 1 engine were removed for further examination and bench testing. The visual inspection of the sensors found that all external surfaces were dirty and covered with black deposits, and the grommets on each connector showed evidence of thermal distress.

All the sensors were electrically tested in accordance with ACMMB 77-19-10-775-801-A01 to assess their functionality. The bench test determined all sensors were functional and all electrical measurements were within the specified limits (Table 7 and Table 8). The N1A sensor was disassembled for further examination; the stator guard appeared discolored and thermally distressed, while the rotor appeared undamaged and turned without difficulty.

Table 7. Electrical Test Results - N1A (Alternator) Sensor

Sensor Name	Part Number	Serial Number	Manuf. Date	Insulation (>20MΩ)	Pin1:Pin2 Resistance (0.27Ω<R<0.33Ω)	Pin1:Pin3 Resistance (0.27Ω<R<0.33Ω)	Pin2:Pin3 Resistance (0.27Ω<R<0.33Ω)
N1A	9550171800	337	1998	> 20MΩ	0.31Ω	0.32Ω	0.31Ω

Table 8. Electrical Test Results (Sensors: N1B, N1C, N2A, N2B, N2C)

Sensor Name	Part Number	Serial Number	Manuf. Date	Insulation (>20M Ω)	Winding Resistance (128.7 Ω <R<157.3 Ω)
N1B	9550165760	2013	Oct. 1998	> 20M Ω	154.64 Ω
N1C	9550165760	2064	Oct. 1998	> 20M Ω	154.4 Ω
N2A	9550165760	2042	Oct. 1998	> 20M Ω	155.87 Ω
N2B	9550165760	2397	Dec. 1998	> 20M Ω	154.97 Ω
N2C	9550165760	2005	Oct. 1998	> 20M Ω	153.46 Ω

4.0 Heat/Temperature (Fire) Sensors

There are two temperature sensors on each of the Arrius 2B1 engines, one located on the forward side of the M01 module at the 12:00 o'clock position, also known as the front or cold zone fire detector. The second sensor is located at the 6:00 o'clock position on the turbine case, which is also known as the rear or hot zone fire detector (Figure 67). The temperature threshold for the cold zone sensor is 210 degrees C and the threshold for the hot zone sensor is 271 degrees C. The sensors for each engine are directly connected to the cockpit warning display unit and if the temperature threshold is exceeded at any one of the sensors, the appropriate engine fire warning indicator will illuminate and an aural warning tone, or gong, will sound in the pilot's headset (Figure 67).

An additional temperature sensor, located in the cold zone on the starter/generator, monitors the starter/generator cooling air with a temperature threshold 205 degrees C. If the temperature threshold is exceeded, a caution message of "GEN OVHT" [generator overheat] will be displayed on the cockpit advisory display (CAD) system (Figure 68).

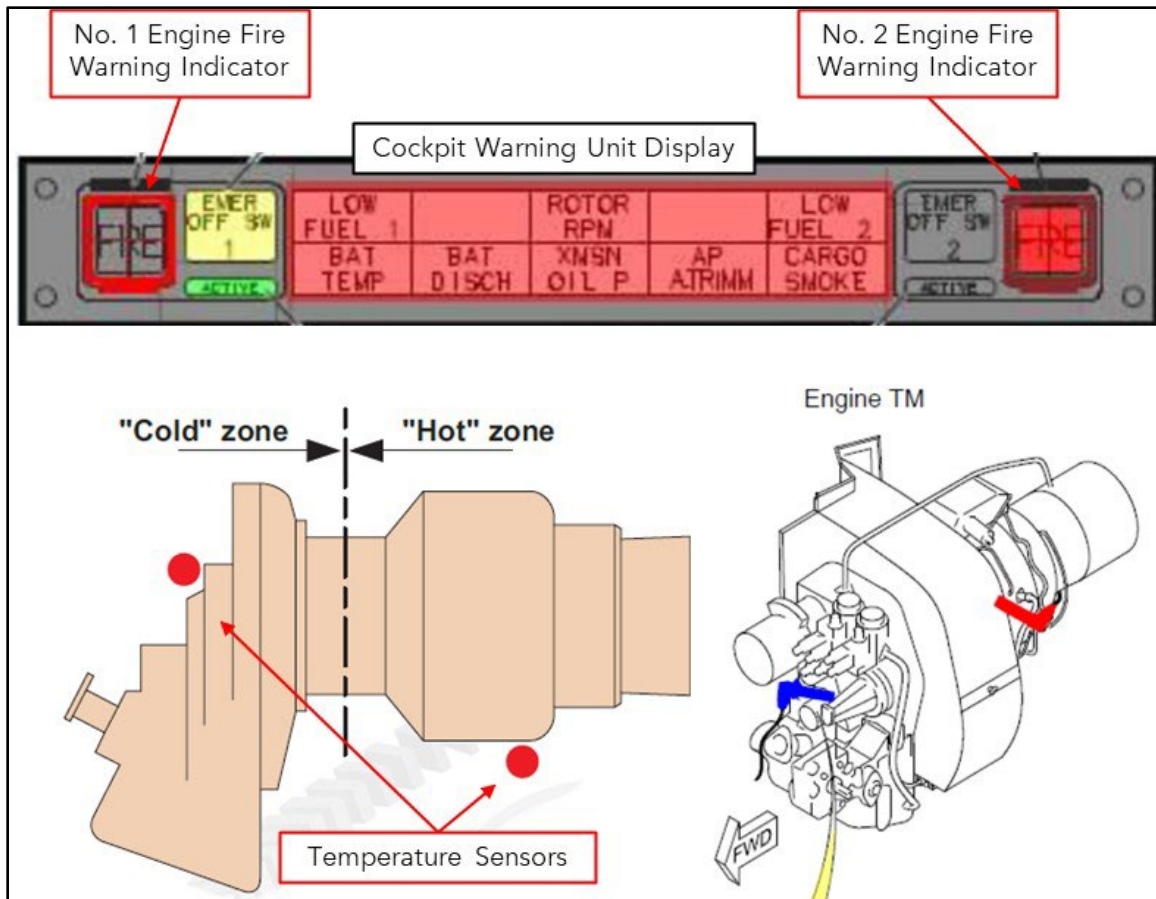


Figure 67. Engine Temperature Sensors and Cockpit Warning Display
(Figure Courtesy of Airbus Helicopters)

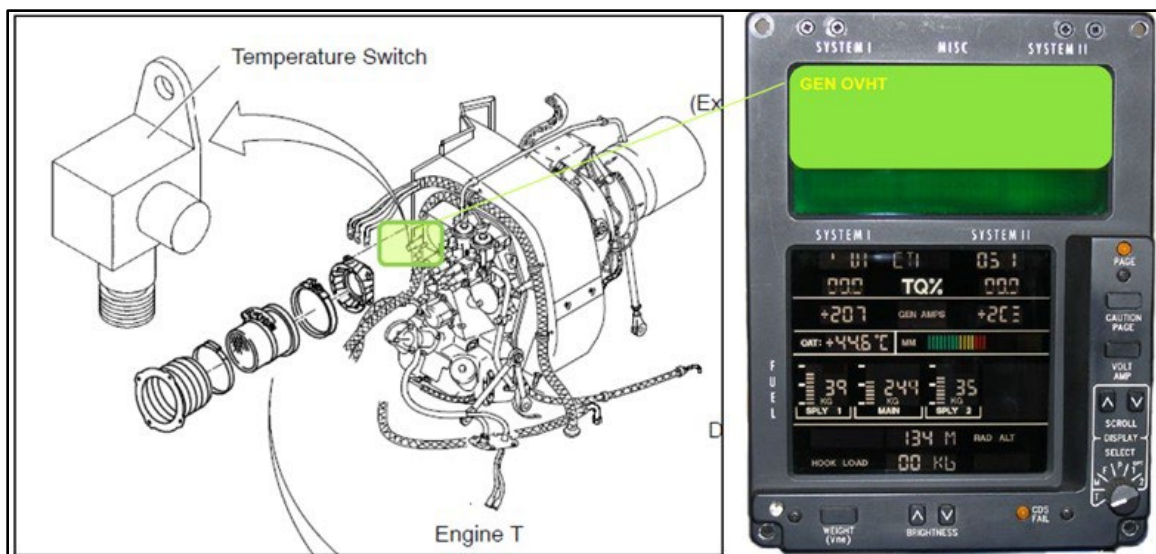


Figure 68. GEN OVHT Temperature Sensor Location and Warning Message
(Figure Courtesy of Airbus Helicopters)

The helicopter contains a temperature sensor in the air duct supplying air to the cabin, which is downstream of the air mixing valve that is supplied by the P3 customer bleed air supply from both engines. This sensor is referred to as the heating, ventilation and air conditioning (HTG) sensor. The cabin air duct temperature sensor is located between the engine compartments along the helicopter centerline, forward of the tail boom attachment flange and is approximately at the same height as the engine decks (Figure 69). This location is in close proximity to the hot zone fire/heat sensors in both engine compartments but separated by the titanium firewall.

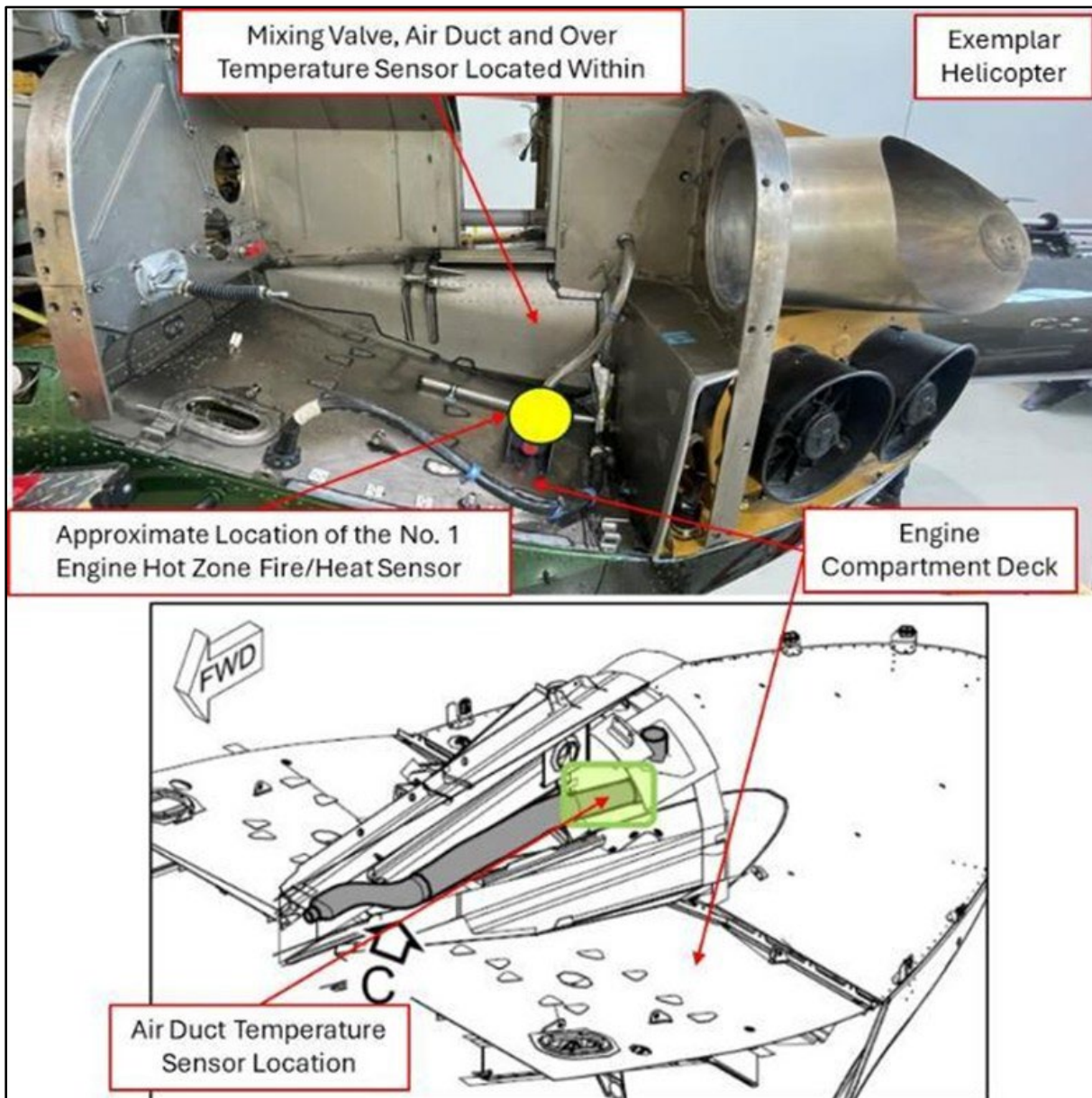


Figure 69. Location of HTG OVTP Sensor and Engine Compartment
(Figure Courtesy of Airbus Helicopters)

This air duct sensor has a temperature threshold 132 degrees C, which when exceeded, a caution message of "HTG OVTEMP" [HTG/temperature overtemperature] will be displayed on the CAD system, an aural warning tone or gong will sound in the pilot's headset and the bleed air shut-off valve will close (Figure 70).

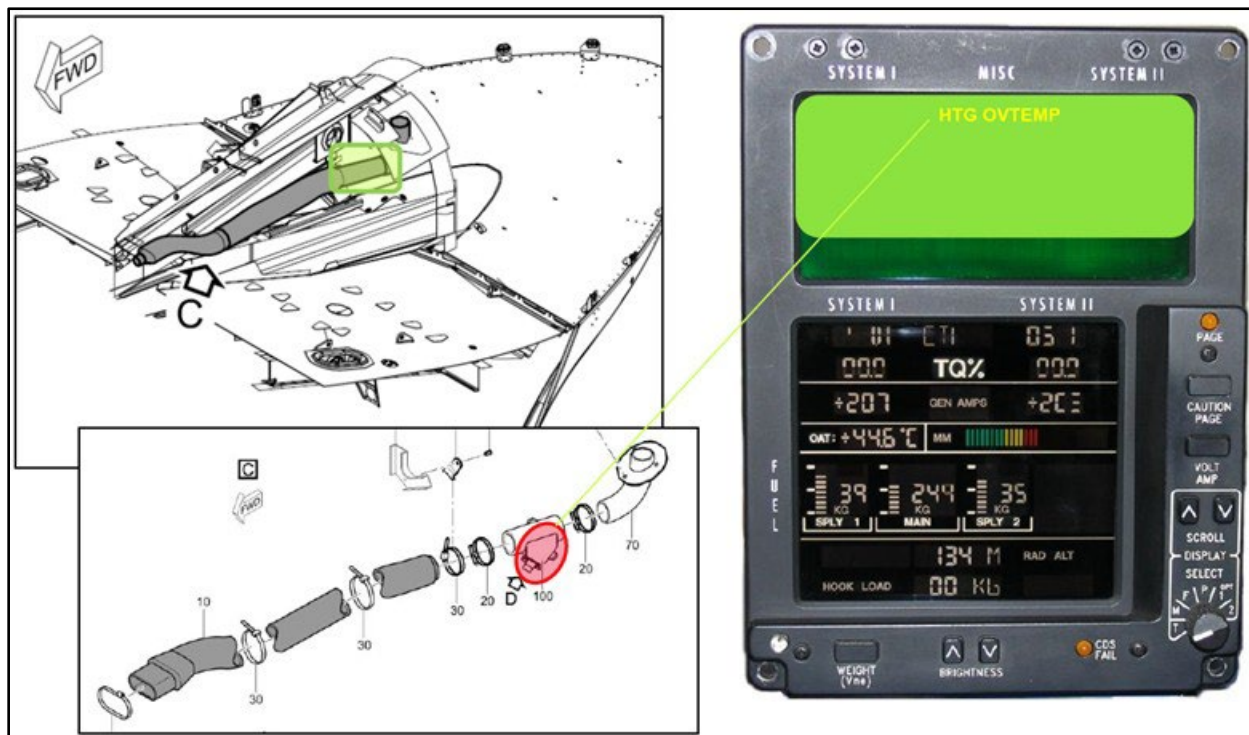


Figure 70. HTG OVTEMP Warning Message

(Figure Courtesy of Airbus Helicopters)

5.0 Electronic Engine Control Unit

5.1 Background/Description

The engine control system is made up of aircraft components, engine components and the EECU. The aircraft components include switches, buttons, indicators, warnings, a 28 volt power supply to the EECU, and start/stop selection logic. The engine components that make up the engine control system include both hydromechanical and electrical components. The hydromechanical components include the low pressure pump, a fuel filter, a high pressure pump, a metering unit (with manual control backup), a check valve, start injectors, and the main injection system. The electrical components include control sensors, indication sensors, a dedicated alternator, a stop electro-valve, a start electro-valve, a neutral position switch, an ignition unit, and the starter/generator⁹. The primary functions of the engine control system include engine starting, engine speed control, storing various

⁹ The starter/generator is considered a helicopter part, not an engine (SafranHE) component.

operational limits, overspeed protection, system monitoring and failure management, and providing a manual training mode.

The EECU hardware and the EECU operating system software (OSS) are provided by Triumph, who provided a system description of the EECUs under investigation as: these are full authority digital electronic control (FADEC), model EMC-35B, consisting of the electronic control unit (ECU) that interfaces with a hydro-mechanical fuel metering unit supplied by SafranHE. The EMC-35B fuel control operating system software (OSS) provides a flexible real-time operating environment and input/output hardware permitting primary electronic control of Safran turbo-shaft engines by SafranHE supplied control system software (CSS). The primary electronics will digitally control engine operation by modulating engine fuel flow, and controlling the various analog, discrete and frequency outputs as requested by the CSS. High power discrete outputs permit ECU control of starter and igniter relays. The separate on-line loader (OLL) software, provided by Triumph, supports EECU reprogramming without disassembling the ECU. The EMC-35B also provides an optional analog electronic over-speed limiter for limiting power turbine speed in the event of a loss of load but this feature was not available for older configurations as with the accident helicopter. The control system is capable of complete engine control in both single engine and twin-engine aircraft applications.

The EECU functions to maintain the PT speed (N2) and subsequently the helicopter main rotor speed (NR) at a nominal value (Figure 71). It does so by assessing the gas generator speed (N1) and the power delivered, which is then modified by adjusting the fuel flow to the engine.

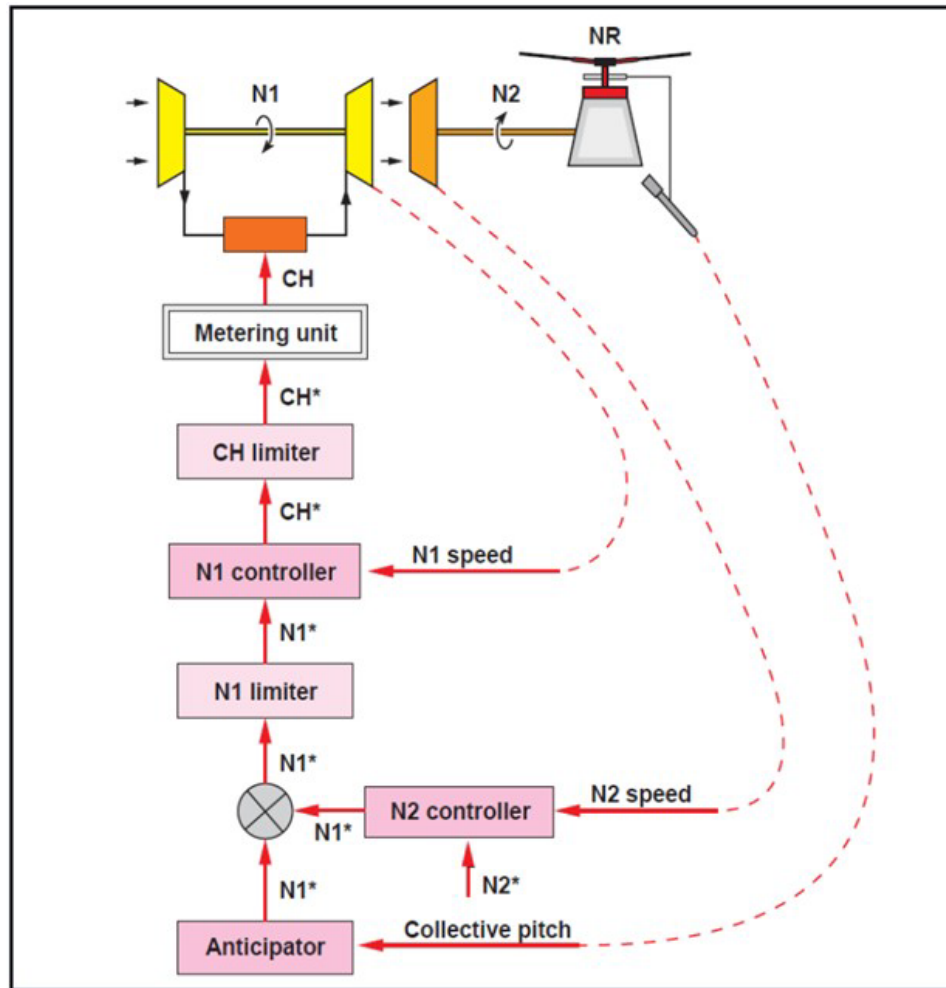


Figure 71. Engine Speed Control Diagram

(Figure Courtesy of SafranHE)

The EECU¹⁰ PN 70EMF01290, was manufactured by Goodrich¹¹, and is a digital single channel configuration, designed for engine control, engine operation monitoring and for use during maintenance activities. There is one EECU for each engine, connected to the engine by the P201 wiring harness, and each engine is linked to the helicopter's cautionary display system (CDS) for failure indications and failure messages through an RS-422 data link within the P202 wiring harness.

The J101/P101 indicating and control wiring harness provides the electrical link between the engine and the CAD for: N1C, N2C, electrical magnetic plugs M01 & M02, low oil & fuel pressure, oil & fuel pre-clogging indicator, T4.5 match, raw T4.5, fire detection, starter temperature, oil temperature and Pressure, supply of fuel start/stop electro-valve (EV), and high energy box.

¹⁰ Sometimes referred to as a FADEC (full authority digital electronic control) but the EECUs installed in N109BC are not full authority type control units.

¹¹ The current manufacturer is now Triumph

The P201 wiring harness provides the electrical link between the engine and its EECU for: N1A, N1B, N2A, N2B, raw T4.5, Torque (TQ), fuel metering unit resolver position (XR), fuel metering unit P3, alternator, neutral notch switch, T1, and raw T4.5.

The P202 wiring harness provides the electrical link between the EECU and the CAD for: torque, $\Delta N1$, discrete outputs, discrete inputs, pitch collective (XPC) and the RS-422.

The EECUs for both engines are mounted to the roof in the aft cabin of the avionics compartment (underneath the engine deck/compartments) (Figure 72 and Figure 73).

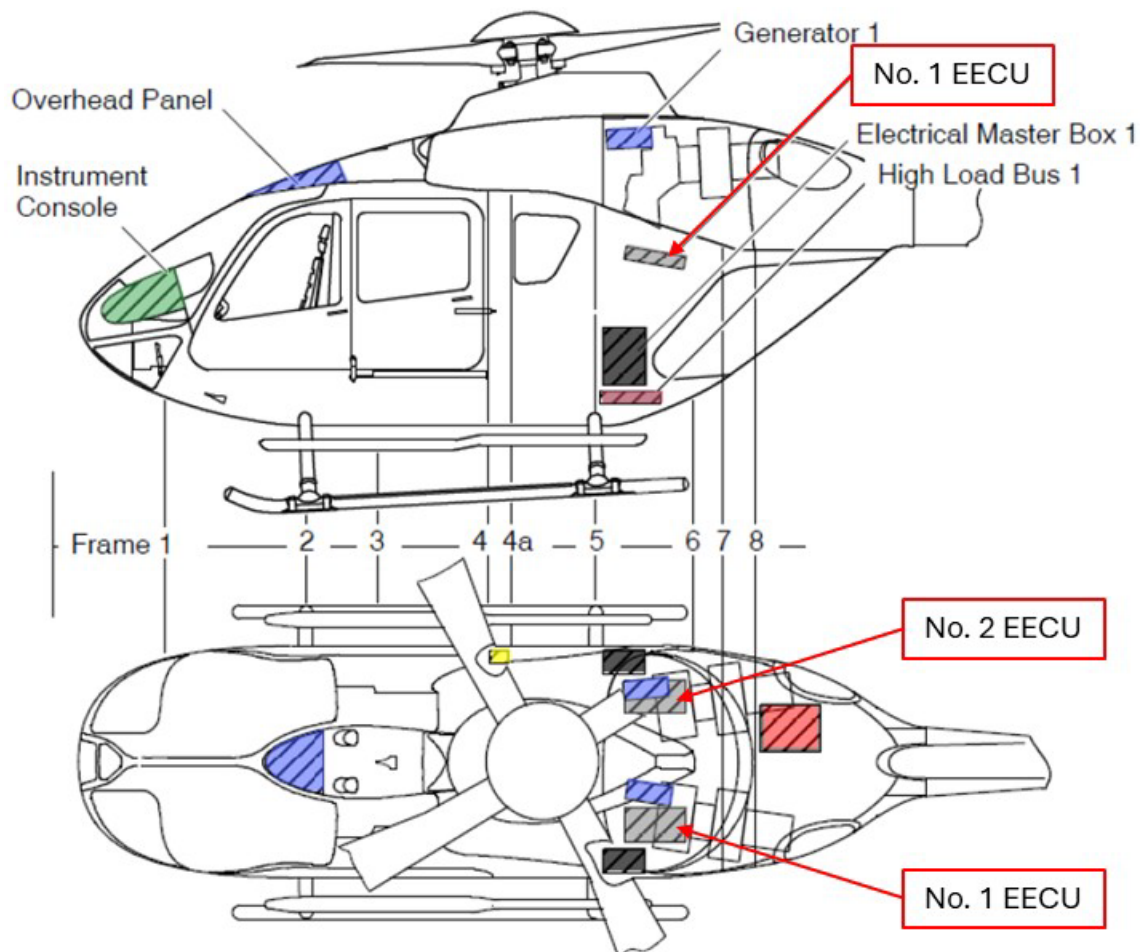


Figure 72. EECU Installation Locations

(Figure Courtesy of Airbus Helicopters)

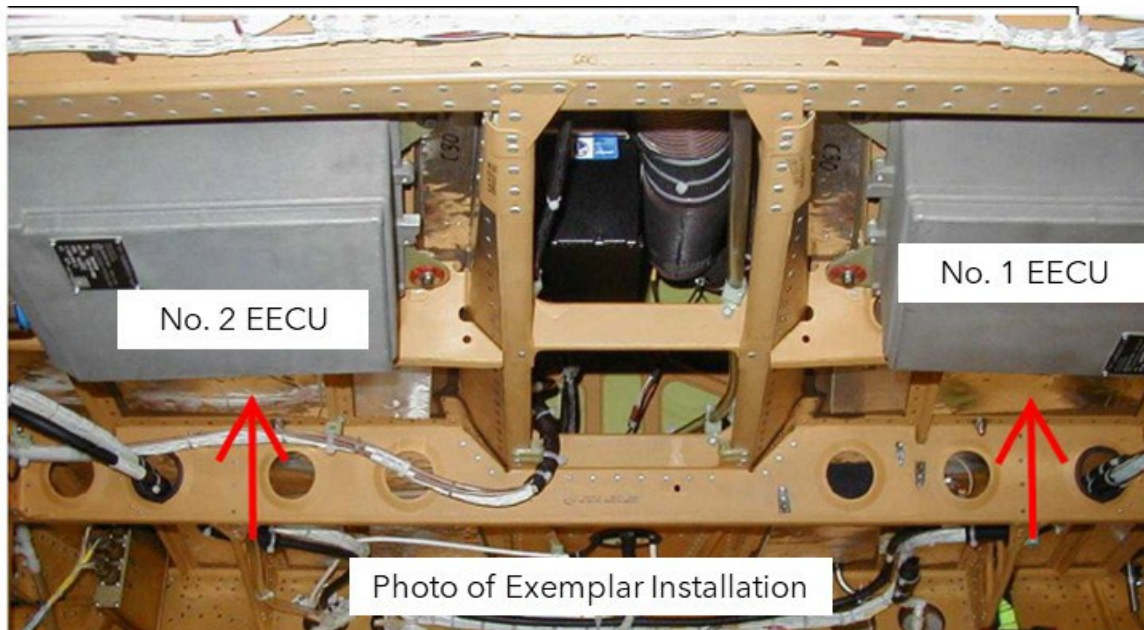


Figure 73. Exemplar Installation of EECUs

(Photo Courtesy of Airbus Helicopters)

5.2 EECU Inputs and Outputs

The inputs into the EECU and the EECU outputs for the EECU configuration in the accident helicopter are listed below in Table 9 and within the following diagram.

Table 9. EECU Inputs and Outputs

ELECTRONIC ENGINE CONTROL UNIT - INPUTS AND OUTPUTS	
Inputs from Aircraft Stop-Idle-Flight Selector Manual-Auto Selector 2B: N1 & N2 Trims 2B: FADEC On/Off 2B: Collective Pitch & Tail Rotor Pitch	Inputs from Engine T1 Temperature Probe Speed Sensors (N1, N2) Torque Confirmation Box P3 Pressure Transmitter T4.5 Thermocouple Probes Metering Needle Position Transmitter Neutral Notch Position Switch 2B: Torque Sensor
Outputs to Aircraft Start Accessory Relay Level 3/2/1 Failures Neutral Notch Position ΔN1 Indication OEI Operation Manual Mode Torque Indication	Outputs to Engine Fuel Metering Needle Stop Electro Valve P3 Pressure Transmitter Metering Needle Position (XMV) T1 Temperature Probe
Power Supply Aircraft 28 Volt DC Power Supply Engine Alternator (Backup Power Supply)	

5.3 EECU - Failure Modes

A primary function of the engine control monitoring system is to ensure proper fuel flow metering to the engine through the use of both primary laws (primary software logic) and backup laws (secondary software logic) in the case of a failure event. In the event of a failure of a primary control input (example double N1 failure) and depending on the level (e.g. Level 1, 2, or 3) of the failure event, the engine control system may automatically reconfigure to ensure the essential functions (primary control laws, N1 and N2 speed control, acceleration, and deceleration control) remain functional.

According to SafranHE, there are three categories of failure defined within the engine control monitoring system: a Level 1, a Level 2, and a Level 3. A Level 1 failure is associated with a minor anomaly such as a loss of a system/control redundancy where there is no effect on engine operation or fuel flow metering (example loss of one N1 sensor signal). For a Level 1 failure, the control system will continue using the primary control laws and the failure will be indicated on the cockpit display system (CDS) display with a message of "REDUND" meaning there has been a loss or failure of a redundant system.

A Level 2 failure is associated with a minor failure where the response time of the control system may be affected but the essential engine control functions are still functioning. For a Level 2 failure, the control system will use the necessary back up law if a failure affects a primary control law (example loss of the P3 signal). A Level 2 failure will be indicated on the CDS display as an amber caution with a message of "DEGRADE".

A Level 3 failure is a major failure, or a total failure (example loss of both N2 sensor signals). For a Level 3 failure, the control system will freeze the fuel flow stepper motor if the failure or failures affects a primary control law and the backup law. The fuel flow stepper motor will be frozen at the last set position and the Level 3 failure is indicated on the CDS display as an amber caution with a message of "FADEC FAIL".

The loss/failure of both N1 speed sensors or the loss/failure of both N2 speed sensors are considered a Level 3 failure. Likewise, the loss of communication with both N1 and N2 speed sensor signals or a compromised wiring harness between the speed sensors and the EECU would also be considered a Level 3 failure.

5.4 EECU - Data Parameters Recorded

An EECU is provided power from the helicopter when the EECU On/Off switch is set to the On position. Once an EECU is powered up, the internal time counter is

activated and tracks the EECU total time of operation in hours and also tracks the time since the EECU was last powered up (known as Power On Time parameter); the time counter stops when power to the EECU is turned off.

The EECU will automatically record engine parameters, the fuel control mode, the logic input/output state and failure messages into a maintenance block when an eligible event occurs. The EECU will record maintenance blocks into 4 different memory zones within the Electrically Erasable Programmable Read-Only Memory (EEPROM), allowing up to 11 individual maintenance blocks to be retrieved by directly reading the binary file from the EEPROM whereas only 8 of the maintenance blocks can be retrieved when using the standard dump software tool. From the EECU total time and power on time, the recording sequence of the maintenance blocks can be determined, with the last recorded block specifically identified by the software and the last memory zone used in the EEPROM also identified by a counter.

The EECU configuration used in the accident helicopter/engine configuration does not include an overlimit block or OEI (one engine inoperative) block recording function. Overspeed shutoff function is an optional feature for the 2B1 engine but was not installed on the accident helicopter.

5.5 EECU - Examination & Data Download

The EECUs were packaged and shipped to NTSB's Vehicle Recorders Laboratory in Washington D.C. for examination.

No. 1 Engine EECU

The NTSB's Vehicle Recorders Laboratory found the damage to the No. 1 engine EECU appeared limited to the exterior of the EECU and was repackaged and shipped to the Bureau d'Enquêtes et d'Analyses pour la Sécurité de l'Aviation Civile (BEA) in Paris, France for data download/recovery and decoding with the support of SafranHE.

The BEA performed X-ray and visual stereoscopy and found two components that were partially unsoldered from their circuit cards. Subsequent visual examination of the circuit cards revealed the circuit cards were generally in good condition, but with several notable observations. Attachment 1 contains BEA Technical Document No. BEA2023-0379_tec01, for additional information regarding observations during the X-ray inspection of the No. 1 engine EECU. Attachment 2 contains BEA Technical Document No. 2023-0379_tech04, for additional information regarding observations during the visual examination of the No. 1 engine EECU.

The memory chip from the No. 1 engine EECU containing the recorded maintenance block data was unsoldered from the circuit card to facilitate

download/recovery of the binary data that was stored on the chip. The recovered binary data was decoded by the BEA with the support of SafranHE, contained a total of 9 maintenance blocks of data. See APPENDIX 1 for a summary of the data recovered from the No. 1 engine EECU.

One pertinent data block, block No. 1, was recorded at an EECU Total Time of 5,194.6 hours and a Power On Time of 195 seconds (approximately 3 minutes, 15 seconds after the EECU was powered on for the accident flight). The information recorded in block No. 1 shows the engine was in flight mode and that a simultaneous double N1 and N2 failure (both are a Level 3 type failures) occurred. The data recoded in block No. 1 for the remaining parameters appear valid and are not the auto generated recovery values¹².

No. 2 EECU

The NTSB Vehicle Recorders Laboratory found the No. 2 engine EECU had sustained damage to its internal components. The nonvolatile memory (NVM) chip, P/N Amtel AT28C64, was removed and a binary image of the chip was created using laboratory tools. The binary image was provided to the BEA for decoding with the support of SafranHE.

The recovered binary data from the No. 2 engine EECU was decoded by the BEA and SafranHE. There was a total of 8 recorded maintenance blocks, 5 of the 8 recorded maintenance blocks were consistent with the accident flight, with the remaining 3 blocks unrelated to the accident flight. See APPENDIX 2 for a summary of the data recovered from the No. 2 EECU.

The 5 pertinent data blocks were recorded within a period of 40 seconds based on the EECU Power On Time (recorded in seconds). Failures were recorded for the following parameters: XPC, 28V power, T1, T4, P3, Level 1, Level 2 and External (APPENDIX 2).

According to BEA and SafranHE representatives, an XPC failure is typically the result of the helicopter impacting the ground.

6.0 Metallurgy

SafranHE performed metallurgical analyses of the engine hardware at their facility in Pau France with NTSB oversight of the workscope and review of the findings. The NTSB Laboratory performed metallurgical analyses on the engine inlets,

¹² Each EECU parameter has a defined range of values, but if the value is outside this range (non-valid) and/or in the event of a failure condition, a predefined value known as an auto recovery value will be used/recorded for that parameter.

engine compartment firewalls and the exhaust ejectors with those respective factual reports being available within the docket.

6.1 No. 1 Engine - Inlet and Compressor Hardware

The impeller, P/N 0319213400, contained areas of varying color discoloration: 1) a small area of blue-purple coloring on the backside of the impeller along the outer diameter (Figure 74), 2) an area of blue coloration on the rear bearing inner race, and 3) an area of yellow-blue discoloration next to the curvic coupling teeth. The aft curvic coupling teeth exhibited fretting wear and material deformation at all locations on the coupling teeth and around the whole circumference. The diameters of the 4 labyrinth seals on the front of the gas generator shaft were measured; the diameter values varied between 43.974 and 43.945 millimeters, and all were within the part drawing specification of 44.0 -0.5/+0 millimeters. The diameters of the 7 rear lips (rear and nearest the impeller) on the gas generator shaft were measured, the diameter values varied between 43.905 to 43.925 millimeters, which are within the part drawing specifications. The length (dimension denoted as W1 in CRM 24140) of the gas generator shaft was measured at 277.087 millimeters, which is within the part drawing specification of 277.5 -0.55/+0 millimeters. The length measurement performed by SafranHE on this part at the time of engine assembly was between 277.41 millimeters and 277.44 millimeters. Hardness measurements were made at various locations on the impeller in both the gas path and non-gas path for comparison. Hardness values varied between 470 HV¹³₃₀ and 480 HV₃₀. Material analysis of the impeller found the chemistry to be within the material specification. No metallurgical transformation or material abnormalities were observed during metallographic analysis of the impeller at various locations (Figure 75).

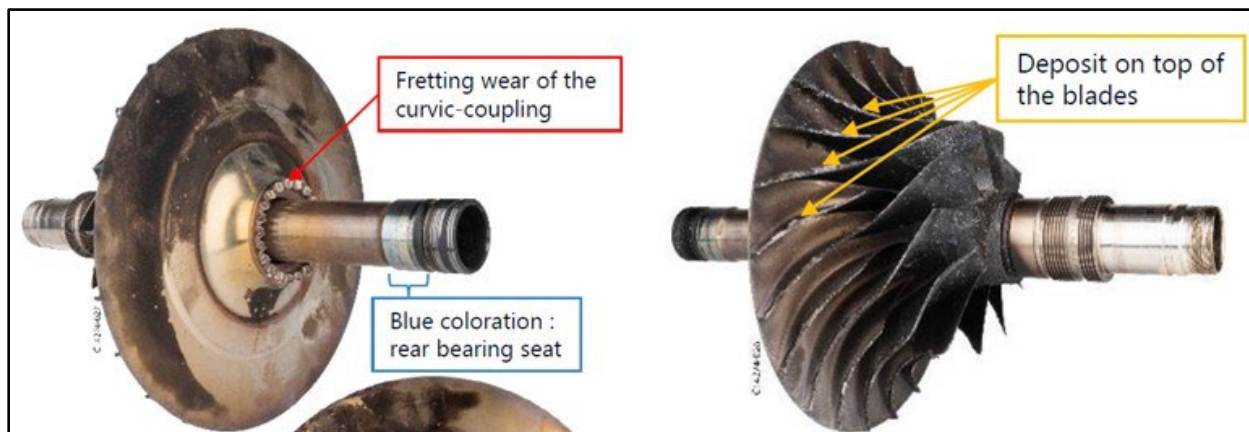


Figure 74. Impeller

(Figure Courtesy of SafranHE)

¹³ The 470 is the harness value, HV means per the Vicker's scale, and 30 means using a test load of 30kilogram-force to make the impression.

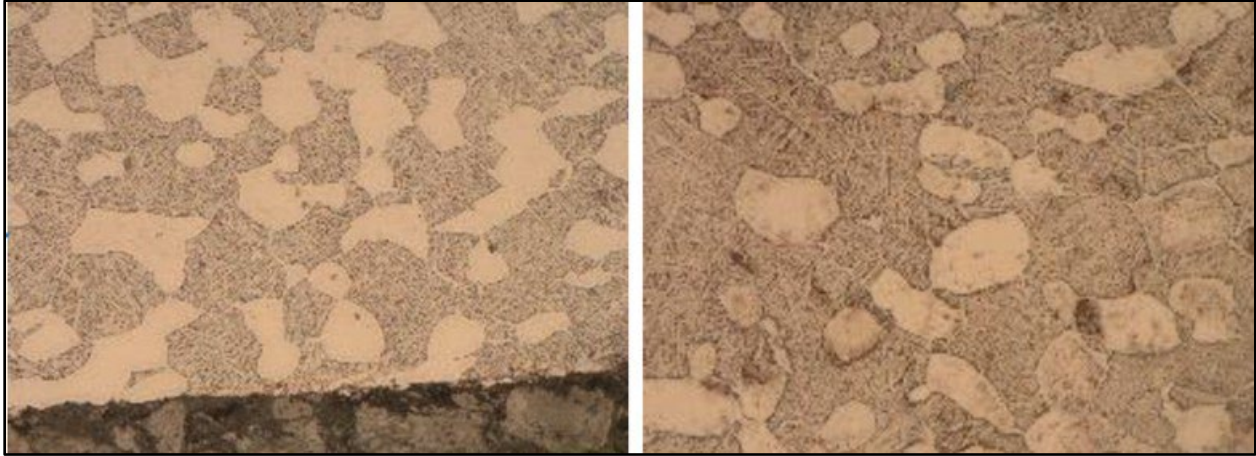


Figure 75. Impeller Metallography - Exit (L), Near Curvic Coupling (R)
(Figure Courtesy of SafranHE, Etchant: hydrofluoric acid reagent)

The centrifugal diffuser, P/N 0319210050, a martensitic stainless steel alloy, contained an area of pinkish discoloration around inner flange (Figure 76). Material analysis of the rear outer flange surface found the presence of a singular microstructure with fewer and smaller carbides with finer martensitic needles as compared to the nominal material specification. A black spot, approximately 15 millimeters in diameter, was present at the 5:00 o'clock position on the external ferrule surface, which corresponds to the location of the air tube that transmits compressor discharge pressure/air (P3 pressure) to the P3 sensor and HMU.

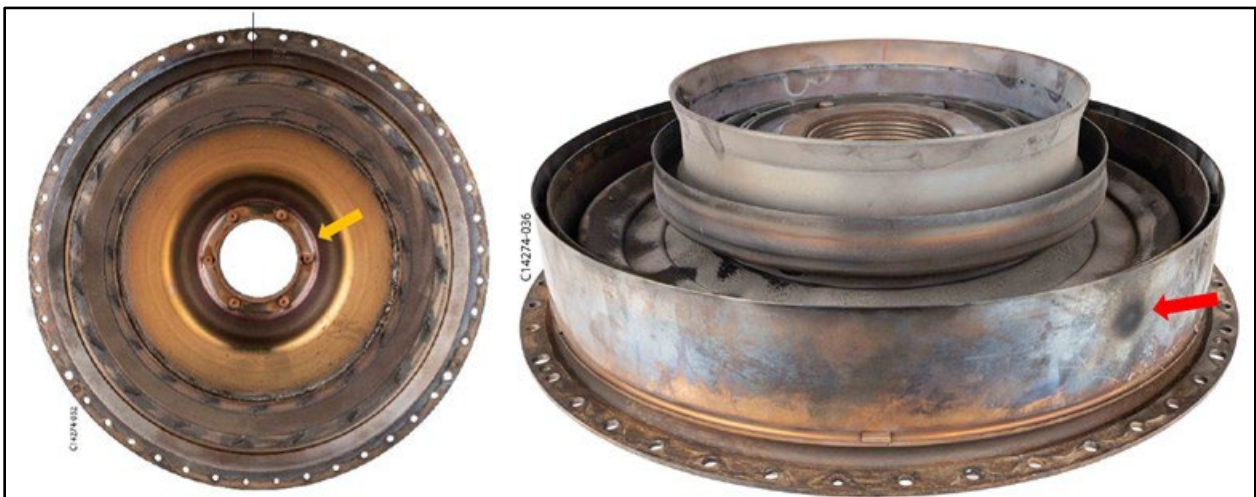


Figure 76. Compressor Diffuser
(Figure Courtesy of SafranHE)

6.2 No. 1 Engine - Gas Generator Turbine Hardware

Material analysis of the gas generator turbine shroud, a nickel-based alloy, found the presence of a singular microstructure with tangled eta phase needles (Figure 77).

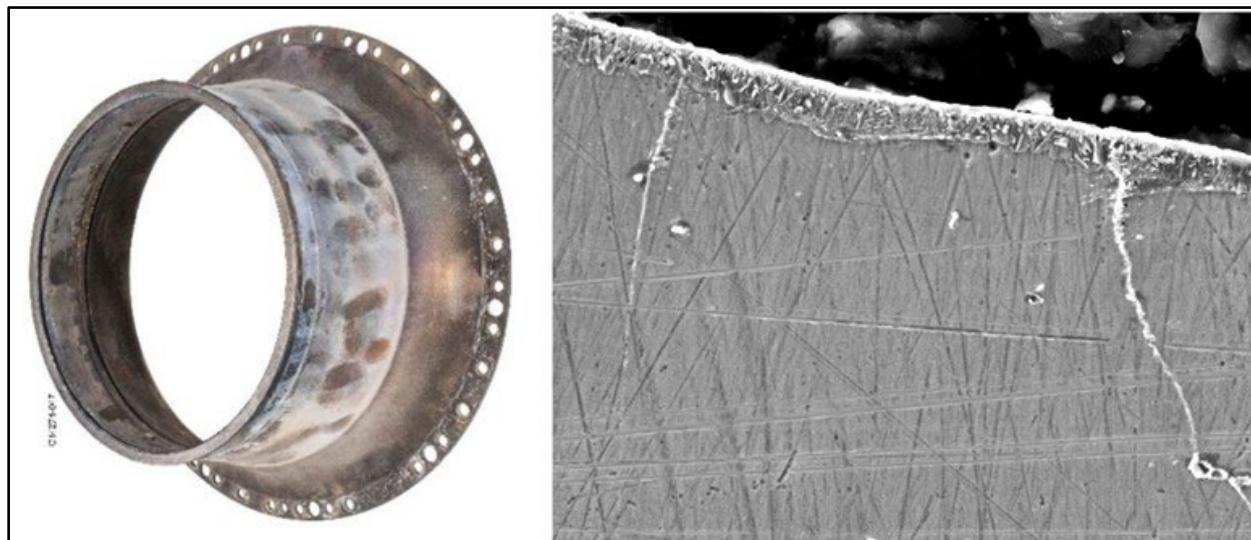


Figure 77. Gas Generator Turbine Shroud

(Figure Courtesy of SafranHE)

Borescope inspection of the combustion chamber flame tube assembly found all pre-vaporization tubes to be in good condition and there were no obstructions.

Examination of the gas generator turbine NGVs, P/N 0319217780, a nickel-based alloy, found the presence of multiple cracks on the inner air flow ring and all visible cooling holes and paths appeared free of obstruction. Multiple vanes exhibited both impact and thermal damage to the airfoil trailing edges. Axisymmetric circumferential rubbing of the labyrinth rings was observed. Two NGVs exhibited cracks at the inboard side of the airfoil leading edge. One of the two cracks was excised and opened for the purpose of performing fractography. The fractographic examination found a dark, oxidized and grainy fracture surface consistent with an interdendritic fracture morphology. Thermal and secondary damage was consistent with post cracking damage.

Metallurgical analysis was performed on the NGV airfoils, a nickel based alloy, at the 6:00 and 12:00 o'clock positions. The outer surface of the airfoils exhibited microstructural transformation to varying degrees but the largest transformation was observed at the trailing edge of the airfoils (Figure 78, Figure 79, and Figure 80). The inner walls of the airfoils exhibited dissolution of material precipitates within a band of approximately 5 to 10 micrometers in depth (Figure 81 and Figure 80). The microstructural transformation was due to an overheating/over temperature condition.

Based on the material's properties and a solvus temperature of 1,132 degrees C, this layer of gamma prime dissolution is indicative of a gas path temperature that is at least 300 degrees C above the typical temperature calculated for a take-off power setting.

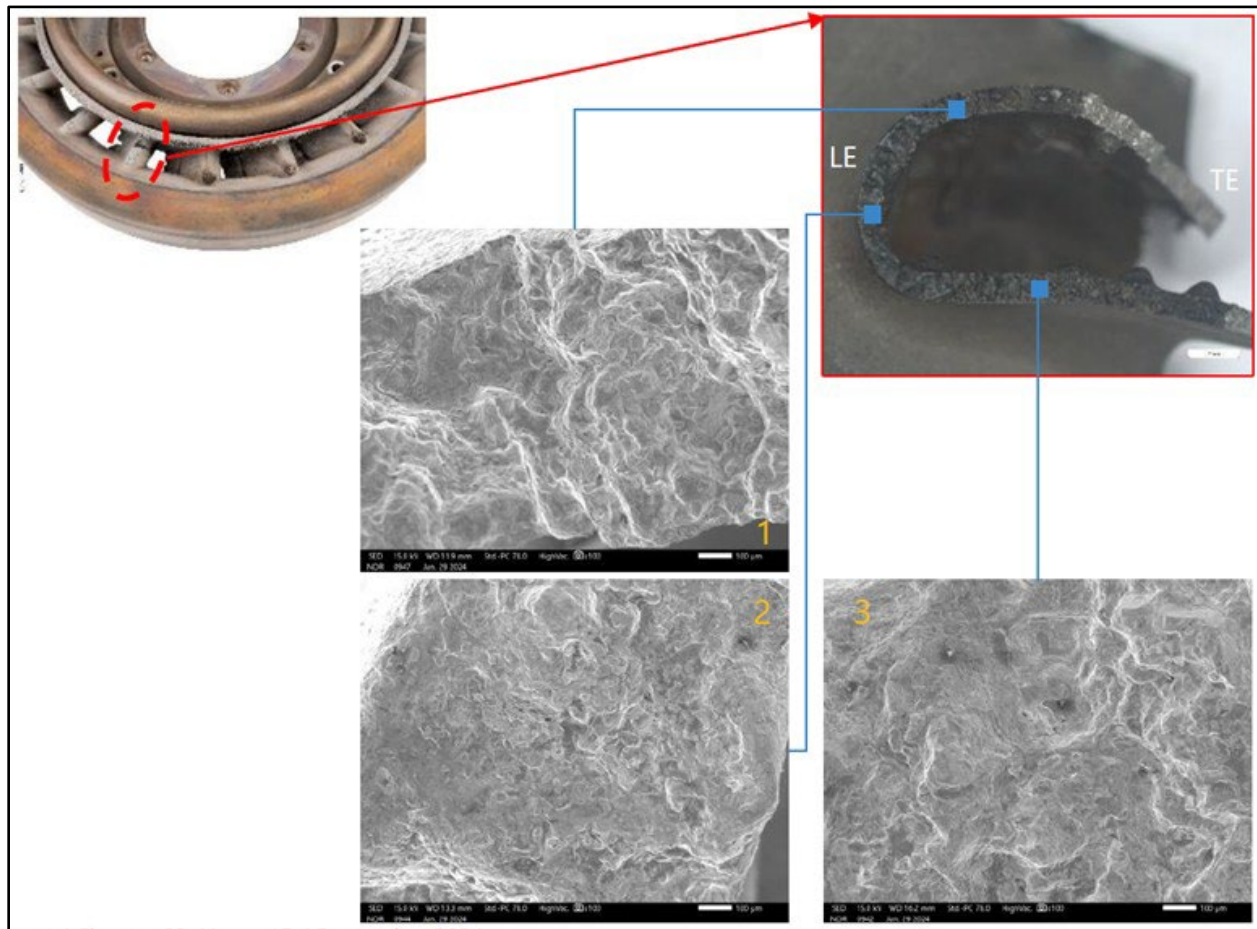


Figure 78. Gas Generator Turbine NGV Metallography - Cracked Vane
(Figure Courtesy of SafranHE, Etchant copper sulfate reagent)

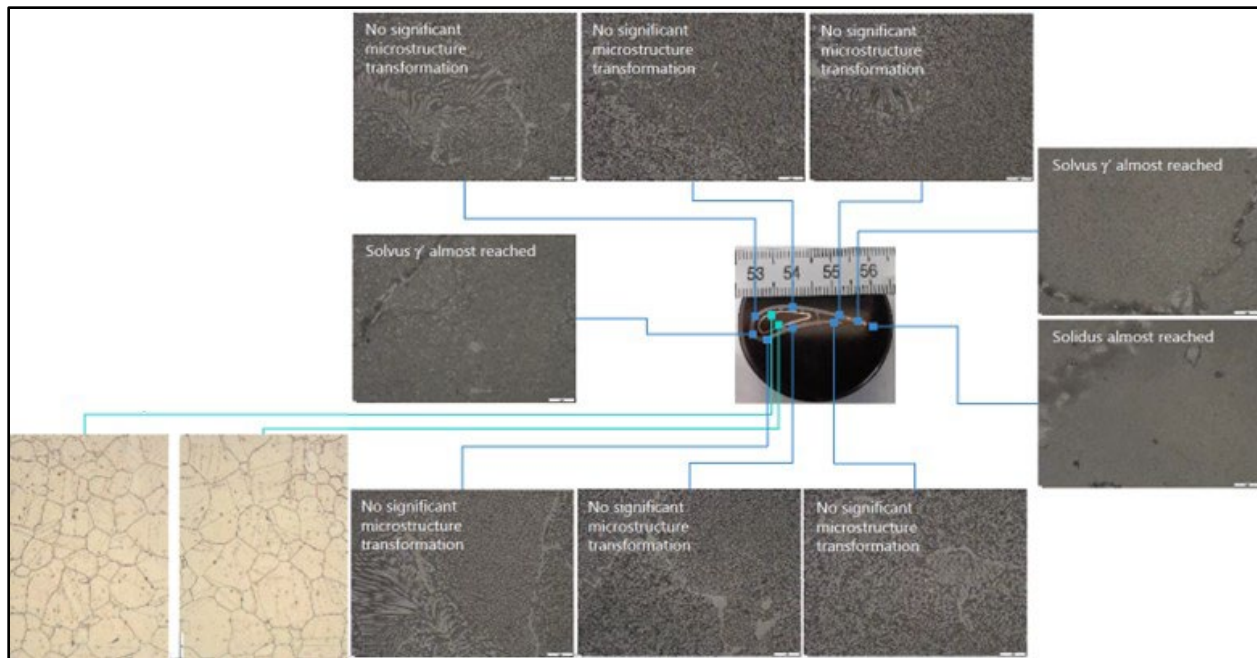


Figure 79. Gas Generator Turbine NGV Metallography - Vane at 6:00 O'clock
(Figure Courtesy of SafranHE)

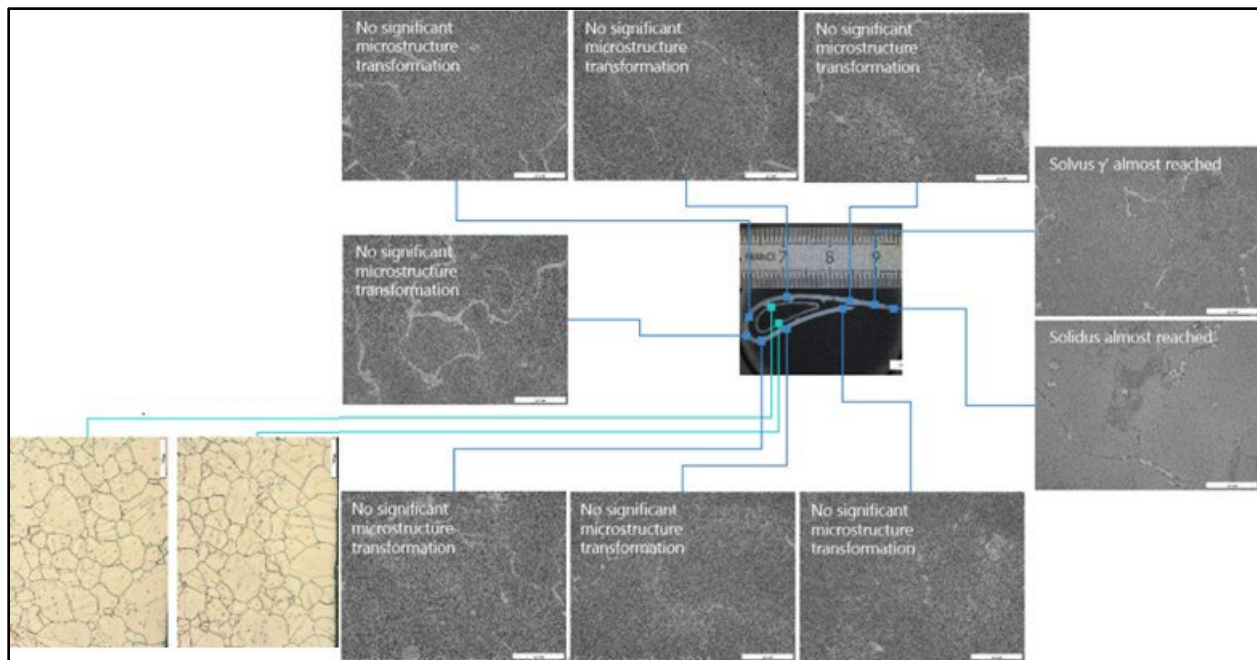


Figure 80. Gas Generator Turbine NGV Metallography - Vane at 12:00 O'clock
(Figure Courtesy of SafranHE, Etchant: copper sulfate reagent)

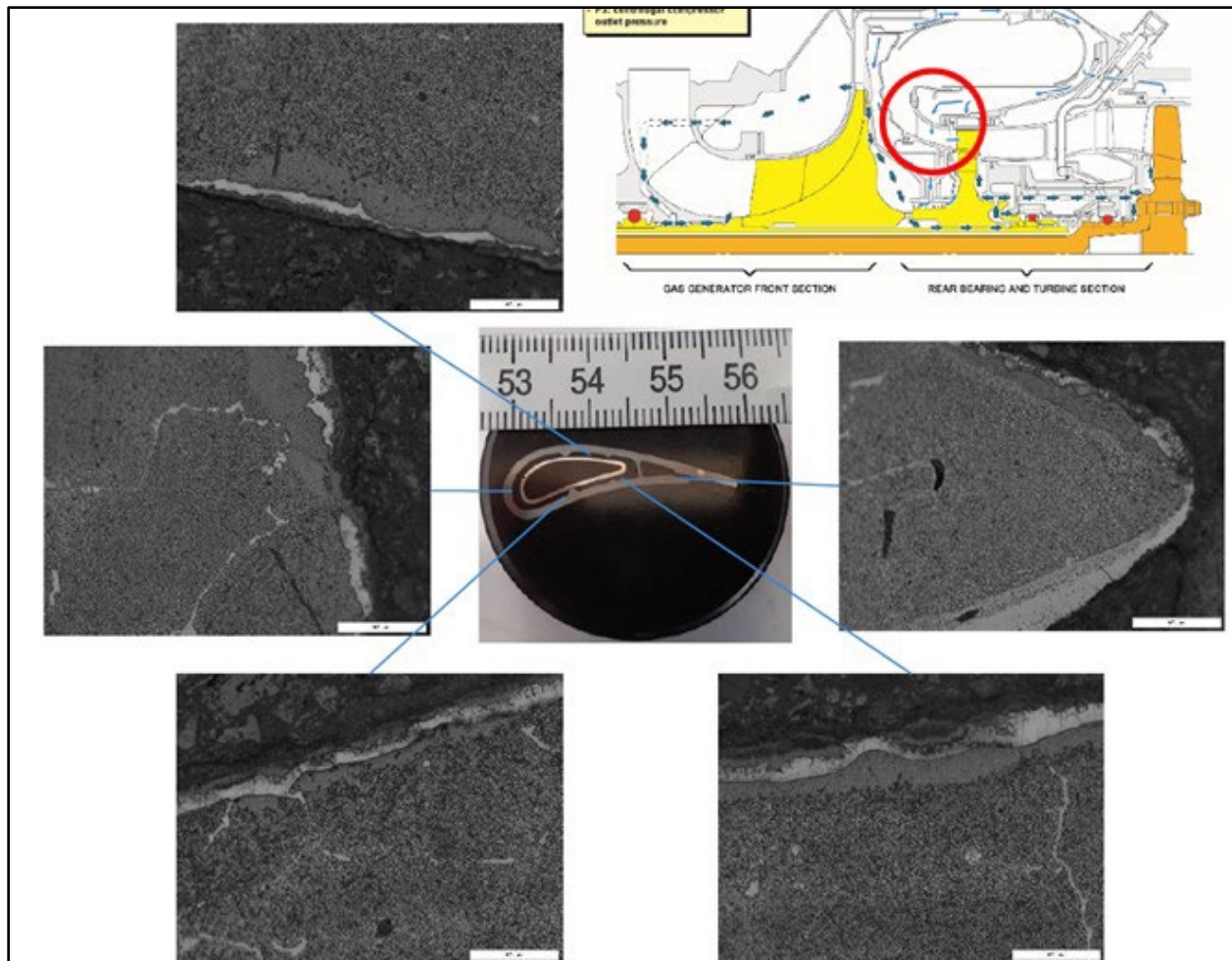


Figure 81. Gas Generator Turbine NGV Metallography - Vane Inner Walls
(Figure Courtesy of SafranHE, Etchant: copper sulfate reagent)

The turbine case, P/N 0319257210, a nickel based alloy, exhibited an impact mark at approximately the 11:00 o'clock position which corresponds to the perforation in the PT shroud (Figure 82). The inner diameter of the turbine case on the forward side exhibited an area of yellow to blue discoloration



Figure 82. Turbine Case (L), Impact Mark at PT Containment Ring (R)
(Figure Courtesy of SafranHE)

The gas generator turbine disk, P/N 2319212280, is a nickel-based alloy composed mainly of grains with gamma double prime precipitates.

The front side of the gas generator turbine disk exhibited multiple areas of discoloration. The diameter next to the curvic coupling teeth exhibited a purple discoloration and the diameter material between the labyrinth seals exhibited a purple discoloration (nearest the curvic coupling) that transitioned to a blue discoloration at the rear of the last front labyrinth seal (Figure 83). The rear side of the gas generator turbine disk exhibited a blue discoloration on the face of the disk between the rear labyrinth seals and the hub. Based on the total time of the event flight, the blue and purple discolored regions indicate an engine operating temperature that was at least 120 degrees C above the typical operating temperature (calculated temperatures at each location). Metallurgical analysis of the gas generator turbine disk surface found a band of delta phase dissolution that was approximately 1 micrometer in depth which, based on the material properties, is indicative of a temperature above 1,000 degrees C for a very short period of time (Figure 84).

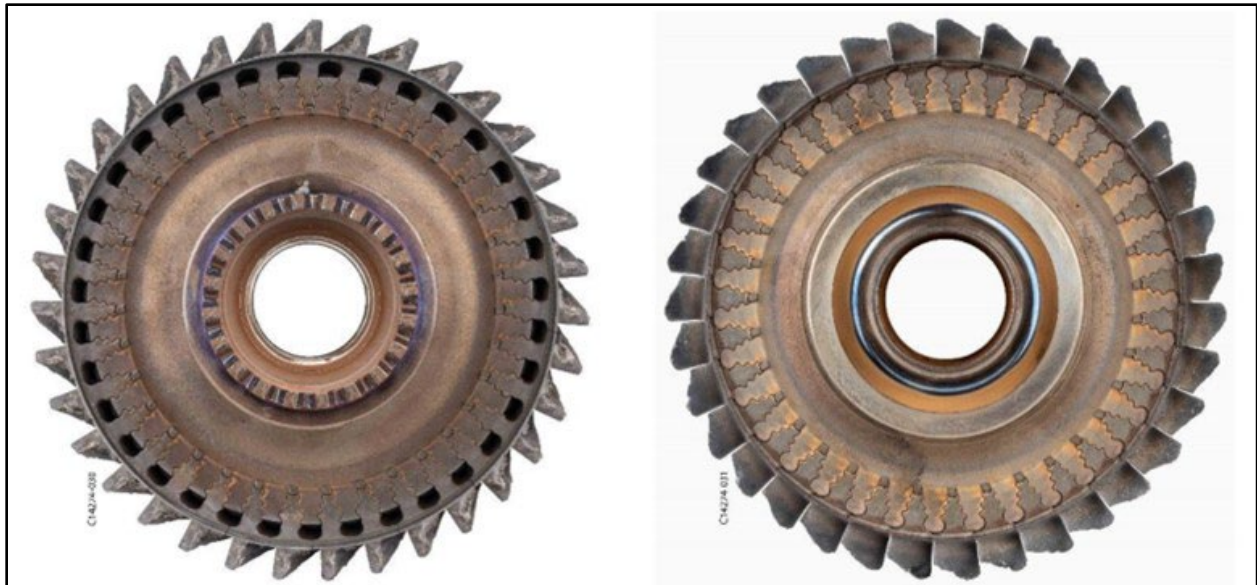


Figure 83. Gas Generator Turbine Disk - FWD Side (L), AFT Side (R)
(Figure Courtesy of SafranHE)

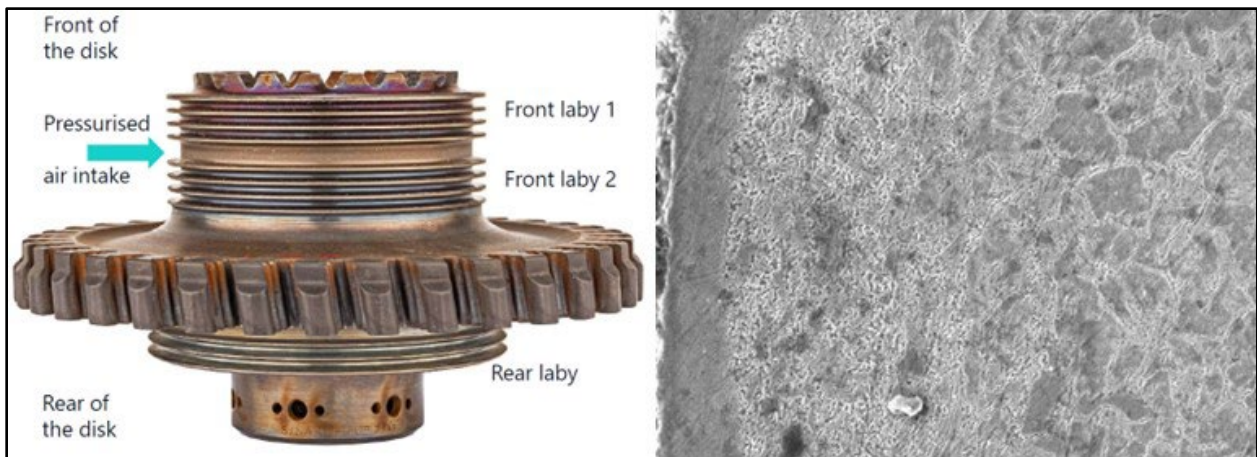


Figure 84. Gas Generator Turbine Disk - Side View (L), SEM Image of Disk Face Surface (R)

(Figure Courtesy of SafranHE, Etchant: CuSO4 reagent)

The gas generator turbine blades, P/N 231921A0E0, made of a single crystal nickel alloy, were all present with the blade locks still installed. All blades were fractured at approximately mid span height, and the fracture surfaces were consistent with overtemperature fractures. Various deposits that appeared white, grey, and black were found on the fracture surfaces.

Metallurgical analysis of the No. 11 blade, S/N 45988, was performed below the blade platform and found a portion of the forward and aft faces of the blade root exhibited dissolution of gamma prime precipitate within a depth of 5 to 10 micrometers (Figure 85). Based on the material's properties and a solvus temperature

of 1,295 degrees C, this layer of gamma prime dissolution is indicative of an operating temperature above 1,295 degrees C for a very short period of time.

The airfoil region exhibited inter-dendritic melting at the fracture surface, the total dissolution of the gamma prime precipitates at a location on the airfoil near the fracture surface and similarly indicates a gas path temperature that is at least 400 degrees C above the typical temperature calculated for a take-off power setting. The nominal microstructure of dense single crystal cuboidal gamma/gamma prime structure was found on the airfoil near the blade fillet (Figure 86).

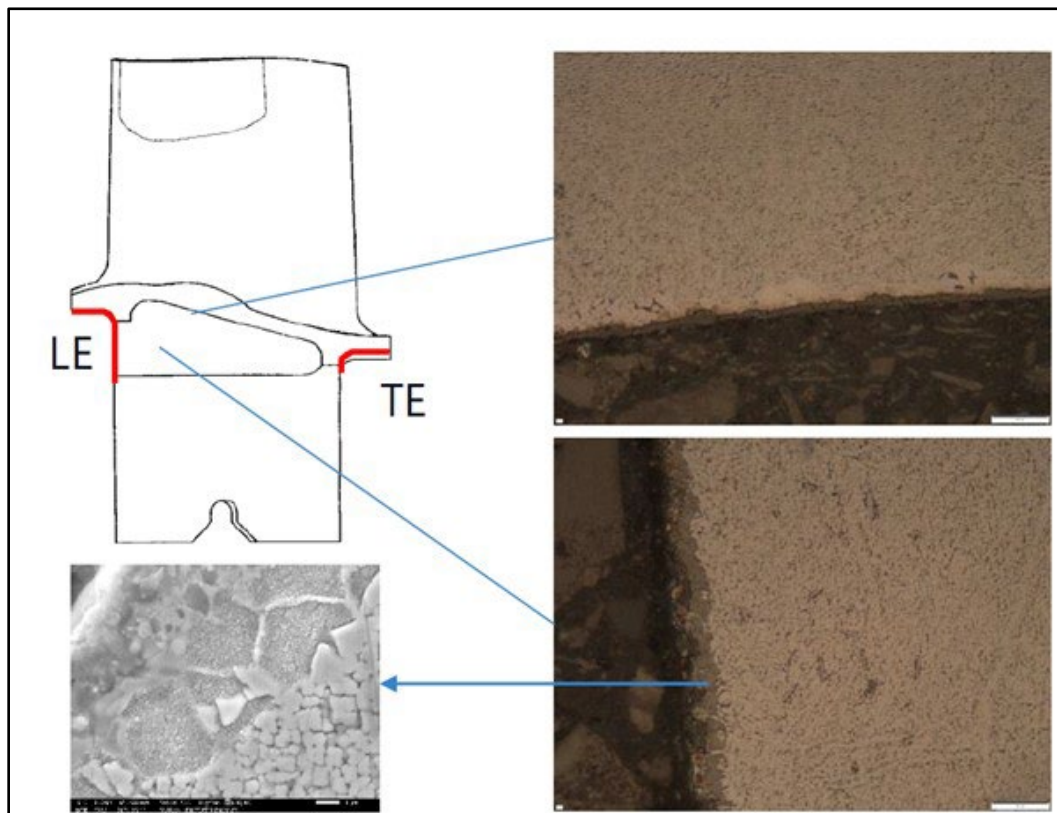


Figure 85. Gas Generator Turbine Blade S/N 45988 Metallography - Below Platform
(Figure Courtesy of SafranHE, Etchant: copper sulfate reagent)

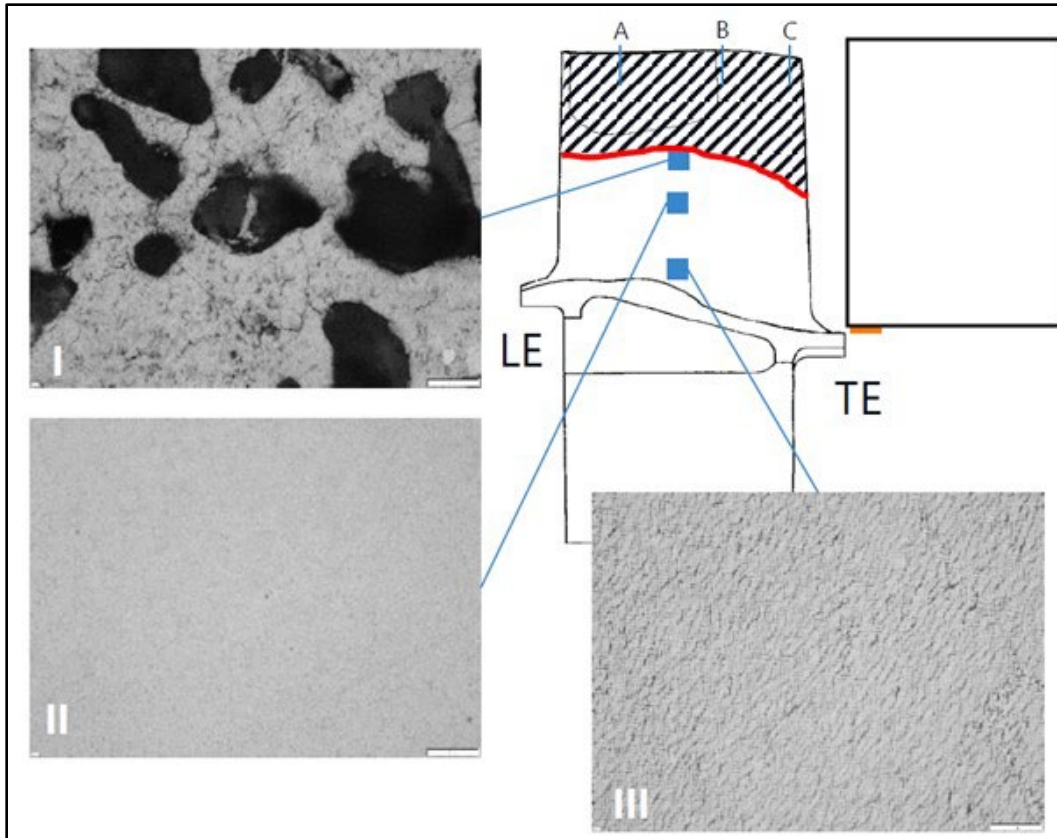


Figure 86. Gas Generator Turbine Blade S/N 45988 Metallography - Above Platform
(Figure Courtesy of SafranHE, Etchant: copper sulfate reagent)

The rear bearing support, P/N 0319250420, was hardness tested at four locations, equidistant apart, on the inner front flange. The hardness values at the 9:00, 12:00, and 3:00 o'clock positions measured 460 HV₃₀ and the hardness value at the 6:00 o'clock position measured 400 HV₃₀.

6.3 No. 1 Engine - Power Turbine Hardware

The PT NGV, P/N 0319250420, a nickel based alloy, was hardness tested at multiple locations on the airfoils at both the 12:00 and 6:00 o'clock positions as well as the inner flange at the same clock positions. The hardness values of the airfoil and the inner flange at the 12:00 o'clock were approximately 190 HV₁₀. The hardness value of the vane airfoil and the inner flange at the 6:00 o'clock were approximately 200 HV₁₀. For comparison, the hardness test performed on the No. 2 engine's PT NGV inner flange, resulted in approximately 240 HV₁₀, indicating the No. 1 engine's PT NGV was exposed to elevated temperatures that reduced the material's typical hardness.

The PT disk, P/N 2319411650, a nickel based alloy, exhibited a blue to purple discoloration of the fir trees around the entire circumference (Figure 87). Metallurgical analysis of the disk on the front face near the fir trees showed evidence of total dissolution of the gamma prime precipitates on the front and rear faces of fir trees to a depth of approximately 1.0 micrometer. Based on the material's properties, this layer of gamma prime dissolution is indicative of an operating temperature above 1,000 degrees C for a very short period of time. Hardness measurements were made at multiple locations on both the front and rear sides that included the fir tree faces (top and bottom) and the disk center. The hardness measurements on the front face of the fir trees were between 550 HV and 560 HV₃₀, the hardness measurements on the rear face of the fir trees were between 520 HV₃₀ and 530 HV₃₀, the hardness measurements on the front disk center were approximately 470 HV₃₀, and the hardness measurements on the rear disk center were approximately 475 HV₃₀.



Figure 87. No. 1 Engine PT Disk and Blades

The PT blade set was comprised of 31 power turbine blades, P/N 231941A0D0, a nickel based alloy, with 5 out of the 31 blades fractured. The five fractured blades

were: blade No. 4, S/N 164640,, blade No. 18, S/N 164644, blade No. 23, S/N 164635, blade No. 24, S/N 16464, and blade No. 29, S/N 164643 (Figure 88).

The fractographic characteristics of all five blades were similar (Figure 88). The fatigue cracking initiated under the blade platforms in the shank region at the leading edge of the pressure side. The initiation sites were mainly crystallographic in nature, consistent with fatigue and cyclic loading, but determination between high cycle fatigue (HCF) and low cycle fatigue (LCF) was not possible.

Initial macro-fractographic inspection of the 5 fractured blades with a binocular magnifying glass found the initiation sites and fatigue zones were oxidized, there were no observable front line across the fracture surface, and crystallographic planes were present at the initiation site. Electron dispersive spectroscopy (EDS) analysis of the initiation site showed no chemical abnormalities.

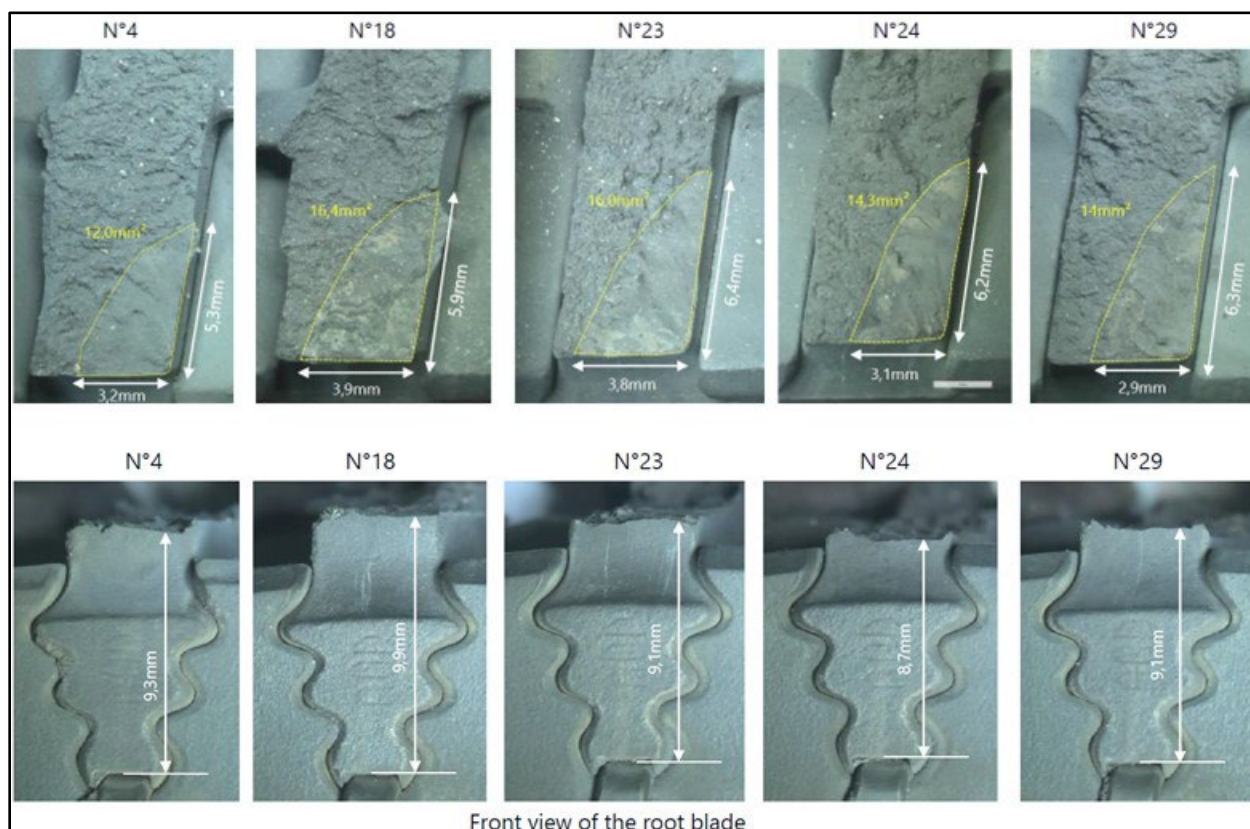


Figure 88. Fractured PT Blades

(Figure Courtesy of SafranHE)

The fracture surfaces were cleaned by brushing with Ethylic Alcohol and then deoxidized with a 5% solution of Alconox® at 80 degrees C for 45 minutes. The following visual inspection found that most of the oxide layer was removed, large striations were still visible, the fine striations were less visible due to the deoxidation, and the gamma/gamma prime microstructure was visible. Following cleaning and

deoxidation, fractography using field emission gun – scanning electron microscope) (FEG-SEM) found areas of large (wide) striations of irregular spacing near the crack propagation front and fine striations were observed throughout the fracture surface. Chemical analysis of the initiation sites using energy dispersive spectroscopy (EDS) found no evidence of any metallurgical abnormalities but phosphorus was present on all fracture surfaces.

Fluorescent penetrant inspection was performed on 24 of the 31 blades (the other 7 blades included the 5 broken blades and 2 other blades, these were set aside for preservation purposes). Crack indications were observed within the fir tree fillets on all 24 blades; however, S/Ns 164622, 164626, and 164631, had crack indications in the shank region on the leading edge of pressure side, with crack lengths measured between 0.5 and 2 millimeters, these indications do not meet the acceptance criteria defined in ACMMB repair task 72-42-22-222-801-A01.

The crack located in the shank of blade S/N 164631 (blade No. 20) was excised and opened under controlled lab conditions to perform fractography of the crack faces, and this was performed with representatives from the BEA and Airbus in attendance. Fractographic analysis found crystallographic facets, consistent with initiation sites of the 5 broken blades; no characteristics of fatigue propagation were observed (Figure 89).

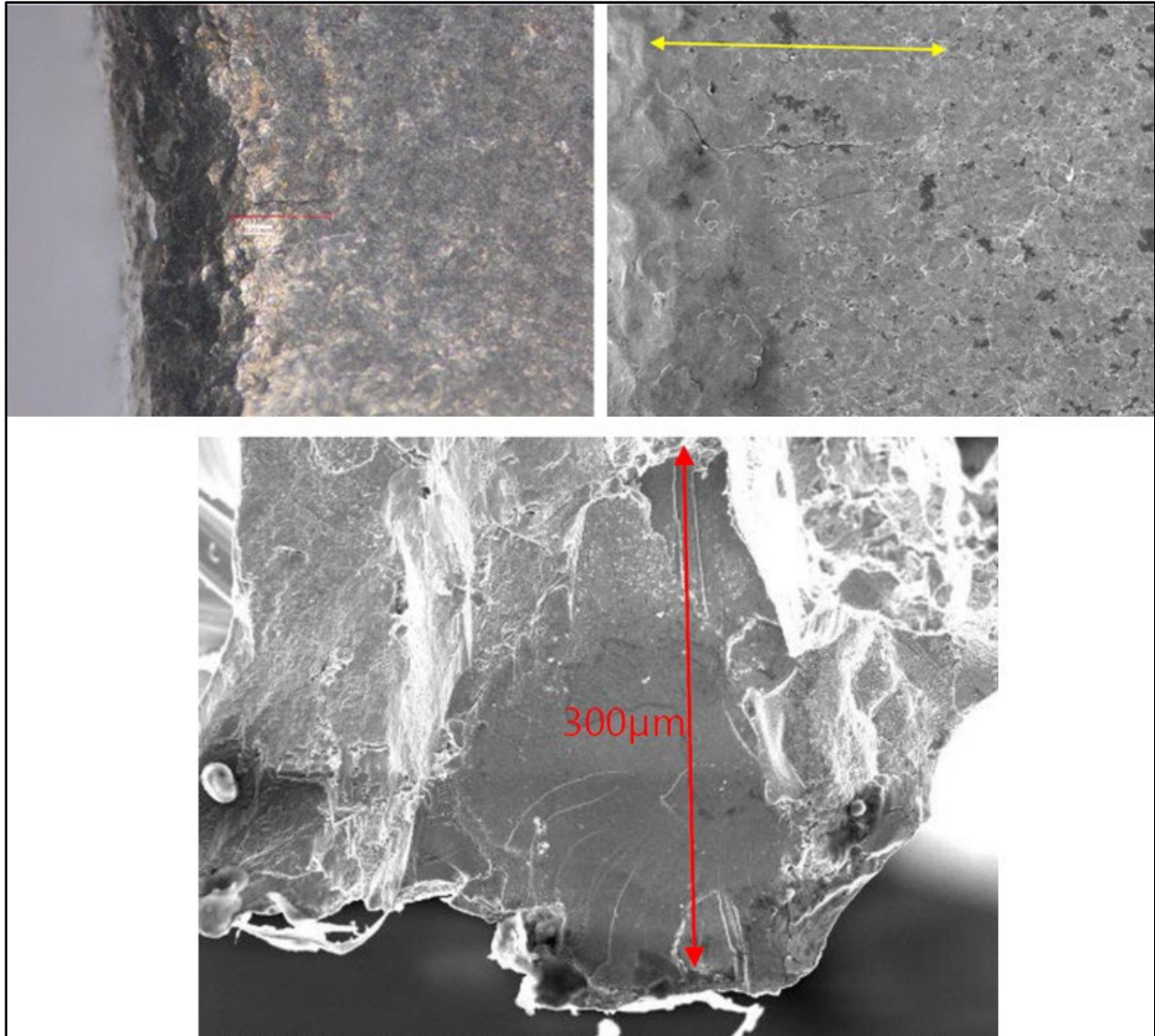


Figure 89. PT Blade S/N 164631

(Figure Courtesy of SafranHE)

Metallurgical analysis was performed on blade S/N 164637 (blade No. 21), which found the total dissolution of the material/alloy precipitates and eutectic phases at the fracture surface and significant dissolution of the material/alloy precipitates and eutectic phases at a location above the blade platform (approximately mid-span and mid-chord) (Figure 90). Based on the material's properties and a solvus temperature of 1,132 degrees C, this dissolution layer indicates a gas path temperature that is at least 400 degrees C above the typical temperature calculated for a take-off power setting. The nominal microstructure of dense gamma/gamma prime structure was found at a location on the airfoil near the blade fillet.

Metallurgical analysis was performed on blade S/N 164628 (No. 12), S/N 164649 (No. 29), S/N 164640 (No. 4), and S/N 164644 (No. 18), which found dissolution of

gamma prime precipitates within a band of approximately 5 to 15 micrometers in depth on the blade root surfaces at both the leading and trailing edge sides (Figure 91). Similarly, this dissolution indicates a gas path temperature that exceeds the material's solvus temperature.

The blade root of blade S/N 164640 (No. 4) was sectioned and metallurgical analysis performed across the root profile just below the fracture surface found the nominal microstructure of dense gamma/gamma prime structure, with no dissolution of the gamma prime precipitates observed (Figure 93).

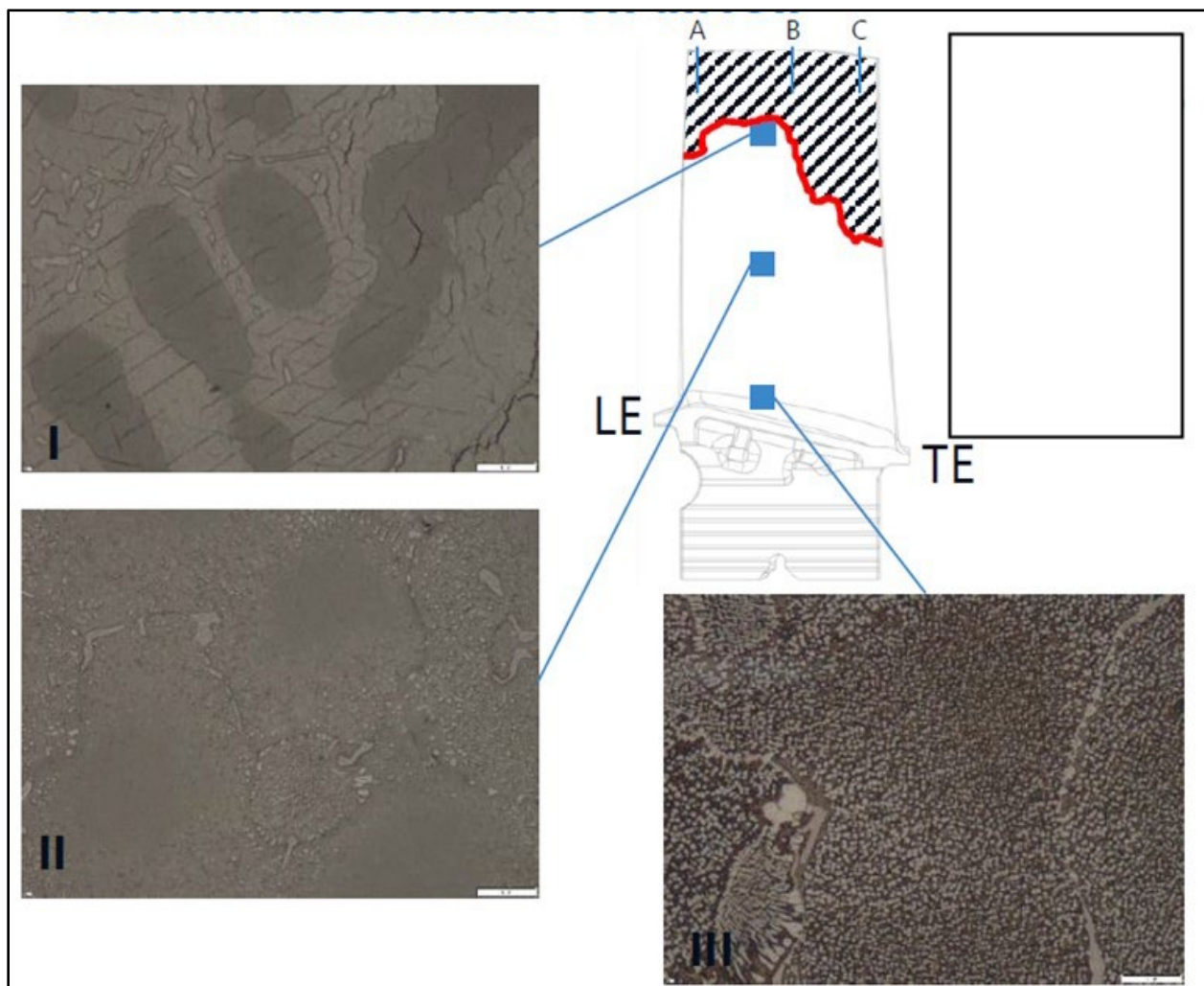


Figure 90. PT Blade S/N 164637 Metallography - Above Platform

(Figure Courtesy of SafranHE, Etchant: copper sulfate reagent)

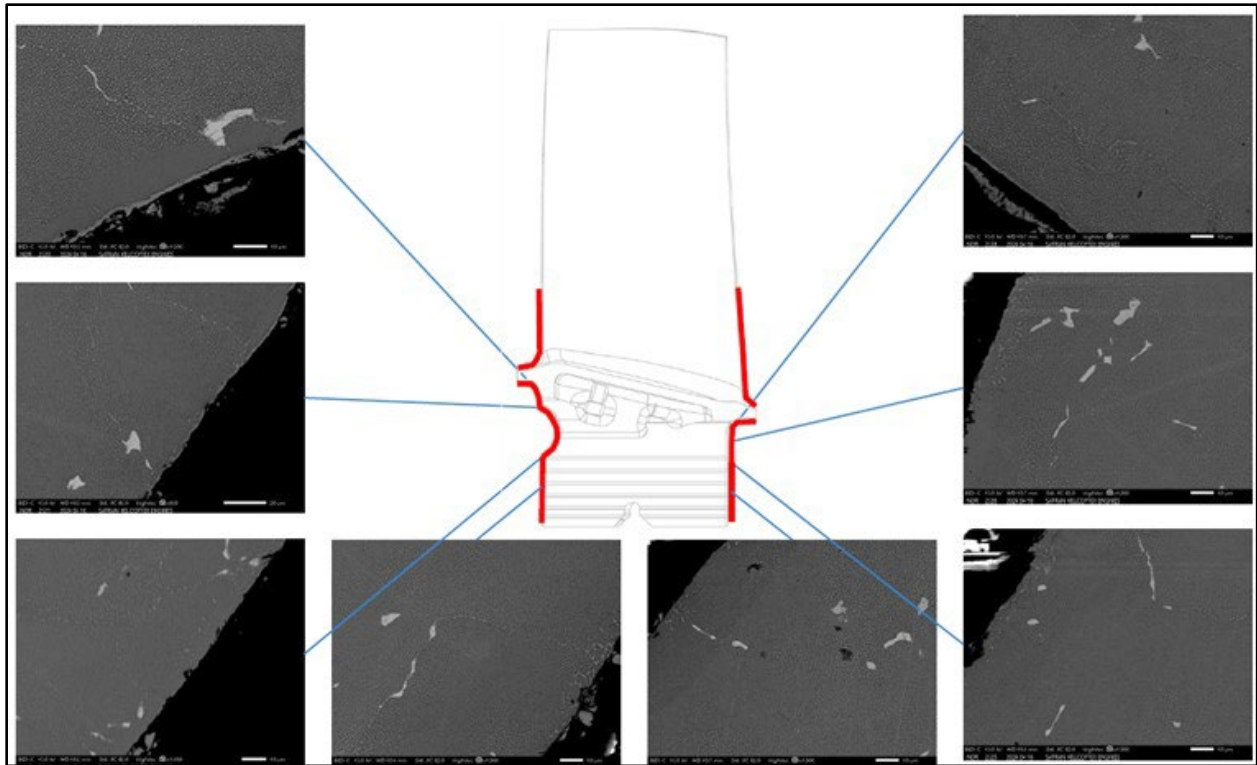


Figure 91. PT Blade S/N 164628 Metallography - Below Platform
(Figure Courtesy of SafranHE, Etchant: copper sulfate reagent)

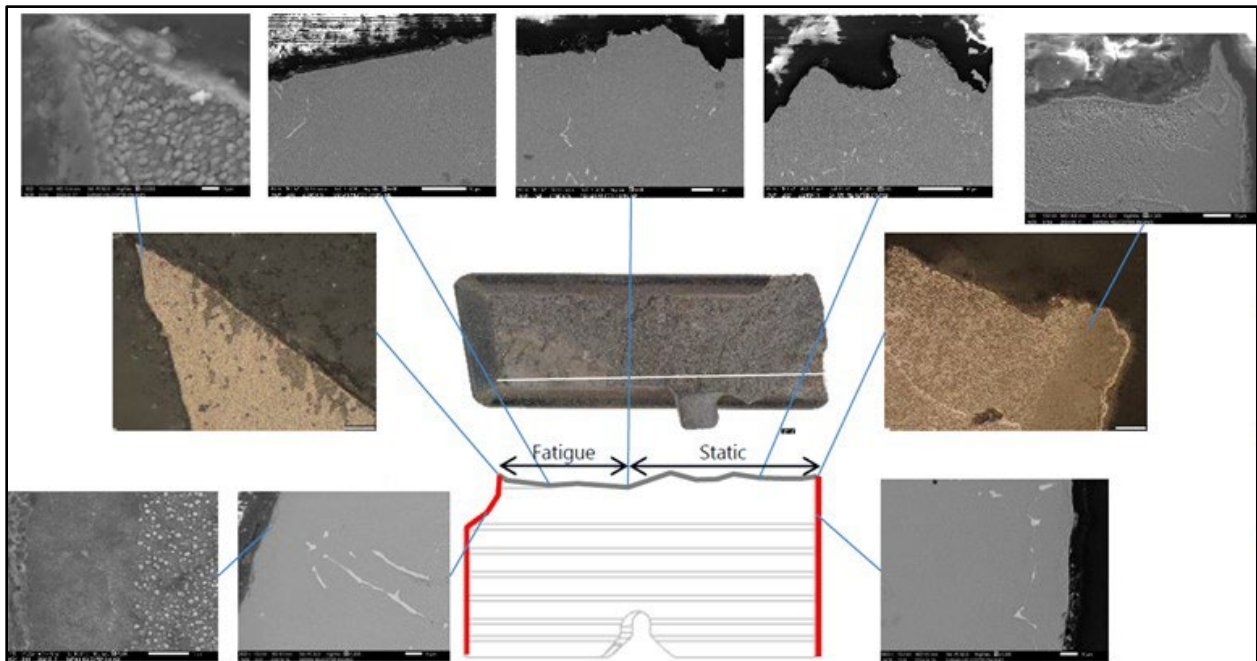


Figure 92. PT Blade S/N 164649 Metallography
(Figure Courtesy of SafranHE, Etchant: copper sulfate reagent)

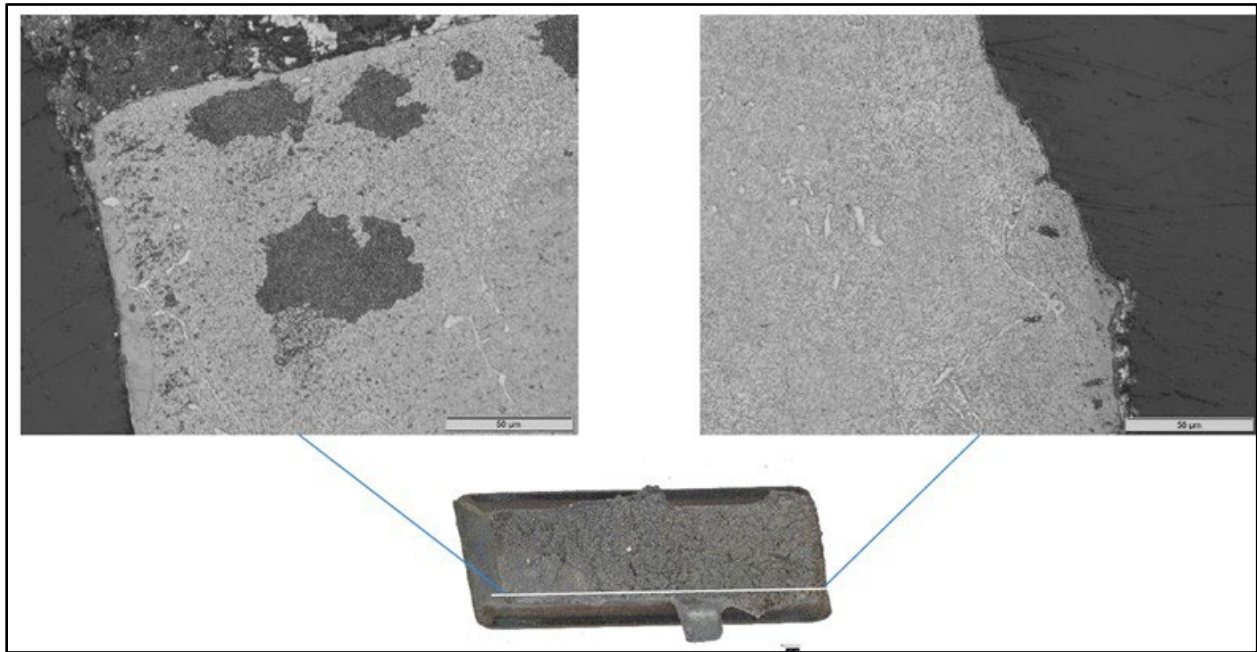


Figure 93. PT Blade S/N 164640 Metallography

(Figure Courtesy of SafranHE, Etchant: copper sulfate reagent)

The PT shaft, P/N 0319417690, a stainless steel alloy, exhibited multiple areas of discoloration; a blue discoloration around the inner diameter at the rear of the shaft and a pink to purple discoloration around the inner bore of the at the rear of the shaft (Figure 94). Based on the total time of the event flight, the discolored regions indicate an engine operating temperature that is at least 700 degrees C, which is 300 degrees C above the typical operating temperature (calculated temperatures at each location) for a take-off power setting.

The rear labyrinth seals exhibited damage on approximately one third of the circumference, with the seals completely worn down at the center of the damaged area, while the seals opposite this location appeared undamaged. The shaft splines appeared undamaged, but the rear half of the shaft was discolored black with soot.



Figure 94. PT Shaft

(Figure Courtesy of SafranHE)

6.4 No. 2 Engine Hardware

The internal hardware appeared sooted, consistent with an engine running in a post crash fire condition.

The gas generator turbine disk contained purple discoloration of the disk's outer web region, just inboard of the disk posts, around the entire circumference (Figure 95).

All gas generator turbine and PT blades were intact and full length (Figure 96).

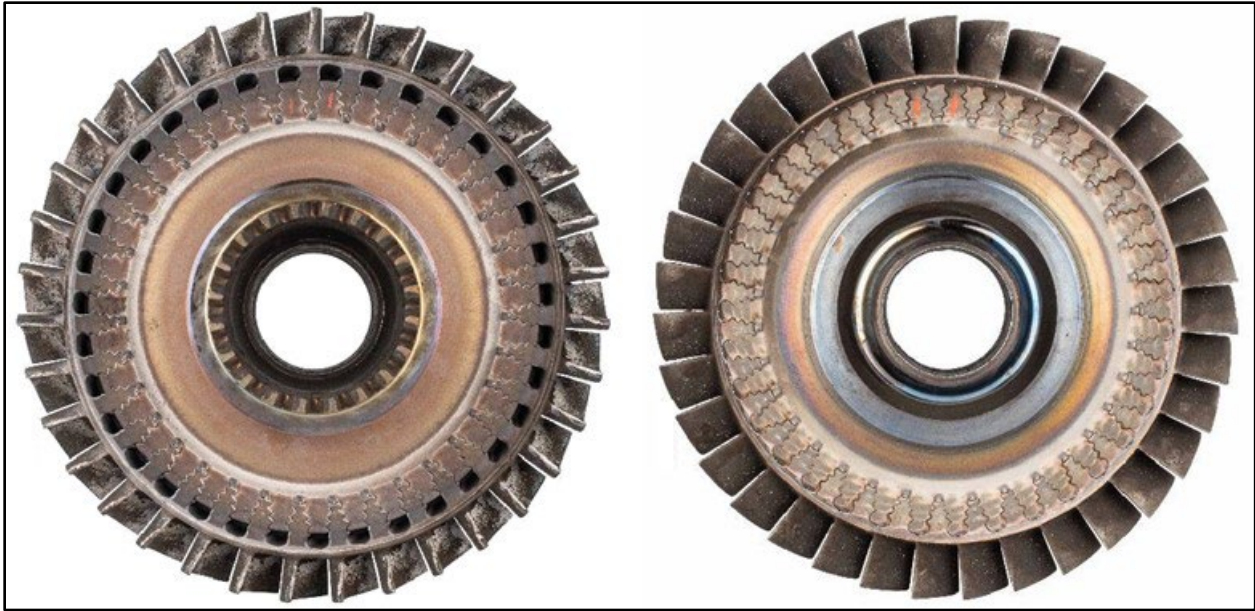


Figure 95. Gas Generator Turbine Disk and Blades

(Figure Courtesy of SafranHE)

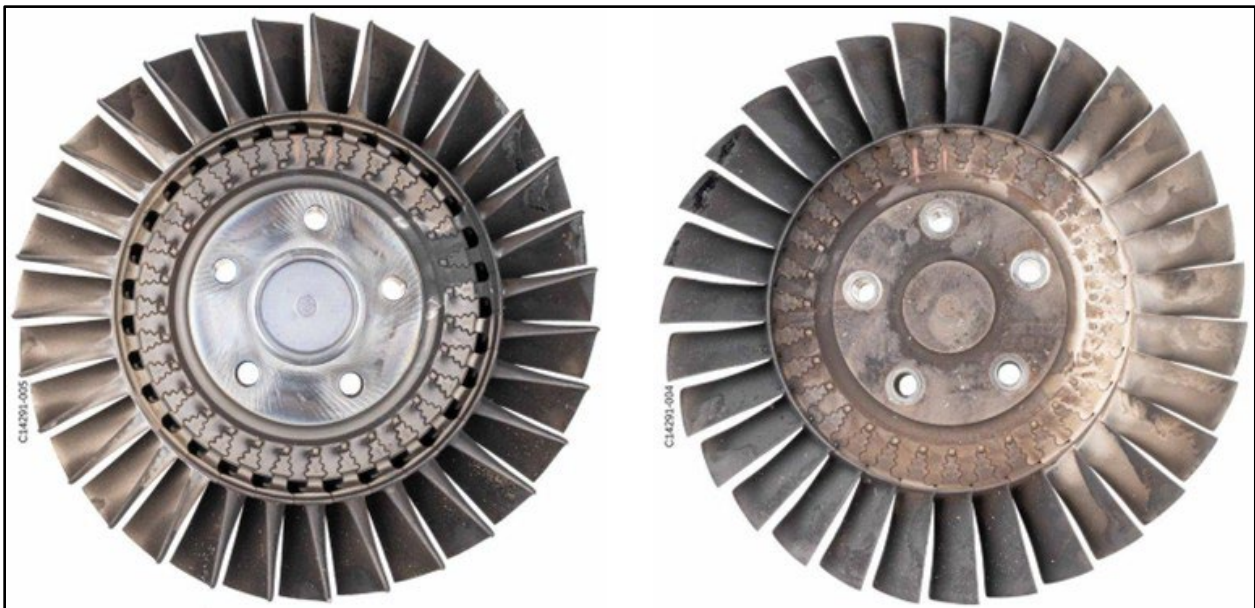


Figure 96. Gas Generator Turbine Disk and Blades

(Figure Courtesy of SafranHE)

Metallurgical analysis was performed on multiple gas generator turbine blades, both above and below the blade platform. Above the blade platform, there was evidence of quasi dissolution of the gamma prime precipitates at the top of the airfoil and significant dissolution of the gamma prime precipitates at the middle of the airfoil (Figure 97). There was no evidence of creep rafting of the gamma prime precipitates. Based on the material's properties and a solvus temperature of 1,295 degrees C, this layer of dissolution indicates a gas path temperature that is at least 300 degrees C

above the typical temperature calculated for a take-off power setting. The nominal microstructure of dense gamma/gamma prime structure and equiaxed grains were found on the airfoil near the blade fillet. Below the blade platform, there was no evidence of dissolution of the gamma prime precipitates and the presence of a secondary reaction zone (SRZ) is consistent with used, in service, blades (Figure 98).

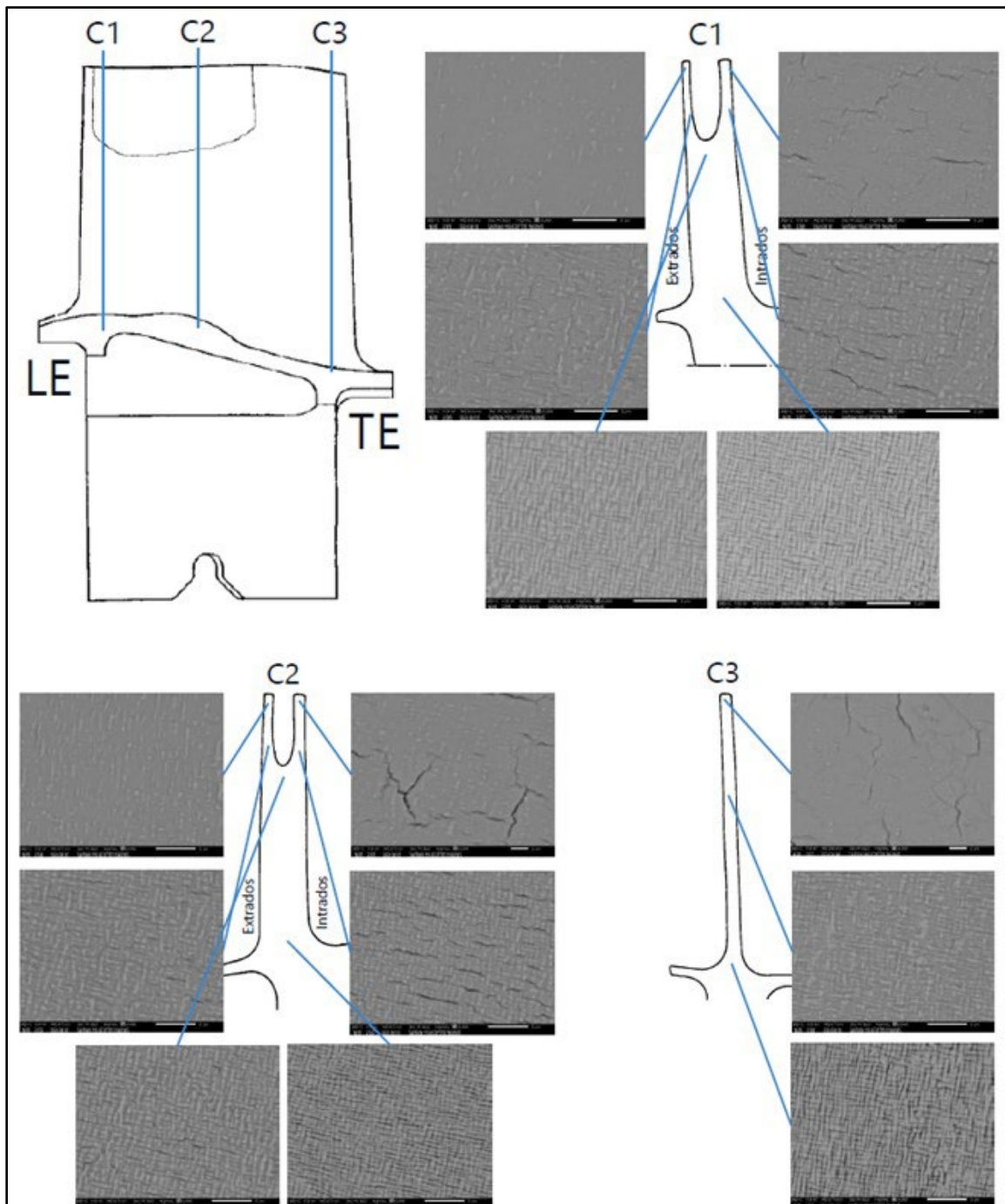


Figure 97. Gas Generator Turbine Blade Metallography - Above Platform
(Figure Courtesy of SafranHE, Etchant: copper sulfate reagent)

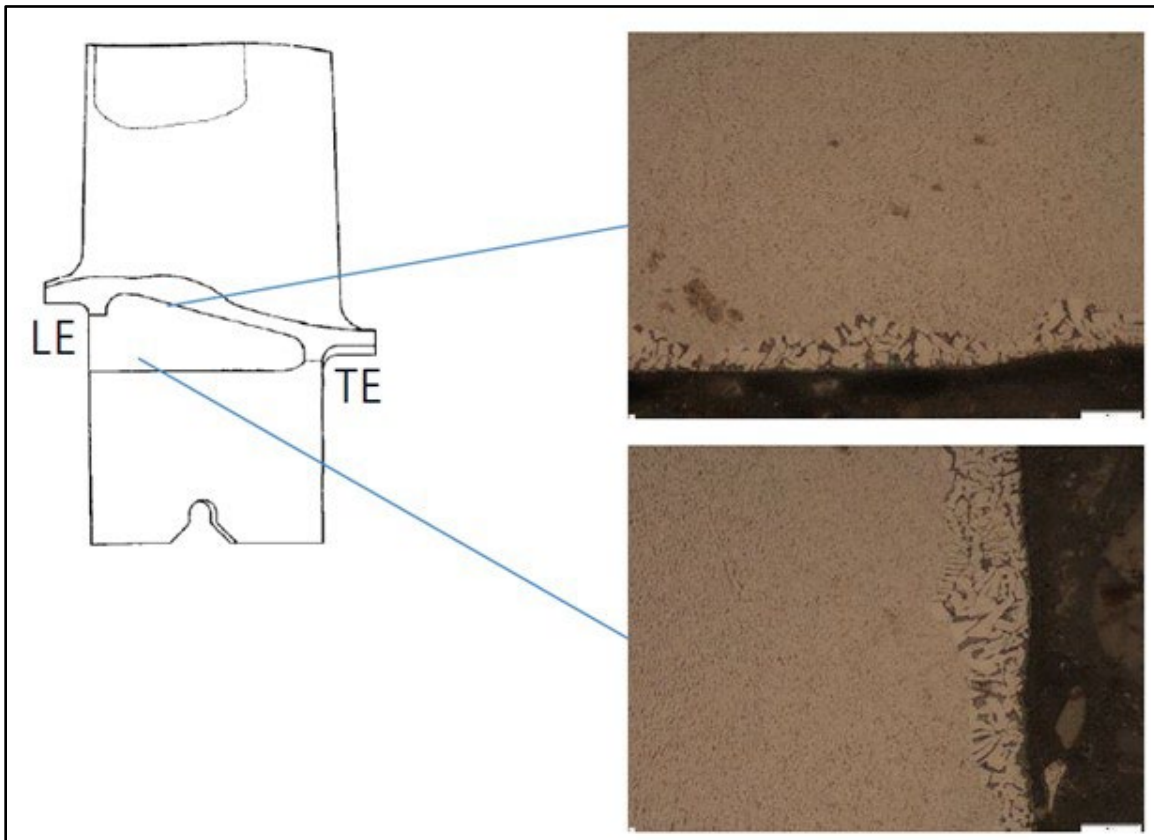


Figure 98. Gas Generator Turbine Blade Metallography – Below Platform

(Figure Courtesy of SafranHE, Etchant: copper sulfate reagent)

Visual inspection of the PT blades found corrosion on both the suction and pressure sides of the platform and along the profile of all blades, contact wear marks on the blade tips and loss of material (erosion) along the leading edges. Fluorescent penetrant inspection was performed on 28 PT blades of the 31 blade set (3 blades were preserved). The presence of cracks were found within the third fillet of the blade fir tree on all blades, blade Nos. 2, 8, 10, and 17 contained cracks that exceeded the limits defined in CMM task 72-42-22-222-801-A01 for continued operation.

Metallurgical analysis was performed on select PT blades, both above and below the blade platform. Above the blade platform, there was evidence of partial dissolution of the gamma prime precipitates and eutectic phases at the trailing edge tip of the airfoils, similarly indicating a gas path temperature that is at least 300 degrees C above the typical temperature calculated for a take-off power setting. The nominal microstructure of dense gamma/gamma prime structure and equiaxed grains were found on the airfoil near the blade fillet (Figure 99). Below the blade platform, there was no evidence of dissolution of the gamma prime precipitates (Figure 100).

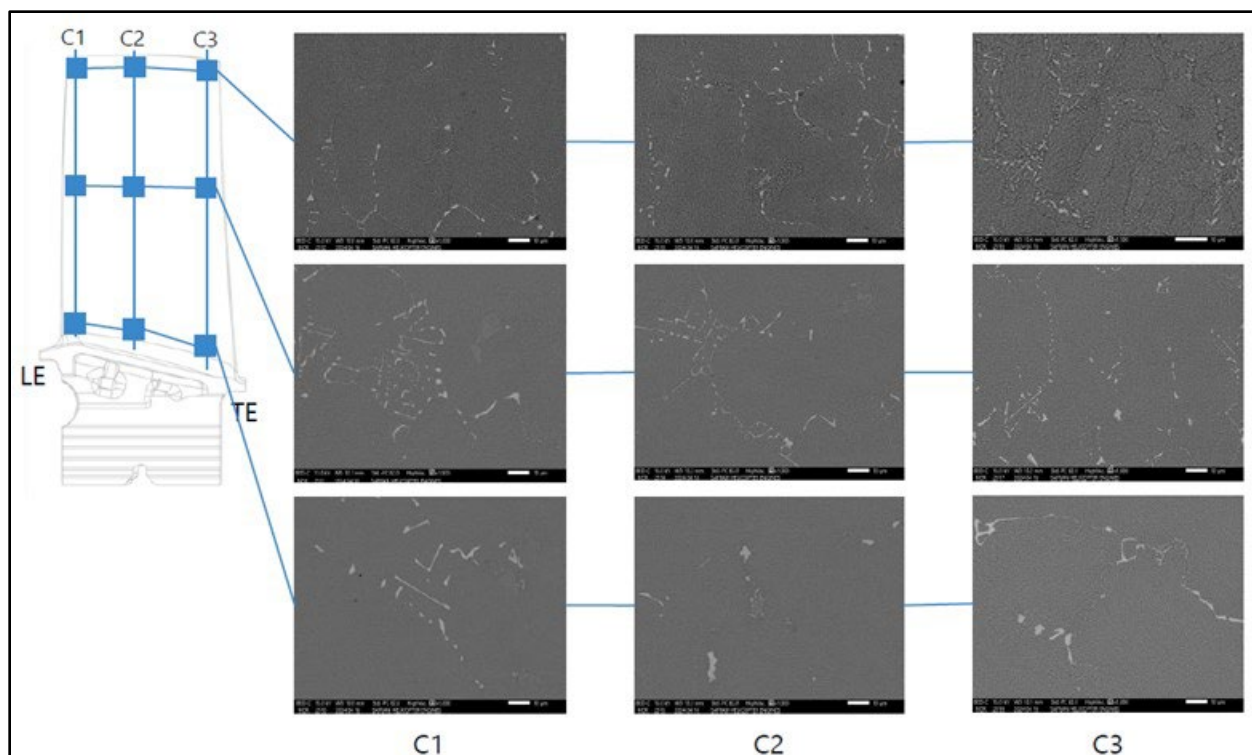


Figure 99. PT Blade Metallography - Above Platform
 (Figure Courtesy of SafranHE, Etchant: copper sulfate reagent)

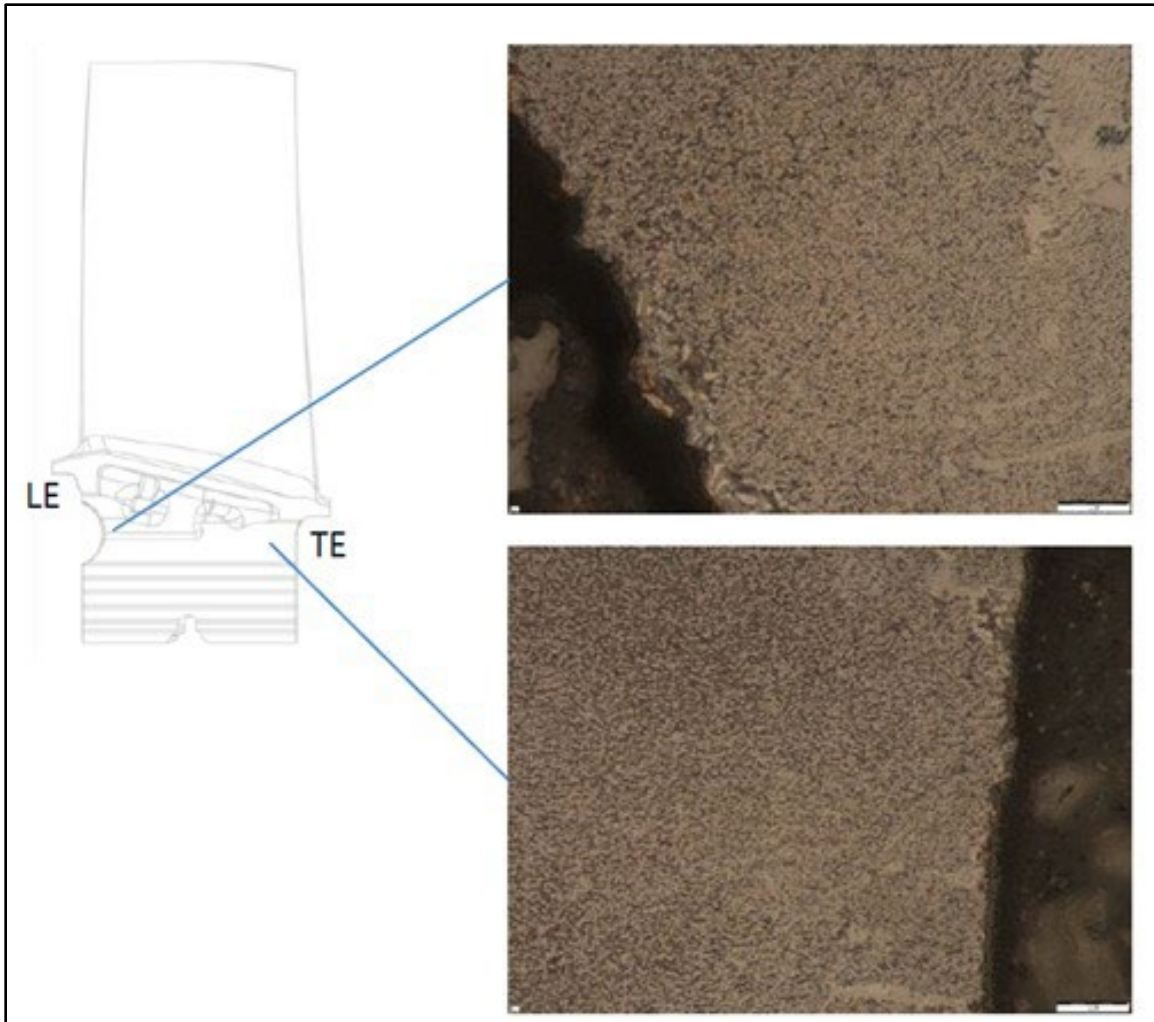


Figure 100. PT Blade Metallography – Below Platform

(Figure Courtesy of SafranHE, Etchant: copper sulfate reagent)

7.0 Arrius 2B2 Engine - PT Blade Rupture History

For reference, the engines installed in the accident helicopter were Arrius 2B1 engines. The primary differences between the 2B1 and 2B2 engine applications include; the design and material system used for the gas generator turbine hardware, increased power and operating temperatures for the 2B2, a high Ng rating for the 2B2 and the availability of the high NR mode.

Helicopter configurations with the Arrius 2B2 engine, and the Arrius2K series engines, may have an Airbus provided NR selector option that allows the helicopter pilot to select an NR speed of Normal or High. The Normal NR mode allows the helicopter's main rotor to operate within a rotational speed range of 100% to 101%, whereas the High NR mode allows the helicopter's main rotor to operate within a rotational speed range of 102% to 103.5%.

The EC135-T1 helicopter with Arrius 2B1 engines is not a configuration with the capability of a high NR mode, but the Arrius 2B1 EECU software includes an N2 law (software logic) that will start to increase N2 speed once it is above 4,000 feet pressure-altitude. At the location and time of the accident, the pressure altitude was less than 4,000 feet, thus the NR speed would be governed within the NR speed range of 100% to 101%.

SafranHE stated in Service Letter No. 3030/19/ARS2 Revision 3, titled "ARRIUS 2B1, 2B1A, 2B2, 2K1, 2K2, 2G1 Power turbine blade rupture events" and dated December 15, 2020, there has been 5 in-service PT blade rupture event on the Arrius 2B2 engines investigated by SafranHE. The 5 events occurred at different phases of flight, at varying power levels and in all 5 events, only a single PT blade failed.

The first PT blade rupture event was determined to be the result of a material inclusion. SafranHE identified similar fracture surface characteristics in the remaining four PT blade rupture events and determined the root cause was attributable to multiple contributors. The contributors include HCF stresses due to an inner blade vibration that occurs at an N2 speed of approximately 102.5%, a cumulative effect of the HCF and LCF (LCF due to N2 cycles) stresses, and a change in the blade root position in the manufacturing tolerance. The N2 operational speed of approximately 102.5%, in parallel with the other contributing factors, creates an environment where the Mode 1 frequency/fundamental vibratory resonance was excited and increased the stress exerted on the blade above the calculated/expected stress and impacting the dynamic margin (material endurance, crack initiation capability vs. the actual stress level). It was also determined that the locations of the maximum static stress and maximum dynamic stress were in the same location of the blade's shank.

The corrective actions taken by SafranHE included a PT blade redesign effort, known as TU164¹⁴, that modified the shape of the blade root and shank regions to decrease the maximum stress and improve the dynamic margin of the PT blade's endurance limit (Figure 101). The TU 164 redesign increases the mechanical strength of the PT blade shank relative to HCF and separates the HCF maximum stress location from the LCF maximum stress location.

SafranHE issued Service Bulletin (SB) No. 319-72-2164, the initial release of SB No. 319-72-2164, Version A, dated December 11, 2020, was applicable to all Arrius 2B1, 2B1A(1), 2B2, 2G1, 2K1, 2K2 engines. SafranHE released Version D of SB No. 319-72-2164, dated March 11, 2024, which updated the compliance period for the completion of this SB to be within 500 flight hours from release of this SB or before 30 December 2026, whichever is reached first.

¹⁴ "TU" refers to Turbomeca Modification Standard

This SB calls for replacement of the PRE TU 164 PT blades with POST TU 164 PT blades. The geometry of the POST TU 164 blade separates the HCF maximum stress area from the LCF maximum stress area. This geometry increases the mechanical strength of the PT blade against HCF. POST TU 164 PT blades are either new blades or existing PRE TU 164 blades changed into POST TU 164 blades.

The original part numbers of the PT blade used in Arrius engine applications were P/N 231941A0D0, 2319411630, and 2319411640. The TU164 redesign effort introduced new blade P/N 231941A7J0 (new manufacture) and P/N 731941A0D1 (reoperated from 231941A0D0) for incorporation into all Arrius engine applications.

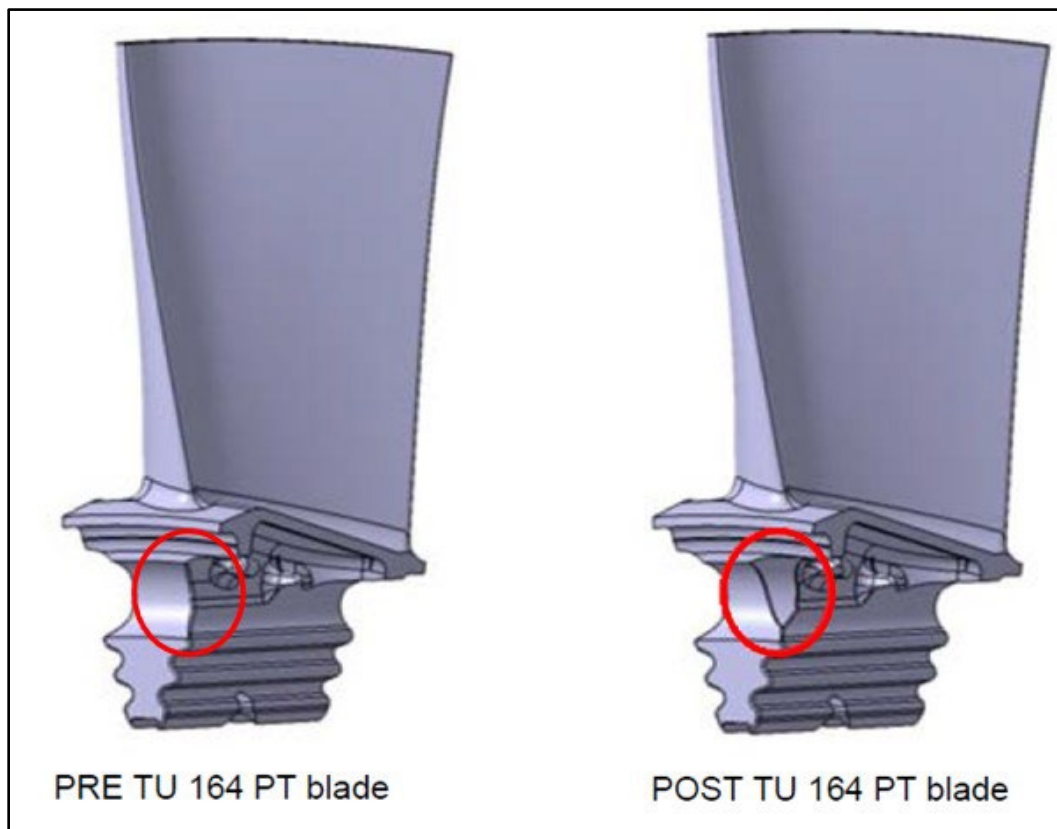


Figure 101. PT Blade Model - Original Design (L), TU164 Re-design (R)
(Figure Courtesy of SafranHE)

For reference: The PT blades on the accident helicopter, installed new in 2016, were P/N 231941A0D0 (pre-TU164)

Table 10. Comparison of Accident Flight and Arrius 2B2 PT Blade Rupture Events

	Arrius 2B2 PT Rupture Events	N109BC - Arrius 2B1 Event
Mission	EMS	EMS
Flight Phase	Decent, Landing, Hover	Initial Climb
Environment (OAT)	TO: Low, Medium	TO: High
High NR Mode Active	Yes	N/A
Number of Ruptured Blades	1	5
N2 Cycles on PT Blades	7,566; 8,084; 5,049; 6,842; 7,658	3,601
EECU N2 Double Failure	Yes	Yes
EECU N1 Double Failure	No	Yes

F ATTACHMENTS

Attachment 1. BEA Technical Document No. BEA2023-0379_tec01

Attachment 2. BEA Technical Document No. 2023-0379_tech04

Submitted by:
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APPENDIX 1

No. 1 Engine EECU Recovered Data

Maintenance Block Number	ECU Total Time (hrs)	Power On Time (sec)	P0 (mbars)	T1 (Celsius)	P3 (bars)	T4 (Celsius)	XPC (%)	XR	N1 Demand	Fuel Flow Demand (L/hr)	N1 (%N1)	N2 (%N2)	Torque (m.daN)
1	5194.6	195	986.69	31.31	4.07	758.5	60.12	56.16	94.57	123.75	89.41	99.78	28.57
2	5045.1	0,00	1009.56	29.08	1.04	1100	39.35	10.45	0	0	111	0	506.2
3	5045.1	40,00	1009.56	29.14	1.06	25.13	39.4	9.93	0	38	0	0	506.2
4	5045.1	0,00	1012.19	29.27	1.08	1100	39.42	9.14	0	36.56	0	0	506.2
5	5045.1	0,00	1010.06	29.34	1.09	1100	39.43	10.45	0	39.44	0	0	506.2
6	5045.1	0,00	1009.69	29.65	1.08	23.38	39.37	9.14	0	36.56	0	0	506.2
7	5045.1	0,00	1009.69	29.97	1.09	1100	39.32	9.05	0	0	0	0	506.2
8	5047.6	785	1013.3	31.18	2.89	600.44	22.32	14.59	68.53	49.19	68.54	75.22	4.5
9	5045.1	0	1010.81	29.34	1.11	1100	39.34	10.45	0	0	111	0	506.2

Maintenance Block Number	Stop Input	Idle Input	Flight Input	Training Mode Input	Manual Mode Input	Self Test Ongoing	Normal Idle Output	Level 1 Failure	Level 2 Failure	Level 3 Failure	OEI 2'30" Consumption	External Failure	T1 Failure	T4 Failure	Alternator or Double N1 Failure	N1 or Double N1 Failure	NTL_A or Double NTL Failure	NTL_B or Double NTL Failure	Collective Pitch Failure (XPC)	N2 Trim Position Failure	P3 Drift Failure	A/C 28Volt Power Supply Failure	28 Volt Alternator Failure (<10V)	SWM_ON_V BIT	SML_BIT_VALI D	SML_BIT_RAW <0.15A
1	0	0	1	0	0	0	0	0	0	1	0	1	0	0	1	1	1	1	0	0	0	0	0	0	0	0
2	1	0	0	0	0	0	0	0	1	1	1	1	0	1	1	1	0	0	0	0	0	0	1	1	0	0
3	1	0	0	0	0	0	0	0	1	0	0	1	0	0	0	1	0	0	0	0	0	0	1	0	0	0
4	1	0	0	1	1	1	0	1	1	0	0	1	0	1	0	0	0	0	0	0	0	0	0	1	1	1
5	1	0	0	1	1	1	0	1	1	0	0	1	0	1	0	0	0	0	0	0	0	0	0	1	1	1
6	1	0	0	1	1	0	0	1	1	0	0	1	0	0	0	0	1	0	0	0	0	0	0	1	1	1
7	1	0	0	0	0	0	0	0	1	1	1	1	0	1	1	1	0	0	0	0	0	0	1	1	0	0
8	0	1	0	0	0	0	1	1	0	0	0	1	0	0	0	0	0	0	0	0	0	1	0	0	0	0
9	1	0	0	0	0	0	0	0	1	1	1	1	0	1	1	1	0	0	0	0	0	0	1	1	0	0

APPENDIX 2

No. 2 Engine EECU Recovered Data

Maintenance Block Number	ECU Total Time (hrs)	Power On Time (sec)	P0 (mbars)	T1 (Celsius)	P3 (bars)	T4 (Celsius)	XPC (%)	XR	N1 Demand	Fuel Flow Demand (L/hr)	N1 (%N1)	N2 (%N2)	Torque (m.daN)
3	7369.17	0	1020.13	30.48	1.11	1100	38.89	9.14	0	0	111	0	506.2
4	7369.7	0	1019.31	30.41	1.15	1100	38.87	10.37	0	0	111	0	506.2
5	7372.2	799	1023.94	31.76	2.98	558.63	21.41	10.43	68.59	50.44	68.52	74.98	5.34
6	7519.2	307	1015.06	43.05	4.19	880.69	48	12.21	73.36	41.94	96.1	107.25	67.69
7	7519.2	307.01	1015.5	43.75	4.11	887.5	48	11.42	55.36	40	92.87	112.5	73.22
8	7519.2	336	1014.19	78.32	2.75	1100	48	11.6	78.34	46.06	79.14	105.46	81.13
1	7519.2	346	1015.5	75.98	2.79	1100	48	5.09	76.28	23.63	81.85	108.07	81.03
2	7519.2	347	1014.19	15.15	13	1100	48	25.84	84.86	77.94	78.3	103.61	81

Maintenance Block Number	Stop Input	Idle Input	Flight Input	Normal Idle Output	Level 1 Failure	Level 2 Failure	Level 3 Failure	OEI 2'30" Consumption	External Failure	T1 Failure	T4 Failure	Alternator or Double N1 Failure	N1 or Double N1 Failure	Collective Pitch Failure (XPC)	N2 Trim Position Failure	P3 Drift Failure	A/C 28Volt Power Supply Failure	28 Volt Alternator Failure (<10V)
3	1	0	0	0	0	1	1	1	1	0	1	1	1	0	0	0	0	1
4	1	0	0	0	0	1	1	1	1	0	0	1	0	0	0	0	0	1
5	0	1	0	1	1	0	0	0	1	0	0	0	0	0	0	0	1	0
6	0	0	1	0	1	1	0	0	1	0	0	0	0	1	0	0	1	0
7	0	0	1	0	1	1	0	0	1	0	0	0	0	1	1	0	1	0
8	0	0	1	0	1	1	0	0	1	0	1	0	0	1	1	0	1	0
1	0	0	1	0	1	1	0	0	1	1	1	0	0	1	1	0	1	0
2	0	0	1	0	1	1	0	0	1	1	1	0	0	1	1	1	1	0