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

December 22, 2021

AIRWORTHINESS GROUP CHAIRMAN'S FACTUAL REPORT

NTSB No: WPR21FA143

A. ACCIDENT

Operator: Soloy Helicopters
Aircraft: Airbus Helicopters AS350 B3, registration N351SH
Location: Palmer, Alaska
Date: March 27, 2021
Time: 1834 Alaska daylight time

B. GROUP

Group Chairman: Chihoon Shin
National Transportation Safety Board
Washington, District of Columbia

Member: Jean-Pierre Scarfo
National Transportation Safety Board
Washington, District of Columbia

Member: Ted Kohlstedt
Federal Aviation Administration
Anchorage, Alaska

Member: Chris Lanphier
Soloy Helicopters
Wasilla, Alaska

Member: Arnaud Blanc
Bureau d'Enquêtes et d'Analyses pour la Sécurité de l'Aviation Civile
Le Bourget, France

Member: Nicolas Holvoet
Bureau d'Enquêtes et d'Analyses pour la Sécurité de l'Aviation Civile
Le Bourget, France

Member:	Julien Ballester Bureau d'Enquêtes et d'Analyses pour la Sécurité de l'Aviation Civile Le Bourget, France
Member:	Seth Buttner Airbus Helicopters Grand Prairie, Texas
Member:	Michel Martin Airbus Helicopters Marignane, France
Member:	Bryan Larimore Safran Helicopter Engines Grand Prairie, Texas
Member:	Jacques Marquie Safran Helicopter Engines Bordes, France

LIST OF ACRONYMS

ADC	alternate direction current
ATT	aircraft total time
BEA	Bureau d'Enquêtes et d'Analyses pour la Sécurité de l'Aviation Civile
CFR	*Code of Federal Regulations*
CSN	cycles since new
DECU	digital engine control unit
EBCAU	emergency backup control auxiliary unit
ELT	emergency locator transmitter
ETT	engine total time
FAA	Federal Aviation Administration
NF	power turbine speed
NG	gas generator speed
NR	rotor speed
NTSB	National Transportation Safety Board
NVM	nonvolatile memory
N1	gas generator speed
N2	power turbine speed
OAT	outside air temperature
OSS	operating system software
PAWS	Wasilla Airport
P/N	part number
S/N	serial number
TCDS	type certificate data sheet
TRDS	tail rotor drive shaft
TSO	time since overhaul
T4	power turbine inlet temperature
UTC	Coordinated Universal Time
VEMD	vehicle engine monitoring display
XPC	collective position

C. SUMMARY

On March 27, 2021, about 1834 Alaskan daylight time, an Airbus AS350 B3, N351SH, was substantially damaged when it was involved in an accident near Palmer, Alaska. The pilot and four passengers were fatally injured, and one passenger was seriously injured. The helicopter was operated under Title 14 *Code of Federal Regulations* (*CFR*) Part 135 as an on-demand air charter flight.

From April 6-8, 2021, members of the Airworthiness Group performed an examination of the recovered wreckage at a hangar at the Lake Hook airpark, Anchorage, Alaska. On April 9, 2021, members of the Airworthiness Group reviewed the aircraft logbook and maintenance records for the accident helicopter.

D. DETAILS OF THE INVESTIGATION

1.0 HELICOPTER INFORMATION

1.1 HELICOPTER DESCRIPTION

The Airbus AS350 B3 helicopter has a three-bladed main rotor system[1] and a two-bladed tail rotor system. The accident helicopter was configured to seat six persons with two seats in the cockpit and a four-place bench seat in the cabin. The helicopter flight controls are hydraulically assisted. The helicopter has a high skid-type landing gear. A single Safran Helicopter Engines Arriel 2B1 turboshaft engine provides power to the rotor system. The AS350 B3 is type certificated under Federal Aviation Administration (FAA) Type Certificate Data Sheet (TCDS) No. H9EU. The Arriel 2B1 engine is type certificated under FAA TCDS No. E00054EN.

The terms “left”, “right”, “up”, and “down” are used when in the frame of reference of looking forward from the aft end of the helicopter, i.e. aft looking forward (ALF). All locations and directions will be viewed from ALF unless otherwise specified. Additionally, clock positions are in the ALF frame of reference unless otherwise specified.

1.3 HELICOPTER AND ENGINE HISTORY

The accident helicopter, airframe serial number (S/N) 4598, was manufactured on October 31, 2008 and was acquired by Soloy Helicopters in 2011 (**Figure 1**). At the time of the accident, the helicopter had an aircraft total time (ATT) of about 5,675.2 hours.

The accident engine, S/N 46173, was installed onto the accident helicopter at its manufacture. The engine was last overhauled in September 2015 and installed back onto the accident helicopter October 2, 2015 at an ETT of 3,328.4 hours, an engine time since overhaul (TSO) of 0.0 hours, and an ATT of 3,434.0 hours. At the time of the accident, the engine total time (ETT) was about 5,569.6 hours. According to an aircraft logbook entry dated March 26, 2021, the day prior to the accident, the engine gas generator had accumulated 11,273.70 cycles since new (CSN) and the power turbine had accumulated 3,951.05 CSN.

[1] The main rotor blades rotate in a clockwise direction when looking down at the main rotor disk from above.

Figure 1. An exemplar photo of the accident helicopter. **(Image courtesy of Soloy Helicopters)**

2.0 POST-RECOVERY WRECKAGE EXAMINATION

The accident helicopter impacted mountainous terrain and came to rest around 61°26’56” N by 148°23’20” W. Photographs of the accident site were taken by the Alaska State Troopers and provided to investigators. These photographs showed a debris trail starting from the vicinity of a mountain peak down to where the helicopter came to rest near the base of a mountain (**Figures 2 and 3**). Due to difficulty in accessing the accident site, investigators did not travel to the accident site to document the wreckage. The wreckage was removed from the accident site and transported to a hangar at the Lake Hood airpark, Anchorage, Alaska. Wreckage documentation was conducted after recovery of it from the accident site.

Figure 2. The bulk of the main wreckage (yellow arrow) located near the base of a mountain.
(Images courtesy of the Alaska State Troopers)

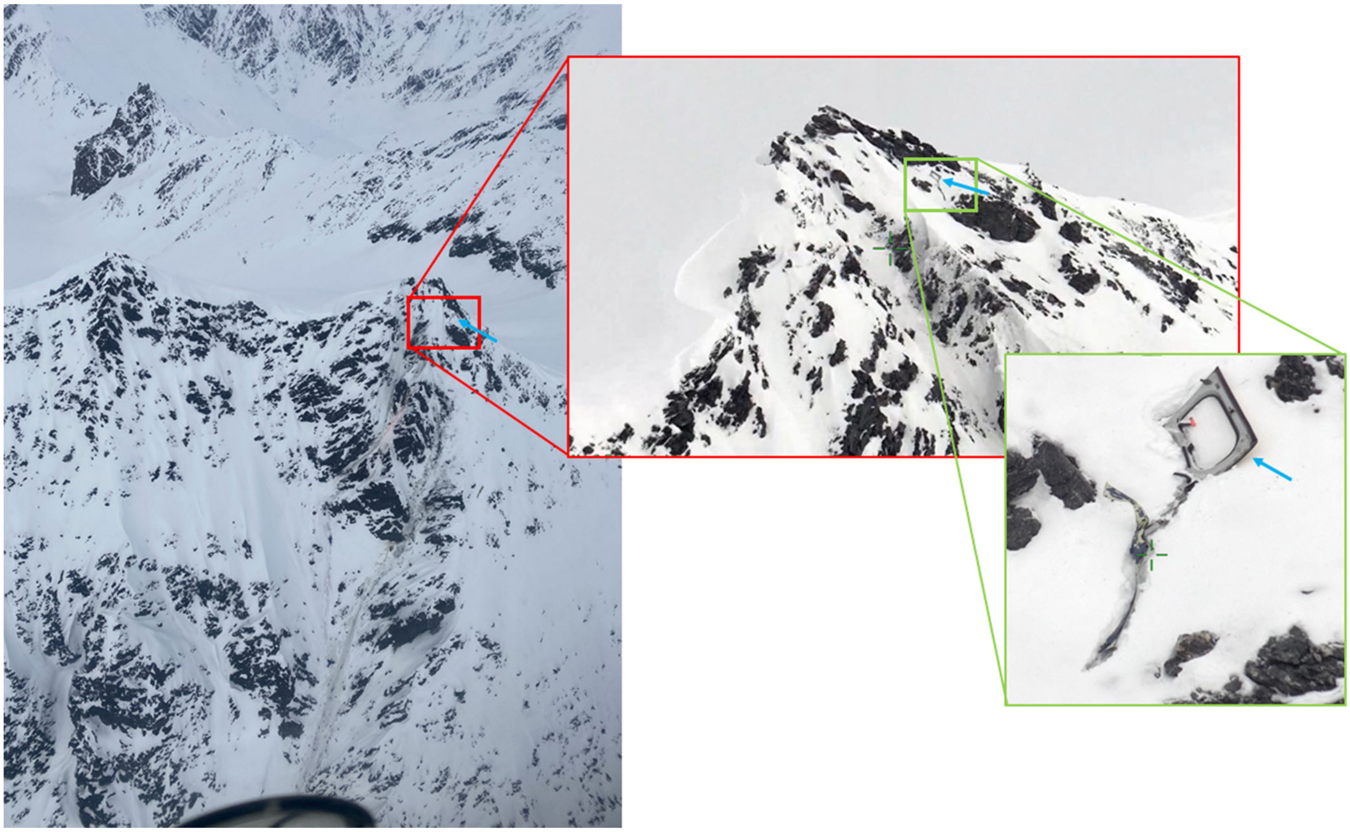

Figure 3. A cockpit door (blue arrow) near the peak of a mountain, the highest altitude piece of wreckage identifiable by investigators. **(Images courtesy of the Alaska State Troopers)**

From April 6-8, 2021, members of the airworthiness group examined the recovered wreckage at a hangar at Lake Hood airpark in Anchorage, Alaska. As of the date of this report, the majority of the helicopter was recovered with the exception of the following: the landing gear skids; outboard segments of the main rotor blades; right cabin sliding door; left and right cockpit doors; aft tail boom section with horizontal stabilizer and vertical fins; tail rotor system; and the cargo basket.

2.1 STRUCTURES

2.1.1 OVERVIEW

The helicopter structure consists of the main fuselage, tail boom and empennage, and skid-type landing gear. The main fuselage comprises: the [central] body structure, primarily supporting the fuel tank, main transmission, and landing gear; the rear structure, primarily supporting the engine and baggage compartment; the bottom structure, primarily supporting the main cabin; and the canopy, primarily supporting the doors and windows. The tail boom is attached to the rear structure and supports the tail gearbox, horizontal stabilizer, tail rotor drive shafts, and the vertical fin.

2.1.2 OBSERVATIONS

The main fuselage, with the forward portion of the tail boom still attached, was found near the bottom of the mountain. The canopy, including the windscreen, cockpit doors, and cabin doors, had separated from the main fuselage (**Figure 4**). The windscreen center post and the upper-forward portion of the canopy structure, containing the rotor brake and emergency fuel shutoff levers, was recovered separated from the main fuselage as a single assembly. The structure to the left and right of the two main [floor] longitudinal beams was fractured and separated. Additionally, the structure forward of the main beams, forward of the pedal controls, was fractured and separated. The instrument panel was separated from the structure but remained connected the main fuselage via wiring.

Figure 4. The recovered main fuselage wreckage.

Photographs from the accident site showed the forward belly panel was in the debris trail leading up to the main fuselage, but it was not recovered. The center belly panel remained attached to the main fuselage but was impact damaged with the forward-left corner detached and the forward-right portion not present. The engine inlet barrier filter was found separated from the airframe. The barrier filter element was present with no evidence of blockages or anomalous damage aside from impact deformation. Sections of the upper cowlings were also recovered separated from the airframe.

The two front (cockpit) seat bases (buckets) remained attached to the airframe via its sliding track, and their seat backs were fractured and separated just above the inertia reel housing. The helicopter had provisions for a dual high-seat back installation under supplemental type certificate (STC) SR01620NY, but the dual seats were not installed at the time of the accident; the stock (standard) seat remained installed. The four rear (cabin) seats were composed of two benches, each with two seats. Both benches remained installed on the airframe. On the right-side bench, measurements taken from the cabin floor to the top of the bench seat showed a height of 8.5 inches at the outboard leg and 8.25 inches at the inboard leg. On the left-side bench, measurements taken from the cabin floor to the top of the bench seat showed a height of 8 inches at the inboard leg and 7.25 inches at the outboard leg.

The fuel tank was fractured in multiple locations, but the bulk of the tank remained within the center structure (**Figure 5**). The X-brace on the left side of the airframe exhibited deformation and was fractured near the center of the "X". The X-brace on the right side of the airframe exhibited deformation but no major fractures. The transmission deck was deformed downward into the area of the fuel tank and exhibited fractures in multiple locations. Additional information of the main transmission attachment to the structure can be found in Section 2.2.3 of this report.

Figure 5. The left side of the main fuselage wreckage.

The forward section of the tail boom was attached to the aft fuselage but exhibited deformation on its right side, adjacent to the aft fuselage's ring frame, also called the tail boom-to-fuselage bulkhead (**Figure 6**). The remnant section of tail boom was bent to the right. The tail boom was fractured and crumpled about mid-length, about 12 inches forward of where the leading edge of the horizontal stabilizer is normally located. The horizontal stabilizer, vertical fins, and the aft section of the tail boom were not recovered, but images from the Alaska State Troopers showed their condition at the accident site (**Figures 7 and 8**).

Figure 6. The right rear area of the main fuselage wreckage.

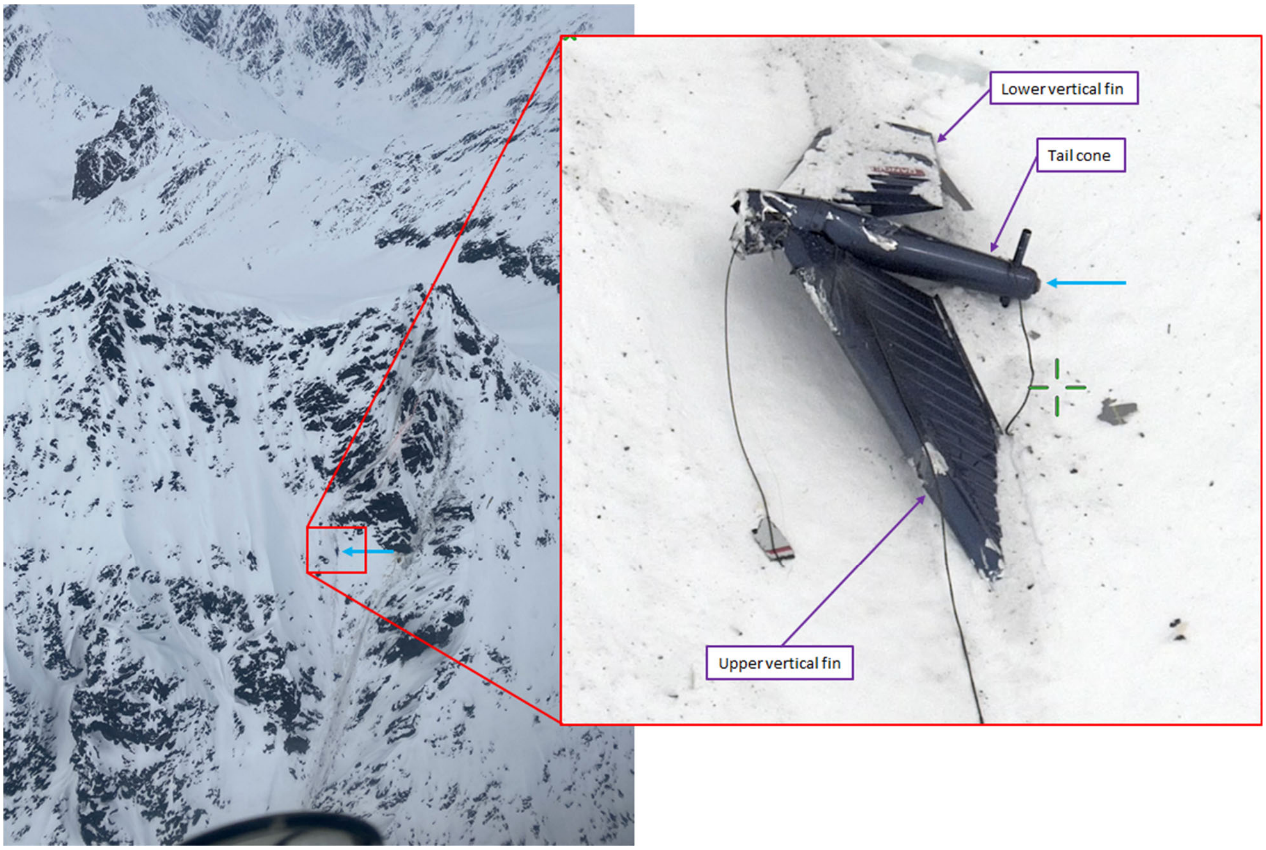

Figure 7. The vertical fin as seen at the accident site. (Images courtesy of the Alaska State Troopers)

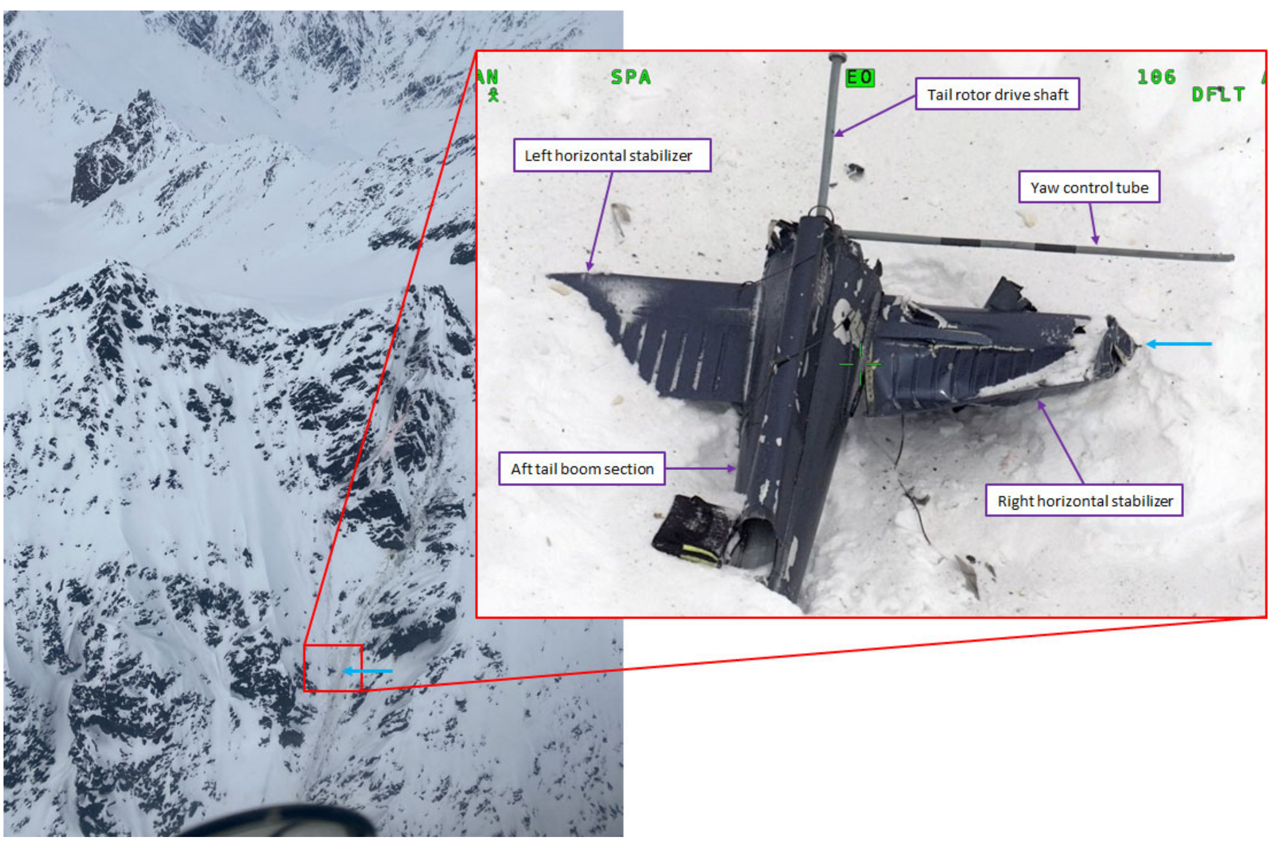

Figure 8. The horizontal stabilizer and aft tail boom section as seen at the accident site. (Images courtesy of the Alaska State Troopers)

The landing gear's forward crosstube remained attached to the airframe but had rotated about 180 degrees such that the crosstube legs were pointing generally upward. The forward crosstube was fractured about 12 inches from the left skid tube attachment point. The forward landing gear oleo-strut was fractured and separated from the forward crosstube but remained attached to the aft cabin bulkhead structure. On the right side of the forward crosstube, a portion of the right skid tube remained attached to the crosstube but was fractured immediately forward and aft of its attachment bolts.

The landing gear's aft crosstube remained attached to the airframe but had rotated about 90 degrees such that the crosstube legs were pointing forward. A portion of the cargo basket attachment structure was present on the crosstube. The aft crosstube had fractured at its interface with the left skid tube. On the right side of the aft crosstube, the right skid tube remained attached and was continuous from its aft heel to about mid-length of the skid tube. The bear paw remained attached to the right skid tube but had rotated.

Photographs from the accident site showed the skid-mounted cargo basket was in the debris trail leading up to the main fuselage wreckage. The cargo basket, normally installed on the left skid, was not recovered.

2.2 MAIN ROTOR SYSTEM

2.2.1 OVERVIEW

Power from the engine reduction gearbox is transferred to a power transmission shaft, the forward end of which is connected to a freewheel shaft. The freewheel shaft is connected to the engine-to-transmission drive shaft (also known as the *input driveshaft*) via a splined adapter. Flexible couplings on both ends of the engine-to-transmission shaft allow for minor misalignment. The engine-to-transmission shaft is connected to the main transmission input pinion pulley flange, which drives the main transmission input pinion, the aft hydraulic pump, and air conditioning unit, the latter two of which are belt driven via the pulley flange. The main transmission contains a single-stage sun and planetary gear system that turns the main rotor shaft. The main rotor shaft is attached to the Starflex via 12 bolts. The main transmission is attached to the airframe via four rigid suspension bars and an anti-torque bi-directional crossbeam with laminated pads installed between the lower transmission housing and the airframe.

The three main rotor blades attach to the Starflex via blade sleeves (two sleeves per blade). An elastomeric bearing connects the inboard end of the sleeves to the Starflex, while an elastomer block (also known as the "frequency adapter") is located near the outboard end of the sleeves and is attached to the outboard end of each Starflex arm. The blade is secured to the outboard end of the sleeve via blade pins. The elastomeric bearing allows for the blade to move in the flapping, lead-lag, and pitch change directions. The Starflex arms are flexible in flapping, but rigid in lead-lag and pitch change directions. The frequency adapter is flexible in lead-lag, but rigid in the flapping and pitch change directions. Each set of main rotor blades, sleeves, and pitch change links are assigned a color for identification purposes; the assigned colors are 'blue', 'red', and 'yellow'.

2.2.2 MAIN ROTOR OBSERVATIONS

The Starflex remained attached to the main rotor shaft and all three Starflex arms were fractured (**Figure 9**). The 'red' and 'blue' Starflex arms exhibiting 45 degree lateral fracture angles while the 'yellow' Staflex arm exhibited a laterally flat fracture. All three main rotor blade sleeves exhibited impact damage and a broomstraw appearance.[2] The inboard portion of the 'blue' and 'red' sleeves remained attached to the Starflex and their thrust bearing. The 'blue' thrust bearing was rotated laterally in the direction opposite of normal rotor rotation and the 'red' thrust bearing did not exhibit a similar lateral rotation. The inboard portion of the 'yellow' sleeve, along with its thrust bearing, were not present and not recovered. The vibration absorber remained installed on the main rotor head and all three springs were present. The vibration absorber housing was deformed due to impact. The droop stop ring remained installed underneath the Starflex.

Figure 9. The main rotor head, shaft, and main transmission. The red arrow points to the inboard portion of the 'red' main rotor blade.

The root end of the 'red' main rotor blade remained attached to remnant portions of its blade sleeves via attachment pins. The 'red' blade was fractured about 32 inches outboard of its sleeve attachment pins and an additional 6-foot section of blade outboard of this fracture remained partially attached. The 6-foot section of blade was lodged within, and extracted from, the fuel tank. The remainder of the 'red' main rotor blade was not recovered.

A 56-inch inboard section of the 'blue' main rotor blade, measured from its sleeve attachment pins outboard to a fracture on its leading edge, was recovered separated from the main wreckage. The sleeve pins remained attached to the blade. The fracture on its

[2] Fractures that exhibit a broomstraw appearance are consistent with high tension stress and overload.

leading edge exhibited inward deformation, i.e. in the direction opposite of normal rotation. The remainder of the 'blue' main rotor blade was not recovered.

An inboard segment of the 'yellow' main rotor blade was recovered separated from the main rotor head. This inboard section measured about 33 inches in length from its sleeve attachment pins to a leading edge fracture outboard of the pins. Remnant pieces of its upper and lower blade sleeves were attached to the blade's inboard end with the frequency adapter still installed. The remainder of the 'yellow' main rotor blade was not recovered.

The detailed findings for the main rotor control can be found in Section 2.4.3 of this report.

2.2.3 MAIN ROTOR DRIVE SYSTEM OBSERVATIONS

The main transmission remained attached to the airframe via the bi-directional beam but the left side of the transmission structure had separated from the transmission deck (**Figure 10**). The four suspension bars remained attached to the main transmission upper housing but all four bars were fractured about mid-length of the bar. The lower portion of all four suspension bars remained attached to their transmission deck attachment mounts. All fractures exhibited signatures consistent with overload.

Figure 10. The main transmission and the transmission deck.

The upper and lower main transmission chip detectors were removed and neither exhibited magnetic chips or debris. The gimbal assembly was found separated, with the forward gimbal mount and gimbal ring attached to the main transmission and the aft gimbal mount attached to the engine power transmission shaft housing. The gimbal ring interior surface exhibited rotational gouging consistent with contact with the input driveshaft flange.

The input driveshaft was recovered within the engine power transmission shaft housing. The input driveshaft was straight with no evidence of anomalous deformation. The aft flexible coupling assembly remained attached to the aft flange of the input driveshaft. The forward flexible coupling assembly was not present and the forward input driveshaft flange, as well as the main transmission input flange and the forward section of the universal coupling, exhibited rotational scoring on their surfaces (**Figure 11**). One remnant bolt shank was present within the main transmission input flange bolt hole. Rotation of the main transmission input flange resulted in a corresponding rotation of the main rotor head, confirming continuity of drive within the main transmission.

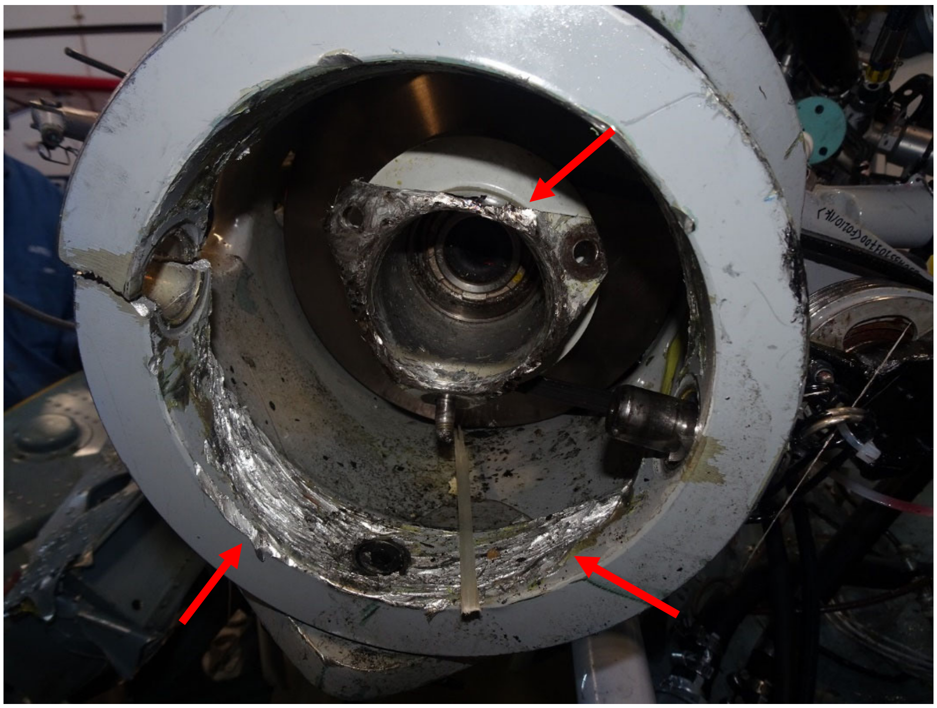

Figure 11. Rotational scoring (red arrows) observed within the forward section of the universal coupling and on the main transmission input flange. **(Image courtesy of Airbus)**

2.3 TAIL ROTOR SYSTEM

2.3.1 OVERVIEW

Engine power is transferred to the tail rotor via two tail rotor drive shafts (TRDS) and a tail gearbox. The forward TRDS, made of steel, is connected to a flange connected to the aft end of the freewheel shaft. The aft TRDS, made of aluminum, connects to the forward TRDS via a splined, steel flange adapter. Flexible couplings are located between each drive shaft attachment point to allow for minor misalignment. Five hanger bearings, mounted within support brackets along the tail boom, support the TRDS. The tail gearbox provides gear reduction and changes the direction of drive. The tail rotor hub, connected to the tail gearbox output shaft, provides final drive to the tail rotor.

The two tail rotor blades share a common composite spar that is flexible in both the flapping and pitch change (torsional) directions. Two metal half-shells are clamped to the center of the spar. The inboard half-shell connects to the tail rotor hub and allows for the tail rotor to teeter. Blade pitch is changed via pitch change links mounted between a pitch horn and a pitch change assembly (also known as the "spider"). The pitch horns rotate about a set of elastomeric bearings at the root end of each blade. The spider slides along the tail gearbox output shaft and is controlled by a pitch change bell crank. Each set of tail rotor blades and pitch change links are assigned a color for identification purposes; the assigned colors are 'red' and 'yellow'.

2.3.2 TAIL ROTOR OBSERVATIONS

The tail rotor was not recovered. Images from the Alaska State Troopers showed a potential section of one tail rotor blade near the peak of the mountain (**Figure 12**).

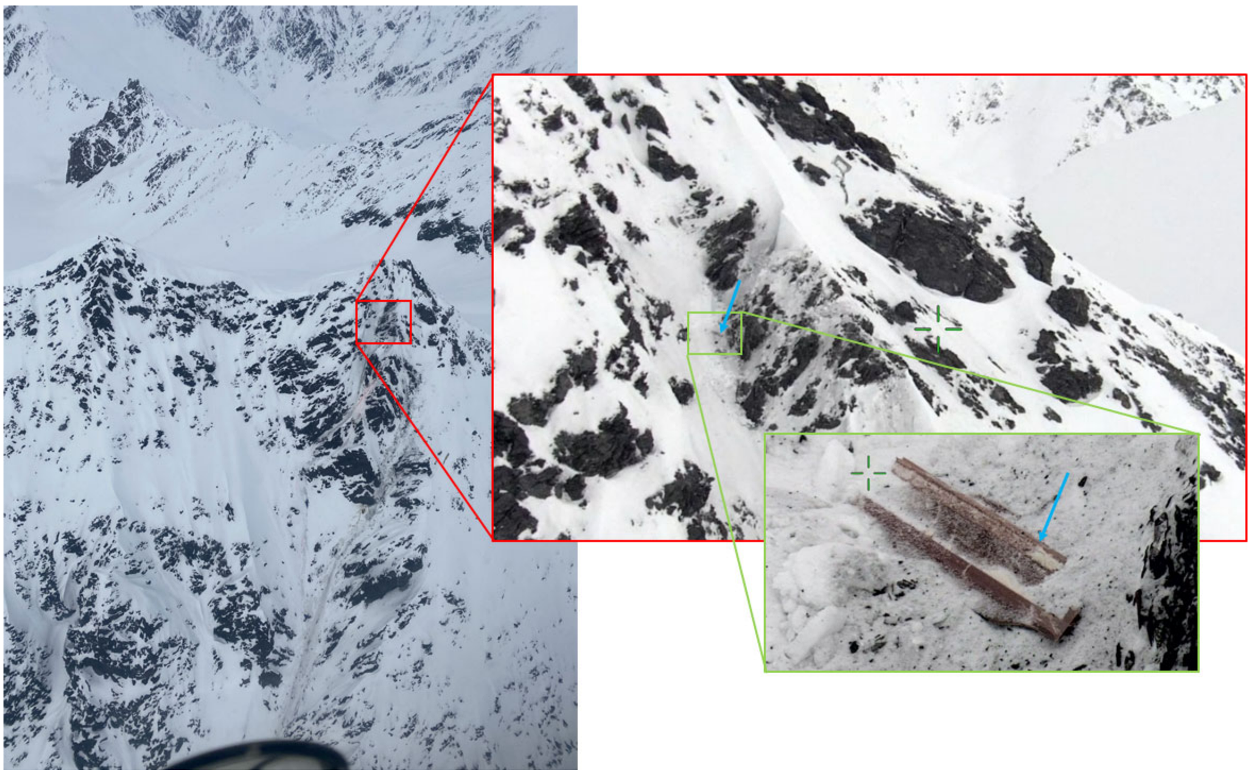

Figure 12. A potential section of one tail rotor blade (blue arrow) at the accident site. (Images courtesy of the Alaska State Troopers)

2.3.3 TAIL ROTOR DRIVE SYSTEM OBSERVATIONS

The tail rotor drive flange remained attached to the aft side of the engine reduction gearbox. Remnant pieces of the flexible coupling, as well as the flexible coupling attaching hardware, remained attached to the tail rotor drive flange. Two of the three tail rotor drive flange tangs were deformed in the aft direction.

Rotational scoring was observed on the underside of the forward tail rotor drive shaft cover. None of the tail rotor drive shafts nor their hanger bearings were present nor recovered. Portions of the four hanger bearing supports were present on the tail boom. The Nos. 1 and 2[3] hanger bearing supports were missing their bearing holders, and the vertical

[3] The hanger bearings are identified Nos. 1 through 4, with the No. 1 hanger bearing being the forward hanger bearing and the No. 4 hanger bearing being the aft hanger bearing.

support exhibited rotational scarring and were deformed in the aft direction. The No. 3 hanger bearing support was partially fractured and its bearing holder was present but deformed inward and in the aft direction. The No. 4 hanger bearing support was fractured at the tail rotor control tube bore and was missing its bearing holder as well as its vertical support. The tail rotor gearbox was not recovered.

2.4 FLIGHT CONTROL SYSTEM

2.4.1 OVERVIEW

The cyclic and collective control inputs are transmitted from the pilot controls to the stationary swashplate through a series of push-pull tubes and bell cranks. The main rotor cyclic and collective controls are hydraulically assisted via three single-cylinder main rotor servo controls: fore/aft, right-roll, and left-roll. The main rotor servo controls contain accumulators which provide backup hydraulic assistance in the event of a loss of hydraulic pressure in the system. The main rotor servo controls are mounted to the transmission upper housing and the stationary swashplate. The pedal control inputs are transmitted to the single-cylinder tail rotor servo control through a series of control linkages, bell cranks, and a flexible ball control cable. A yaw load compensator, with an accumulator, is connected to the tail rotor servo control output piston via the compensator connecting link, which actuates a push-pull tube connected to the pitch change bell crank mounted to the tail gearbox. The pilot seat is normally located on the right side of the helicopter. The AS350 B3 helicopter can be equipped with single or dual flight controls for one or two pilots, respectively. The accident helicopter was equipped only with single flight controls installed on the forward-right seat position.

2.4.2 COCKPIT FLIGHT CONTROL SYSTEM OBSERVATIONS

The pilot's cyclic, collective, and pedal controls remained installed in the cockpit (**Figure 13**). The cyclic control was bent, primarily in the forward-right direction, and fractured at its base. The cyclic grip remained attached to the control stick. The cyclic control friction knob was present and was able to be manipulated by hand. The friction knob was in a higher friction ("tight") setting, but there was deformation of the structure near its base.

The collective control was found in a low collective position with about 1 inch of downward travel remaining until it hit is lower stop. The collective-mounted twist grip throttle was found in the "flight" position. The collective head was fractured and separated from the collective control but remained attached via wiring. The hydraulic cut-off switch was separated from the collective head and was not found in the recovered wreckage. The collective lock remained attached to the airframe but its attachment base was deformed. The collective lock hole exhibited normal wear around its edges but no cracks or fractures.

For the pilot's pedal controls, both pedals were found in approximately the neutral position. The pedals could be manually moved. The pedals remained attached to the quadrant and interconnected to the left seat pedal mounts. The lateral interconnect tube was bent such that the pedal position between the left seat and right seat pedals were not aligned. The left seat pedal controls were not installed and a cover was present over the left seat pedal mounts.

Figure 13. The pilot's cyclic control (red arrow), collective control (blue arrow), and pedals (yellow arrows).

2.4.3 ROTOR CONTROL SYSTEM OBSERVATIONS

The mixing unit remained installed on the airframe (**Figure 14**). Multiple impact fractures and deformation was observed on the control tubes between the mixing unit and the cyclic, collective, and pedal controls. The three vertical push-pull tubes remained connected to the mixing unit but were fractured near their transmission deck side. There was no evidence of disconnection between the control linkages. The collective position (XPC) link remained connected to its mixing unit arm but was separated from the collective position sensor.

Figure 14. The flight control mixing unit installed on the airframe.

The rotating and stationary swashplates remained installed on the main rotor shaft. The rotating and stationary scissors remained attached to their respective swashplates and to the main rotor drive link and the main transmission housing, respectively. The tube of the swashplate guide was separated from its lower flange. The swashplate guide lower flange remained attached to the main transmission upper case. The swashplate rubber cover (boot) remained installed on the swashplate.

The 'yellow' main rotor blade pitch change link remained attached to the rotating swashplate and its pitch horn, but the pitch horn was separated from the 'yellow' main rotor blade. The 'yellow' pitch change link exhibited a 90-degree bend about mid-length of the link. The 'red' pitch change link remained attached between the rotating swashplate and its pitch horn. The 'red' pitch change link exhibited a slight inward deformation. The 'blue' pitch change link remained attached between the rotating swashplate and its pitch horn. The link body did not exhibit deformation, but the lower rod end was bent in the direction opposite of normal rotor rotation.

The tail rotor control Teleflex cable was present but was fractured at its aft threaded connection to the rod end (installed on a bellcrank). The tail rotor control linkages aft of the Teleflex cable remained installed and connected within the aft portion of the center structure and the forward portion of the tail boom. The tail rotor control tube remained connected to the yaw servo but was fractured about 43 inches aft of its forward attachment point to the yaw servo. The yaw control position link remained installed at both ends and to the yaw position sensor.

2.4.4 HYDRAULIC SYSTEM OBSERVATIONS

The fore/aft servo remained connected to the stationary swashplate but its piston was fractured (**Figure 15**). The lower portion of the piston remained connected to its main transmission attachment point. The piston fracture surfaces exhibited signatures of overload. The input linkage to the fore/aft servo remained connected to the vertical control tube, but the control tube's rod end was separated from its bellcrank, with the rod end attaching hardware remaining connected to the rod end. The hydraulic lines remained attached to the fore/aft servo.

The right roll servo remained connected to the stationary swashplate and its piston to its main transmission attachment point. The input linkage remained connected to the vertical control tube, the latter of which remained connected to its bellcrank. The hydraulic lines remained attached to the right roll servo.

The left roll servo remained connected to the stationary swashplate and its piston to its main transmission attachment point. The input linkage remained connected to the vertical control tube, but the control tube was fractured near its lower end. The lower end of the vertical control tube remained connected to its bellcrank. The hydraulic lines remained attached to the left roll servo.

The yaw servo and the yaw load compensator, and its accumulator, remained installed on the forward portion of the tail boom. The hydraulic lines remained connected to the yaw servo and yaw load compensator.

The hydraulic pump remained installed on the main transmission. The hydraulic pump drive belt remained installed with no evidence of breaks. On the pump's pulley, the drive belt was partially displaced in the forward direction (**Figure 16**). The hydraulic reservoir remained attached to the aft side of the main transmission housing and exhibited impact deformation. The hydraulic reservoir filler cap was not present.

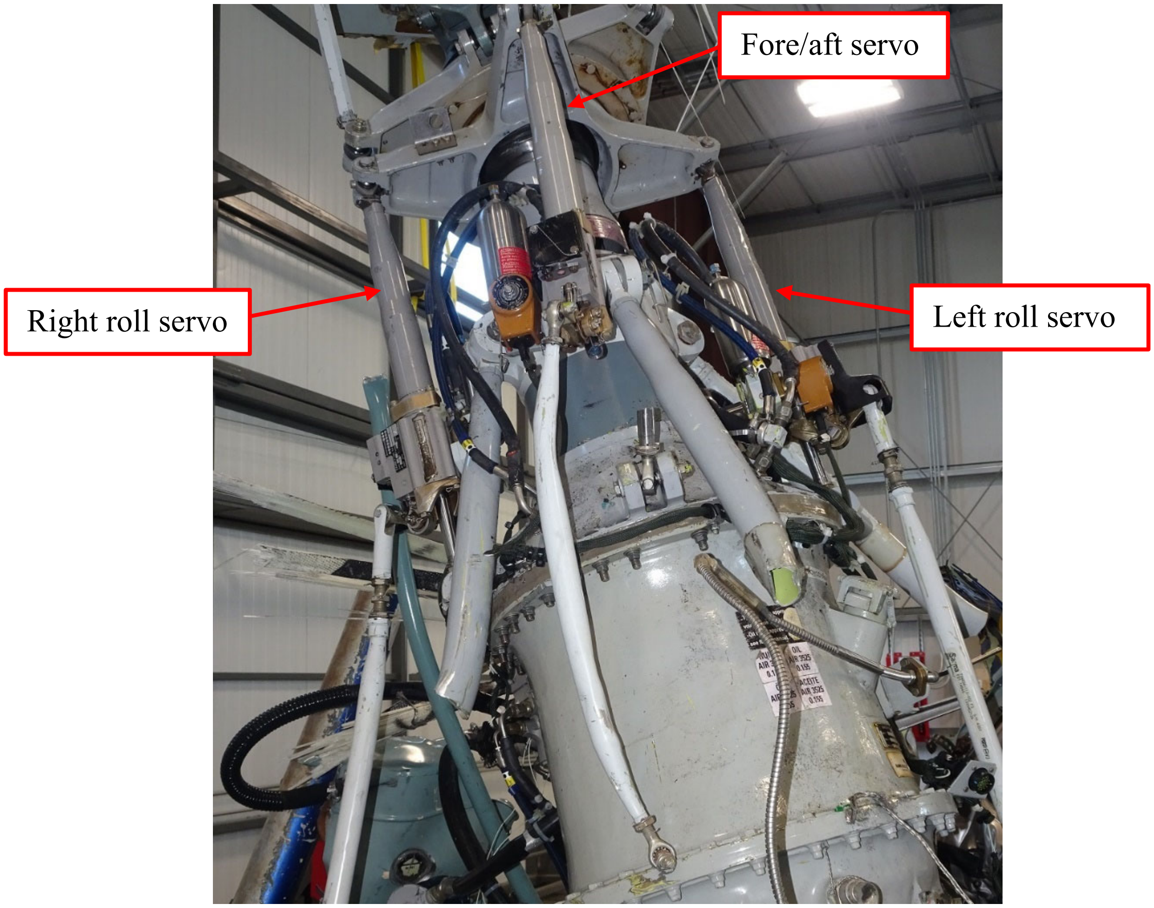

Figure 15. The three main rotor servos.

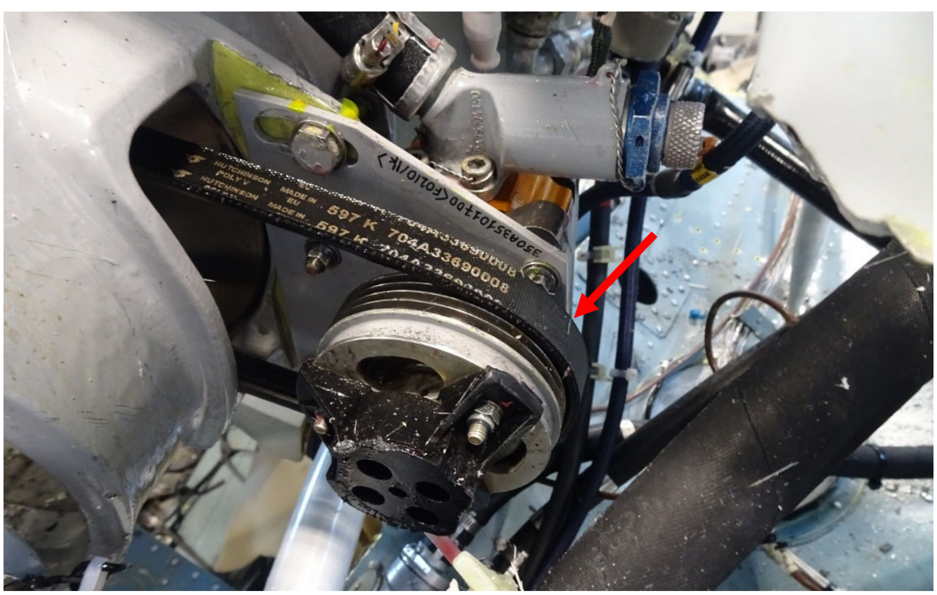

Figure 16. The red arrow points to the hydraulic pump drive belt that was displaced forward.

2.5 ENGINE

2.5.1 OVERVIEW

The Arriel 2B1 turboshaft engine features an axial air intake, a single-stage axial compressor and a single-stage centrifugal compressor, an annular combustor, and a single-stage turbine rotor that drives the compressor, and a single-stage free (power) turbine rotor (**Figure 17**). A reduction gearbox, mounted to the rear of the engine, is driven by a splined coupling (also known as a "muff coupling") connected to the free turbine rotor. The reduction gearbox provides final drive to the power transmission shaft. The freewheeling unit is mounted to the forward end of the power transmission shaft. The Arriel 2B1 contains a dual-channel digital engine control unit (DECU) that controls a hydromechanical fuel metering unit. Additionally, an electrical engine backup control auxiliary unit (EBCAU) is fitted to the fuel metering unit and is mechanically linked to the metering needle bypass valve.

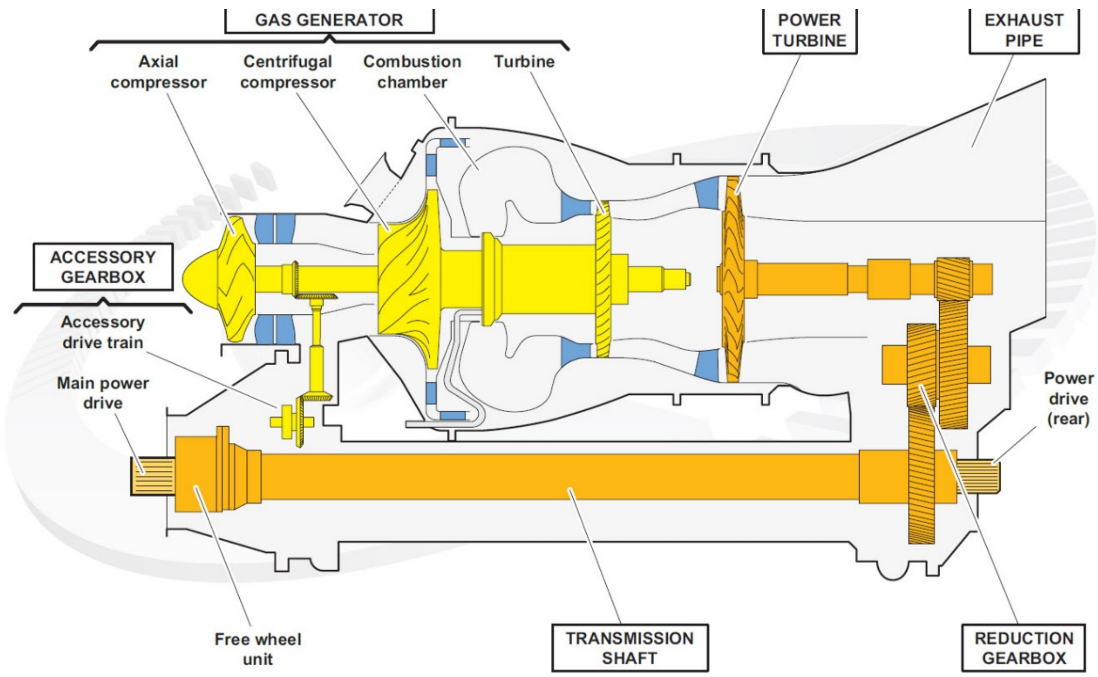

Figure 17. A cross-sectional view of the Arriel 2B1 engine. **(Image courtesy of Safran Helicopter Engines)**

The Arriel 2B1 is of modular design where each subassembly (module) is an independent unit that can be removed, inspected, and replaced on-site without complex tooling or full disassembly of the engine. The engine has five modules, each with its own identification plate, which are identified in **Table 1** and seen in **Figure 18**.

Table 1. Engine module descriptors.

Module No.	*Nomenclature*
Module 1 (M01)	Transmission shaft and accessory gearbox
Module 2 (M02)	Axial compressor
Module 3 (M03)	Gas generator
Module 4 (M04)	Power turbine
Module 5 (M05)	Reduction gearbox

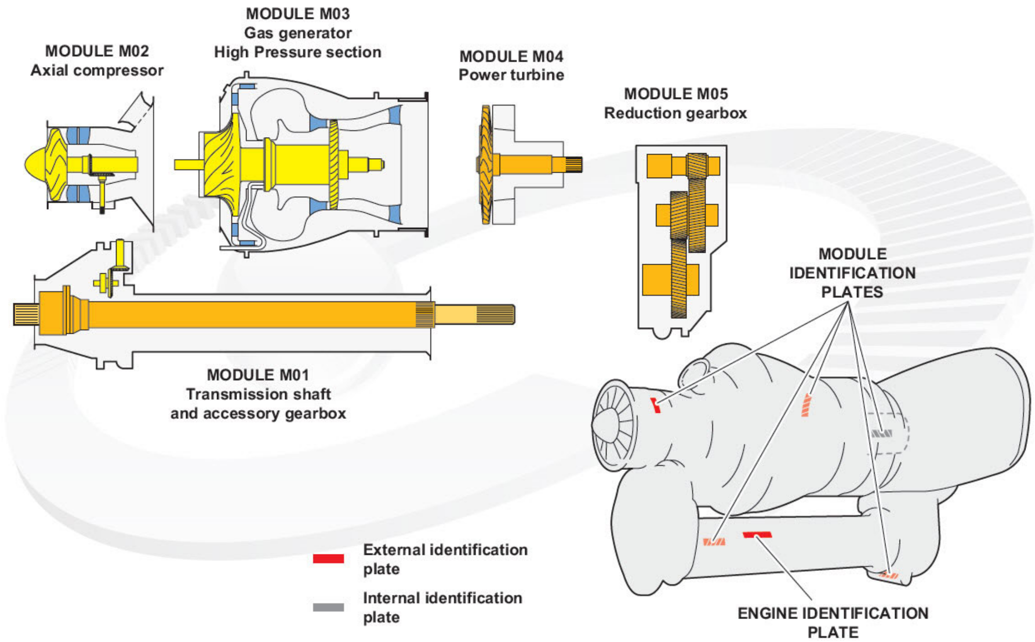

Figure 18. The Arriel 2B1 engine modules. **(Image courtesy of Safran Helicopter Engines)**

2.5.2 ENGINE MOUNTING

The engine was found loose within the area of the engine compartment and only remained attached to the helicopter structure via fuel lines and electrical cables (**Figure 20**). There was no evidence of thermal or fire damage on the engine. The exterior of the engine did not exhibit any signs of case breaches or uncontained engine failures. The power turbine containment ring was intact and undamaged.

Figure 20. The engine resting on the recovered wreckage.

The forward half of the engine rear straddle mount was distorted downward and in the aft direction, while the aft half appeared to exhibit no distortion or bending. Only one of the two clamp straps was still intact and attached to the M01 tube; the strap was distorted down and forward and was located where the forward strap was normally installed. The protection tube exhibited wear indications that both straps were in place and secure, and exhibited inward impact damage on the bottom of the protection tube. The rear saddle mount remained installed on the engine deck.

The locking castellated nut, adjacent to the splines on the forward end of the power transmission shaft, exhibited rubbing wear consistent with contact with the inner diameter of the engine-to-transmission (input) driveshaft. The inner diameter of the input driveshaft exhibited corresponding contact wear.

2.5.3 COMPRESSOR SECTION

All axial compressor blades were present and of full length. Every axial compressor blade exhibited some combination of hard- and soft-body leading edge impact damage and airfoil material loss (**Figure 21**).[4] The axial compressor turned freely and smoothly when rotated by hand. The air inlet cone was intact and exhibited light circumferential scoring. The flow path of the compressor case exhibited light circumferential scoring along the blade running position.

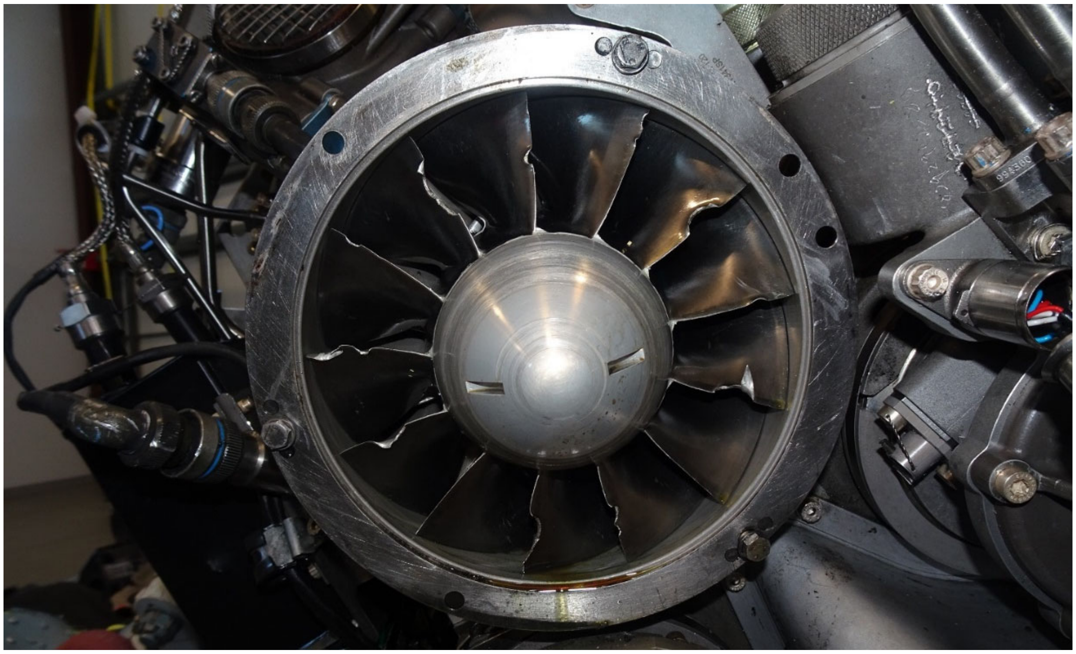

Figure 21. A view of the leading edge damage on the engine axial compressor blades.

The starter-generator was removed and manual rotation of the starter-generator pad resulted in rotation of the axial compressor, confirming continuity of drive between the accessory gearbox and the gas generator.

[4] Hard-body impact damage is characterized by a serrated appearance and deep cuts or tears to a blade's leading edge. Hard-body impact damage can result from ingestion of metal parts, concrete, asphalt, or rocks. Soft-body impact damage, which results from impacts with pliable objects, is characterized by a large radius of curvature or curling deformation to the blade and can cause curling deformation of the blades in the direction opposite of normal rotation.

2.5.4 TURBINE AND REDUCTION GEARBOX SECTIONS

Examination through the engine exhaust pipe revealed all power turbine blades were present, full length, and intact (**Figure 22**). Continuity of drive between the power transmission shaft, reduction gearbox, and the power turbine was confirmed by rotating the power transmission shaft (at the front of the engine) and observing a corresponding rotation of the power turbine. Normal operation of the freewheeling unit was also confirmed.

Figure 22. A view of the leading edge of the power turbine blades (after removal from the engine).

The reduction gearbox and power turbine modules were removed from the engine. The index mark between the input pinion gear and its splined nut was offset by about 3 millimeters in the counterclockwise (tightening) direction (**Figure 23**).[5] The power turbine was rotated by hand and exhibited smooth movement. The power turbine shroud exhibited light circumferential scoring along the blade running position and the blades had no anomalous leading or trailing edge airfoil damage. The power turbine nozzle guide vane assembly was intact and exhibited no impact damage. All of the power turbine nozzle guide vanes were covered with metal splatter that had adhered to the suction (convex) side of the airfoils.

All gas generator turbine blades were present, full length, and intact. Looking past the gas generator turbine blades, the gas generator nozzle guide vanes were visible. All gas generator nozzle guide vanes were present and intact. The airfoil suction side of all gas generator nozzle guide vanes airfoil suction side were covered with metal splatter. Continuity of drive between the axial compressor and gas generator turbine was confirmed via manual rotation of the axial compressor.

[5] Within the reduction gearbox is a splined nut that secures the input pinion gear and also transfers drive from the muff coupling to the input pinion gear. After the splined nut is installed, an index mark is inscribed between the splined nut and the input pinion gear. The quantity and direction of an offset between the two index marks can be measured to indicate certain circumstances experienced by the engine's power transmission system, such as an overtorque. The splined nut has left-handed threads that rotates in the counterclockwise direction for tightening.

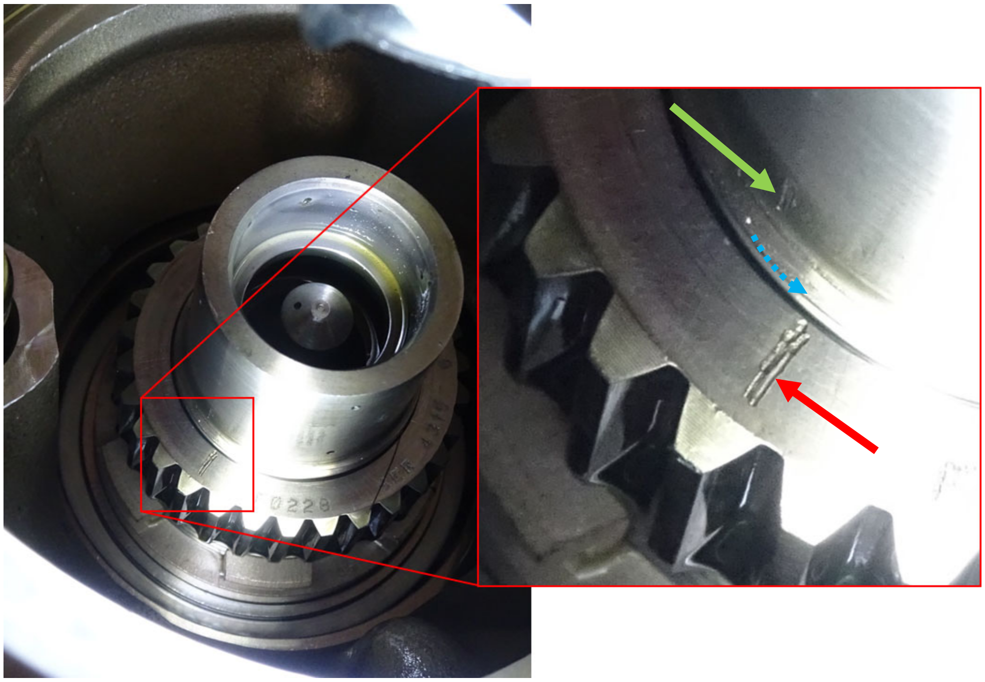

Figure 23. The index mark on the splined nut (red arrow) and the pinion gear (green arrow) was offset by about 3 millimeters. The blue arrow shows the direction of the index mark's movement.

2.5.5 LUBRICATION

The lubrication system magnetic chip detectors from the accessory and reduction gearboxes as well as the electrical magnetic chip detector from the accessory gearbox were removed and revealed no evidence of magnetic chips or debris. The reduction gearbox magnetic chip detector was fractured and missing its locking assembly, but the magnetic plug itself was still installed within the reduction gearbox. The oil filter cover's bypass indicator was not extended ("popped").[6] The fuel filter bypass indicator, similar in function to the oil bypass indicator, was not extended. Both oil and fuel filter elements were removed and exhibited no evidence of debris within the filter pleats. The oil appeared to be in good condition and did not have an acrid smell. The fuel appeared clear and smelled of jet fuel. No debris was noted in the samples collected from the low pressure fuel inlet line (to the hydromechanical unit) as well as the fuel filter bowl.

2.5.6 ENGINE CONTROLS

The DECU remained installed and secured within the airframe baggage compartment. The DECU was removed from the helicopter and appeared intact with no evidence of damage or breaches to the box. All electrical connectors to the DECU were secure. The data plate for the DECU showed it was manufactured by Thales in June 2008 and was part number (P/N) 70BMF01020 and S/N 7089. The DECU was shipped to Safran Helicopter Engines in Grand Prairie, Texas to download its onboard data. See Section 3.1 of this report for additional details of the data download.

[6] When the filter bypass indicator is extended, there is a visible red-colored bypass indicator to alert personnel that there may be a current or impending clog within the filter.

The EBCAU input has a keyway that indicates that the bypass valve is in the neutral position if at 12 o'clock relative to the fuel metering unit, indicative that the engine fuel is being metered by the DECU. The EBCAU was removed from the fuel metering unit and its input shaft was observed to be in the 12 o'clock position.

2.6 MISCELLANEOUS ITEMS

2.6.1 COCKPIT INSTRUMENTS

The instrument panel was generally intact with all instruments still installed on the panel (**Figure 24**). The pitot tube was found lodged within the instrument panel. The hour-meter displayed 5,673.7 hours. The vehicle engine monitoring display (VEMD) remained installed within the instrument panel. The VEMD was removed and its data sticker showed P/N B19030MD05 and S/N 6109. The VEMD was shipped to the Bureau d'Enquêtes et d'Analyses pour la Sécurité de l'Aviation Civile (BEA) in Le Bourget, France to download its onboard data. See Section 3.2 of this report for additional details of the data download.

Figure 24. The recovered instrument panel. The VEMD is composed of the two screens at the center of the instrument panel.

The 30-alpha panel assembly, containing multiple buttons and circuit breakers, was partially separated from the airframe and leaning forward. The upper cover of the 30-alpha panel was partially separated on its left side. The blue-colored circuit breakers were present except for four on the forward end. For the circuit breakers on the left side of the 30-alpha panel, the "TDR" and "ALT C" circuit breakers were in the open ("popped") position while the remainder of the circuit breakers remained in the closed position. All circuit breakers in on the right side of the 30-alpha panel were in the closed position. The helicopter battery was not within the airframe and the battery panel was deformed outward.

The control cables for both the emergency fuel shutoff and rotor brake levers were fractured from their swage fitting in the area of the canopy fracture. Both levers were found in the stowed position. The engine start switch was found in the "on" position. The EBCAU test button was in the "out" position with the button cover intact. Investigators pushed the EBCAU test button; it did not pop out fully, but did pop out to some degree when not depressed.

3.0 DATA DOWNLOADS

3.1 DECU DOWNLOAD

The DECU is equipped with nonvolatile memory (NVM) that can be downloaded for analysis.[7] The DECU has two independent channels, Channel A and Channel B, and each channel runs the same software. A channel selector within the DECU will select which channel will control the engine, depending on the signals it receives from both channels. The normal channel in command is Channel A. If Channel A fails, the system will automatically switch to Channel B. Discrete airframe and engine inputs are received and shared by the two channels, and the discrete outputs are controlled by the selected channel.

On April 8, 2021, the recovered DECU was downloaded by Safran Helicopter Engines in Grand Prairie, Texas with oversight from the FAA. The downloaded data comprised of Fault Context[8] files for both Channels A and B as well as Technical Occurrences[9] files for both Channels A and B. Engine parametric data included gas generator speed (NG), power turbine speed (NF), engine torque, and power turbine inlet temperature (T4).[10]

Review of the Channel A Fault Context data file showed that a Collective Pitch Fault (Fault Block 1/8) occurred at 11 minutes 35 seconds (HH:mm:ss) after PowOn Time and 16,200 total Number of Power-Ons with the Channel A in command and the engine in Flight Mode; the engine torque was 96.03%, NG was 86.52%, and NF was 44.57%. With the same PowOn Time and same total Number of Power-Ons, a raw T4 fault (Fault Block 2/8) occurred along with the Collective Pitch Fault from Fault Block 1; the engine torque was 111.19%, NG was 90.70%, and NF was 25.63%. The data shows an increase in engine torque with a corresponding decrease in NF at the time of the faults (**Figure 25**). The remaining six Fault Blocks were correlated with PowOn Times and total Number of Power-Ons that preceded the first two Fault Blocks by many operations/flights.

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[7] NVM is a type of memory that retains stored information after power is removed.

[8] The Fault Context files record the last 8 fault blocks in chronological order. When a fault is recorded, a snapshot of the engine parameters and logic words at the time of the fault (time = n), 20 milliseconds before the fault (time = n-1), and 20 milliseconds after the fault (time = n+1) are also recorded. If more than 8 fault blocks are recorded, the most recent fault block will overwrite the oldest fault block. Thus, any of the 8 Fault Context fault blocks could be the most resent fault block recorded. The order of the faults can be determined by other DECU-specific counting parameters, such as power-on time and total number of power-ups.

[9] The Technical Occurrence files record a maximum of 32 fault blocks in chronological order but does not record engine parameter during, before, or after the fault. The most recent fault block is block No. 1 and the oldest fault block is block No. 32. The DECU clock time starts when the DECU is energized, referred to as *PowOn Time*, and has no correlation to the helicopter's flight time nor any standard clock time, such as Coordinated Universal Time (UTC). Another DECU reference indicator is the number of times the DECU has been powered-up, referred to as total Number of Power-Ons, which is a running tally of all the times the DECU has been turned on.

[10] NG is also known as N1. NF is also known as N2.

Block 1/8 Ack/Status

Numeric parameters - Block 1/8

Designation	Value
[74] PowOnTime (HH:mm:ss)	00:11:35
[306] Power-on time counter (h)	7867
[76] Number of power-on	16200
[261] T0 engine (DegC)	-13.43
[212] P0 engine (mbar)	796.8
[53] Torque (%)	96.03
[346] N1 (%)	86.52
[344] N2 filtre (%)	44.57
[260] T4 conforme (DegC)	650.5
[336] Rudder bar anticipation (%)	53.46
[264] P3 (bar)	5.208
[332] XR (Deg)	43.17
[237] Actuator position offset (Deg)	-0.42
[54] Corrected N1 demand (%)	89.12
[7] Flow demand (l/h)	179.7
[232] Torque conformation slope	1.05
[235] Torque conformation offset (%)	1.84
[233] T4 conformation slope	0.98
[234] T4 conformation offset (DegC)	-28.3
[56] Collective pitch (XPC) (%)	12.98
[66] dN1/dt (%)	10.25
[203] dN2/dt (%)	-38.21
[327] N1 max since power-on (%)	97.47

[360] Fault flags 1 channel A at cycle n+1 - Block 1/8

1 Watchdog trip	0	9 Raw torque fault	0
2 N2 trim fault	0	10 T4 conformation fault at power-up	0
3 Start selector fault	0	11 Torque conformation fault at power-up	0
4 At least one fault on channel A	1	12 T4 conformation fault after power-up	0
5 Collective pitch fault	1	13 Torque conformation fault after power-up	0
6 Raw T4 fault	0	14 EECU internal fault	0
7 T0 helicopter fault	0	15 P3 drift or engine flame-out	0
8 P3 fault	0	16 P0 engine fault	0

Numeric parameters - Block 2/8

Designation	Value
[74] PowOnTime (HH:mm:ss)	00:11:35
[306] Power-on time counter (h)	7867
[76] Number of power-on	16200
[261] T0 engine (DegC)	-13.43
[212] P0 engine (mbar)	796.8
[53] Torque (%)	111.19
[346] N1 (%)	90.70
[344] N2 filtre (%)	25.63
[260] T4 conforme (DegC)	644.4
[336] Rudder bar anticipation (%)	53.45
[264] P3 (bar)	5.692
[332] XR (Deg)	49.66
[237] Actuator position offset (Deg)	-0.41
[54] Corrected N1 demand (%)	92.99
[7] Flow demand (l/h)	204.6
[232] Torque conformation slope	1.05
[235] Torque conformation offset (%)	1.84
[233] T4 conformation slope	0.98
[234] T4 conformation offset (DegC)	-28.3
[56] Collective pitch (XPC) (%)	14.55
[66] dN1/dt (%)	7.79
[203] dN2/dt (%)	-15.34
[327] N1 max since power-on (%)	90.70

[360] Fault flags 1 channel A at cycle n+1 - Block 2/8

1 Watchdog trip	0	9 Raw torque fault	0
2 N2 trim fault	0	10 T4 conformation fault at power-up	0
3 Start selector fault	0	11 Torque conformation fault at power-up	0
4 At least one fault on channel A	1	12 T4 conformation fault after power-up	0
5 Collective pitch fault	1	13 Torque conformation fault after power-up	0
6 Raw T4 fault	1	14 EECU internal fault	0
7 T0 helicopter fault	0	15 P3 drift or engine flame-out	0
8 P3 fault	0	16 P0 engine fault	0

Figure 25. DECU Channel A Fault Context Data – engine parameters and logic words. (Courtesy of Safran Helicopter Engines)

Review of the Channel A Technical Occurrences data file showed that the 4 seconds (11 minutes 39 seconds after PowOn Time and 16,200 total Number of Power-Ons) after the Collective Pitch Fault and the Raw T4 fault occurred, a Stop, Idle, and Flight selector fault occurred along with a NF measurement sensor fault (Fault Block 1/32). Two seconds (11 minutes 37 seconds after PowOn Time and 16,200 total Number of Power-Ons) after the Collective Pitch Fault and the Raw T4 fault occurred, an alternate direction current (ADC) Channel B operating system software (OSS) fault occurred (Fault Block 2/32) (**Figure 26**). Based on the same timestamps, these two sets of Fault Blocks were likely part of the accident flight. The remaining 30 Fault Blocks were correlated with PowOn Times and total Number of Power-Ons that preceded the first two sets of Fault Blocks by many operations/flights.

Block 1/32 Ack/Status

Numeric parameters - Block 1/32

Designation	Value
[74] Diagnostic time (HH:mm:ss)	00:11:39
[75] Block number	1
[76] Power up number	16200
[77] Fault over flow	0

Failures - Block 1/32

	Identification	Localisation
Panne 1	T4 Measurement / AS	T4 Sensor
Panne 2	Collective pitch measurement / AS	Collective pitch potentiometer
Panne 3	E/L ARR / AS	DECU
Panne 4	E/L RAL / AS	DECU
Panne 5	E/L VOL / AS	DECU
Panne 6	N2 measurement sensor 1 / AS	DECU
Panne 7	No failure	No failure
Panne 8	No failure	No failure

Stop

Idle

Flight

Block 2/32 Ack/Status

Numeric parameters - Block 2/32

Designation	Value
[74] Diagnostic time (HH:mm:ss)	00:11:37
[75] Block number	2
[76] Power up number	16200
[77] Fault over flow	0

Failures - Block 2/32

	Identification	Localisation
Panne 1	ADC channel B / OSS	DECU
Panne 2	Collective pitch measurement / AS	DECU
Panne 3	No failure	No failure
Panne 4	No failure	No failure
Panne 5	No failure	No failure
Panne 6	No failure	No failure
Panne 7	No failure	No failure

Figure 26. DECU Channel A Technical Occurrences. (Courtesy of Safran Helicopter Engines)

3.2 VEMD DOWNLOAD

The VEMD was received at the BEA on April 26, 2021. The unit was opened and the two main circuit boards (one per module) were removed and were in good condition. The two memory chips, one per circuit board, were unsoldered and its data recovered. The data relative to the last recorded flight was associated with Flight No. 8585, the accident flight.[11] During Flight No. 8585, five failures were recorded on both modules of the VEMD: *TEST FADEC 4*; *TEST FADEC 5*; *TEST FADEC 3*; *SURV REC NF A*; and *TEST FADEC 28 EC130/B3 2B1*. **Table 2** shows the details of the five failures recorded to the NVM and the failures are described in detail below.

The *TEST FADEC 4* failure occurs when a *collective pitch failure* is received from the DECU. A failure of the collective pitch anticipator potentiometer position triggers when the measurement of the relative position of the potentiometer reaches its minimum (5%) or maximum (95%) thresholds or when its variation is above 350% per second. The position of the collective pitch anticipator potentiometer is used by the DECU for power anticipation in order to maintain rotor speed (NR) within an acceptable range. In case of failure, the DECU remains in automatic mode and uses a backup law for the collective pitch values.

The *TEST FADEC 5* failure occurs when a *raw T4 failure* is received from the DECU. A raw T4 failure is triggered when there is a loss of redundant T4 information from the VEMD or if the T4 measurement reaches an out-of-limit value. In case of failure, the DECU uses a backup law for the T4 values.

The *TEST FADEC 3* failure occurs when a *selector failure* is received from the DECU. This failure corresponds to a discrepancy of the STOP/IDLE/FLIGHT selector. A combination different from 1/0/0, 0/1/0, or 0/0/1 leads to the discrepancy detection. When this discrepancy is detected in flight, the control system remains in FLIGHT mode.

The *SURV REC NF A* failure occurs when there is an invalid value of NF received by the VEMD. Example causes of an invalid value of NF include a failure of the power turbine speed sensor or wire harness damage.

The *TEST FADEC 28 EC130/B3 2B1* failure occurs when a *NF failure* is received from the DECU. This failure can occur when the NF value is out of limits, the rate of change of NF is out of limits, or if the difference between both channels is higher than 3%.

According to the downloaded VEMD data, the last engine power check was performed during Flight No. 8571. No discrepancies were noted on the last 8 engine power checks.

[11] A VEMD flight number begins when NG is above 60% and ends when NG is below 50%. The flight data is recorded to the NVM every 20 seconds.

Table 2. The recorded failure data from Flight No. 8585 recovered from the VEMD.[12]

Flight Time (*mm:ss*)	Failure Label	NG (%)	NG FADEC (%)	Torque FADEC (%)	T4 (°C)	T4 FADEC (°C)	NF (rpm)	NR (rpm)	OAT (°C)
10:05.50	TEST FADEC 4	85	87	96	644	655	181	--	--
Fault 1 DECU - At least one fault on Channel A - Collective pitch failure		**Fault 2 DECU**		**Logic Out, 1 DECU** - Self-test completed - Governor on FLIGHT		**Logic Out, 2 DECU** - Channel A monitoring - Automatic mode		**Logic Input DECU** - Neutral position - Selector on FLIGHT - Bleed valve opened - TRIM2 input - TRIM 3 input	
10:06.00	TEST FADEC 5	88	90	111	662	644	92	--	--
Fault 1 DECU - At least one fault on Channel A - Collective pitch failure - Raw T4 failure		**Fault 2 DECU**		**Logic Out, 1 DECU** - Self-test completed - Governor on FLIGHT		**Logic Out, 2 DECU** - Channel A monitoring - Degraded operating mode - Automatic mode		**Logic Input DECU** - Neutral position - Selector on FLIGHT - Bleed valve opened - TRIM2 input - TRIM 3 input	
10:07.50	TEST FADEC 3	94	94	72	749	737	302	--	--
Fault 1 DECU - At least one fault on Channel A - Collective pitch failure - Raw T4 failure - Selector failure		**Fault 2 DECU** - At least one fault on Channel B		**Logic Out, 1 DECU** - Self-test completed - Governor on FLIGHT - Bleed valve closed		**Logic Out, 2 DECU** - Channel A monitoring - Degraded operating mode - Automatic mode		**Logic Input DECU** - Neutral position - TRIM2 input - TRIM 3 input	
10:08.00	SURV REC NF A	94	--	68	752	--	316	174	-13
10:08.00	TEST FADEC 28 EC130/B3 2B1	94	94	54	753	748	310	--	--
Fault 1 DECU - At least one fault on Channel A - Collective pitch failure - Raw T4 failure - Selector failure		**Fault 2 DECU** - At least one fault on Channel B - NF failure		**Logic Out, 1 DECU** - Self-test completed - Governor on FLIGHT - Bleed valve closed		**Logic Out, 2 DECU** - Channel A monitoring - Automatic mode		**Logic Input DECU** - Neutral position - TRIM2 input - TRIM 3 input	

[12] NG and T4 parameters are recorded twice, one with the value coming from a sensor dedicated to the VEMD and one from another sensor dedicated to the FADEC.

4.0 MAINTENANCE

According to the operator, the accident helicopter was maintained under the manufacturer's recommended maintenance program. The helicopter was based out of Wasilla Airport (PAWS), in Wasilla, Alaska. The accident helicopter was parked at, and departed from, PAWS at the start of the day on March 27, 2021. According to the operator, a mechanic would have been at PAWS in order to perform an 'after last flight' inspection and perform any necessary maintenance actions after the helicopter's return. The operator utilized a computerized system for tracking maintenance, overhauls, inspections, airworthiness directives, service bulletins, component times, and inventory. According to the maintenance tracking report, there were no overdue items for hour-limited, cycle-limited, and calendar inspections. Additionally, there were no overdue airworthiness directives and service bulletins.

The aircraft logbook was normally stored at the operator's base in PAWS for day trips that originated and ended at PAWS. For trips to other destinations, the aircraft logbook would travel with the helicopter. The accident helicopter's logbook remained at PAWS on March 27, 2021. The aircraft logbook entry dated March 27, 2021 showed a signature in the section that included the 'before first flight' inspection. The entry also showed an ATT of 5,674.1 hours and an engine ETT of 5,568.5 hours. The aircraft logbook entry dated March 26, 2021 showed that, before the first flight of the day, a 7-day/15-hour engine inspection was completed. Additionally, a loose drain line was tightened and the aircraft weight and balance was updated for the installation of the skid-mounted cargo basket. Furthermore, after the last flight of the day, the engine was serviced with ½ quart of oil. **Attachment 1** to this report contains the aircraft logbook entries for March 26 and 27, 2021.

The last annual inspection of the accident helicopter was completed on March 3, 2021 under Soloy Helicopters work order No. 5286. Additional inspections completed under work order No. 5286 included the 30-hour, 50-hour, 100/150-hour, 300-hour, 12-month, 600-hour, 600-hour/24-month, 1200-hour/48-month, 72-month, 5000-hour/72-month, and 144-month airframe inspections as well as the 7-day/15-hour, 30-hour, 300-hour, and 600-hour engine inspections. At the completion of work order No. 5286, the helicopter had an ATT of 5,651.0 hours and an ETT of 5,545.4 hours. Additionally, the gas generator had accumulated 11,196.40 CSN and the power turbine had accumulated 3,912.65 CSN at the completion of work order No. 5286.

Chihoon Shin
Aerospace Engineer – Helicopters