Ministry of Economy and Sustainable Development of Georgia

Civil Aviation and Maritime Transport Accident Incident Investigation Bureau

Final Report on accident to Bell 505 helicopter in Kazbegi municipality, Georgia on 6 June 2019

Tbilisi 2020

Helicopter type BELL-505

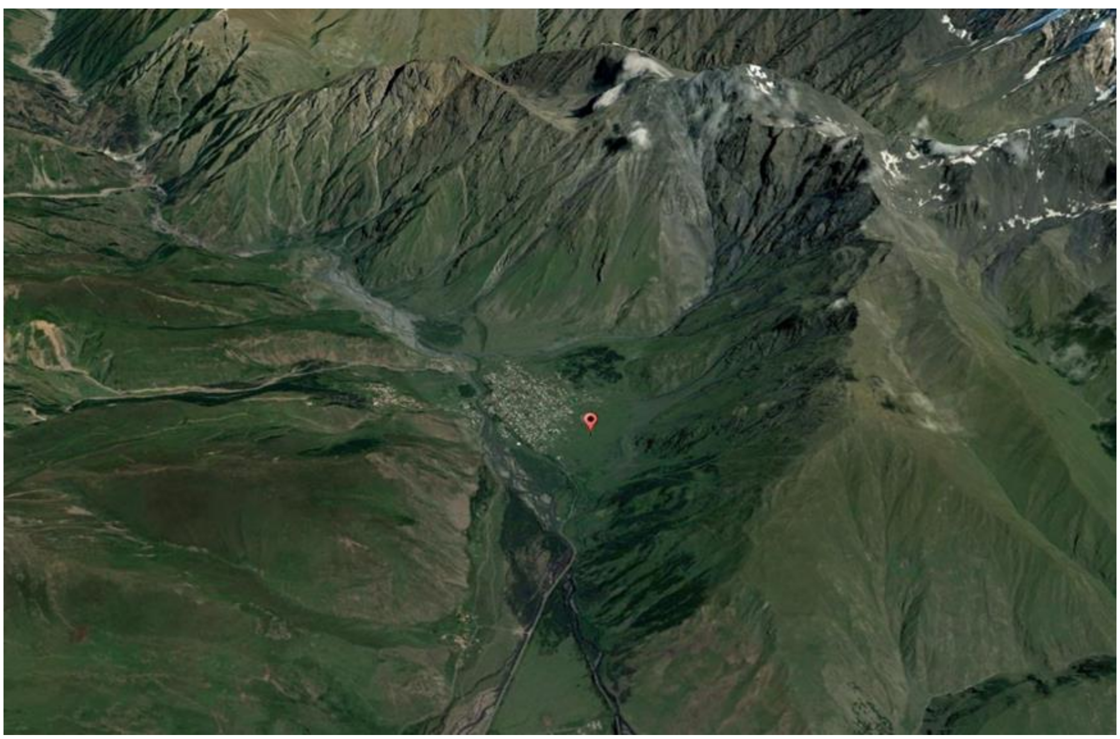
Nationality and registration marks 4L- ADJ

Operator "Aviator+" LLC

Time and date of the accident 10:57 (local time) 06.06.2019

Accident location "Kazbegi" municipality settlement "Stepantsminda" (Georgia).

Geographical coordinates N 42°38' 59'' E 44°38' 49''



The sole objective of the investigation of an accident or incident shall be the prevention of future accidents and incidents. It is not the purpose of this investigation to apportion blame or liability. (Convention on International Civil Aviation (Chicago, 1944), Annex 13, Article 3.1)

Content

Introduction

On 6 June 2019, at 12:00 local time (UTC+4) The Civil Aviation and Maritime Transport Accident/Incident Investigation Bureau received a notification about an aviation accident, involving a helicopter Bell-505. The aircraft was operated by “AVIATOR +” with Georgian nationality and registration marks 4L-ADJ. The helicopter departed from the concrete helipad at the base of“Kazbegi Helicopters” in the direction of “Stepantsminda” Hotel “Rooms”. Approximately 2 minutes after departure the helicopter collided with the terrain. All 3 occupants (the pilot and 2 passengers) received fatal injuries.

According to the Air Code of Georgia, Provision of The Ministry of Economic and Sustainable Development, Provision of the Investigation Bureau, “Rule on Civil Aviation and Maritime Transport Accident/Incident Investigation”, and ICAO Annex 13 to the Convention on International Civil Aviation, an investigation of this occurrence was performed by Civil Aviation and Maritime Transport Accident/Incident Investigation Bureau.

In accordance with the ICAO Annex 13 requirements, for participation in an investigation, the France as the state of Manufacture of the Bell-505 helicopter engine appointed an accredited representatives and advisers from the “Bureau of Enquetes et D’Analyses” (BEA), as well as an consultant-adviser in technical matters from the engine manufacturer „Safran Helicopter Engines“.

In addition, “Bell Helicopters”- USA (on behalf of the helicopter manufacturer Bell Canada) provided experts to the investigation.

Investigation started	June 2019
Investigation finished	November 2020

Factual information

1.1 History of the flight

The preflight events were captured on CCTV video footage at the Kazbegi Helicopters base. On 6 June 2019, the occurrence helicopter being hand pulled to helipad, located next to the hangar by the helicopter PIC and 2 hotel employees, who were also the passengers on the occurrence flight.

The preflight inspection was performed by the PIC. At about 10:55 the helicopter flew up with 2 employees of the Hotel “Rooms” in the direction of the Hotel “Rooms”. As the video footage, which was taken by the mobile phones of the eye-witnesses show, the helicopter took off vertically, afterward at high altitude and turned in the direction of the settlement “Stepantsminda”.

As the recordings of the CCTV cameras located on the base “Kazbegi Helicopters”, also “Truck Terminal” and “Socar” petrol station shows, after takeoff in about 2 minutes, the helicopter suddenly started descend till collision with the ground.

1.2. Injuries to persons

Helicopter pilot and 2 passengers were killed on the accident site

Injuries	Crew	Passengers	Other persons
Fatal Injuries	1	2	0
Serious Injuries	0	0	0
Minor/Uninjured	0	0	0

1.2 Damage to the aircraft

The helicopter was destroyed upon impact. There was no post impact fire.

The helicopter Emergency Locator Transmitter (ELT) automatically was activated and start send signals.

1.3 Other damages

The Helicopter collided into ground occurred in the entrance of the settlement “Stepantsminda”, on a slope, away from residential homes and outside the land fenced with wire and iron poles. Subsequently the inertia collided with the iron poles of the fence and, as a result of several rotational movements, collapsed and landed in the aforementioned fenced land. The helicopter did not damage any other objects.

1.5. Information about PIC (Pilot in command)

PIC	Georgian citizen
Age	45
Profession	Pilot
Gender	Male
First issue of the pilot license	15.03.2012
ATPL №	GE-ATPL-00119
Date of issue	01.08.2017
ATPL validity rating	01.08.2019
Medical examination Class I	№456/18
Medical license validity	01.08.2019
Flight technic and aeronautical Examination	19.04.2019
Total flight hours	1413:01 hrs.
Flight hours on BELL-505	85:37 hrs. (right seat)
Flight hours on BELL-505 PIC	85:37 hrs.
Flight hours on AS 350	107:25 hrs.
Flight hours on A 22	7 hrs.
Flight hours on AW109	141 hrs.
Flight hours on MI-8T	172:24hrs
Flight hours on MI-8MTV	78:03hrs
Flight hours on MI-24	416:31hrs
Flight hours on TH-67	210 hrs.
Flight hours on UH-1H	142:50 hrs.
Flight hours on BELL-212	12 hrs.
Flight hours on BO-105	4:11 hrs.
Flight hours on 2B-24	36 hrs.

Flight and duty time Information:

Last 90 days flight hours	27 hrs. 05 minutes
Last 30 days flight hours	21 hrs. 48 min
Last 24 hrs. Flight hours	0.25 hrs.
Previous accidents	none

1.6 Engine and helicopter information

Helicopter type	BELL-505
Factory identification number	65074
National registration number	4L-ADJ
Manufacturer	BELL
Manufacturing date	March of 2018
Total hours since manufacturing	162 hours.
Number of landings	415

Information about engine

Type	Arrius 2R
Manufacturer number	50065
Manufacturer	Safran Helicopter Engines
Manufacturing date	22.03.2017
Work hours since manufacturing	162 hours.
Remaining resource	2838 hrs.
Remaining cycles	9585

Helicopter Main transmission EPICYCLIC REDUCTION GEAR

Type ARRIUS 2R

P / n 206-040-004-117

S / n HB 571947

The overhauling resource is 4500 hrs.

The remaining resource is 4338 hrs.

TAIL GEAR BOX

P / n	206-040-402-111
S / n	BH 507663
The overhauling resource is	6000 hrs.
The remaining resource is	5833 hours 8 min.
Last check was provided	21.05.2019

Technical log. No 35587. Line Maintenance Work Order # 65074/19/10

Helicopter airworthiness certificate	**№ 476**
Validity Date	29.06.2019
The issuing authority	Civil Aviation Agency of Georgia
Issued on	29.06.2018
Owner of the helicopter	LTD "Aviator +"
Helicopter operator	LTD "Aviator +"
Helicopter Operator certificate	**No 086**
The issuing authority	Civil Aviation Agency of Georgia
Date of issue	21.11.2018
Validity Date	22.11.2019

1.7. Meteorological information

The actual weather during the occurrence in the surrounding area of the flight was: Visibility over 6-7 km, the lower cloud base was 4000 meters AGL, wind direction 220^0/10 knots.

The weather was not a factor in this accident.

1.8 Navigation aids

Navigation aids were not a factor in this accident.

1.9 Communication

Communication was not a factor in this accident. The flight did not take place in controlled airspace (CTR)

There was no MAYDAY call by the pilot.

1.10. Aerodrome information

The base of “Kazbegi Helicopter” in “Kazbegi” Municipality Settlement Stepantsminda meets the existing requirements.

The helipad arranged in the area of the Rooms hotel meets the existing requirements.

1.11. Flight recorders

Flight data recorder (FDR) and cockpit voice recorder (CVR) on the helicopter BELL-505, national registration 4L- ADJ are not required by the international regulation and were not installed on this aircraft.

1.12. Wreckage and impact information

The scene of the accident site indicated the following:

The helicopter after the first collision with the ground, by inertia, collided with the iron poles of the fence and, as a result of several rotational movements, collapsed and landed In 43 Meters from the first bounce, it turned opposite the flight direction (North-east direction 30 °), nose-to-west in the direction of the 275 °, at 1797 meters above sea level (ASL). The geographical coordinates were :

N 42°38’ 59’’ E 44°38’ 49’’

Around the perimeter of the helicopter wreckage, there were scattered parts consisting of glass, plastic, and metal debris.

About 43 meters southwest from the helicopter, there is a trace of a 40-centimeter-long (northeastward direction), Caused by helicopter collision with the ground (Probably, the place of touch of the vertical tail stabilizer during the attempted landing).

In the south-west in 41 meters from the helicopter, there are two parallel two-meter long trails, were landing skid ground scars, that were essentially oriented in the direction of travel since they pointed to the final resting location of the helicopter main wreckage.

In the south-west in 35 meters from the helicopter, there are two parallel excavated trails (04 and 01 meters), which are directed towards north-east (Most likely touchpoint of landing skid).

In 30 meters southwest from a helicopter, on the ground Observed black metal construction (Left landing skid of the helicopter).

From the helicopter about 30 meters southeast, There is a black metal construction on the ground, Left landing skid of the helicopter, Next to it lies a metal tubing detail of a dark red color (landing skids connector and holder with the fuselage of the helicopter), Which has external longitudinal damages.

From the helicopter, about 25 meters southwest, on the ground lies a black metal rod, Part of the right landing skid.

In 155 and 158 meters southeast from the helicopter, on the ground, lies the tail rotor driveshaft (connecting shaft of the main rotor transmission and tail gearbox), which is divided into two parts.

In the south-east in 91 meters from the helicopter, there were two details of the dark red rectangular helicopter details with a penetrating hole on the ground (a part of the horizontal stabilizer).

In 66 meters east from a helicopter, on the ground lies a rectangular dark-red detail, part of helicopter's tail (Part of the horizontal stabilizer).

The helicopter's tail after collision with the ground was removed from the fuselage and split into two parts. The first part is 9 meters in the south direction from the fuselage, and the other part is 57 meters in the south-east direction from the fuselage.

In the south-east in 55 (fifty-five) meters from the helicopter, there is a dark red rectangular detail of the helicopter tail (a part of the horizontal stabilizer) on the ground.

From the helicopter to the southeast within a 36 meter, on the ground is white electrical wiring, both ends of which are cut off, also next to it there is a small dark red rectangular casing detail of the helicopter.

In the east in 30 meters from the helicopter, there is a part of the tail boom on the ground, a metal cable, which is torn in the end. Next to the tail boom, there is a covering of the tail boom on the ground.

In the east in 19 meters from the helicopter, there is a fragment of the metal cable, approximately 2.7 meters long, both ends of which are torn.

In 12 meters northeast from the helicopter, on the ground lies Helicopter's tail rotor and blades ("white blades"), the blades are damaged, the earphones are attached to it with black wires.

In the south in 9 meters from the helicopter, there is a torn tail boom on the ground.

In 2 meters to the right side from the fuselage, on the ground lies a black, rubber elastic fuel tank.

In the north in 6 meters from the helicopter, there is the left door of the helicopter. Its glass is cracked

In distance, 4 meters south from the helicopter lies damaged the right door of the helicopter.

The location of the tail rotor driveshafts, tail rotor assembly, and stabilizers that were found on the right side of the helicopter wreckage (as defined by direction of travel) are indicative of a main rotor strike to the tailboom area during the impact sequence. The main rotor strike propelled these items away from the main wreckage a significant distance to the right of the wreckage trail due to high energy contact from the main rotor.

The helicopter fuselage construction was substantially damaged and broken in 2 parts. The main part of the fuselage rests on the ground with the left-hand side. The part of the fuselage where is the engine and main rotor are located, Lies with the main rotor on the surface of the earth. The second part with the floor and the cockpit came from above and lies on the main part of the fuselage. Pilot cockpit glasses are crushed, the doors are broken, tools and all devices located in the cockpit are damaged, the pilots instrument panel is moved from the cockpit and lies on the left side of the helicopter. The right display of the flight control system (G1000) is damaged. The front two chairs are damaged and installed in place. The passenger's waist and the shoulder seatbelts were fixated in a buckled condition when the corpse was removed from the helicopter. The PIC waist and the shoulder seatbelts are undamaged, when the pilot's corpse was removed from the helicopter, the wait seatbelts were in a fixed and shoulder seatbelts was not fixed position.

Helicopter cyclic control stick is broken in the top section, also in the cyclic stick holder's section. The collective pitch lever is broken in the attachment point with the floor, the plastic ending of the lever on which is located "fly- idle" mode selector is also broken.

One main rotor blade, the White blade, was found fractured in one meter from a place of fixation consistent with bending overload forces at ground impact and lies in 14 m from the fuselage. The second is firmly fixed and along with the main rotor is under the helicopter.

The helicopter's main rotor's hydraulic pipes and devices are damaged, one of the pitch links is broken in the middle part, and the second one is stuck around the main rotor mast. The helicopter engine is in place of installation, the external protection cowlings are less damaged. The engine exhaust pipe has been deformed by collision.

All aft three backrests of the passenger chairs are fixated, undamaged. The body of the left last chair seat is broken. The passenger's seatbelts on the left last chair are in buckles condition, attachment points on the backrest are fixated and are cut from the attachment when the corpse was removed.

1.13. Medical and pathological information

„Levan Samkharauli National Forensics Bureau" conducted a medical chemical-toxicological examination of the bodies of the pilot and passengers, the results of which are:

Expert opinion №003887619 - 08/10/2019 - "Methyl, ethyl, isopropyl alcohols were not found in the blood taken from the corpse of the helicopter pilot. No drugs or psychotropic substances were found in the blood and internal organs.

Cause of death: combined blunt trauma to the body with damage to skeletal bones and internal organs.

Expert opinion №003889919 - 07/10/2019 - “Carbamazepine anticonvulsant was found in the blood and internal organs taken from the corpse of a passenger sitting to the left of a helicopter pilot. Methyl, ethyl, isopropyl alcohols were not found in the blood. No drugs or psychotropic substances were found in the blood and internal organs.

Cause of death: combined blunt trauma to the body with damage to skeletal bones and internal organs.

Expert opinion №003887819 - 08/10/2019 - “Methyl, ethyl, isopropyl alcohols were not found in the blood taken from the corpse of a passenger sitting in the back of the helicopter. No drugs or psychotropic substances were found in the blood and internal organs.

Cause of death: combined blunt trauma to the body with damage to skeletal bones and internal organs.

1.14. Fire

There was no post impact fire.

1.15. Survival factors

After the aviation accident, according to the explanations of the eye-witnesses (citizens and police officers), who came first to the accident site, 3 people in the helicopter received fatal injuries.

1.16. Tests and research

On the 06.06.2020 – 07.06.2020 initial investigation at the site of the accident was carried out jointly by the Civil Aviation and Maritime Transport Accident and Incident Investigation Bureau of Georgia, the investigating authority and forensic experts of the Ministry of Internal Affairs.

The site of the accident and its environs (the place where the helicopter wreckage was scattered) was inspected and photo and video material was recorded.

To send for further research and testing, samples of fuel and lubricants required for the study, as well as blood and DNA, were taken and sealed.

In the interests of the safety investigations, on 07.06.2019, the helicopter debris were completely collected and main parts of fuselage with engine were moved in hangar three hundred meters away from the crash site. The next examination was carried out from 10.06.2019 to 12.06.2019.

The exam was attended by:

- Bureau of Civil Aviation and Maritime Transport Accident Investigation of Georgia,

- French Bureau of Safety Investigations (BEA).

- Technical consultant from the helicopter engine manufacturer "Safran Helicopter Engines".

- Technical Advisor to the TSB of Canada Accredited Representative

- Representatives of the “Aviator +” airline (pilot, technicians).

- Investigative and forensic expert of the criminal police of the Ministry of Internal Affairs.

During the examination was performed the necessary procedures - checking the helicopter and engine details, taking photos, taking different types of samples.

The first studies conducted on the helicopter debris, engine, and CCTV cameras records taken from various sites, located on the perimeter of the flight showed, that during the occurrence of the accident, when colliding with the ground, the helicopter engine didn’t release power required for flight.

In order to determine, why the helicopter engine didn’t develop the power required for flight, upon a decision of the parties the computer equipment’s of the helicopter. Engine electronic control unit EECU, the Garmin 1000 MFD (left) and PFD (right) monitors, were removed and sealed for future examinations in France.

In July, the helicopter engine was sent to the manufacturing factory (SAFRAN HE), also the computer equipment of the helicopter and engine electronic control unit EECU and GARMIN MFD (left) and PFD (right) monitors, by the Georgian Bureau of accident investigation were personally delivered to Bureau of Enquiry and Safety Analysis (BEA).

The next expertise was performed from 11.06.2019 till 19.06.2019.

The Garmin 1000 MFD and PFD monitors, as well as the Electronic Engine Control Unit (EECU) were checked by the French Bureau of Investigation BEA and at the NCE - European office of GARMIN (France). Namely: visual as well as fluoroscopic and microscopic examination-check.

The data of the helicopter Garmin 1000 MFD and PFD monitors, was sent and decoded at the main office of GARMIN (USA).

Engine Control Computer EECU Data was decoded in Safran Helicopter Engines special technical laboratory in France.

The main transmission assembly gearbox, with the LIVE mounts, and engine collective lever with Fly / Idle switch were sent to Bell in the USA for laboratory research to investigate the possibility of helicopter vibration and its source and expertise was performed from 24.02.2020 – 27.02.2020 at the Bell's Field Investigation Laboratory in Forth Worth Texas USA.

Fuel tests - Before supplying the fuel, testing for suitability of the fuel was performed on 17.05.2019 at the testing lab of "Petrocas Fuel Services Georgia" LLC (conclusion of the protocol № 66 – suitable for issuance, conclusion of the protocol №7 – suitable for issuance).

In place of the aviation accident the fuel removed from the tank of the helicopter, also from a tank on the petrol station of the base "Kazbegi Helicopters", also the samples of the oil from the helicopter engine, were sent to Levan Samkharauli National Forensics Bureau for examination and establishment of suitability and compatibility with the requirements. According to the expertise conclusion № 005403619 the samples of the fuel and the oil comply with the requirements, fuel was not contaminated.

1.17. Information about organizations and administrative activities

LLC "AVIATOR+" is registered since 06.04.2016. The air operator certificate № 086, date of issue 22.11.2018 and appendix №1, date of issue 22.11.2018, awarded by LEPL Civil Aviation Agency with the right to perform commercial exploitation, as it is determined in the attached exploitation specifications by observing those requirements determined by the fulfillment manual of the flight, Georgian Legislation and ICAO standards.

Operative and periodic technical service of the helicopter, operated by LLC "AVIATOR+" from 15.05.2018 till 31.08.2018 was performed by BELL helicopter Prague's aviation technical maintenance specialists, certificate № VAL-124 valid from 15.05.2018 till 15.05.2019, than by Aviation Technical Services company LLC “AIRPLANE TECHNIQUES”, certificate № GE.104, valid from 09.11.2018 till 09.11.2019, Allowed by The Civil Aviation Agency to provide aviation maintenance services in accordance with the Annex to this Certificate.

The crew place of residence was in the hotel in the town of “Stepantsminda”. The crew's work and rest time norms correspond to the existing standard requirements.

2. Analysis

Analysis of the accident was done with the details available on factual information and operational as well as mechanical factors. Several discussions were held among the members and experts, especially on the possibility of the human factors, violation of regulations, conditions of crash site, aerodynamics and other technical aspects. It is found that the PIC was properly certified and qualified under the GCAA regulations and training requirements. No evidence was found to indicate any pre-existing medical condition that might have adversely impaired the pilot's performance during the flight. The helicopter’s weight and balance were within the limits for the duration of flight. During the course of investigation the commission analyzed all the possible factors that may relate to the accident categorically. After the detail and extensive analysis the commission has reached to the conclusion accordingly and forwarded safety recommendations accordingly.

Site Examination

Ground scars revealed that the helicopter impacted upright in a relatively level attitude with slight right skid low in an approximate 275° heading. The left and right skid tube marks on the ground were orientated in the direction of forward helicopter travel (See Photo 1). A large gouge in the ground was consistent with hard fuselage belly contact during impact. Impact marks behind the main fuselage impact marks were consistent with tailboom vertical fin contact with the ground. The vertical fin strike marks are consistent with the tailboom being connected to the main fuselage at the time of impact. The helicopter broke through a wire fence just forward of the initial impact point. The helicopter was found approximately 30-40 meters from the initial impact point. Dead grass was observed forward and to the left of the main wreckage, consistent with fuel spray from the fuel cell which was breached on the left side and liberated from the fuselage structure during impact.

Airframe Fuselage Structure

No pre-impact anomalies were observed in the airframe fuselage structure and all observed fractures were consistent with overload forces.

The cockpit and cabin floor fractured at its attachment to the aft fuselage consistent with impact overload forces (See Photo 2). The top of the helicopter was intact. The aft fuselage steel truss was relatively intact but exhibited bent structure at both the aft crosstube mounting locations consistent with a hard ground impact (See Photo 3). The right bottom side of the steel truss was more damaged than the left bottom side. Bottom surfaces of the airframe exhibited impact damage. Additionally, the truss exhibited a fracture just aft of the aft crosstube. No evidence of wirestrike was observed on the helicopter.

Flight Controls & Hydraulics

The helicopter was equipped with single flight controls at the pilot right side position. No pre-impact anomalies were observed in the flight controls other than the Fly/Idle throttle switch found in the idle position. All fractures were consistent with overload fracture.

The Fly/Idle throttle switch on the collective head was found in the Idle position when observed (See Photo 4). The engine selector switch on the console was found in the Start/Run position (See Ph5).

All observed hydraulic lines were intact. A small amount of red colored fluid up to the refill line on the hydraulic reservoir sight glass, consistent with hydraulic fluid, was observed in the hydraulic reservoir and filter module. Leakage evidence of red colored fluid was observed near the hydraulic reservoir and filter when the helicopter was examined resting on its left side (See Photo 6).

The cyclic stick fractured consistent with overload forces at its base (See Photo 7). The cyclic hand grip also fractured consistent with overload forces away from the cyclic stick. A screwdriver was placed at the fractured base and was moved longitudinally and laterally by hand. Movement to the cyclic mixer was observed. Both cyclic vertical control tubes fractured near the bottom of the tubes consistent with overload forces from separation of the cabin floor from the aft fuselage (See Photo 8). The upper fractured vertical cyclic tubes moved freely by hand and the servo input valve levers moved freely with input from the vertical tubes. Both cyclic servo pistons were free to move with hand pressure. Additionally, the left cyclic vertical tube to the non-rotating inner ring swashplate was fractured consistent with overload forces.

The collective stick was fractured at its base consistent with overload forces at impact (See Photo 9). The collective head was also fractured consistent with overload forces at impact (See Photo 10). A screwdriver was inserted into the fractured base and up and down collective movement was applied. Control movement was observed to the overload fractured collective control tube at the cyclic mixer. The collective vertical control tube to the servos exhibited a fracture consistent with overload forces in the bottom rod end near the mixing unit where the cabin floor and aft fuselage separated during impact (See Photo 8—Blue Circle).

The upper fractured portion of the collective vertical control tube was moved by hand and free movement was observed at the collective servo valve input lever. The collective servo piston rod moved freely after the two steel tubes (1 horizontal and 1 vertical) to the collective lever were removed since the collective lever was locked because of impact damage.

The tail rotor pedals exhibited damage consistent with impact forces (See Photo 11). The pedals moved freely by hand and movement was observed in the steel tail rotor cables through the forward pulley at the rear of the cabin floor. Tail rotor cable fractures were observed in two areas of airframe fracturing. First, the aft fuselage between the mid upper and aft pulleys and just aft of the turnbuckles, the left and right pedal cables were observed fractured as result of separation of the main cabin floor from the aft fuselage (See Photo 12). Second, the aft tailboom area at the location of the tailboom fracture from main rotor strike, both left and right tail rotor pedal cables were fractured. Free pitch change was demonstrated from the slider tube through the tail rotor mast to the tail rotor hub.

Main Drive System

No pre-impact anomalies were observed in the main drive system and all observed fractures were consistent with overload forces.

The main drive system exhibited drive continuity when the main rotor blades were rotated by hand. Free rotation of the main transmission, main driveshaft and forward tail rotor drive system was observed. The transmission oil was clean and caramel in color. The upper and lower chip detectors were free of any chips.

For the left and right LIVE mounts, all four LIVE support arm connections to the airframe were connected to airframe truss structure (See Photo 13). Two studs on the inboard right LIVE transmission adaptor exhibited stripped threads and the nuts were missing (See Photo 14). The load direction to strip the threads of the studs is consistent with a heavy downward and lateral impact force. The fore/aft elastomeric restraints, located aft of the transmission mounts, were intact.

The freewheeling unit was checked for proper operation while installed. The main rotor was rotated by hand and the Np turbine wheel was prevented by hand from rotating when a hand was placed in the exhaust. However, the main driveshaft and tail rotor driveshafts connected to the freewheeling unit continued to rotate, demonstrating that the freewheeling unit operated properly.

Tail Rotor Drive and Tail Rotor System

No pre-impact anomalies were observed in the tail rotor drive system nor the tail rotor system and all observed fractures were consistent with overload forces.

Sections of the tail rotor driveshafts were found in two locations. The tail rotor driveshafts located in the aft fuselage remained in the aft fuselage area. The steel shaft under the engine and the oil cooler shaft were both intact and rotated freely when the main rotor was rotated by hand. Four of the tail rotor drive segments normally mounted to the tailboom, were reportedly found approximately 150 meters away from the wreckage (See Photo 15) and exhibited evidence of a main rotor blade strike as did the left side of the tailboom. Several of the hanger bearings were liberated from the tailboom, but all hanger bearings were accounted for and rotated freely by hand. The tail rotor gearbox rotated freely by hand with no unusual noises heard during rotation. The tail rotor gearbox chip detector was free of any debris or chips.

Both tail rotor blades exhibited damage consistent with ground impact forces (See Photo 16). One tail rotor blade fractured approximately 30 cm. from the tip. The other tail rotor blade was bent at an inboard location. The intact tail rotor hub exhibited intact connections at each tail rotor pitch change link and to the tail rotor blades through the intact pitch horns (See Photo 17). The tail hub and blades exhibited free pitch change when manipulated by hand.

Main Rotor Hub & Rotating Controls

No pre-impact anomalies were observed in the main rotor hub and rotating controls and all observed fractures were consistent with overload forces.

The main rotor hub remained attached to the mast splines (See Photo 18). Each main rotor blade grip remained installed on each yoke spindle, evidence that the Tension-Torsion (TT) straps between the grips (outboard) and the main rotor yoke (inboard) were intact. The grips for each blade exhibited over-rotation in the clockwise pitch direction consistent with impact forces. The Red blade pitch change link (PCL) fractured consistent with bending overload forces during impact near the center of the aluminum PCL (See Photo 19). The Red PCL top end remained connected to the Red grip and the bottom end to the red arm of the rotating outer ring swashplate. The White PCL was found wrapped around the mast but connections to the pitch horn on the top and to the swashplate on the bottom were intact (See Photo 20). The splined drive collar that provides drive from the mast to the swashplate through the drive link was disconnected from its mating mast splines consistent with impact forces (See Photo 21). Damage to the collar splines was consistent with impact forces. One of the bolts that holds the two collar halves together had partially stripped threads and the nut was missing consistent with impact forces (See Photo 22). The other collar nut and bolt were intact. The splined drive collar when connected to the mast provides rotational movement to the rotating outer ring swashplate. When the drive collar becomes disconnected from the mast while the mast continues to rotate (hence driving rotation of the swashplate will stop), PCLs may wrap around the mast and possibly fracture. The rotating outer ring swashplate rotated freely by hand.

Main Rotor Blades

No pre-impact anomalies were observed in the main rotor blades and all observed fractures were consistent with overload forces.

The White main rotor blade exhibited a fracture consistent with overload forces at impact (See Photo 23). The blade root end remained connected to the hub but the blade was fractured near the most outboard doubler near the root end of the blade (See Photo 24). The blade was bent back opposite the direction of rotation consistent with ground strike while rotating. The fractured blade section was reportedly found approximately 10 meters from the main wreckage. The trailing edge tip section was bent down and ground strike evidence was found near the final resting place of the blade section. Red paint transfer consistent with fuselage and tailboom paint was observed on the blade leading edge approximately 60cm. from the blade tip (See Photo 25).

The Red main rotor blade remained attached to the hub and the blade was not fractured, but relatively intact (See Photo 26). The outboard section of the blade exhibited several leading edge impact marks consistent with striking the tailboom during impact. Additionally, striated marks observed on the outboard leading edge were consistent with the pattern, size, and shape of the tail rotor control cable that transits the tailboom (See Photo 27). Mud evidence was observed on the bottom outboard surface of the blade and the blade was bent up at the tip, which was further evidence of a ground strike at impact.

Tailboom

No pre-impact anomalies were observed in the tailboom and all observed fractures were consistent with overload forces.

The doublers at the forward end of the tailboom that provide reinforced connection to the aft fuselage were fractured consistent with overload forces at impact (See Photo 28). The tailboom exhibited a front to rear oriented downward linear strike mark (See Photo 29) and fracture to the left side of the tailboom and left horizontal stabilizer consistent with a main rotor strike at impact. The aft portion of the fractured tailboom was reportedly found about 30-40 meters from the main wreckage. The vertical fin exhibited a fracture near the tail rotor gearbox with evidence of severing as result of a tail rotor control cable contact during impact (See Photo 30). The bottom of the vertical fin exhibited evidence of striking the ground with damage observed and dirt deposited on the vertical fin at a fracture location.

Landing Gear

No pre-impact anomalies were observed in the landing gear and all observed fractures were consistent with overload forces.

Both crosstubes were spread out (bent) from impact forces and the four clamps that secure the crosstubes to the fuselage were fractured (See Photo 31). The amount of spread (bending) between the left and right sides was relatively even with slightly more spread on the right side, indicating a relatively level impact that was slightly right skid low. The right skid fractured consistent with overload forces in the middle section of the skid at both forward and aft saddles that attach the skid to the forward and aft crosstubes.

Seats and Restraints

No pre-impact anomalies were observed in the seats or restraints and all observed fractures were consistent with overload forces at impact or the result of first responders cutting.

Both cockpit seats exhibited fractured back supports and were displaced to the right consistent with impact forces to the right (See Photo 32). The seat metal supports on both sides of each cockpit seat exhibited stroking on the right side supports of both the pilot (right front—See Photo 33) and copilot seats. The left seat metal supports on the pilot and copilot seats (left front—See Photo 34) were also stroked. Fractures were observed on all cockpit seat stroking supports consistent with impact forces. Both the pilot and left front seat passengers were reportedly still sitting in their seats when first responders arrived but were fatally injured. The left aft passenger seat containing the third occupant exhibited a seat that had fractured down consistent with impact forces (See Photo 35).

The pilot reportedly was only restrained by the lap belt. It was reported that the pilot did not wear the shoulder harness during flight. The pilot side rotary latch mechanism and attached seat belt were not available for examination and had reportedly been kept by the Georgian Police because of blood contamination. The copilot (left front seat) seat belts, rotary latch, and shoulder harnesses were intact. The left rear passenger's seat belt was cut by a first responder from the Rooms Hotel (driver of the General Manager) (See Photo 36). The left rear shoulder harness was intact.

Fuel System

No pre-impact anomalies were observed in the fuel system and all observed damage was the result of impact damage.

The emergency fuel shutoff valve handle in the cockpit and the lever at the emergency shutoff valve located in the aft fuselage were both in the Off position. No mechanical damage was observed on the cockpit handle to indicate that a hard contact from an object at impact caused the handle to move from On to Off (See Photo 37).

The shutoff cable was continuous from the handle to the shutoff valve. The fuel line to the emergency shutoff valve was removed and it was confirmed that the valve was in the closed position (See Photo 38). The cockpit handle was then moved between On and Off and the emergency shutoff valve positions corresponded or agreed with the lever positions. Data from the FADEC and Garmin show that the engine continued operating at Idle through impact and therefore it is believed that the emergency fuel shutoff was activated to Off during the impact sequence.

Fuel lines were borescoped from the HMU at the engine to the fuel tank to look for obstructions. The only obstruction observed was in the fuel line from the top of the fuel cell to the start pump which is located on an aft fuselage deck under the transmission. An obstruction was found in the fuel line at the point where the fuel line makes a 90 degree turn at the aft fuselage deck (See Photo 39). The aft fuselage deck structure was bent downwards consistent with impact forces. The start pump, which is attached to the aft fuselage deck, also was displaced downward which pulled on and stretched the fuel line at the 90 degree turn area. After this fuel line was straightened, the borescope was still prevented from transiting through the fuel line at the obstruction which appeared to be crimped (See Photo 40). The fuel line was cut during the exam at this location and the interior plastic liner was indeed crimped consistent with impact forces when the fuel line stretched as the aft fuselage deck bent downward (See Photo 41). An uncrimped section of the fuel line was crushed by hand and it resulted in similar crimping. The crimped line is consistent with occurring during the impact as the airframe deformed. The fuel line from the crimped location to the fuel cell was free from obstruction.

The fuel cell top plate was removed to examine the fuel cell. The fuel cell ruptured near the fuel filler area on the left side of the helicopter but was otherwise intact in other locations. The fuel quantity probe was bent from impact. The fuel strainer in the fuel cell was free from any debris. The fractured fuel line near the fuel filler area was borescoped from both directions and no obstructions were observed although a little sand was observed.

Laboratory Exam of Airframe Parts

In March 2019, with 120 hours total time on the helicopter, Bell Prague sent technical support to help with a reported vibration issue, track and balance the main rotor, service the LIVE mounts with additional nitrogen pressure, and replace a tail rotor gearbox assembly which exhibited an oil leak through the seal with the tail rotor output shaft (i.e. tail rotor mast) because of a scratch on the output shaft/mast. The helicopter total time at the time of the accident was 162 hours, so the helicopter flew for 42 hours from the time of the maintenance action to the accident. No vibration issues were officially reported in to have occurred during those 42 hours of flight activity. The following parts were sent to Bell for laboratory examination to explore whether evidence of malfunction was present during the accident:

- Main transmission
- Left and Right LIVE mounts assemblies (LIVE—Liquid Inertia Vibration Elimination)/support arm assemblies
- Fractured collective stick with damaged collective head that houses the Fly/Idle switch

The LIVE support arms exhibited fracturing, and the transmission adaptor exhibited stripped stud threads on a single side, consistent with impact forces. It was found that none of the parts exhibited pre-impact anomalies. Additionally, there is no history of fatigue cracks or fractures on any 505 pylon LIVE mount supports.

History and Related Data:

1) Part Name: See Table 1

2) Part Number: See Table 1

3) Part Serial Number: See Table 1

4) Total Part Time: 162 Hours

5) Helicopter Model: 505

6) Serial Number: 65074

7) Helicopter Time: 162 Hours

8) RMA Number: See Table 1

Lab examination started on February 24, 2020 and was attended by the Georgian Investigator-In-Charge, a representative from Transportation Safety Board of Canada,

Transport Canada Safety Inspector, ITT Enidine (USA vendor of the LIVE units), and the aircraft operator.

The shipping container was received in the Field Investigations Laboratory as shown in Figure 1.

The main transmission assembly and the LIVE mount assemblies were received as well as the pilot's collective stick assembly. The examination objective of the LIVE mounts was to determine if there was any evidence or source of vibrations that would render the pilot frombeing unable to perform an auto-rotation.

The examination objective for the collective stick was to determine if the fly/idle switch for the engine throttle control operated as required.

LIVE Mount Assemblies

Figures 2 and 3 show illustrations of the pylon support and main transmission. In flight, with the LIVE mount attached to the adapter bridge, which is attached to the transmission, the weight of the aircraft rests on the top surface of the adapter bridges while the load path of the main rotor mast is upward. Vertical main rotor vibrations are dampened via the pressurized LIVE units. A fluid inside the upper and lower reservoirs, which are connected with a tube, dampen the vibration along with the elastomeric bearings.

The right-hand LIVE mount adapter bridge had two studs with sheared threads (see Figure 4).
The nuts and safety wire connecting the two studs were missing. The studs and nuts hold the LIVE units in place while the aircraft is on the ground. The stud threads sheared as a result of the ground impact, where the spherical bearing flange stayed in place while the downward momentum of the transmission stripped the stud threads. Remnants of the safety wire were found inside the stud pin holes (see Figure 5), which were sheared by the nuts upon impact.

Figure 6 shows the right-hand spherical bearing housing flange at the sheared stud thread locations. There was no wear on the spot faces, which indicated that the nuts were not loose.

The left-hand LIVE mount adapter bridge studs and hardware were still intact as shown in Figure 7. Safety wire was still in place.

After disassembly, the adapter bridge stud nut was placed back in its original position on the stud to show that the thickness of the stack-up (flange thickness, shim thickness, and the nut's chamfer and first incomplete thread) suggests the nut on the sheared studs were properly located and were not loose.

Energy dispersive x-ray spectroscopy (EDS) determined the alloy composition of the sheared MS51989-105D10 studs were similar to 41xx series steel. The required alloy was 4130 steel.

The hardness of the studs measured 28 HRC, which conformed to the specification's requirement of 26 HRC minimum.

The lower half of the LIVE mount center support cracked at the aft-inboard attachment location along with the aft mounting arm for both the left and right-hand sides (see Figures 8 and 9).

The mounting arm flanges fractured. The cracks and fractures were a result of overload from the ground impact. The cracked LIVE mount center support allowed the dampening fluid to leak out of the lower cavity. This damage did not allow testing of the LIVE units to be performed by the vendor. During disassembly of the LIVE units, the LIVE units were found to no longer be pressurized and the diaphragm in the accumulator on both sides were torn.
Evidence of the dampening fluid was found.

EDS analysis determined the alloy composition of the LIVE unit center supports (SLS-010-211) was similar to the required aluminum alloy. The hardness of the center supports conformed to the drawing's required minimum. The conductivity met the required minimum for the specified alloy and heat treatment.

Main Transmission

The main transmission rotated freely by hand by rotating the main rotor input, which moved the planet pinion carrier assembly. Figure 10 shows the main transmission partially disassembled for inspection. The sun gear revealed an axial feature, which reflected more light than its surrounding area of machining grooves. The feature consisted of axial lines and polishing of the machining grooves that stretched nearly the entire length of the sun gear. A scratch was observed in the polished area. The scratch was eddy current inspected and the entire sun gear was fluorescent magnetic particle inspected. No crack indications were found. The manufacturing planning notes for the sun gear did not reveal any repairs or discrepancies.

Pilot's Collective Stick

It was reported that the aircraft had a collective stick for the pilot and not for the co-pilot seat.

The collective stick was received with the head fractured off as shown in Figure 11. It was reported that the fly/idle switch was found in the “idle” position at the accident scene. Electrical continuity of the switch was evaluated in the lab by using a multimeter at connector pins 1 and 2. As required, when switched from “Fly” mode (open circuit) to “Idle” the circuit became closed. While simulating vibration by hitting the switch against the table in the lab, a wire fractured from a terminal and the switch still operated as required (redundant circuit). There were no exposed wires at the grommet location at the base of the collective stick. The base of the collective stick fractured via overload. The head of the collective stick fractured via overload with the head fracturing in the upward direction relative to the collective stick.

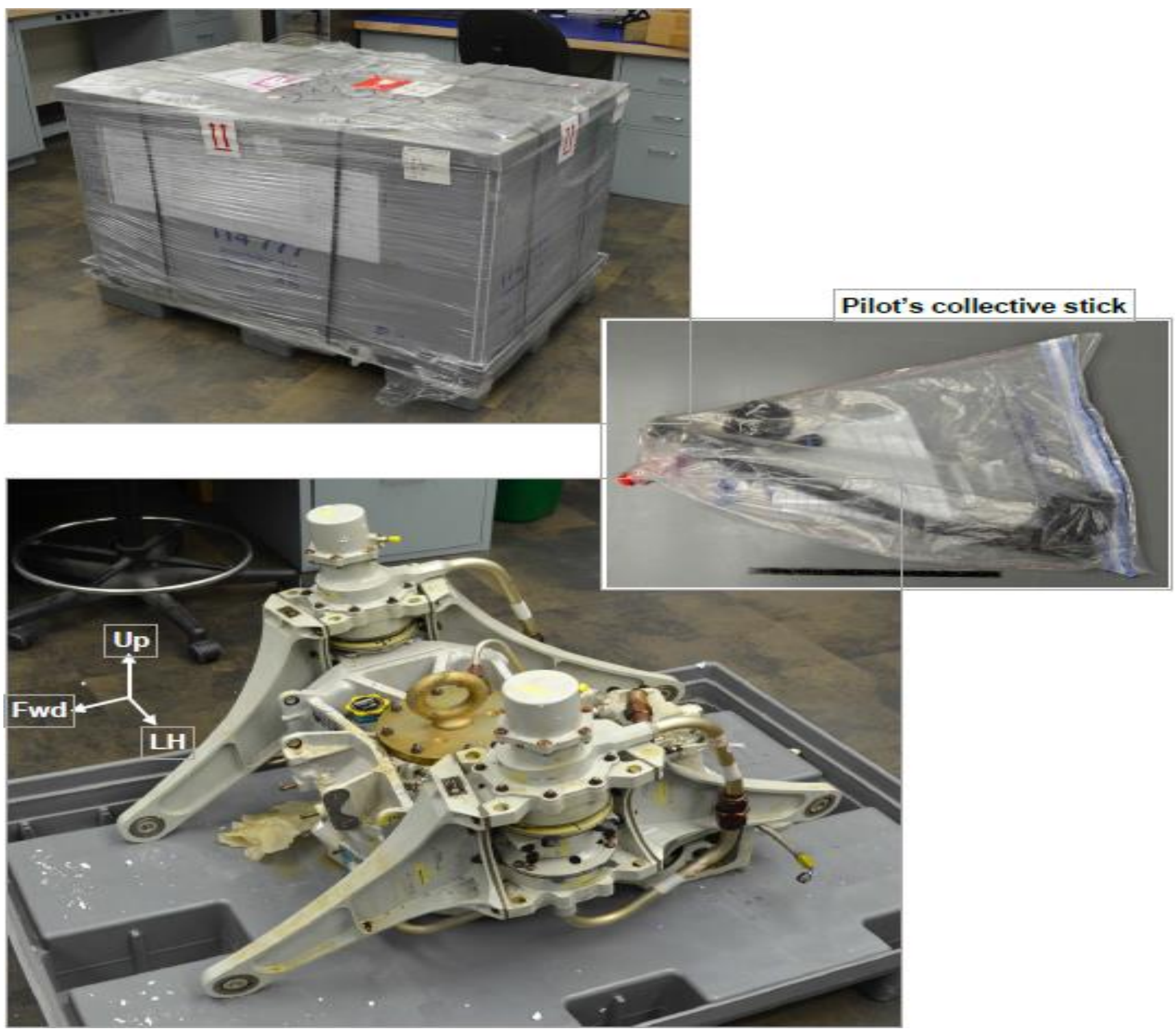


Table 1: Parts received for examination.

Part Name	Part Number	Serial Number	RMA
Live Mount Assy.	SLS-010-200-202	IE000016	65083542
Live Mount Assy.	SLS-010-200-201	IE000018	65083543
Main Transmission	206-040-004-117	BH571947	65083544
Collective Stick	M207-20M478-041	820022405	65083714

Figure 1: Overall view of the shipping container and of the main transmission assembly with the LIVE mounts attached as-received. The pilot's collective stick was also received.

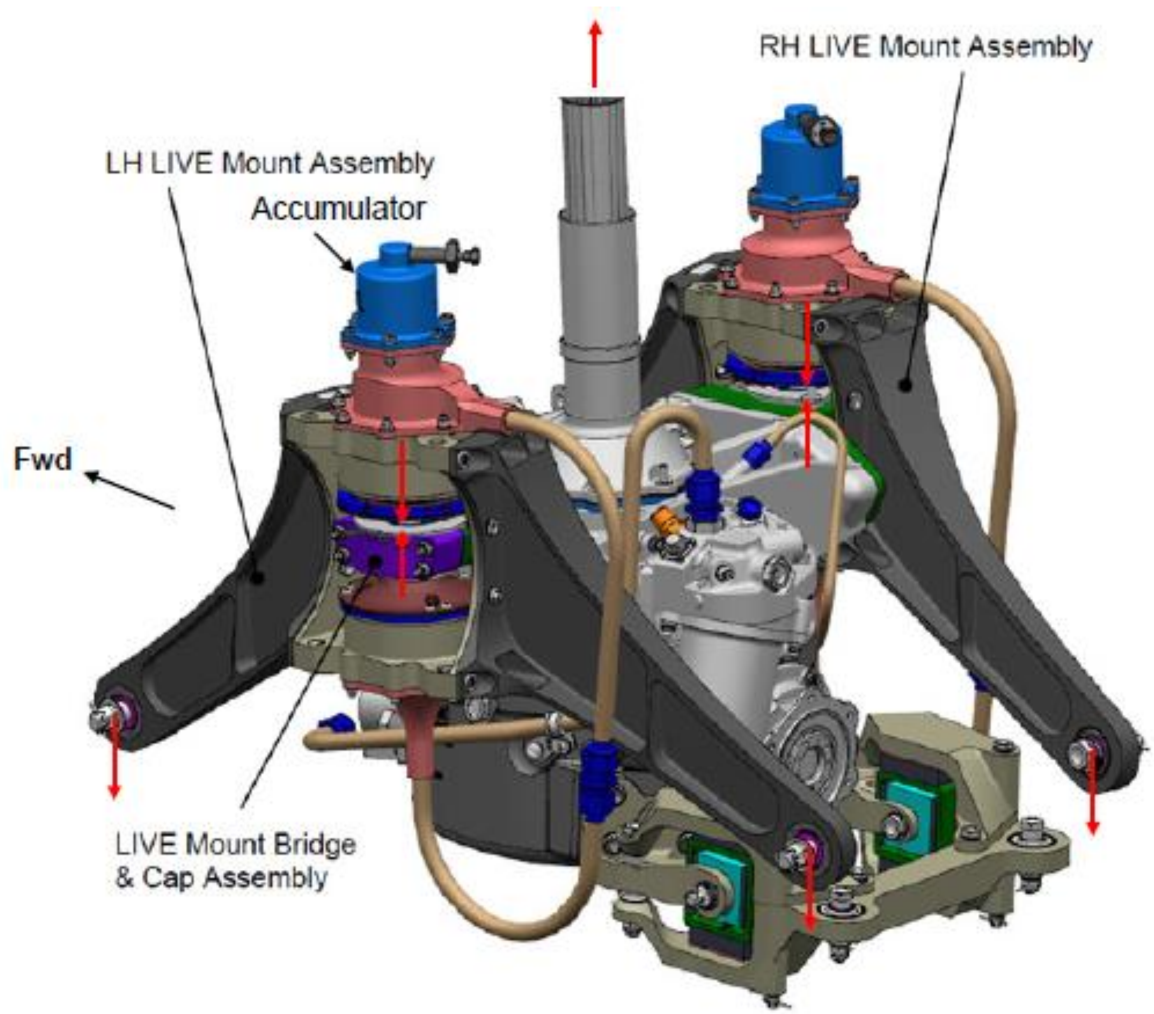


Figure 2: Illustration of the pylon support structure and main transmission. The fuselage attaches to the four corners of the LIVE mount arms (red arrows indicate load path – also see

Figure 3).

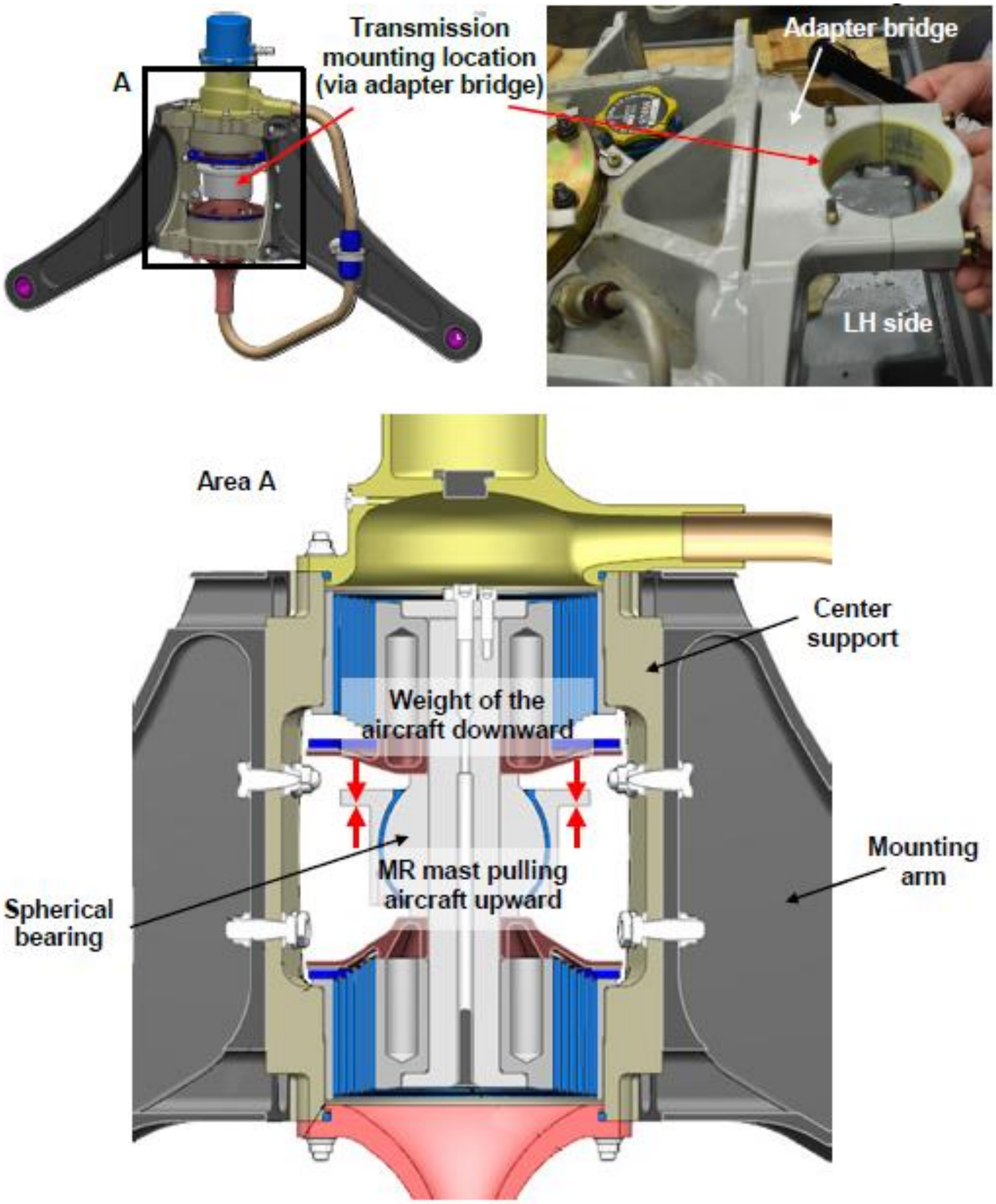


Figure 3: While in fight, the weight of the fuselage rests on the LIVE mount adapter bridge, putting the interface between the spherical bearing flange and adapter bridge in compression. Vertical vibrations are dampened with the load path going through the LIVE units.

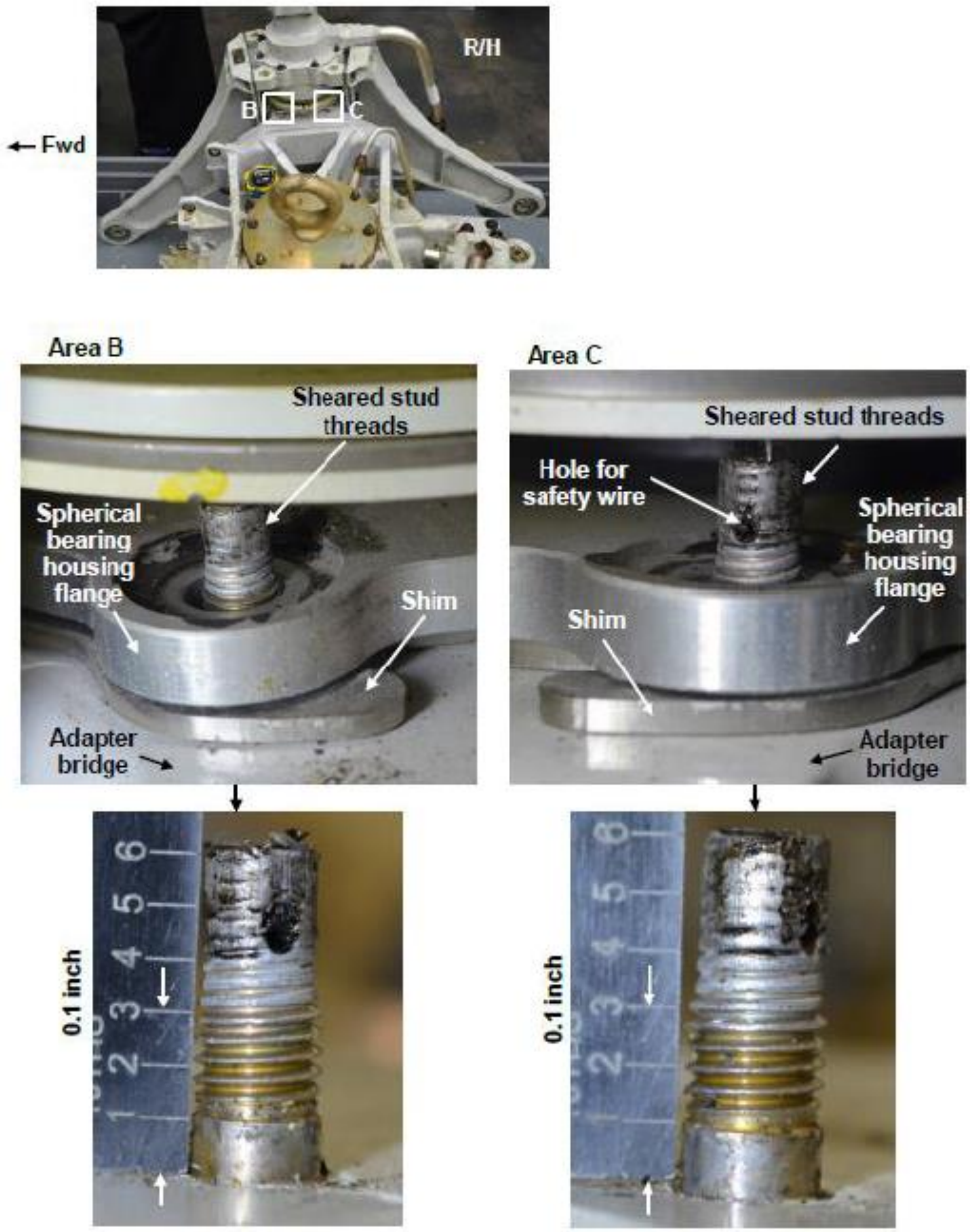


Figure 4: Top photo shows the right-hand LIVE mount assembly. Both adapter bridge studs had sheared threads as a result of the downward force from the impact.

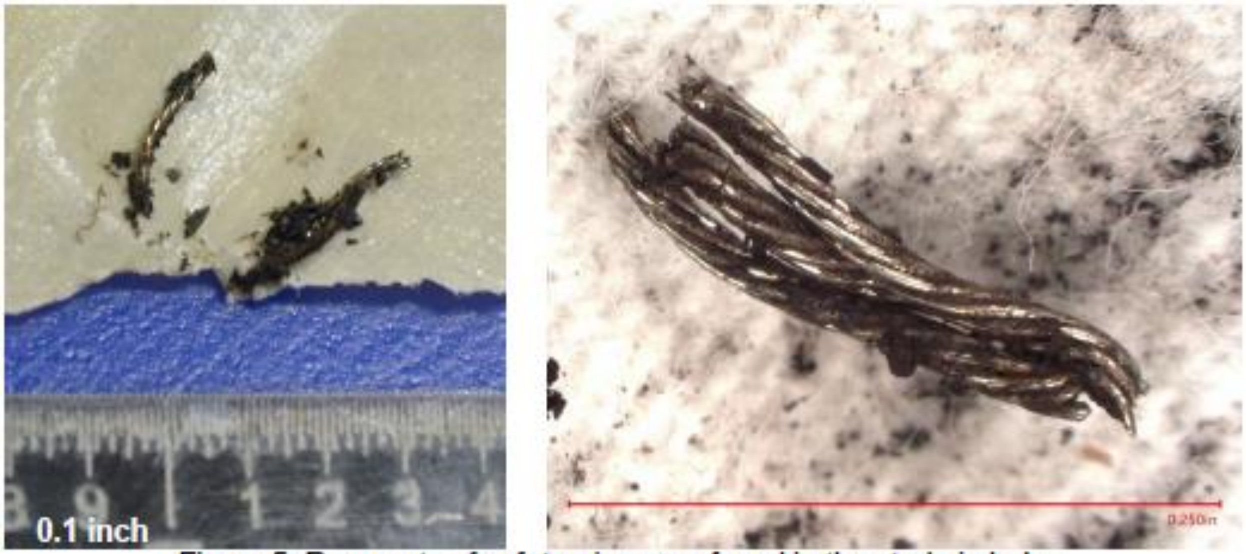


Figure 5: Remnants of safety wire were found in the stud pin holes.

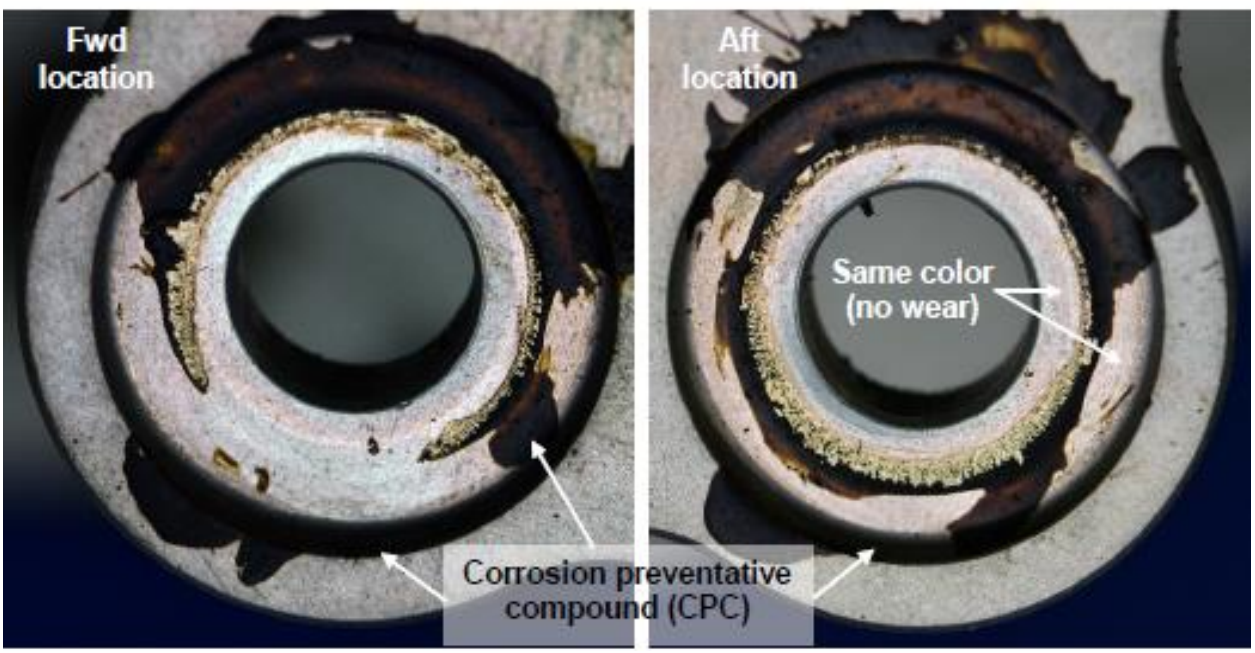


Figure 6: Views of the right-hand spherical bearing housing flange at the sheared stud thread locations. There was no wear on the spot faces, which indicated that the nuts were not loose.

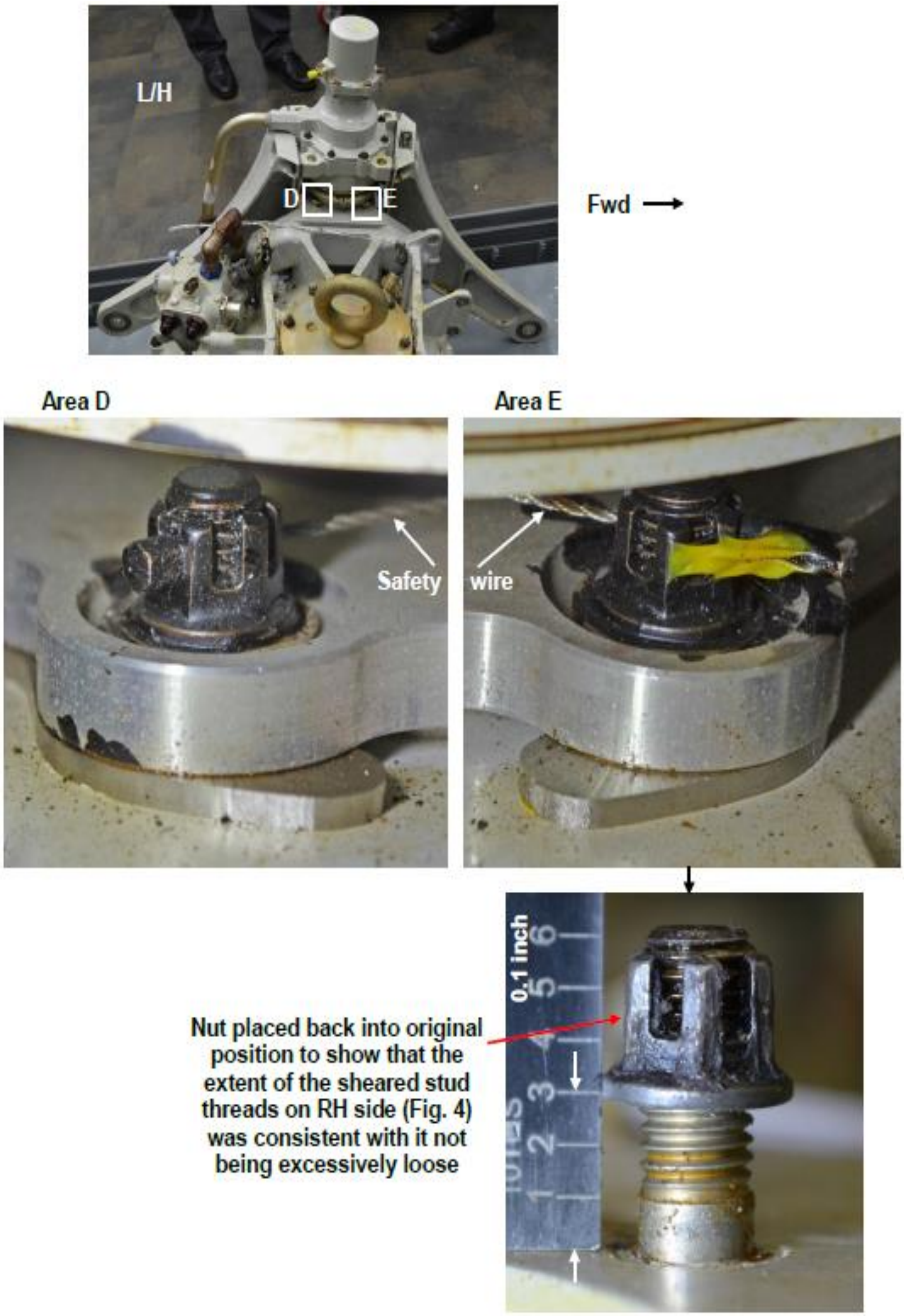


Figure 7: Top photo shows the left-hand LIVE mount assembly. The adapter bridge studs, nuts and safety wire were all intact.

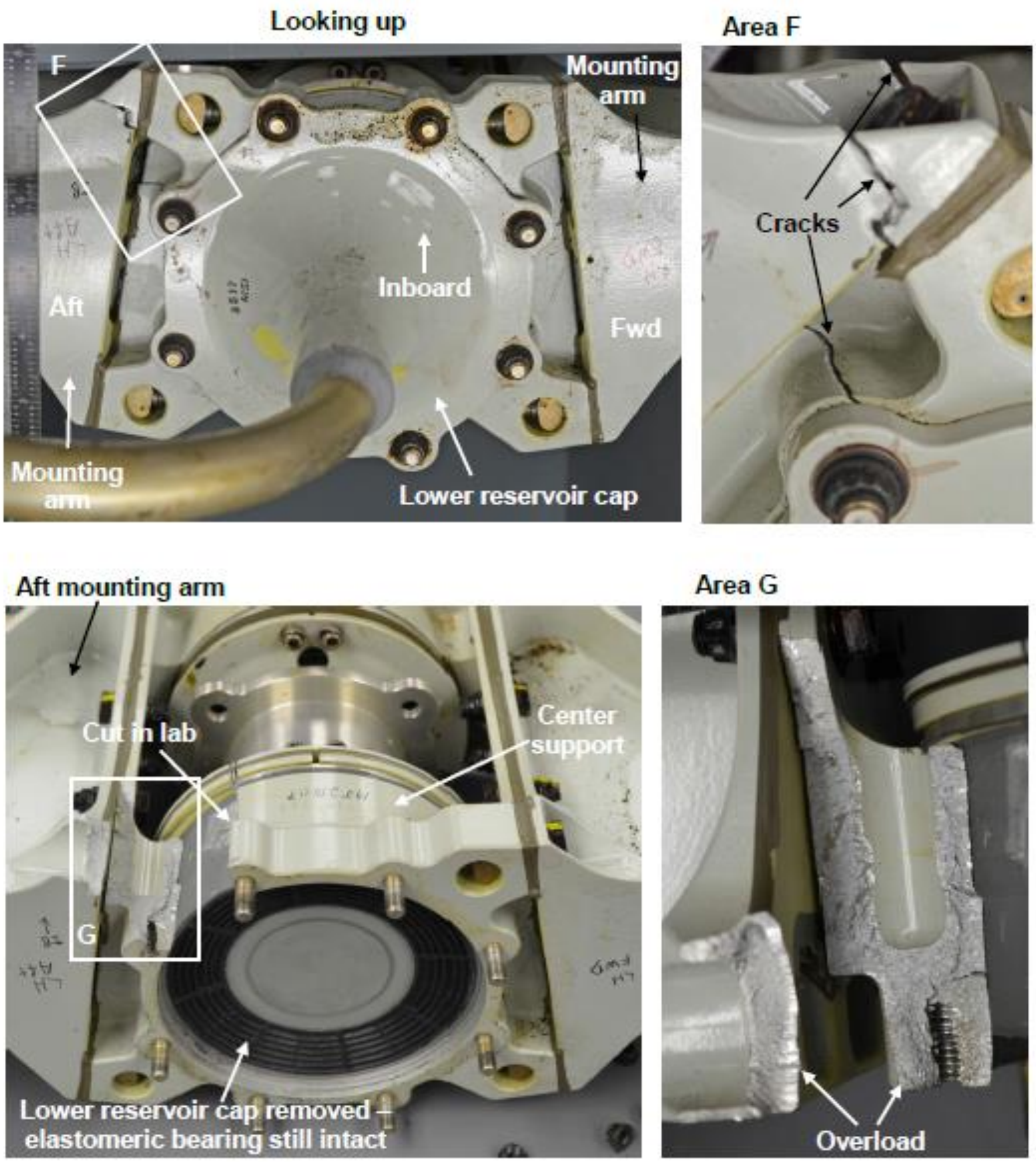
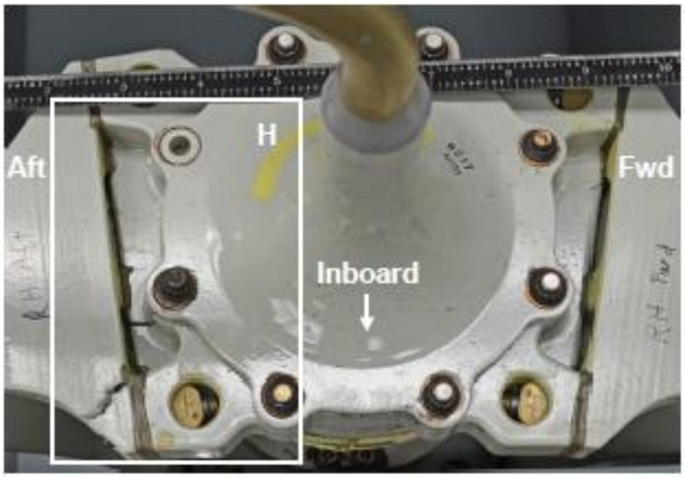
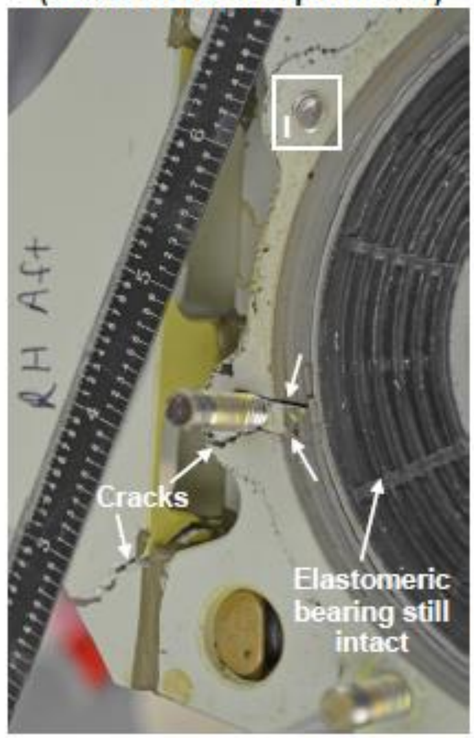
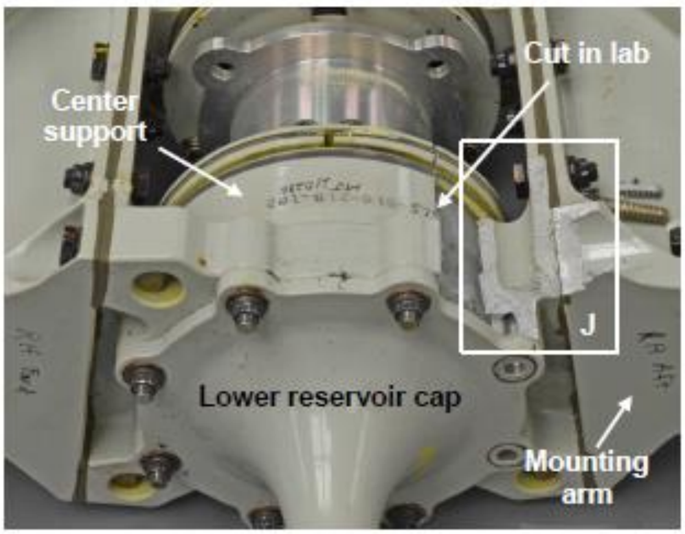


Figure 8: Views of the left-hand LIVE mount with overload cracks in the center support and the aft-inboard mounting arm attachment flange. The overload cracks were the result of the ground impact. The LIVE mount fluid had leaked out at the cracked center support.



Area H
(Lower reservoir cap removed)

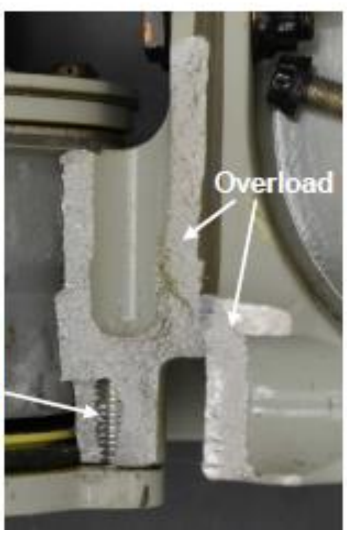




Area I

Shear overloaded stud

Area J



Stud fell out during sectioning

Figure 9: Views of the right-hand LIVE mount with overload cracks in the center support and the aft-inboard mounting arm attachment flange. The overload cracks were the result of the ground impact. The LIVE mount fluid had leaked out at the cracked center support.

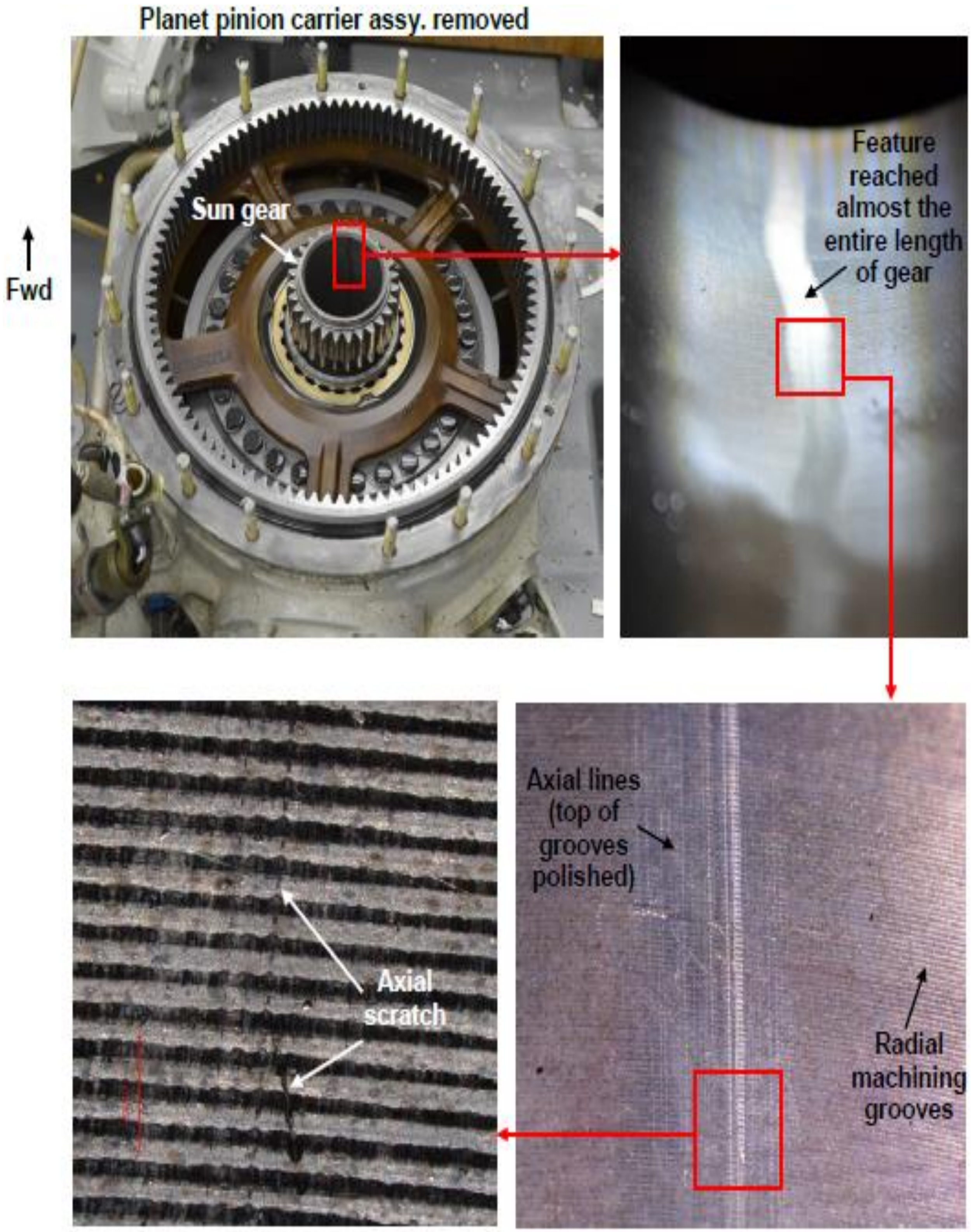


Figure 10: View of the main transmission with the planet carrier assembly removed.
The sun gear had a region of axial polishing. A scratch was found. Magnetic particle and eddy current inspections did not reveal any crack indications. No other anomalies were observed in the transmission.

Figure 11: Views of the pilot's collective stick. The fly/idle switch operated as required, even after a wire fractured from a terminal after hitting the switch against the table in the lab.

Bell Engineering Analysis

A Bell dynamics engineer who was part of the 505 certification efforts and incorporation of the LIVE system into the airframe took part in the lab examination. He provided context on the LIVE installation and Bell engineering efforts to reduce cabin vibrations of the 505 during certification. The Bell 505 was flown safely for 533 hours by Bell test pilots before the LIVE mounts were incorporated in the design to reduce main rotor vertical vibrations to acceptable levels. The dynamics engineer wrote a summary report of the dynamics work done during flight test and discussion about the accident flight. Both the engineering summary report and an American Helicopter Society (AHS) paper from 2017 about the LIVE mount incorporation into the 505 was provided.

It was determined that suspected malfunctioning LIVE units on the accident helicopter (of which no evidence was found) would not have been debilitating to the pilot's control of the helicopter. He also found that according to engineering analysis, the fractures found in the LIVE mount supports and the transmission adaptor stripped stud threads are consistent with occurring from impact loads.

The right-hand LIVE mount adapter bridge had two attachment studs for the LIVE unit with sheared threads as a result of the ground impact. Remnants of safety wire were found inside the stud holes. This proves that the wire was cut by a strong single blow as a result of overload. The corresponding spot face on the LIVE unit attachment flange at the sheared studs did not reveal any evidence of wear from looseness. The lower half of the LIVE mount center support cracked at the aft-inboard attachment location along with the aft mounting arm for both left and right-hand sides as a result of overload from the ground impact.

There was no evidence to indicate there were any issues with the LIVE mount assemblies or any maintenance discrepancies that would have limited its functionality.

The fly/idle switch on the collective stick that controls the engine throttle was tested for electrical continuity. The switch operated as required

Maintenance Trainer and Flight Training Device Simulations at Bell Training Academy

The Bell Maintenance Trainer and FTD (Flight Training Device) at the Bell Training Academy (BTA) were used to simulate the accident flight . The accident profile was flown in the FTD by a Bell instructor pilot to simulate the accident flight at a high altitude location. The Bell pilot flew the FTD at the high altitude Aspen, Colorado airport (airport identifier KASE) for the simulation.

It was found that the main rotor rpm would decay significantly once the throttle switch is placed from fly to Idle and the collective is not reduced completely.

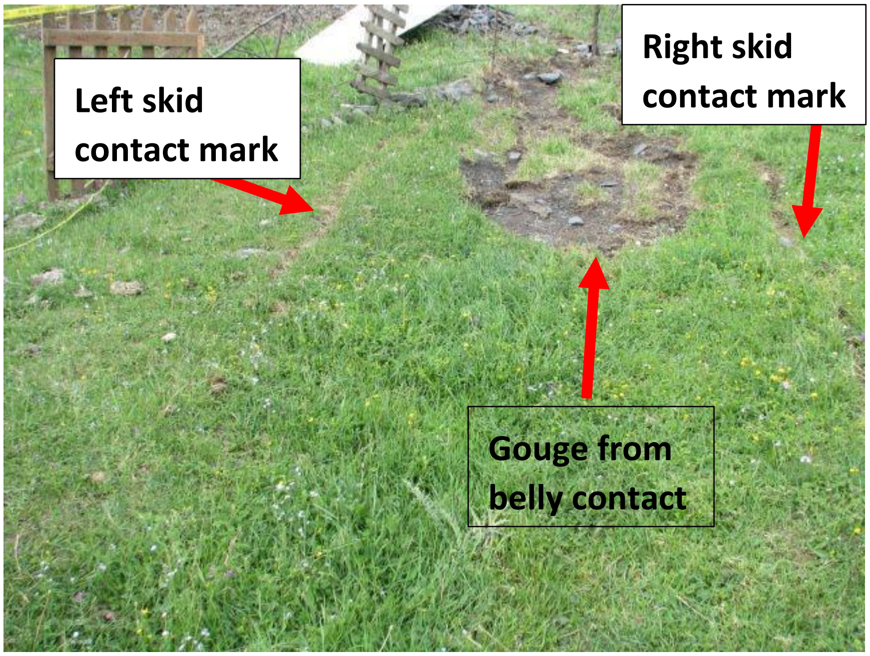


Photo 1
Forward fuselage and skid gear impact marks

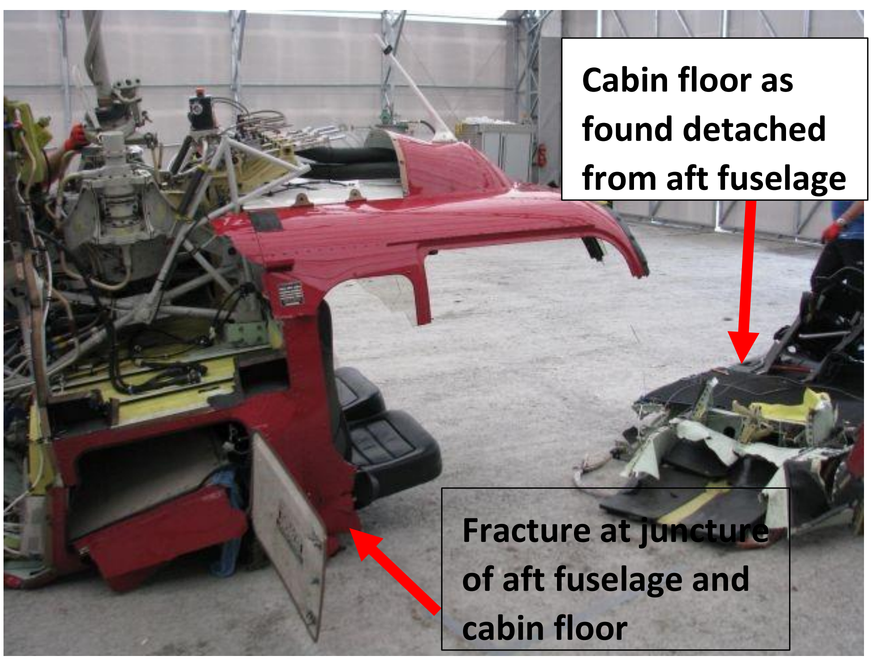


Photo 2
Right side of fuselage with fracture of cabin floor from aft fuselage

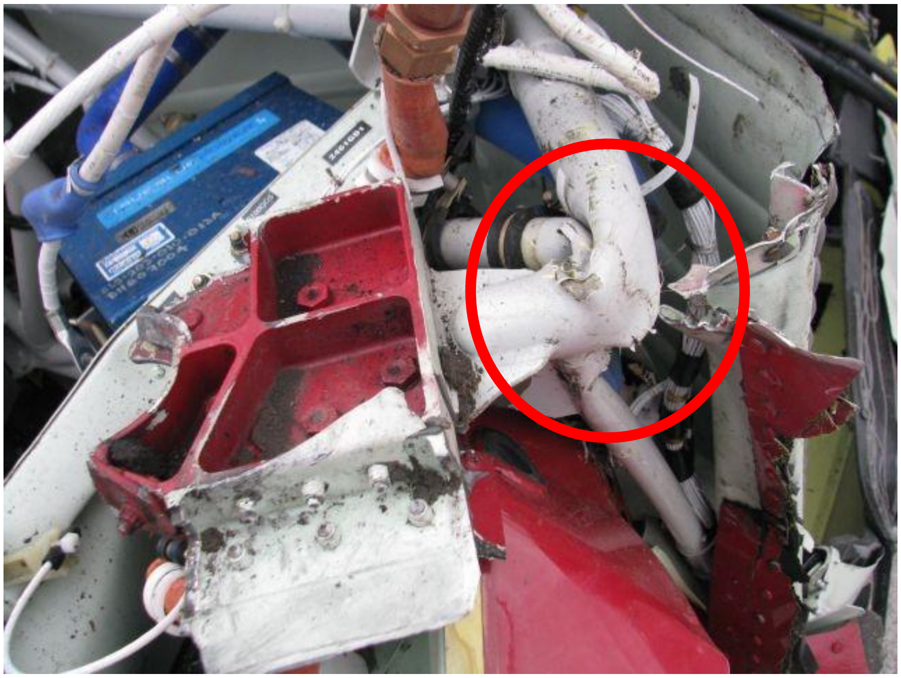


Photo 3
Right bottom side of truss was bent from hard impact of rear crosstube

Photo 4 Fly/Idle throttle switch (fractured, but held in place) on collective head was found in Idle position

Photo 5

Engine selector switch found in Start/Run position

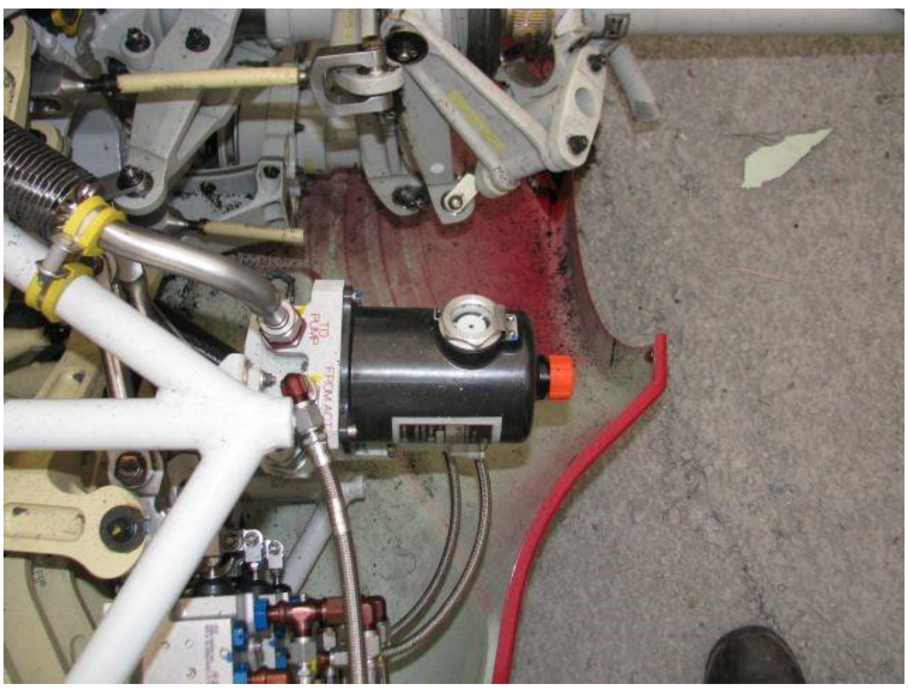


Photo 6

Red colored fluid found on cowling was consistent with hydraulic fluid. Hydraulic reservoir intact.

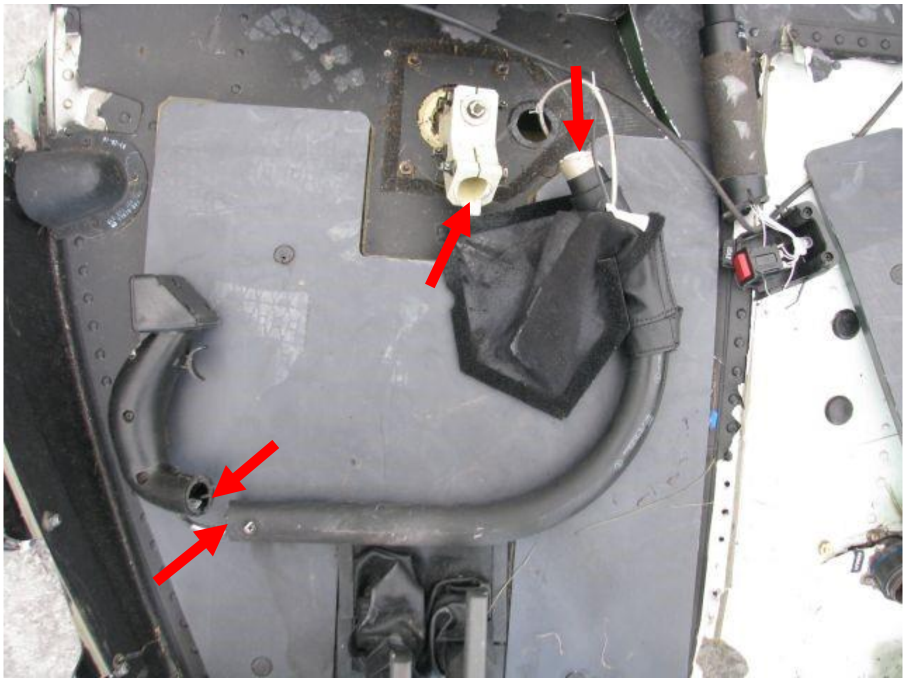


Photo 7

Cyclic stick fractured due to overload at its base and at the hand grip (red arrows)

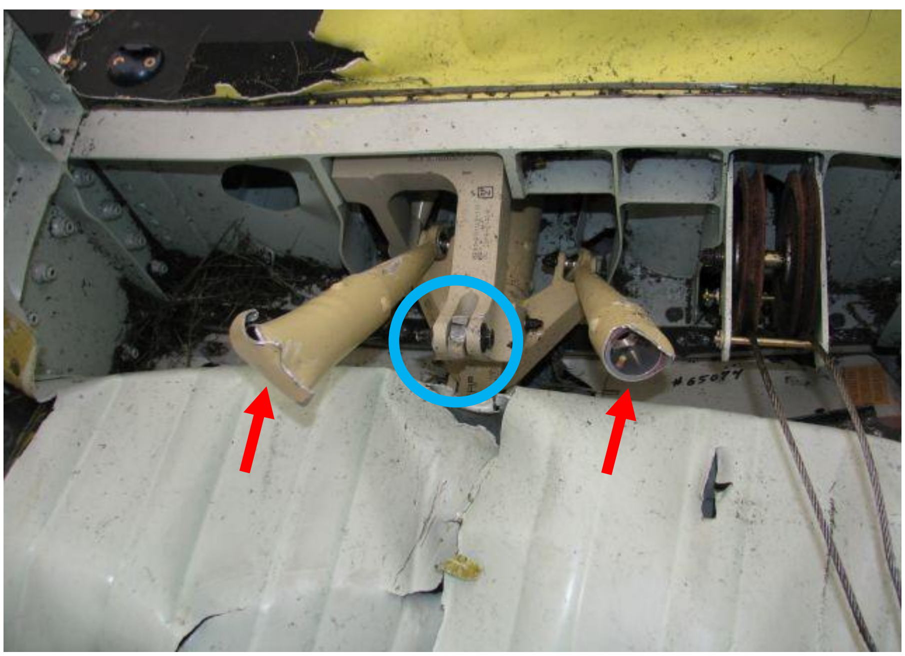


Photo 8. Vertical cyclic (Red) and collective (blue) control tubes fractured consistent with overload forces

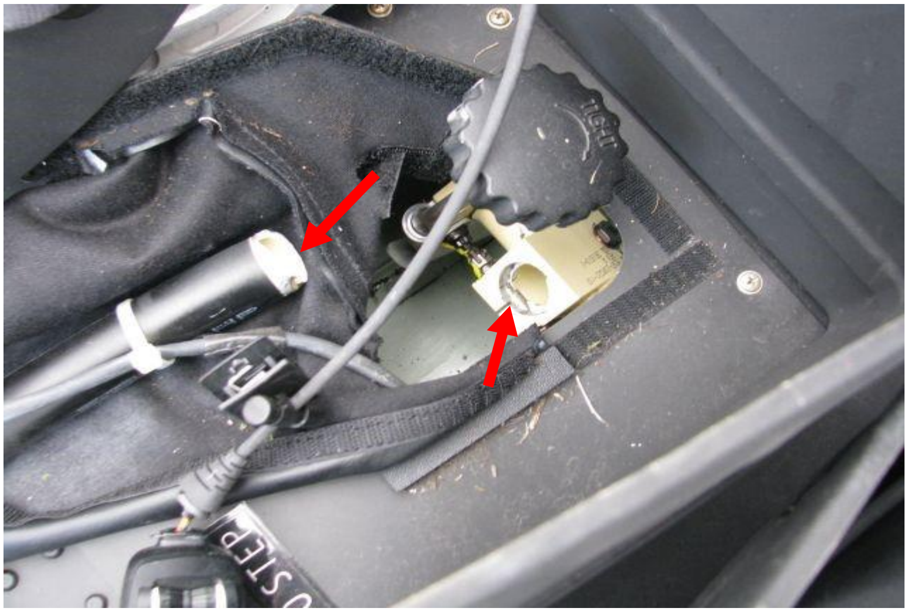


Photo 9

Collective stick fractured at its base consistent with overload forces

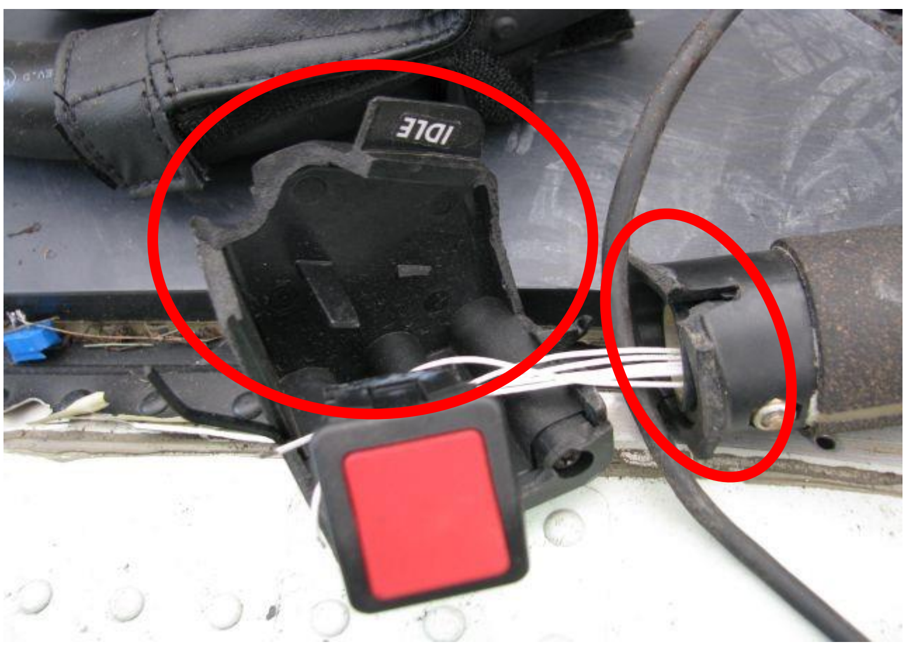


Photo 10

Collective head fractured consistent with overload forces at impact

Photo 11
Damage to pilot tail rotor pedals consistent with impact forces

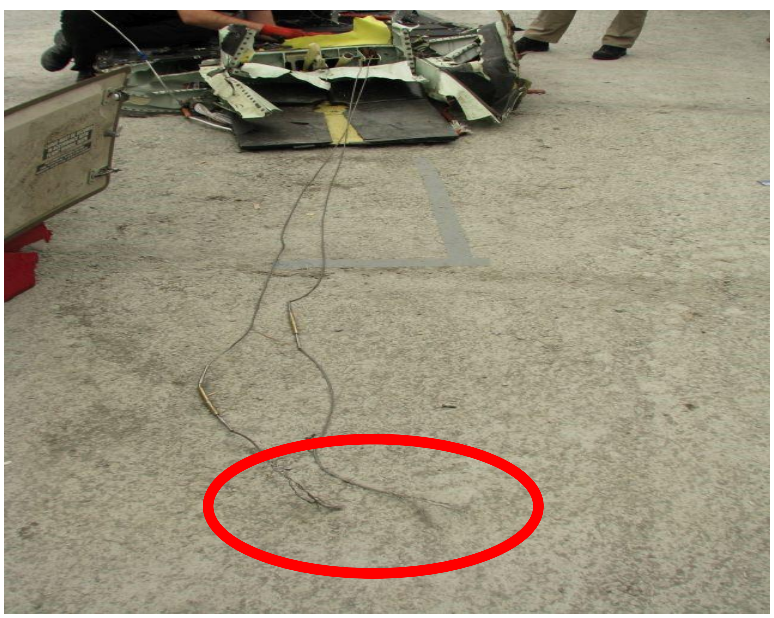


Photo 12 Fractured tail rotor control cables as result of separation of cabin and aft fuselage at impact

Photo 13

Intact nut and bolt securing RH forward LIVE support arm to truss. Other three corners also intact.

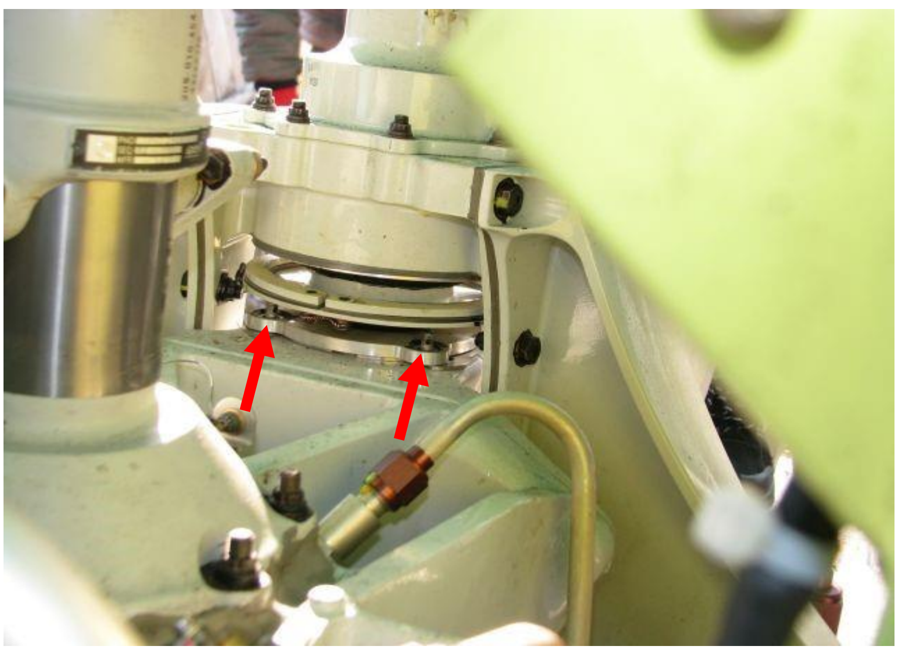


Photo 14

Two studs on RH inboard LIVE transmission adaptor were found stripped and nuts missing

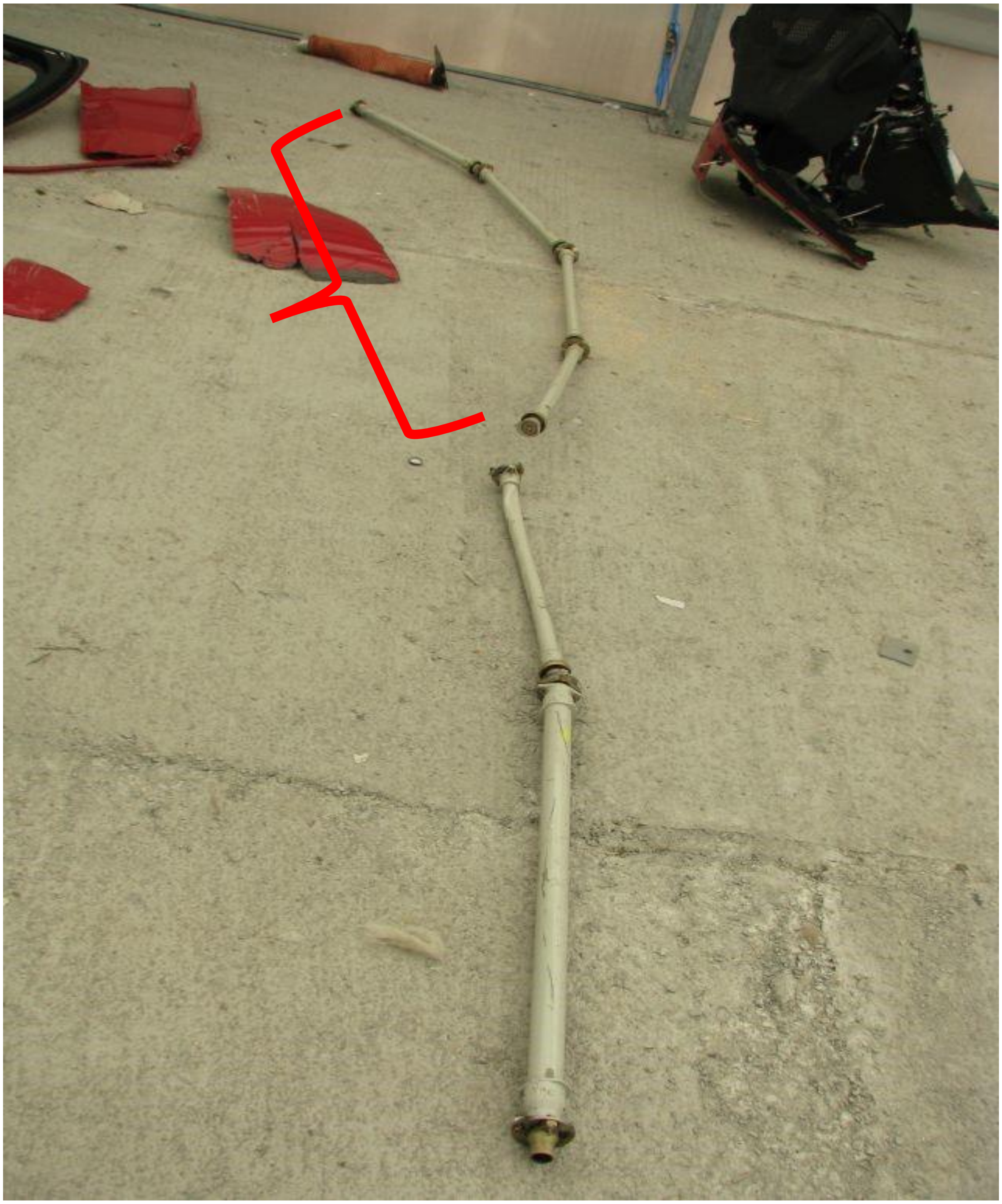


Photo 15

Four aft tail rotor driveshafts found approx. 150 m from wreckage consistent with main rotor strike

Photo 16
Both blades exhibited impact damage

Photo 17
Tail rotor hub with intact connections at pitch change links and blades

Photo 18
Intact main rotor hub remained connected to mast assembly

Photo 19

Red pitch change link (PCL) fractured near center of tube consistent with impact overload force

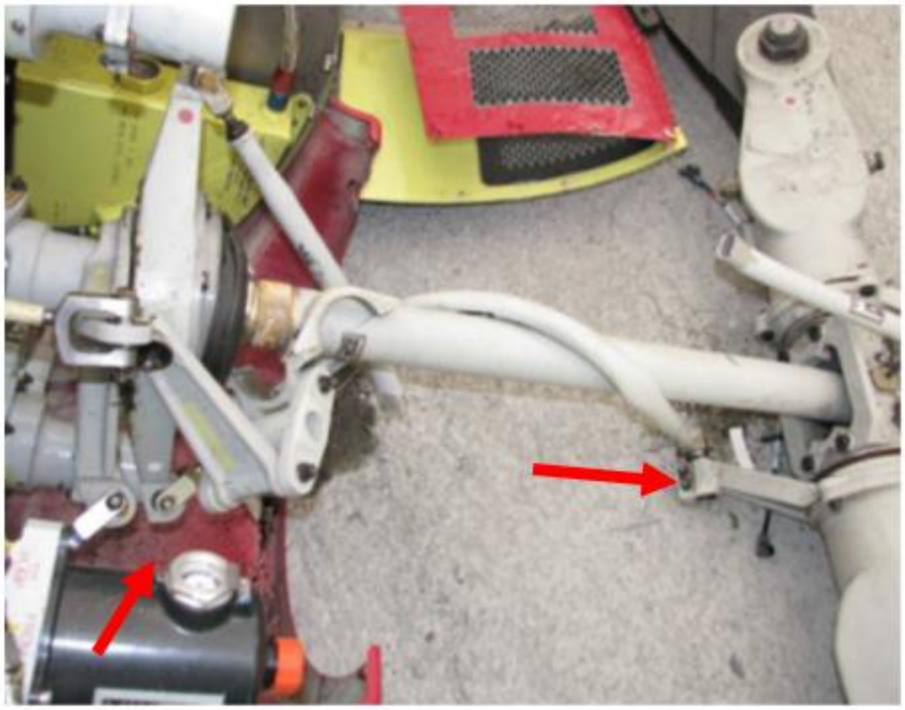


photo 20
White PCL wrapped around mast with intact connections to swashplate (bottom) and pitch horn (top)

Photo 21
Splined drive collar found disconnected from mast splines consistent with impact forces

Photo 22
Missing nut and partially stripped bolt threads on retention bolt for splined drive collar

Photo 23
White main rotor blade fractured at inboard location consistent with overload

Photo 24
White blade fracture through spar and afterbody near most outboard doubler

Photo 25
Red paint transfer on White blade leading edge consistent with tailboom color

Photo 26

Red main rotor blade was relatively intact with no spar fractures

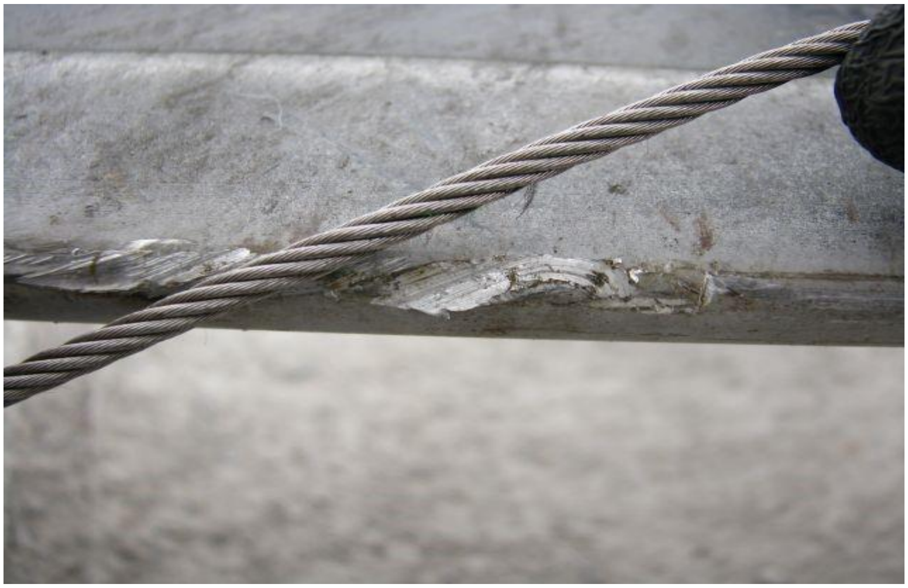


Photo 27

Markings on the Red main rotor blade leading edge were consistent with contact with tail rotor cable

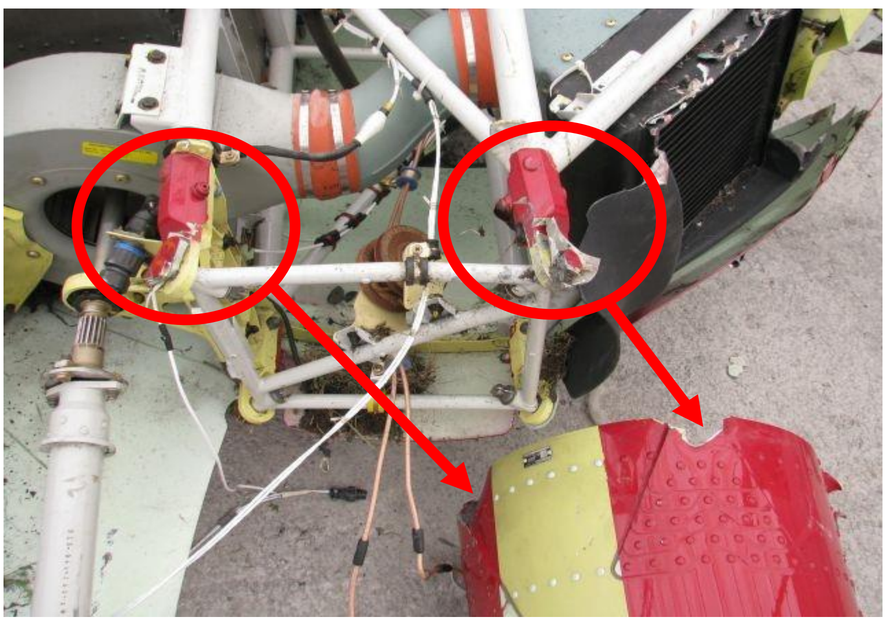


Photo 28

Tailboom separation from aft fuselage with doubler fractures consistent with impact overload forces

Photo 29
Linear strike mark on left side of tailboom consistent with main rotor strike

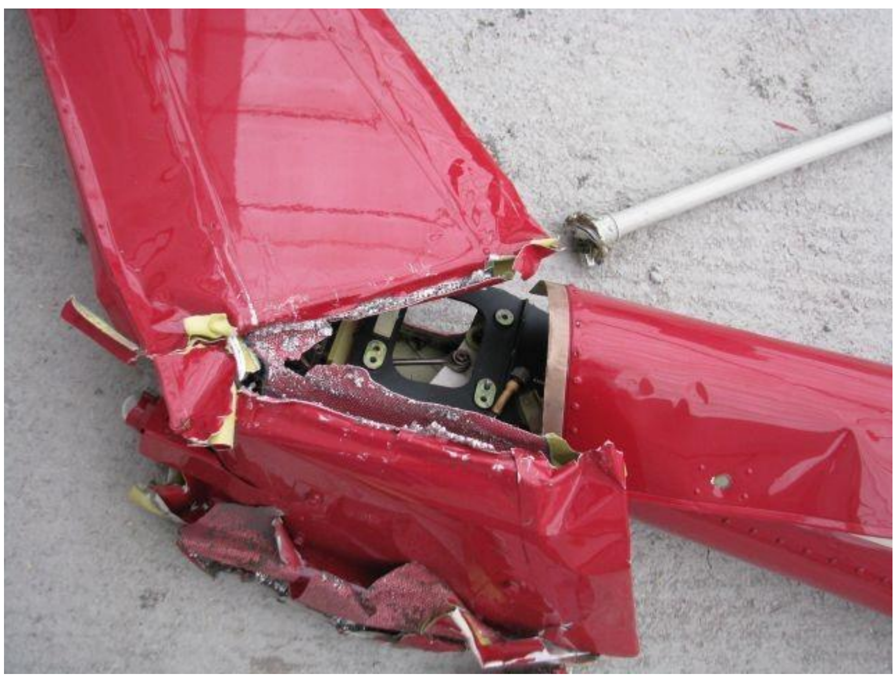


Photo 30
Fracture in middle of vertical fin from cable strike and mud on bottom fracture

Photo 31
Both crosstubes spread out with slightly more spread on right side

Photo 32
Pilot and copilot seats displaced to the right and exhibiting fractured back supports

Photo 33
Pilot (right front) seat was stroked and exhibited fractured support

Photo 34
Copilot (left front) seat was stroked and exhibited fractured support

Photo 35

Left passenger seat with seat structure fractured down

Photo 36
Passenger seat lap belt was reportedly cut by a first responder

Photo 37
Emergency fuel shutoff handle was extended to Off position, but no markings on handle

Photo 38
Emergency fuel shutoff valve was found in Off position

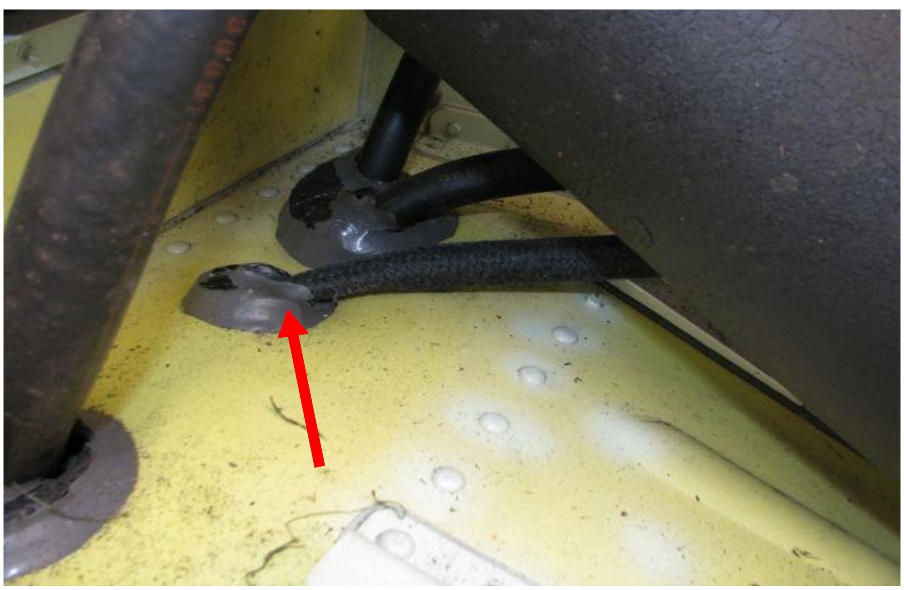


Photo 39
Fuel line was found crimped at location where it makes 90° turn at aft fuselage deck

Photo 40

When fuel line was removed, crimped section was observed at area of 90° turn

Photo 41
Fuel line was cut in area of reduced cross section and crimped area was observed

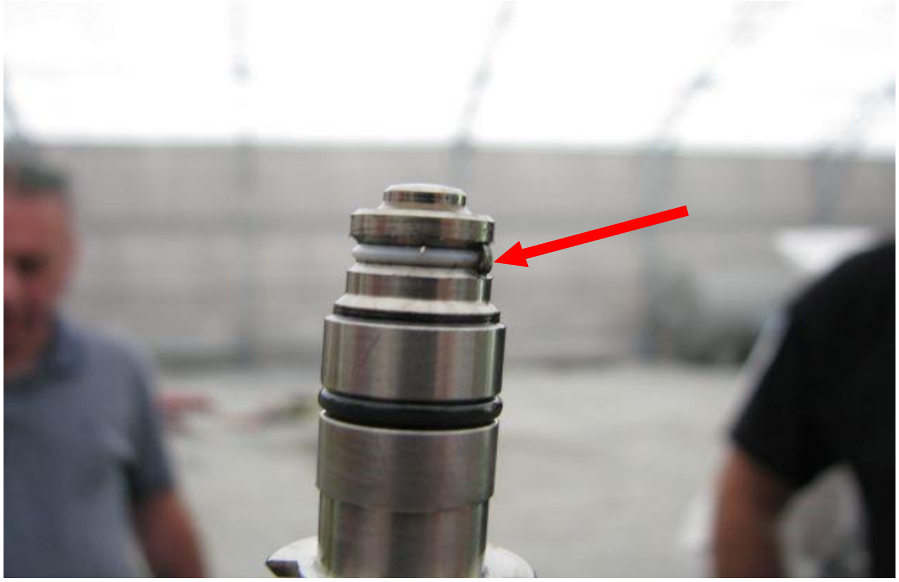
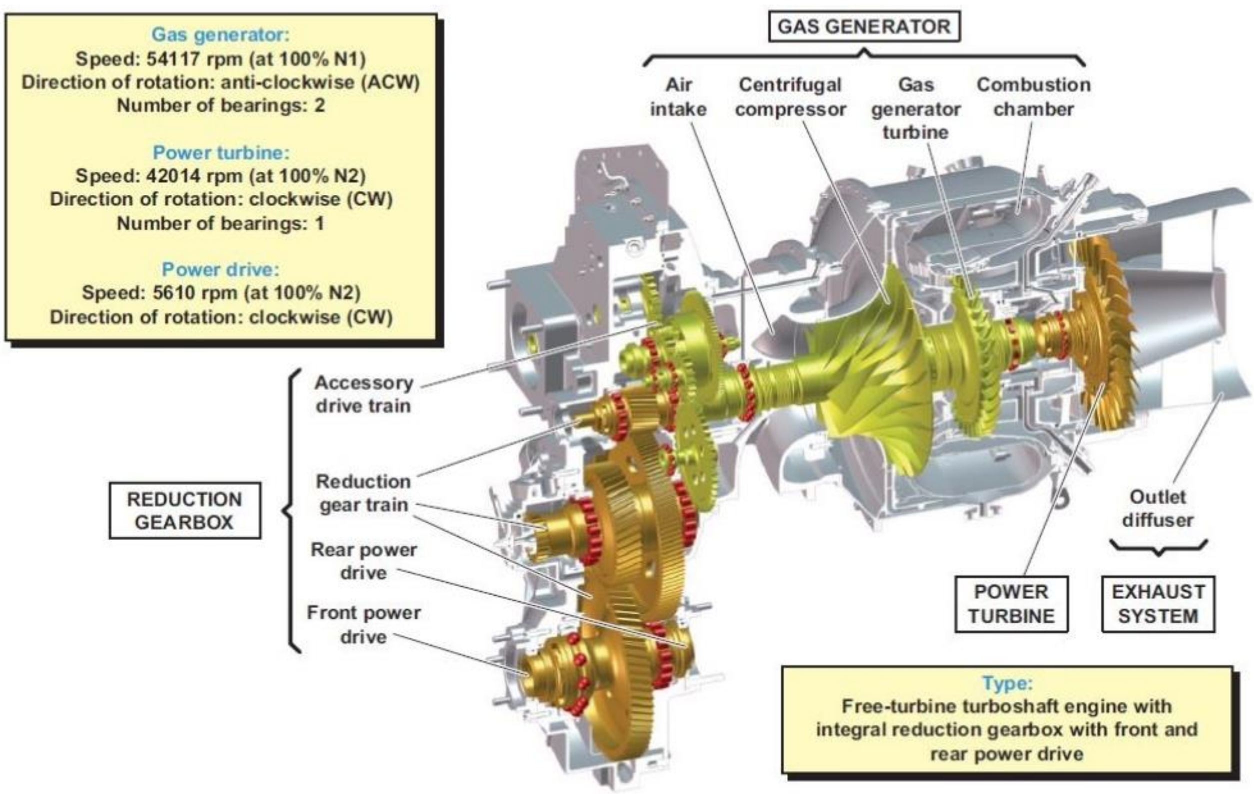


Photo 42
Engine module 01 chip detector with tab type chip found (arrow)

Engine on-site examination

The Arrius 2R is a turboshaft engine with a single-stage centrifugal compressor, a reverse flow annular combustion chamber, a single stage high pressure turbine, a single stage power turbine, and a reduction gearbox with an output at 5834 rpm at 104% Free Turbine rating which is the normal operating rating of Bell505 (100% free turbine rating provides a nominal output at 5610 rpm).



The engine at the accident site was examined without removing any parts, apart from :

- the fuel & oil filters; kept by Georgian criminal police;

- the two engine electrical magnetic plugs, put back into position;

- the left hand side borescope inspection hole cover on the combustion chamber casing, improperly put back in position;

- the left hand side igniter and start injector, improperly put back in position;

- both air entry lips & grid, put back in position;

- engine EECU; kept by Georgian criminal police.

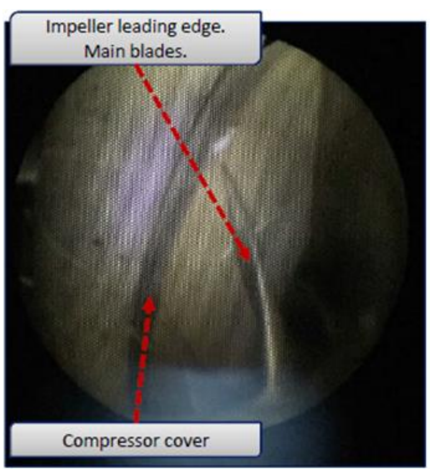
Module 01 : accessory and transmission gearbox

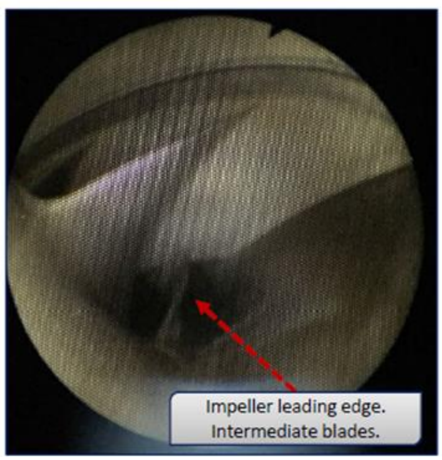
The transmission gear box is operating normally, as it transmits the rotation of free turbine shaft to both main rotor and tail rotor shaft. In order to maintain all evidences in place, it is decided not to remove any other equipment from Module 01.

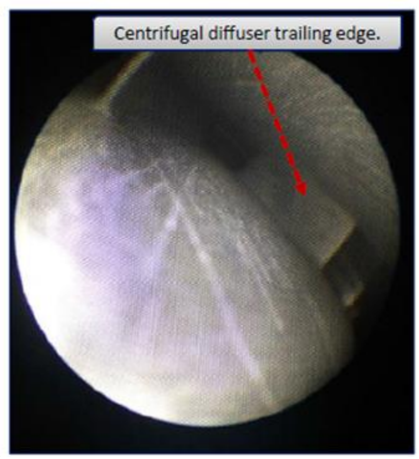
Module 02 : air intake and compressor

The impeller leading edges are found without any visible signs of damage, rub, erosion, nor FOD. The compressor cover does not present visible signs of rub in the inspected sectors.

The difference of color is normal due to the abradable coating in regard with the impeller blades. Centrifugal diffuser trailing edges are found without any visible signs of damage in the inspected sector.

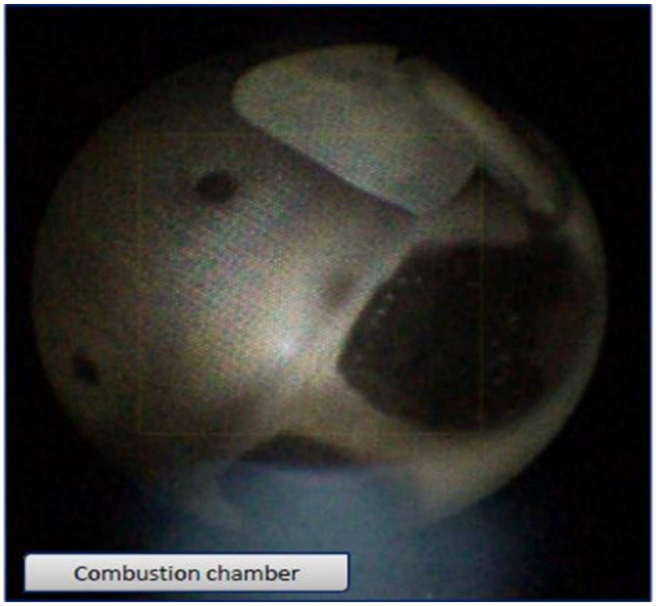


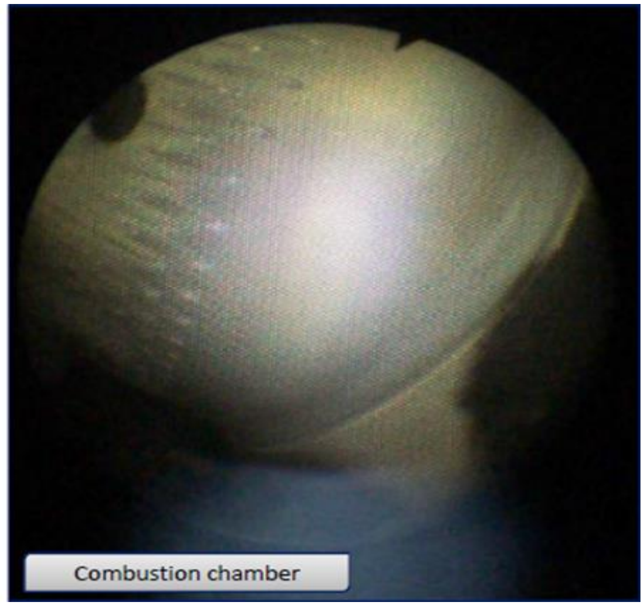




Module 02 : Combustion Chamber

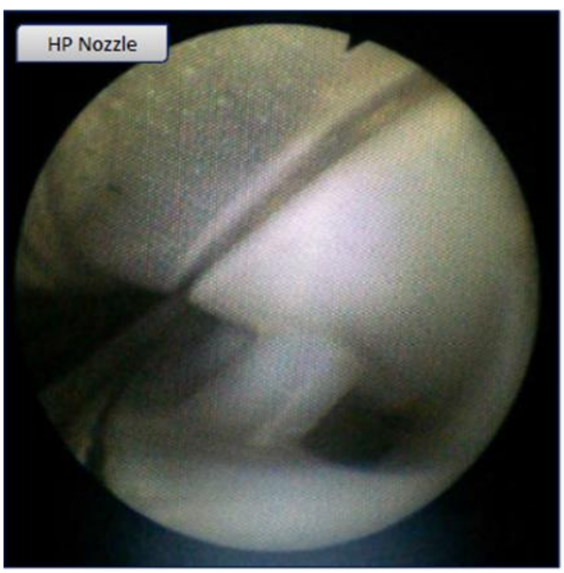
The combustion chamber was found in good condition with no visible sign of damage in the inspected sector.

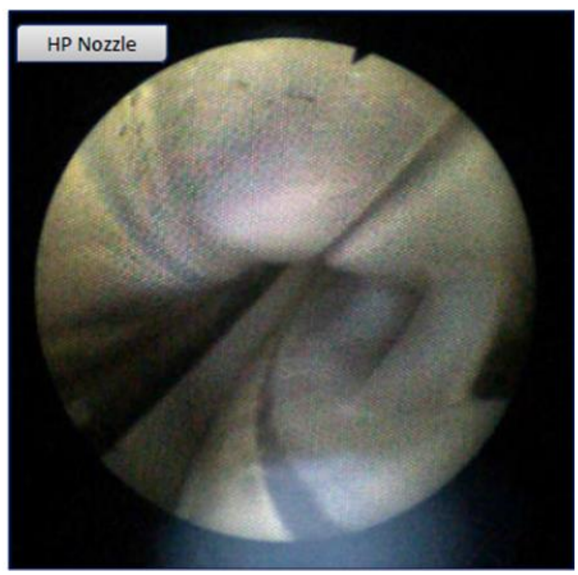




Module 02 : HP Turbine

The HP nozzle was found in good condition with no visible sign of damage in the inspected sector. The HP turbine blades were not accessible

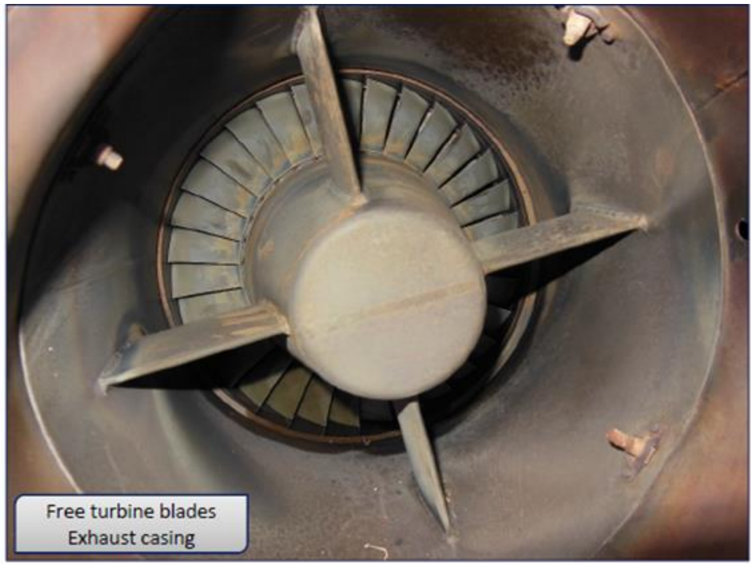




Module 02 : Power Turbine

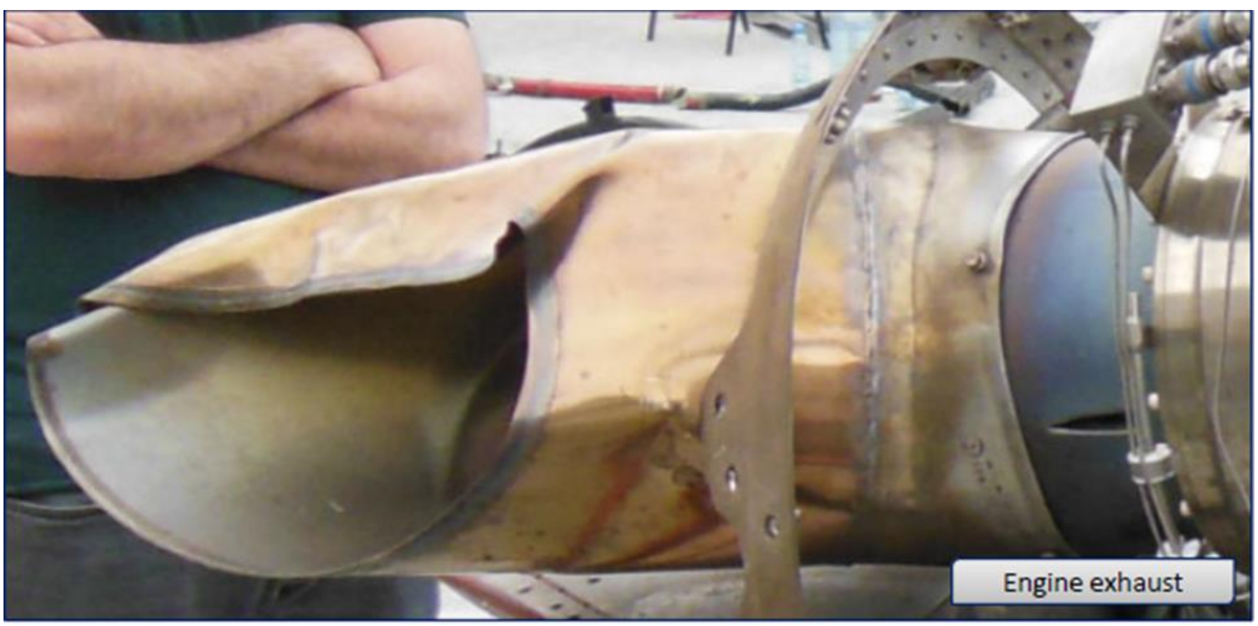
The PT nozzle was found in good condition with no visible sign of damage in the inspected sector. The power turbine blades were found in good condition with no visible sign of damage. A small liquid depot was found on the bottom of the casing (wreckage position).


Free turbine nozzle


Free turbine blades
Exhaust casing

Exhaust diffuser and cone

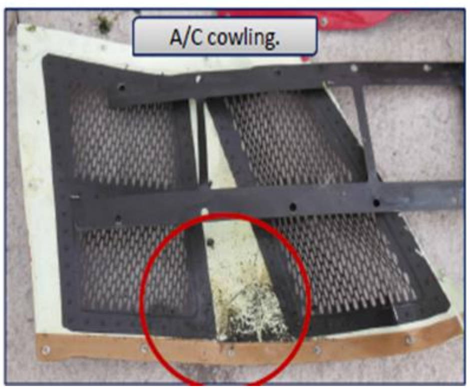
It was found damaged as a consequence of the crash.


Engine exhaust

Dressing (pipes and harnesses) and A/C connections of systems

Oil :
- The oil tank is found empty.
- Two oil samples were taken from the Oil Cooling System scavenge and kept by the Georgian authorities for analysis.
- Traces of liquid leak that may be oil was found on A/C air entry cowlings.
- No other specific finding on pipes and connections.


Oil tank.


A/C cowling.


Engine oil samples.


Engine oil scavenge screw on OCS.

Dressing (pipes and harness**es) and A/C connections of systems** **Dressing (pipes and harnesses) and A/C connections of systems**

- **Ignition** : no specific finding.
- **Harness & connectors** : no specific finding.
- **P3 picking** : no specific finding.

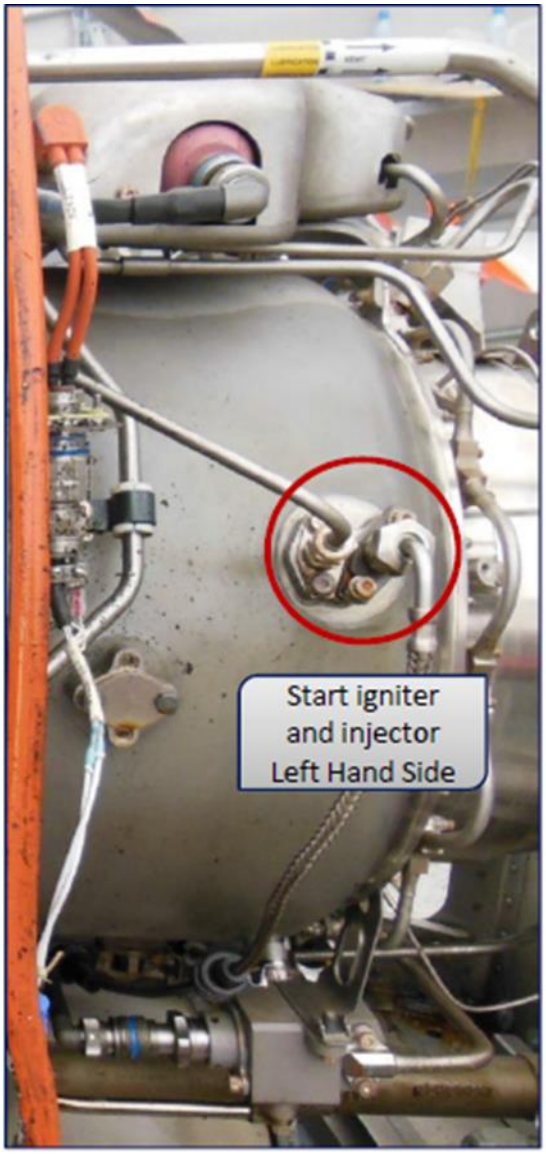
- **Fuel :** no specific finding. The fuel line from engine firewall to HMU was tested clear.

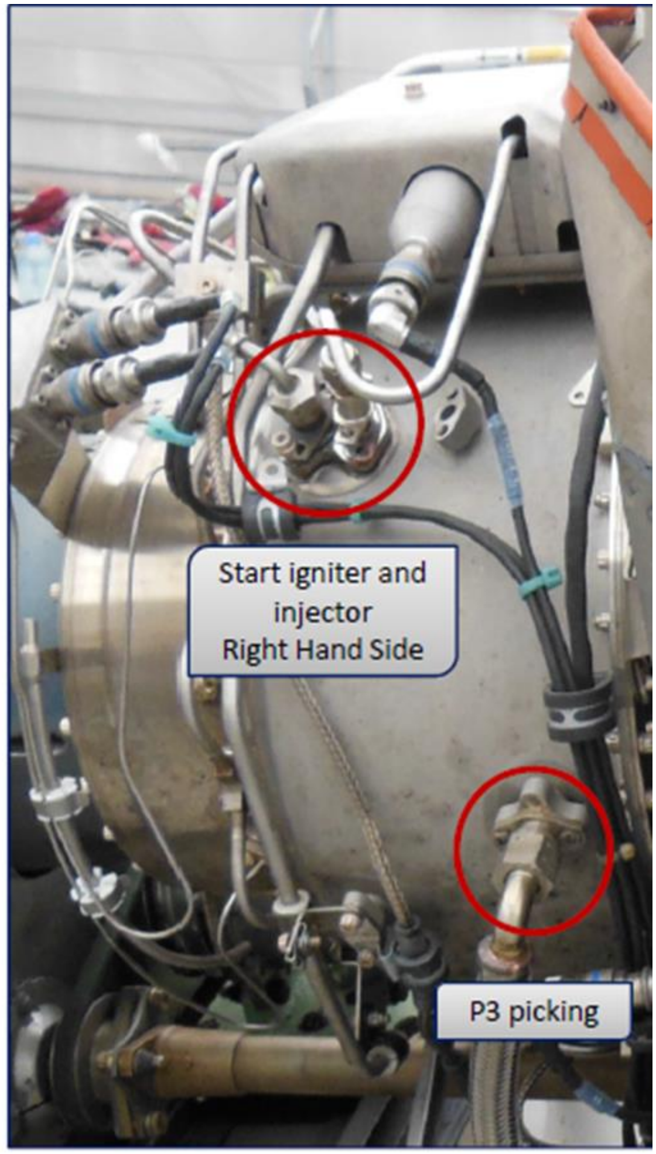
- **Harnesses & Electrical connections :** no specific finding. Especially, EECU connectors seems clean.

HMU + Fuel Line

EECU set on helicopter's deck

EECU connectors





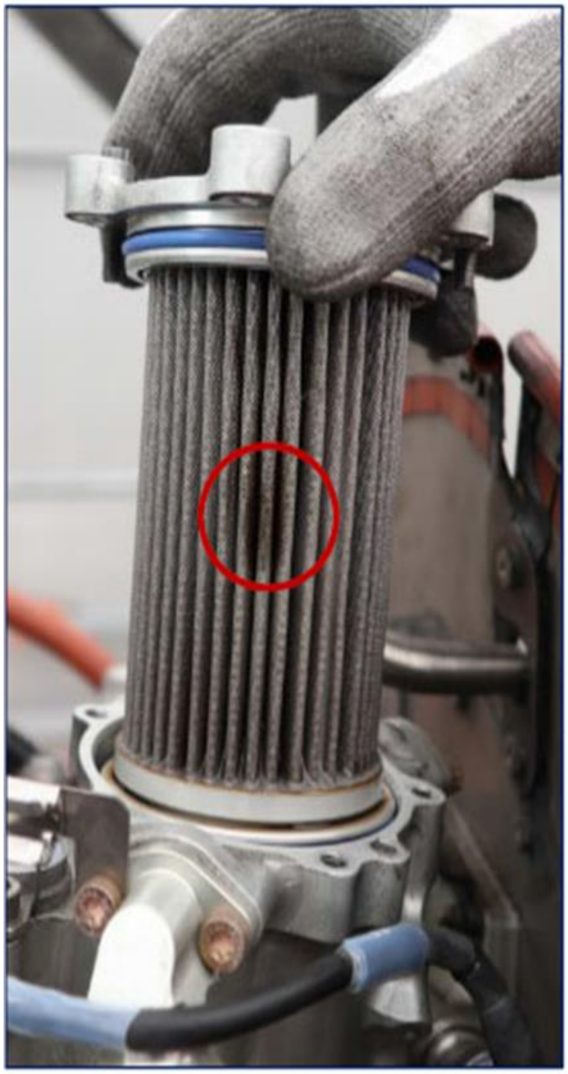
Filters & Popup

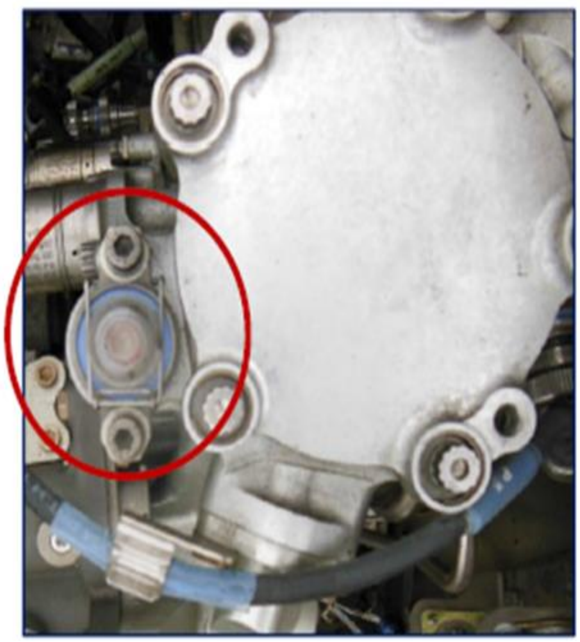
- Fuel filter

The fuel filter was checked and found clean. Traces of rusty depot is visible inside the fuel filter casing.

- Fuel filter blockage indicator

The fuel filter blockage indicator is not activated (off position).





- **Oil filter**

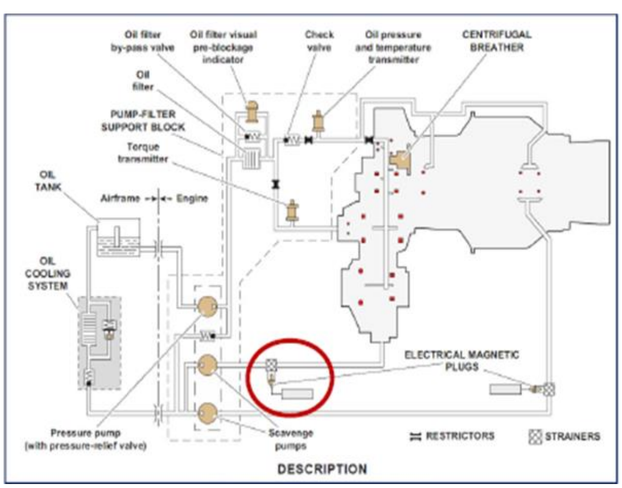
The oil filter was checked and found polluted with one single particle.

- **Oil visual pre blockage indicator**

The oil visual pre blockage indicator is not activated (off position).

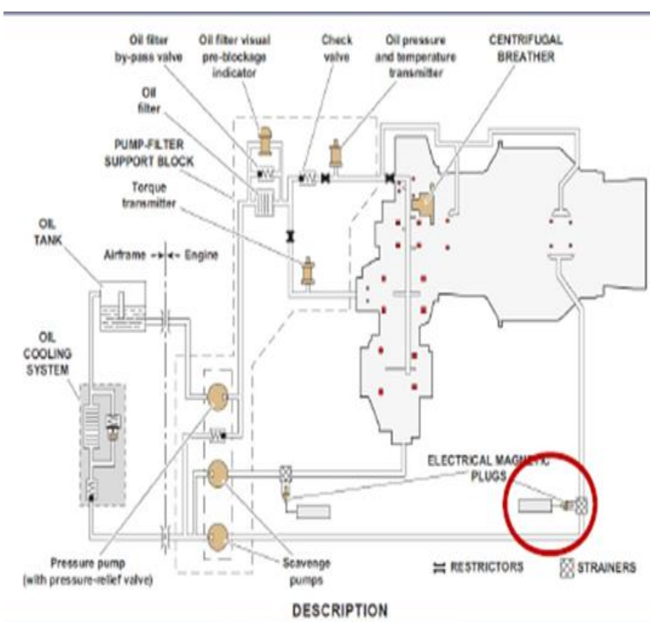
Electrical Magnetic Plugs : Module 01 scavenge line

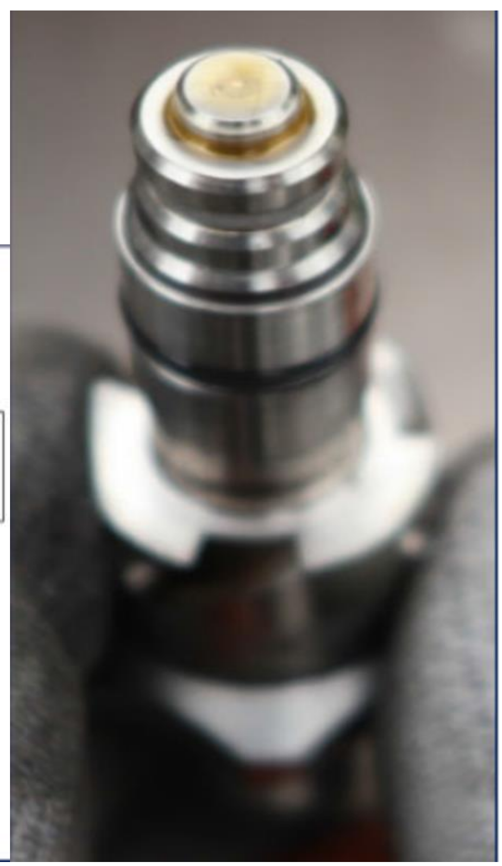
Some metallic (03) parts (bearing tab type) are visible at the junction of the magnetic segment and the main body of the EMP.



Electrical Magnetic Plugs : Module 02 scavenge line

This Module 02 EMP was found clean of any particle





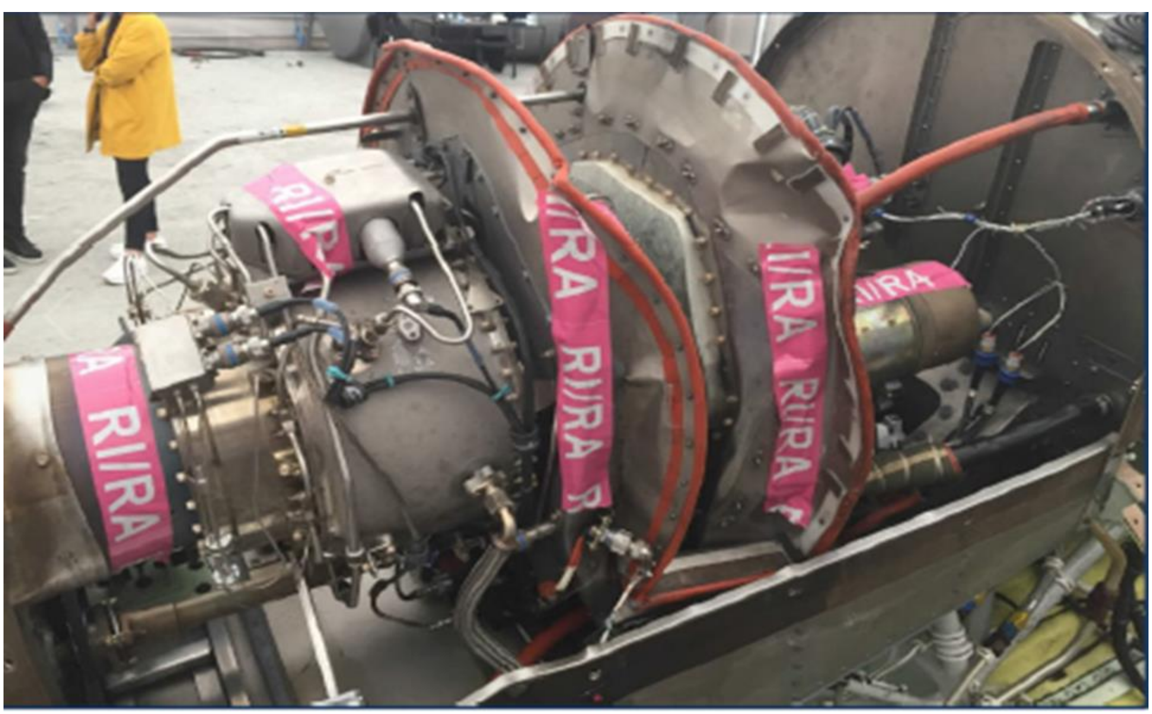
Main Equipments

- **Hydromechanical Metering Unit (including fuel pumps)** : no specific finding.

- **Fuel Valve Assembly :** no specific finding.

- **Oil Pump-Filter Support Block :** no specific finding.

The engine was left on the aircraft and sealed under IIC and Georgian police supervision



The engine exam revealed that tab-type chips were found on the engine lower chip detector (See Photo 42) although no chips were observed on the rear upper chip detector. The Ng and Np turbines rotated freely by hand. The engine was later removed from the helicopter by “Aviator +” personnel and shipped to Safran Helicopter Engines facilities at France for future examination. Inspections of the engine, of the airpath and its rotating parts were performed, as well as checks and tests on the electrical magnetic plugs and on the igniters. The removal of the accessory gearbox (module M01) was then operated to control the state of the drive train.

ABREVIATIONS

EMP : electrical magnetic plug

FH/fh : flight hours

P/N : part number

RPM : revolutions per minute

S/N : serial number

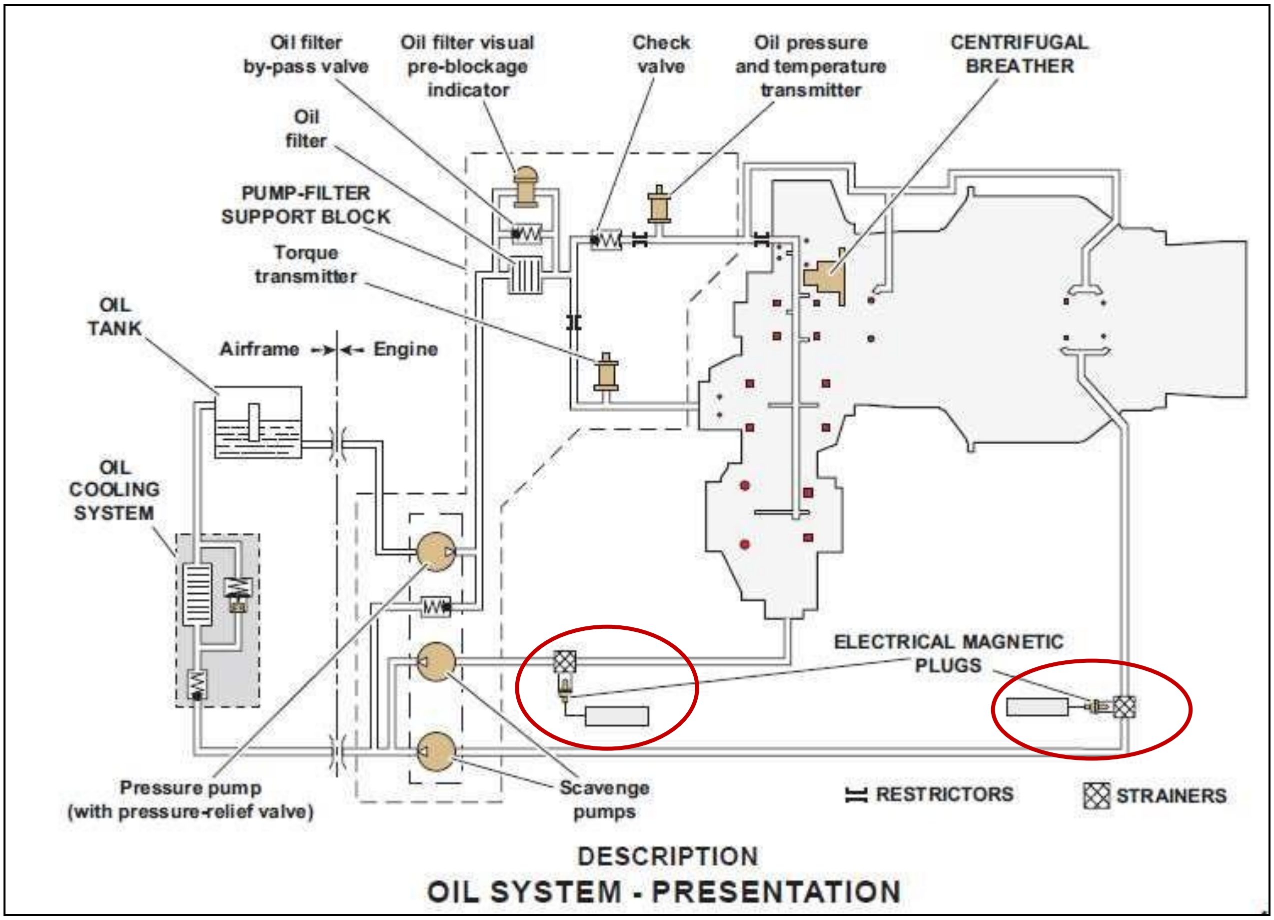
T41 : combustion chamber exit’s temperature

T45 : turbine outlet temperature

HMU : hydromechanical metering unit

TSN/TSO : time since new/time since overhaul

<u>Picture 4 :</u> engine is set on a workshop frame.



Electrical Magnetic Plugs description

Scheme 1 : arrius 2R oil system and localization of both electrical magnetic plugs

The electrical magnetic plugs (EMP) attract and retain any magnetic particles in the oil system, and provide a cockpit indication if particles are detected. They are located in the scavenge lines of each modules : M01 and M02.

Each EMP consists of a magnetic probe made up of two parts, separated by an insulator. These two parts attract and retain any magnetic particle that, if in sufficient number, form a bridge across the insulator, completing the electrical circuit that generates the cockpit indication.

Each plug is connected directly to the avionic via the engine electrical harness. In case of chip detection, an *engine chip CAS message* (**C**rew **A**lerting **S**ystem) is displayed on the avionic warning panel.

Electrical Magnetic Plugs checks

During on site-examination, it was detected that metallic particles were located on Module M01 EMP. On addition, as the examination of the data retrieved from the avionic was progressing, it was established that no chip CAS message was recorded by the avionic. It was therefore necessary to test the functionality of Arrius 2R S/N 50065 module M01 and M02 EMP, with the same protocol.

This test was based on the measurement of the resistive current in three different configurations. It was performed with a dedicated connection tool where a multimeter device can be plugged.

Electrical continuity test directly through the EMP's connector

- Electrical continuity test after EMP removal
- Electrical continuity test with deliberate electrical contact between the two heads of the EMP

This protocol was firstly validated with another EMP than those of the engine. Then it was applied to both engine's EMP.

ELECTRICAL CONTINUITY TEST DIRECTLY THROUGH THE EMP'S CONNECTORS

Both module M01 & M02 EMP's resistive values were measured at approximately 60kΩ. This means the circuit was opened and that no chip was detected.

Therefore, no chip detection could have been made in this configuration by the EMP.

This result allowed to explain no chip detection was recorded by the avionics during the last flight.

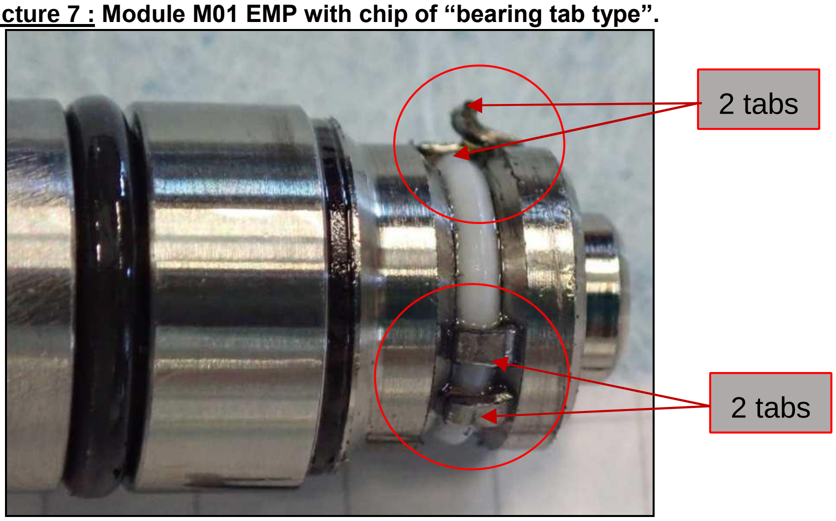
Picture 5 : direct measurement of EMP Module 01 resistance through connector.

<u>Picture 6 :</u> direct measurement of EMP module M02 resistance through connector.

ELECTRICAL CONTINUITY TEST AFTER EMP REMOVAL

Module M01 EMP was removed from the engine and the presence of 4 chips of "bearing tab type" was detected between the two heads of the plug.

<u>Picture 7 :</u> Module M01 EMP with chip of "bearing tab type".



<u>Picture 8 :</u> Module M01 EMP with 4 chip of "bearing tab type".

An electrical continuity test was performed with the tabs left as they were found: the resistance measured (60kOhm) confirmed that, in the tabs position, no current goes between the two heads of the EMP.
This result also allowed to explain no chip detection was recorded by the avionics during the last flight.

Module M02 EMP was then removed from the engine. After it was established that no chip was captured by this EMP, an electrical continuity test was performed: the resistance measured (60kOhm) indicated that the EMP functions were in accordance with the specification.

<u>Picture 9 :</u> Module M02 EMP is clean.

ELECTRICAL CONTINUITY TEST WITH DELIBERATE ELECTRICAL CONTACT BETWEEN THE TWO HEADS OF THE EMPS

The last test performed on the EMPs of both modules consisted on making a deliberate electrical contact between the two heads of the EMP.

Firstly, this was performed with a metallic part to check the EMP's functionality.

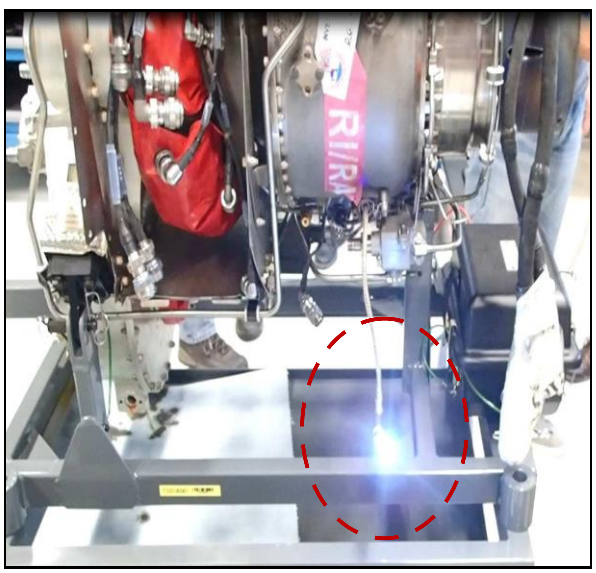
A resistance measured below 10 Ohms indicated that a current was going from one head to another and that both EMP's functions were in accordance with the specifications.

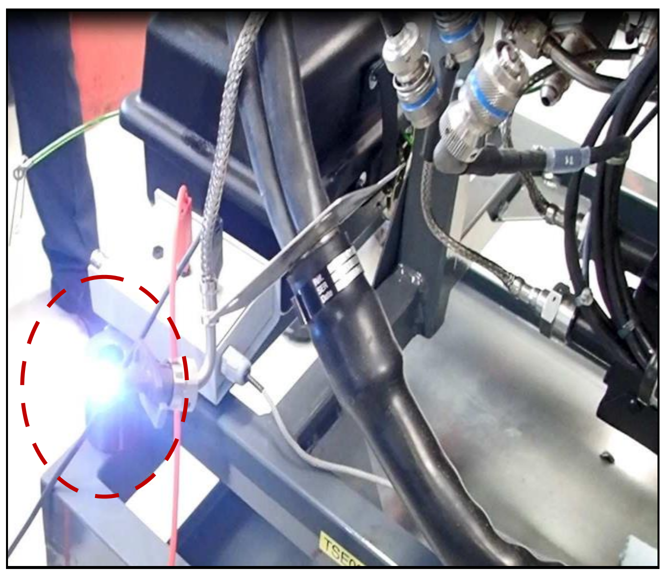
Picture 10 : **deliberate contact between two heads of EMP of Module M01 - still with tabs on.**

Secondly, specifically on Module 01 EMP, it was performed directly by the IIC who was pushing the tabs with its fingers until the electrical contact was confirmed by the multimeter.

Start Igniters check

The start igniter test was requested during the investigation by the IIC. Both start igniters were found functioning as requested.





Pictures 11 & 12 : **Left and Right Hand igniter test.**

Engine evaluation

The engine was evaluated in regard of usual repair criteria by an Arrius specialist.

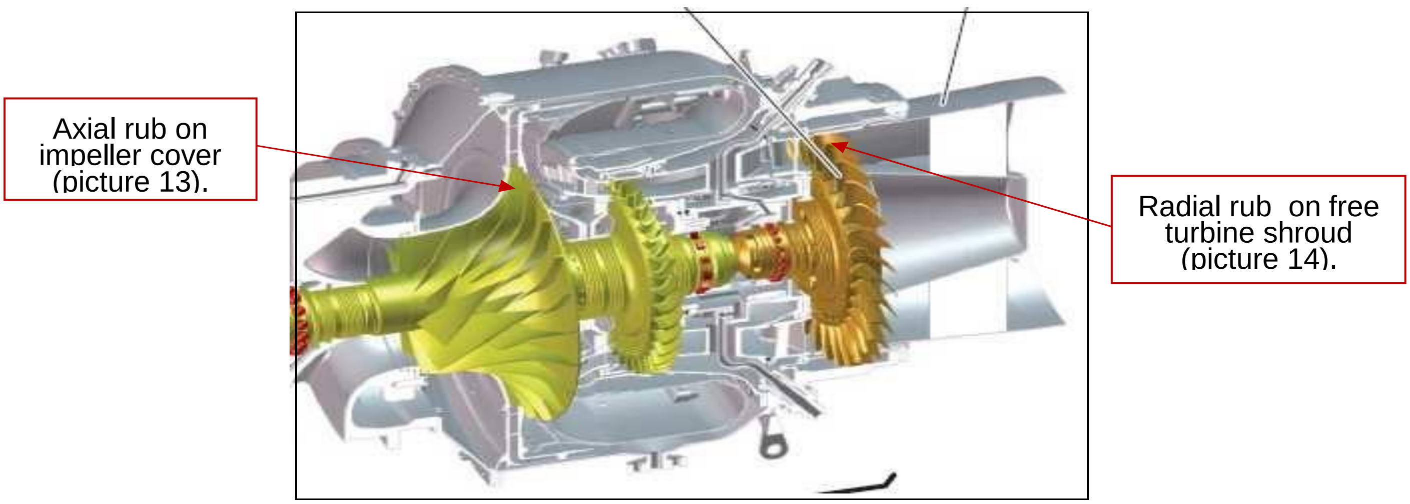
The smooth and free rotation of both N1 and N2 shafts was confirmed with no abnormal noise.

A detailed borescope inspection of all engine stages confirmed the good state of the engine internal parts with similar visual statements as the on-site examination. It was however detected two additional findings:

- an axial rub on the compressor cover at the impeller’s trailing edge.
- a radial rub on the free turbine shroud.

Both rubs were not detected during on-site examination. They are not significant: no visible coating projection with no visible blade damage means low energy on shafts at the moment of the rub.

As the parts are not touching the rubbed zone anymore, it means it is likely that these rubs occurred at impact.

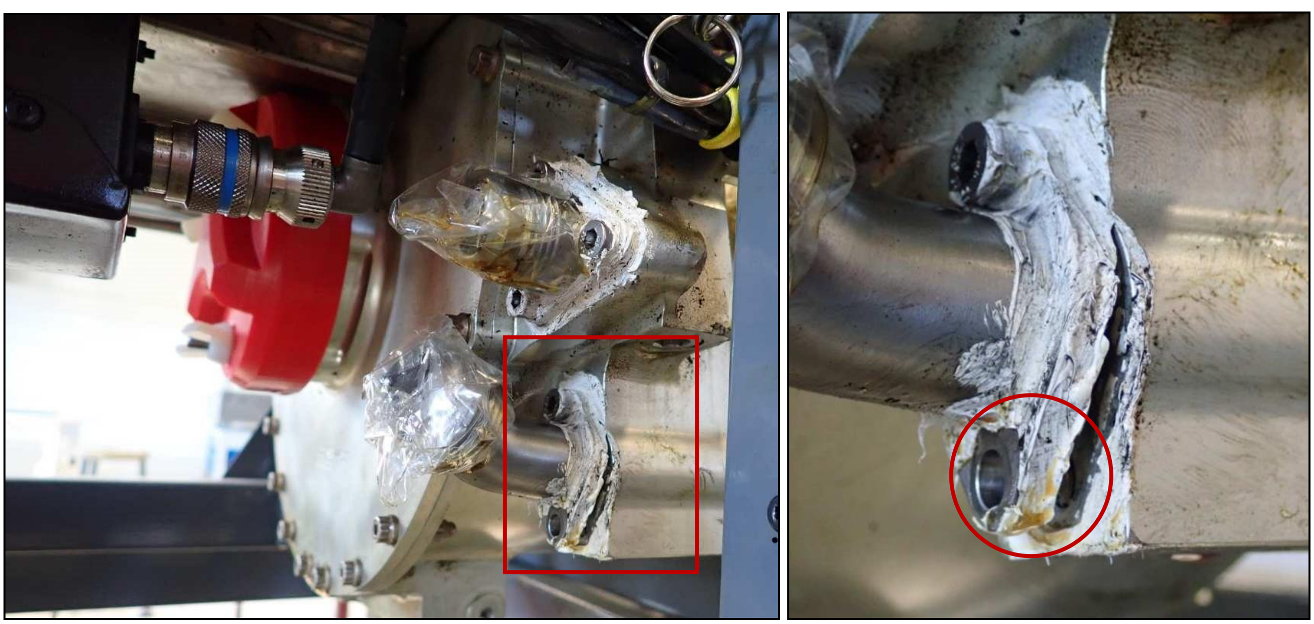


<u>Picture 13 :</u> ligth rub of compressor cover, at impeller trailing edge (red rectangle).

<u>Picture 14 :</u> light rub of free turbine shroud, at free turbine blade tips (red rectangle).

A visual statement is made that the oil connection from the oil tank to the module M01 casing was found partially unplugged. One of the two screws of the flange of the plug was also missing. This may explain why the oil tank was found empty on the accident site.

This kind of damage is repairable in the perspective of an engine run trial.



<u>Pictures 15&16 :</u> oil supply of Module M01 is unplugged and a screw is missing.

A visual inspection was performed on the oil pump which was later removed : no particular findings.

According to this evaluation results, it is established by Safran Helicopter Engines to the IIC and to all the investigation team that an engine run trial on a test bench could be attempted.

IIC has decided not to run the engine at test bench but to open the engine Module M01.

Module M01 removal

Following IIC decision, the engine module M01 is removed after the engine evaluation.

OIL SAMPLE AND OIL FILTRATION

This operation was requested during the investigation by the IIC.

Before any operation on the module M01, the oil recovered from the accessory gearbox was filtered in order to check for particles.

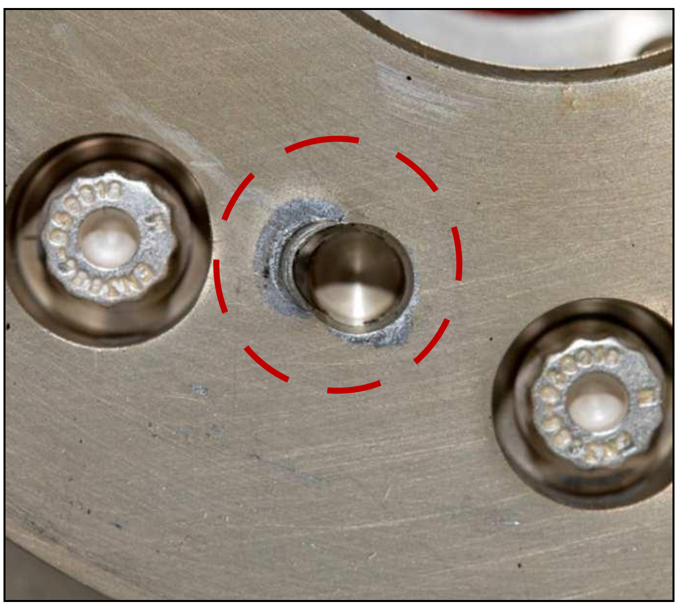
The result of this filtration did not lighted up any significant nor additional information.

Picture 17 : **accessory gear box oil filtered.**

HMU REMOVAL

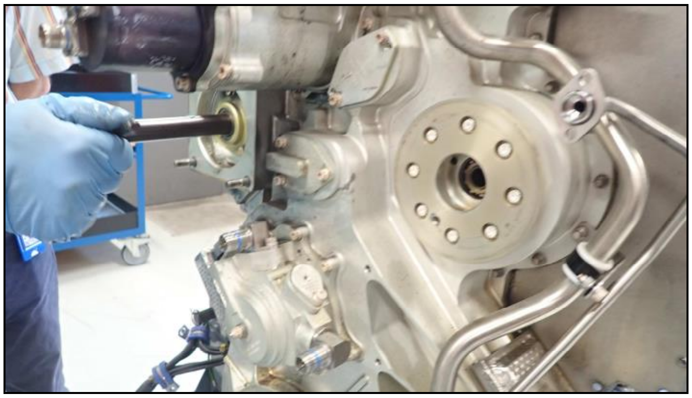
The HMU is removed from module M01. The positioning pin sets the angular position of the HMU on the module M01 casing. A masting of this positioning pin on the module M01 casing was detected. This statement was relevant with the induced metal beading and the ovalization of the drilling on the module M01 casing side. No other traces of displacement were found on the two cover flanges. These findings allow to conclude that the damage of the positioning pin was due to impact forces at the moment of the crash.

Picture 18 : **positioning pin of the HMU below the centering bore and its rubber seal.**



Picture 19 : **casing side, ovalization and metal beading of the drilling of the HMU's positioning pin.**

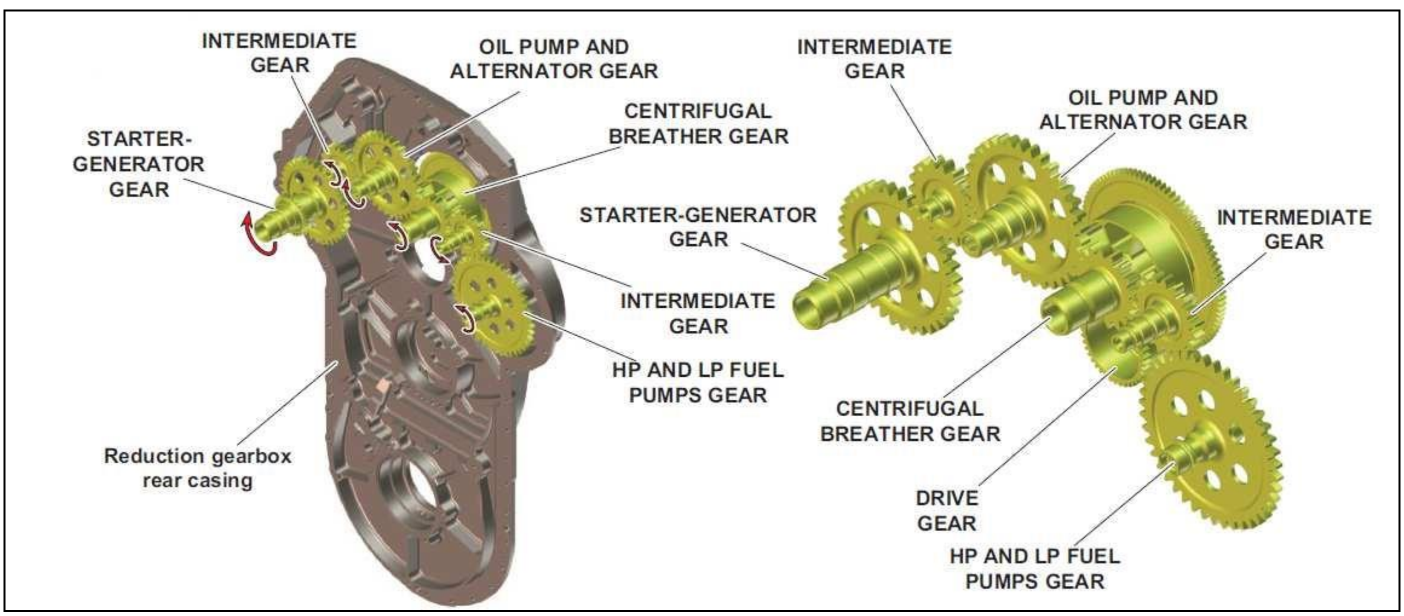
After this operation, a check of the mechanical continuity of the accessory gearbox was performed to confirm the HMU shaft was driven properly by the associated gear train, and especially by the HMU intermediate gear (port side). Smooth and free rotation of the gears is stated with no abnormal noise.



<u>Picture 20 :</u> mechanical continuity check of the accessory gearbox.

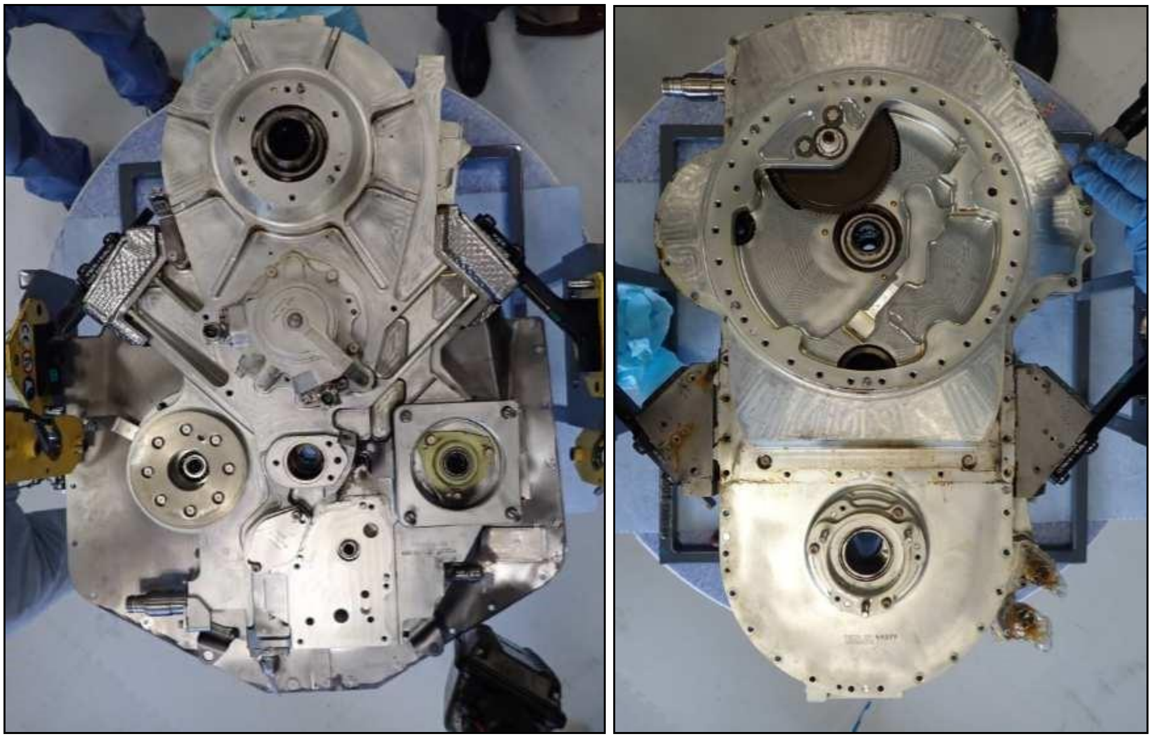
ACCESSORY GEARTRAIN INSPECTION

As tabs were recovered on the module M01 EMP, being aware of an existing in-service experience, it was decided to inspect the accessory gear train to find the origin of these particles, and to look for any visible damage.



<u>Scheme 2 :</u> accessory geartrain details.

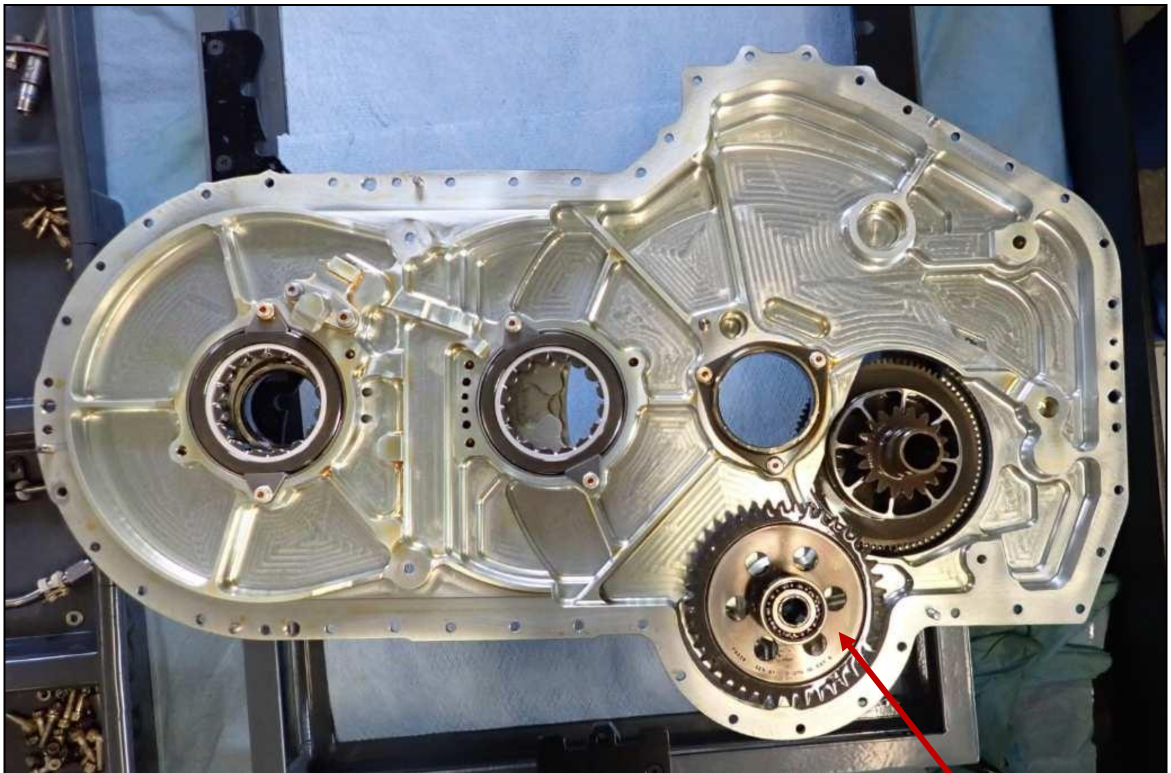
The first operation consisted on removing and then opening the module M01.



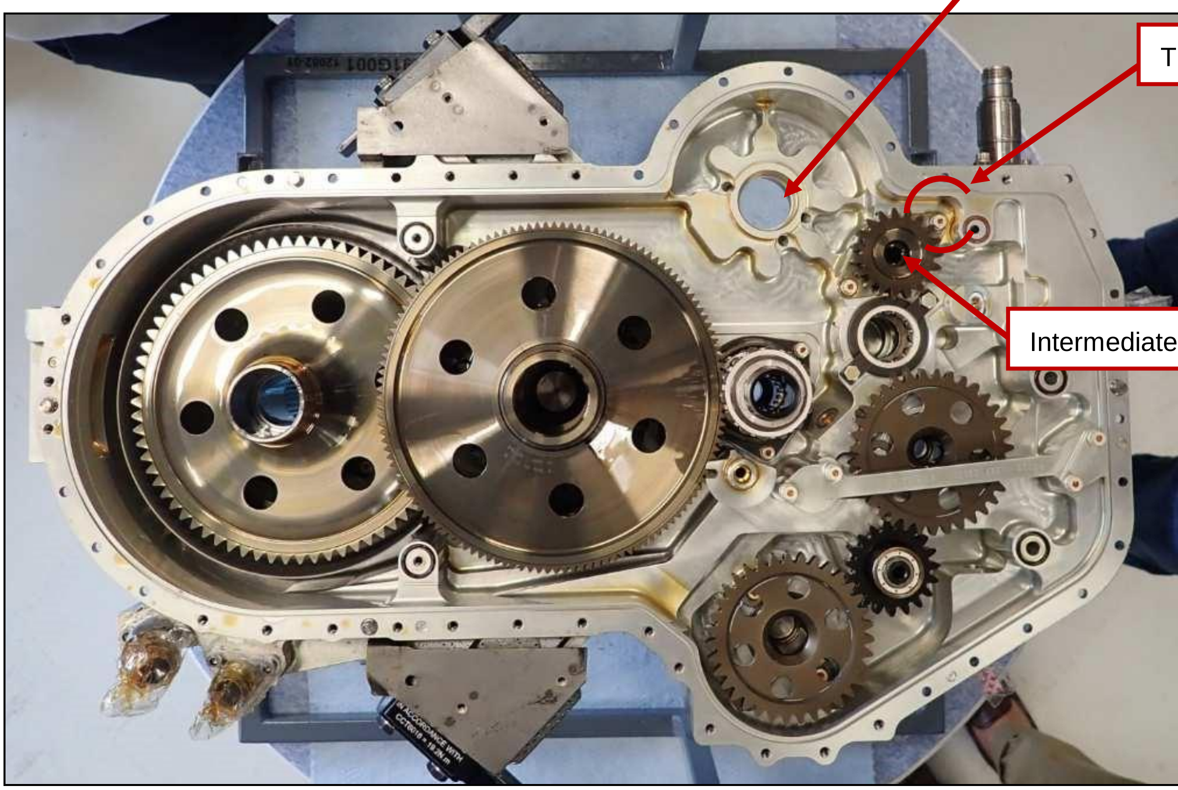
<u>Pictures 21 & 22 :</u> Module M01 front side & M01 rear side.

<u>Picture 23 :</u> Module M01 opening.

Once the front casing of module M01 was removed, an inspection of the gear train was visually performed. No visible damage was detected on any parts. A bearing tab was detected into one housing and was collected later by the IIC along with the HMU intermediate gear (port side).



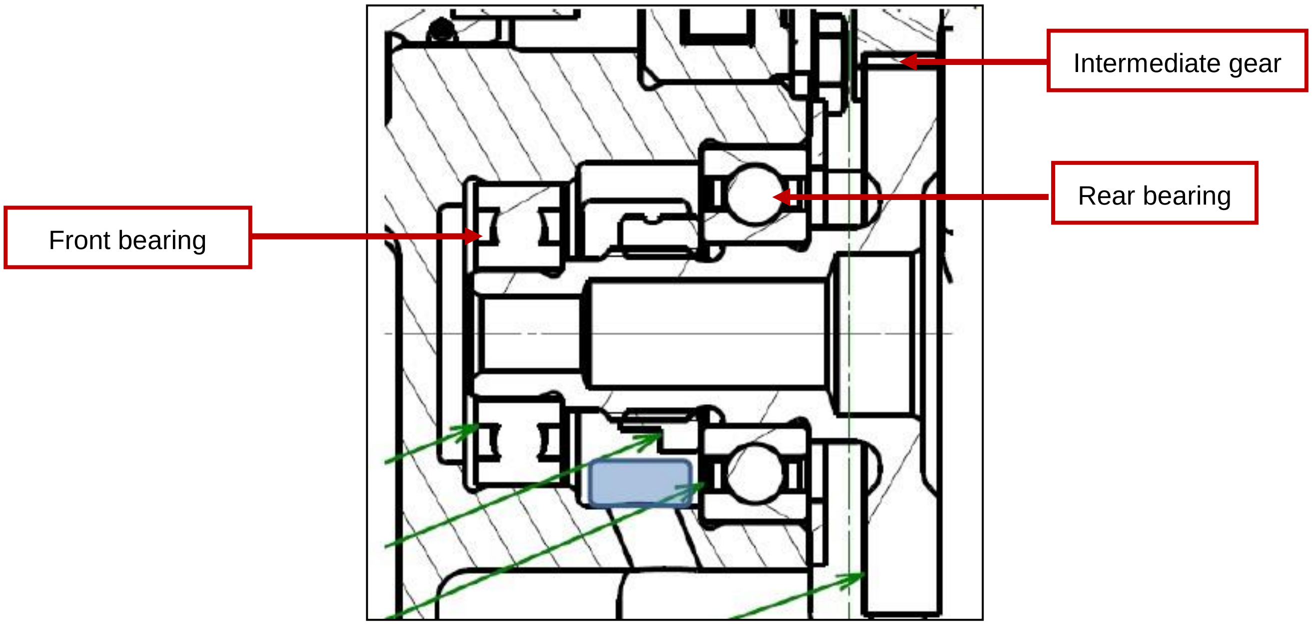
<u>Picture 24 :</u> Module M01 gears, rear casing, front side side.



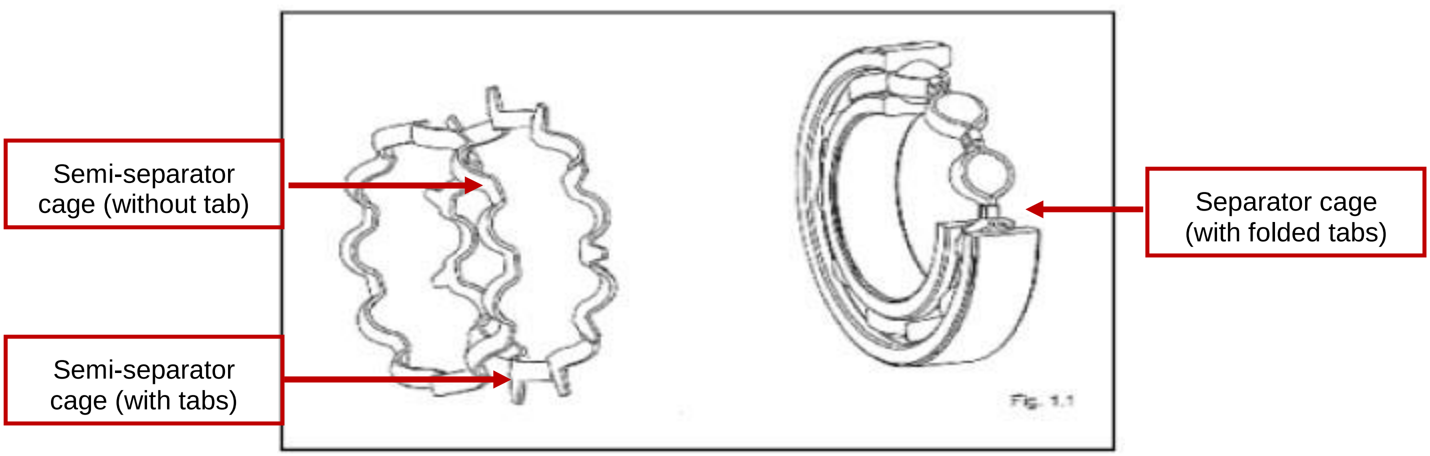
<u>Picture 25 :</u> Module M01 gears, front casing, rear side. Bearing tab located in the red circle.

HMU INTERMEDIATE GEAR

The module M01 front casing removal allowed access to the HMU intermediate gear (port side).



Scheme 3 : **cross section of intermediate gear.**

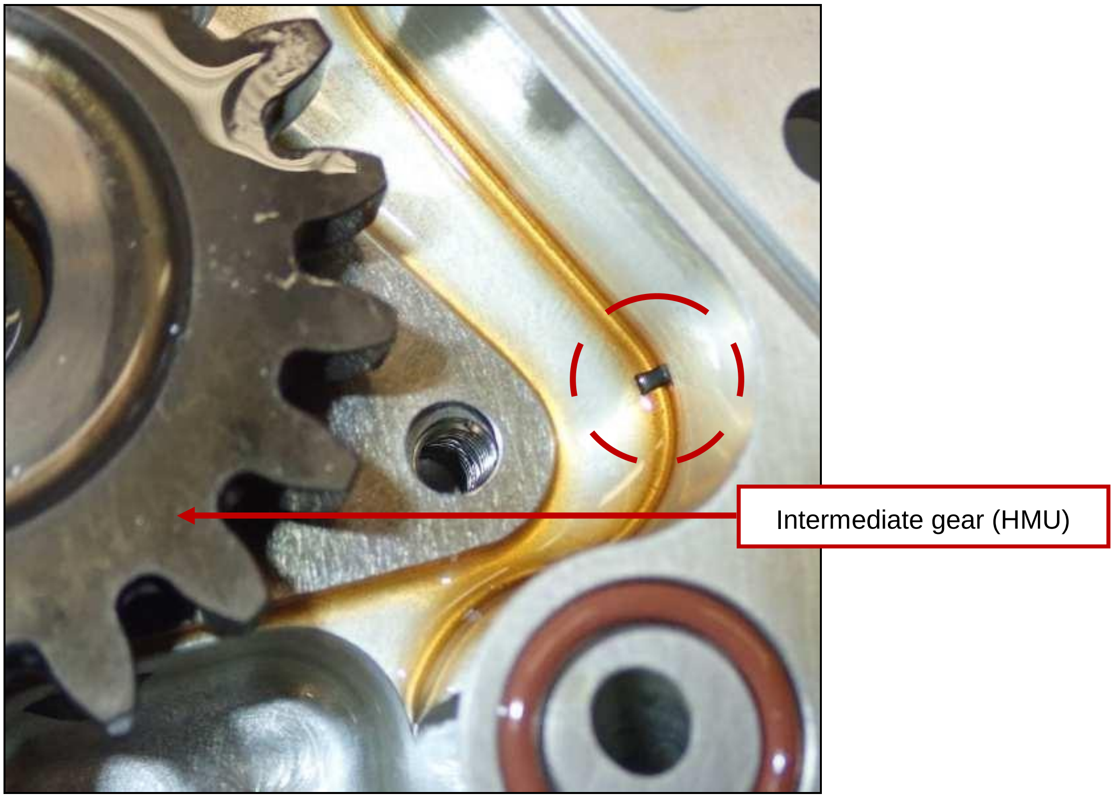


Scheme 4 : front bearing of intermediate gear 3D cross-section

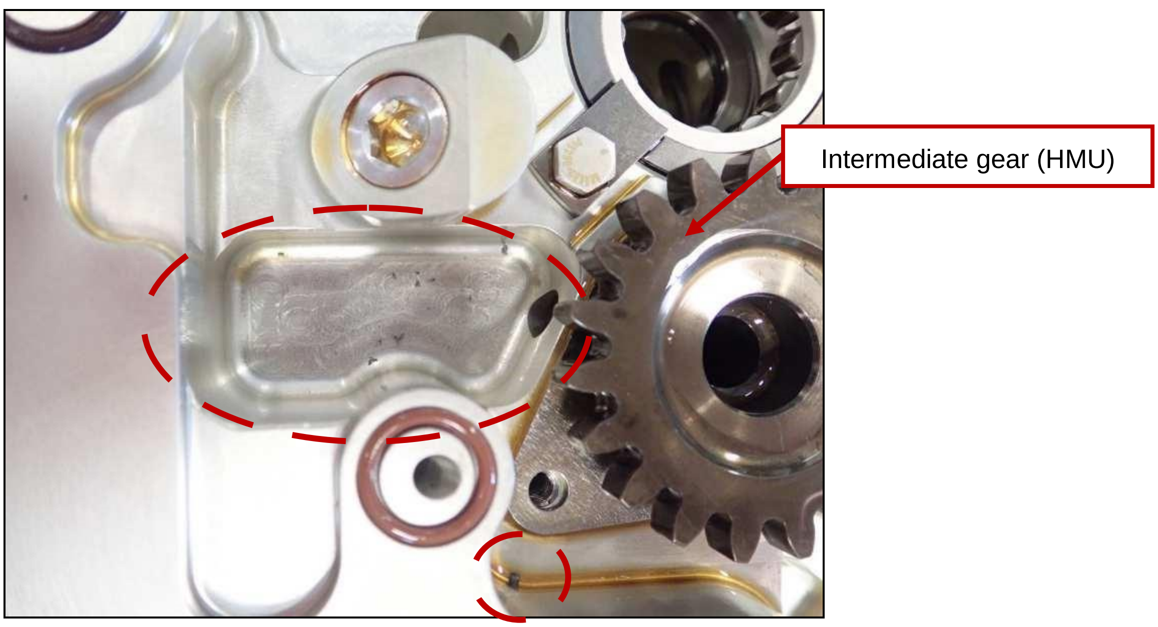
The front bearing of this gear is composed of :

- one inner and one outer race;
- 8 balls;
- two distinct semi-separator cages, one of which presents 8 folded protuberances to bond the two cages together (also called "tabs" when broken), and to imprison the balls.

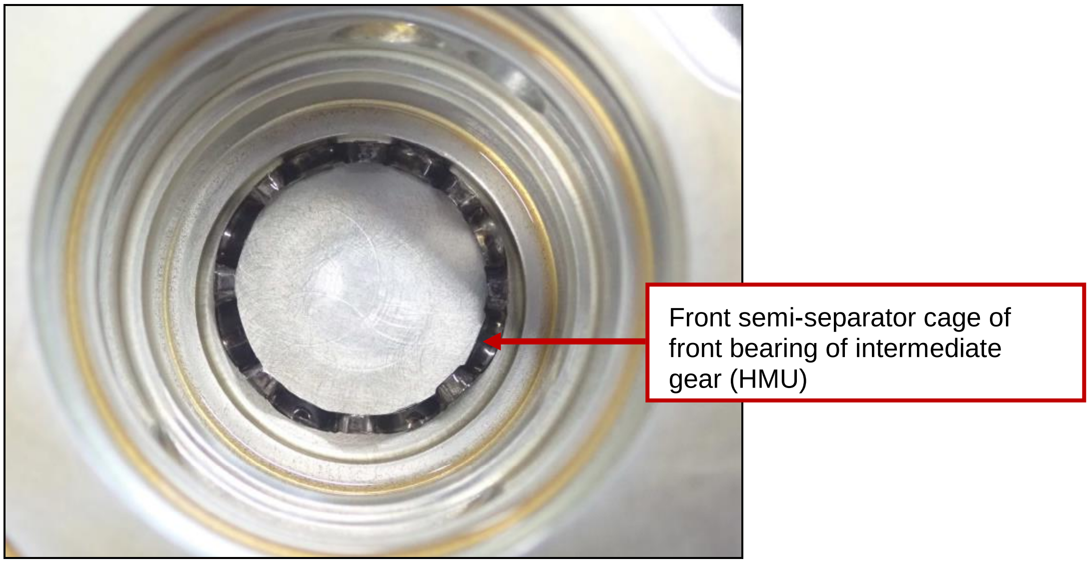
The intermediate gear was removed along with its two bearings.



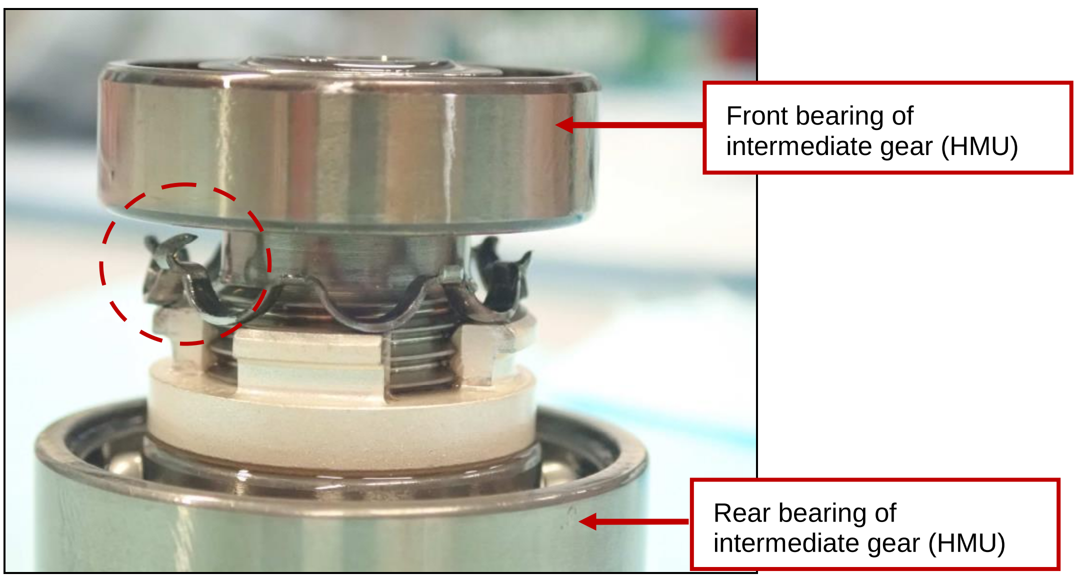
<u>Picture 26 :</u> Bearing tab located in the red circle, in the intermediate gear housing.



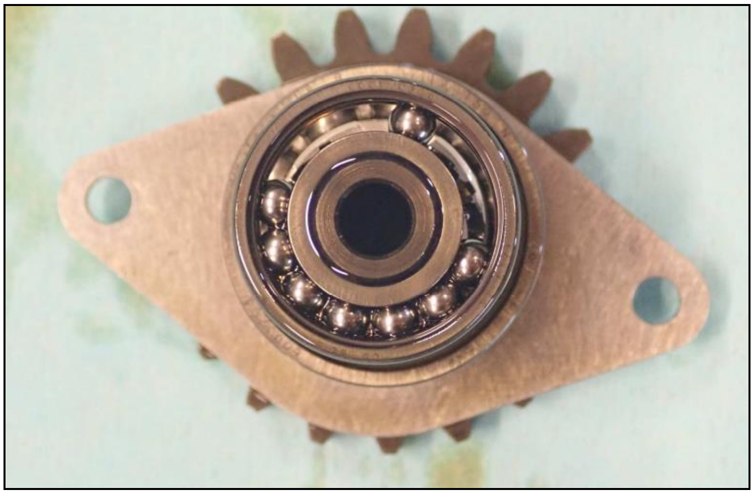
<u>Picture 27 :</u> Dust of unknown material was detected in one housing next to the intermediate gear housing. It was put into a dedicated plastic packet along with the tab found into the casing (previous picture).



Picture 28 : **one of the semi-separator cage is left into the housing of the front bearing of the intermediate gear. Considering the loss of bearing tabs, this is an expected statement once the gear is removed.**



Picture 29 : **the rear semi-separator cage with the only tab that did not brake.**



<u>Picture 30 :</u> the front bearing of the HMU's intermediate gear : no ball was missing.

The loss of 7 of the 8 "bearing tabs" was then established:

- One was still set on the semi-separator cage.
- As previously quoted in, four tabs are recovered on module M01 electrical magnetic plug.
- One is recovered into the intermediate gear housing.
- 2 tabs are missing.

On addition, no visible damage was noticed on the 8 balls (none missing) nor on the races of the front bearing.

The occurrence of loss of bearing tabs is known from Safran Helicopter Engines, who introduced a new bearing design from December 2018 through the publication of the "Tf 90" service bulletin [2] SB 319 72 4090.

In the new bearing design, the separator cage is now riveted and not stamped. Indeed, this former manufacturing process was introducing indications of folded material that was leading to a stress concentration, therefore to a crack initiation and eventually to a fatigue rupture of the stamped metallic parts of the semi separator cage, then called "tabs".

The loss of 7 out of 8 tabs is part of the known in-service experience of Arrius 2R, which allowed to establish that this damage did not alter the engine operation. (see attachment 1. „Safran Helicopter Engines" presentation)

According to the module M01 examination, no relation can be made between the accident sequence and the loss of front bearing tabs of the HMU intermediate gear.

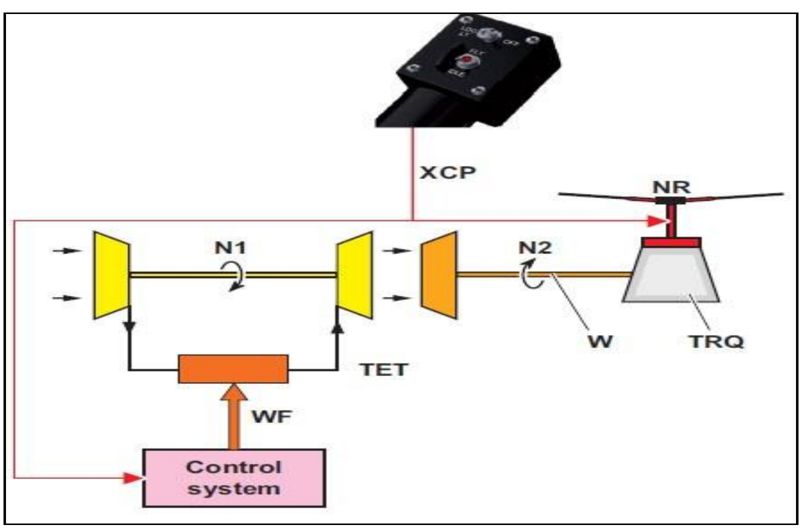
Electronic Engine Control Unit, Garmin Multi-Function Display (MFD) and Primary Flight Display (PFD) examination.

CAS	Crew-Alerting System
DIN	Discrete Input Number
EECU	Electronic Engine Control Unit
IIC	Investigator In Charge
MFD	Multi-Function Display
PFD	Primary Flight Display
PON	Power-On Number
POT	Power-On Time
Safran HE	Safran Helicopter Engines
TAIIB	Transport Accident/Incident Investigation Bureau
UTC	Coordinated Universal Time
WOG	Weight On Ground

Sistem description

ENGINE CONTROL OVERVIEW

The EECU aims to maintain the Power Turbine speed (N2) and hence the rotor speed (NR) at nominal value. It does so by assessing the speed (N1), and hence the power, delivered by the Gas Generator. The target N1 is obtained by modifying the fuel flow (WF) to the engine. Actions on the collective pitch lever (XPC signal) and/or Yaw pedals (XPA signal) can have an effect on the target N1 and fuel flow.



__Scheme 1 :__ **engine control overview**

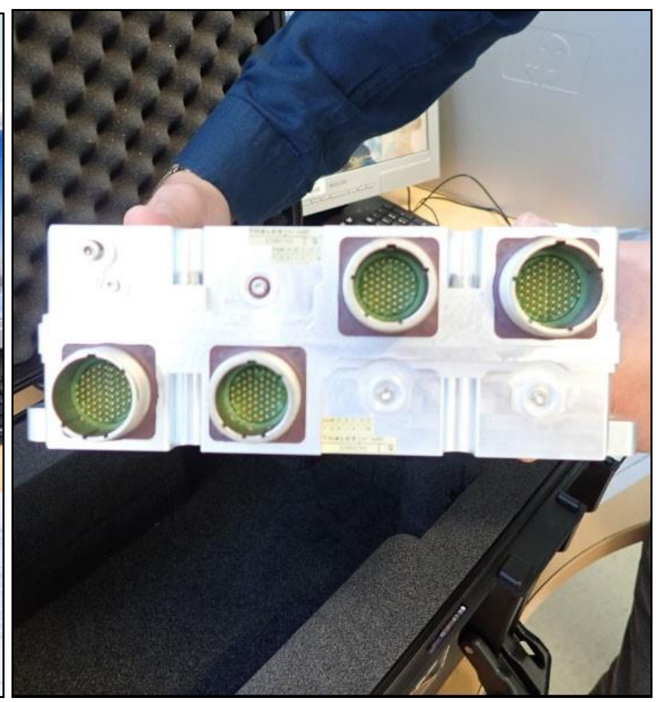
EECU EXAMINATIONS

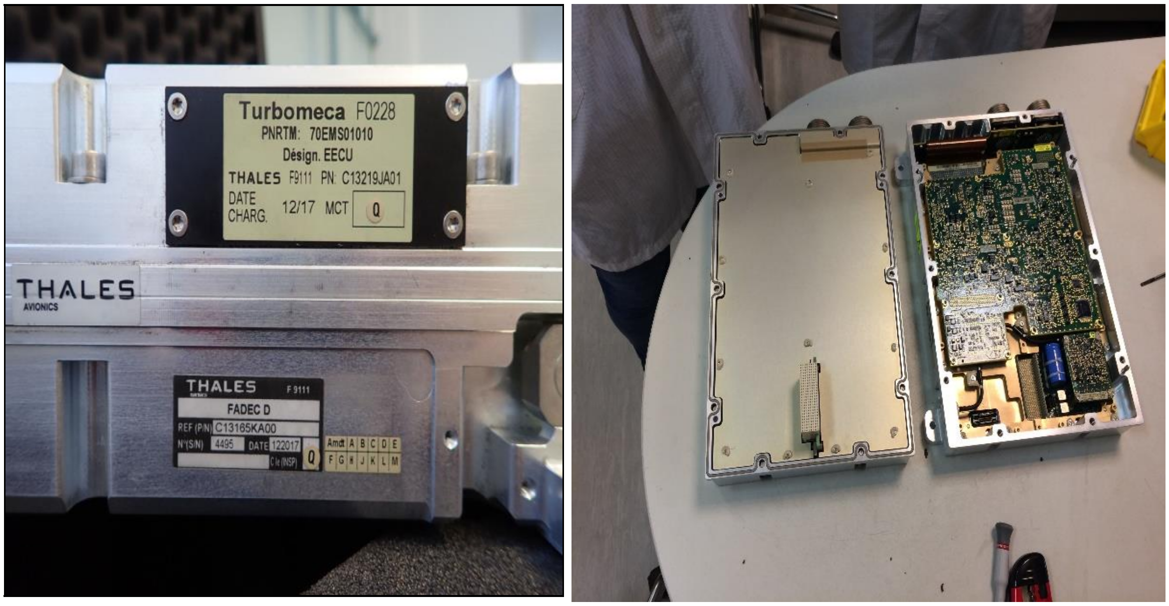
EECU PRELIMINARY CHECKS IN BEA

As part of the meeting organized by BEA in Paris on Monday 15th July 2019, BEA specialists had the EECU integrity checked in order to evaluate that no damage may affect its internal memory during the dump in Safran Helicopter Engines facilities. Following these verifications, BEA granted Safran Helicopter Engines to read and dump the EECU in their lab.

VISUAL CHECKS IN SAFRAN HELICOPTER ENGINES FACILITY

The EECU came in a protective antistatic wrapping (unsealed at BEA premises).

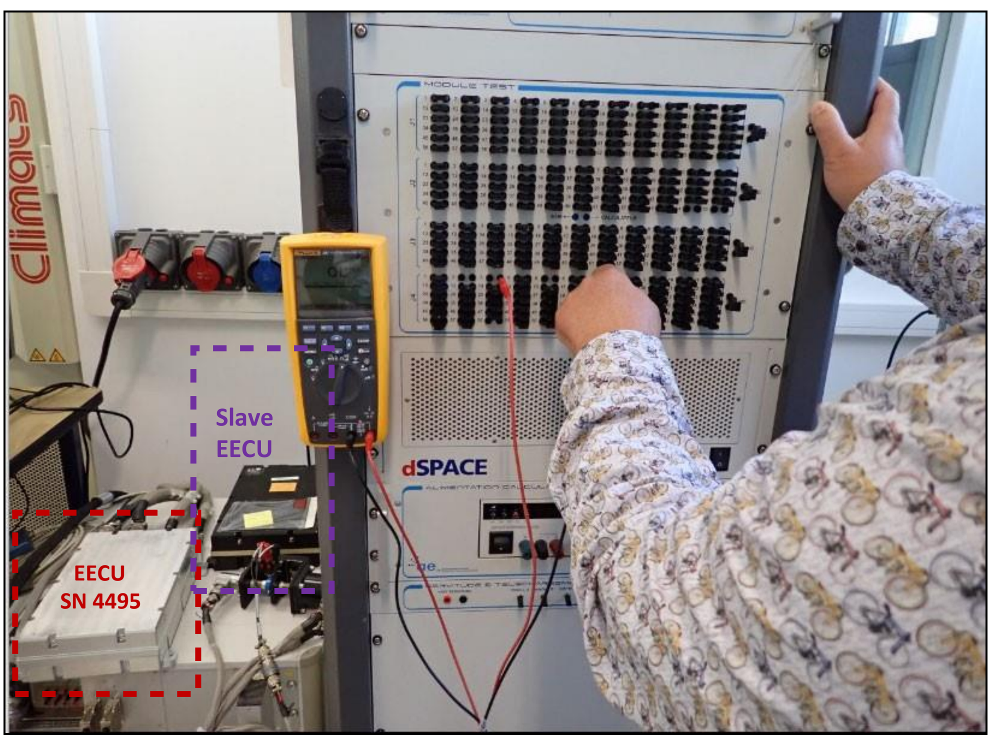




Picture 3 : **EECU reference and part number. Thales sealing is unsealed since the examination at BEA.**

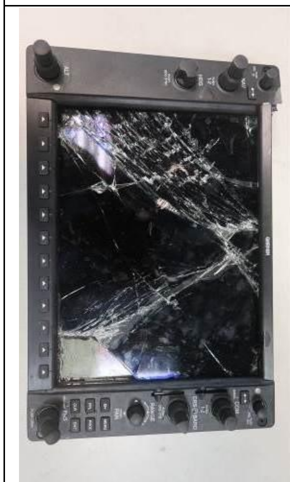
ELECTRICAL PRELIMINARY CHECKS IN SAFRAN HELICOPTER ENGINES FACILITY

Prior to downloading any information, electrical checks were performed on the EECU's connectors to look for potential internal damage. The electrical checks results were satisfactory and the download went ahead.



Picture 4 : **EECU SN4495 connected on electronic test bay and on-going continuity check.**

SYSTEM DESCRIPTION

	G1000 PFD	G1000 MFD	EECU
			
Manufacturer	Garmin	Garmin	Thales
Part number	01100972-20	01100972-20	70EMS010100
Part number and serial number (SD cards) Equipment serial number (EECU)	010-00330-4G 65074 - PFD	010-00330-4G 65074 - MFD	4495

EECU

The EECU is a dual module digital control unit performing fuel regulation, engine parameters management and engine data recording. One module is called channel A, the second one is called channel B. One channel is in control at a given time, and the other channel monitors the system in stand-by, in order to take over in case of an internal failure. On each module, engine parameters, logical words and failure flags are stored in non-volatile memory components for maintenance purposes. EECU software version was L13100_0503.

The following data is expected to be retrieved from the EECU: - Flags: Failures that triggered while the EECU was powered-on

- Limits: Overlimits that were recorded while the EECU was powered-on
- Continuous recording - Channels A and B: Engine data recorded at a sample rate of 1 second

- Context Engine Running - Channels A and B: Engine data and logical words recorded at a sample rate of 20 ms on a limited duration, theoretically from 4s before to 4s after a triggering event (in particular a failure flag), while the engine was running.
- Context Engine Not Running - Channels A and B: Engine data and logical words recorded at a sample rate of 20 ms on a limited duration, theoretically from 4s before to 4s after a triggering event (in particular a failure flag), while the engine was running.
- The EECU records each flight and associates it to a PON (Power-On Number). It uses the number of seconds since the EECU was powered on as a timestamp. This parameter is named POT (Power-On Time).

EECU continuous recording:

- starts when the battery is switched on.
- stops when the collective switch is put in IDLE mode. This design was adopted by SafranHE's engineers to be able to record more flights in its memory.

EECU contextual recordings contain engine parameters, but also logical words that are exchanged between the avionics and the EECU. In particular, two logical words are exchanged between the EECU and the avionics to synchronize the control modes and the regulation status.

G1000H Description

The avionics suite G1000H is composed of two screens, one MFD (left screen) and one PFD (right screen). Both screens contain two SD slots, vertically aligned. The data recorded on the SD cards are:

- For the upper card on the MFD and on the PFD:
- CAS messages, Failures and 1 Hz data in csv files
- For the lower card on the MFD:
- Terrain and navigation databases, CAS messages, Failures and 1 Hz data in a Garmin proprietary format
- For the lower card on the PFD:
- Terrain and navigation databases

Garmin continuous recording:

- starts when the engine starts running
- Garmin equipment is set to record continuous data while the engine is running.
- Garmin time is in UTC time.

For the flight of the event, Garmin continuous recording went on until the impact with the ground.

Recorded Failure

Only relevant failures are described in this section.

The « Low RPM » failure is recorded when the Garmin detects that the Rotor RPM values are too low. This failure is triggered in normal mode when the following conditions are met:

- RPM message: 20% ≤ NR < 97%
- Low RPM aural warning: 80% < NR < 97%

A CAS message is displayed on the instrument panel when this fault is triggered. This CAS message is also recorded by the Garmin.

WORK PERFORMED

Data extraction

EECU

The EECU was in good condition. It was opened and the electronic boards were visually inspected under magnification. They were in good condition. On July 16th 2019, the computer was powered up on the Safran HE test bench in accordance with the BEA/manufacturer procedure.

The following file was downloaded: DUMP_C13165KA00_ECU004495_20190716_115446.dmp

The following data was exported into .csv files:

- Flags - *EXPORT_LGBK_FAIL_C13165KA00_ECU004495_20190716_160802.csv*
- Limits - *EXPORT_LGBK_LIM_C13165KA00_ECU004495_20190716_160823.csv* - Continuous recording - Channels A and B - *EXPORT_LGBK_CONT_A_C13165KA00_ECU004495_20190716_160501.csv* and *EXPORT_LGBK_CONT_B_C13165KA00_ECU004495_20190716_160545.csv*
- Context Engine Running - Channels A and B - *EXPORT_LGBK_TRCRUN_A_C13165KA00_ECU004495_20190716_160649.csv* and *EXPORT_LGBK_TRCRUN_B_C13165KA00_ECU004495_20190716_160621.csv*
- Context Engine Not Running - Channels A and B - *EXPORT_LGBK_TRCNONRUN_A_C13165KA00_ECU004495_20190716_160714.csv* and *EXPORT_LGBK_TRCNONRUN_B_C13165KA00_ECU004495_20190716_160732.csv*

In the EECU data, PON (Power-On Number) 348 was identified as the PON associated with the flight of the event on channel A and B. Channel A was in control for the flight of the event.

Continuous recordings were recorded by both channels of the EECU. Following are the time intervals of recording (given in EECU sample time = POT / 100):

- EECU Channel A: [132.76 – 436.76] s
- EECU Channel B: [130.83 – 436.83] s

The EECU continuous recordings from channel A and from channel B were compared: the recorded values from channel A and from channel B were consistent with each other. Figure 1 below shows an example of this consistency (see Appendix 1 for the comparison of the relevant parameters).

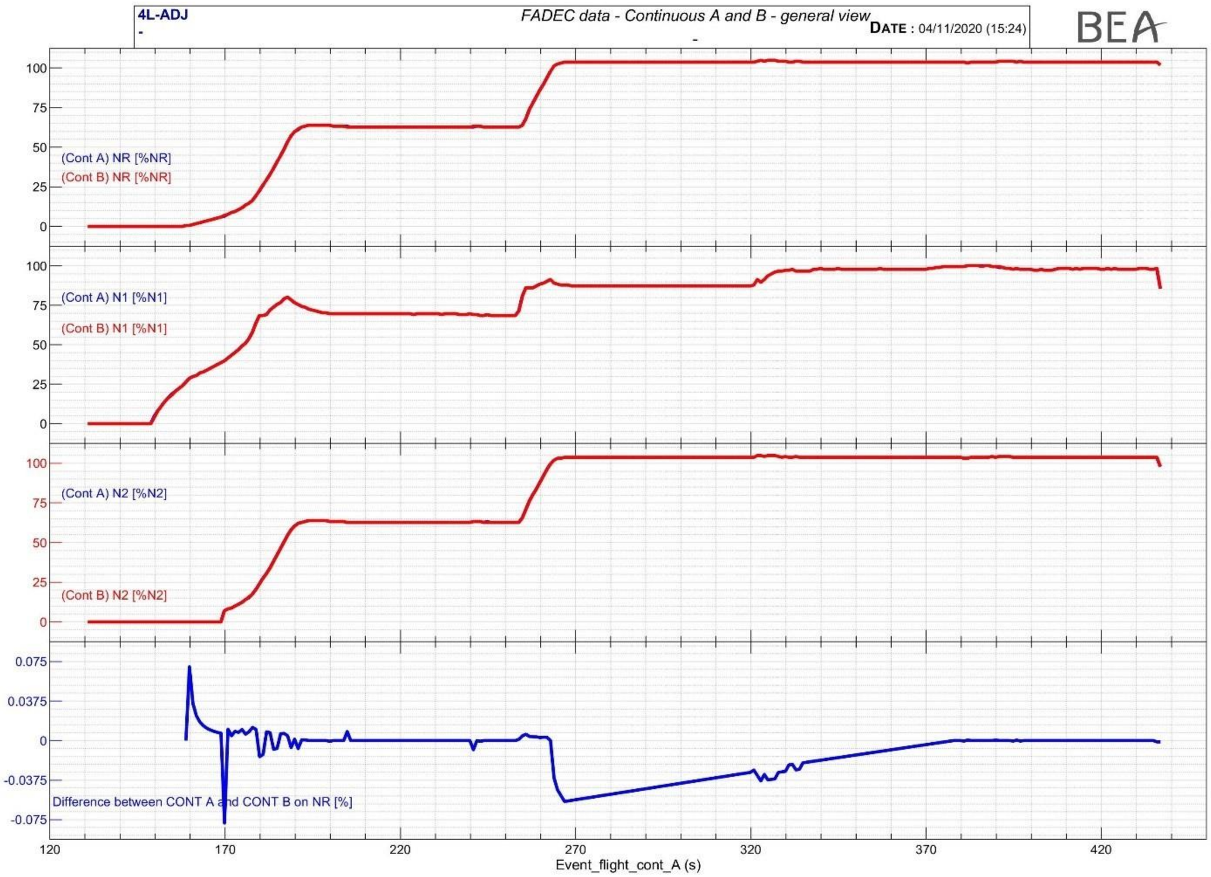


Figure 1: Example of values consistency between channel A and channel B

Contextual recordings for engine running and engine non running, downloaded thanks to the dedicated software, were not consistent. An investigation with Safran HE specialists demonstrated that the most probable cause was an issue with the software that was used to download the data. It was then decided to power-on the EECU again and to download the same data using HyperTerminal. This would allow to gather data in a .txt format without going through the software conversion process.

The download using the HyperTerminal process was done at Safran HE facilities the day after. Four .txt files were gathered:

- NOvram_B_sn4495.TXT
- NOvram_A_sn4495.TXT
- FLASH_B_sn4495.TXT
- FLASH_A_sn4495.TXT

The contextual recordings for engine running of the channels A and B were decoded by Safran HE experts and provided to BEA in two files named *Appendix_7-6_context_records.xlsx* (November,12th2019) for the contexts of the Channel A and *4L-ADJ_Context Record_Channel B.xlsx* (August, 4th 2020) for the contexts of channel B.

Contextual recordings were recorded in six different blocks in channel A:

- [431.9 – 435.9] s

- [445.94 – 449.94] s
- [450.02 – 454.02] s
- [454.1 - 458.1] s
- [459.1 – 463.1] s
- [465.38 – 466.36] s

Contextual recordings were recorded in twenty different blocks in channel B:

- [193.19 – 197.19] s
- [197.27 – 201.27] s
- [201.35 – 205.35] s
- [213.19 – 217.19] s
- [217.27 – 221.27] s
- [377.09 – 381.09] s
- [381.17 – 385.17] s
- [385.25 – 389.25] s
- [431.85 – 435.85] s
- [435.93 – 436.89] s
- [436.97 – 440.97] s
- [441.05 – 445.05] s
- [445.93 – 449.93] s
- [449.97 – 449.97] s
- [450.07 – 450.07] s
- [450.37 – 450.37] s
- [450.01 – 454.01] s
- [455.91 – 457.91] s
- [463.07 – 463.07] s
- [467.47 – 467.47] s

G1000H MFD

The G1000H MFD (left screen) was in good condition. A SD card was present in the lower SD slot and the upper SD slot was empty. The SD card from the lower slot was retrieved and plugged in a computer. The data was copied and sent to Garmin main office in USA for decoding.

The following folders were decoded by Garmin specialists for several flights:

- 1 Hz Log: continuous recording of some engine parameters and some flight data
- CAS History: dated failures that were shown as CAS messages
- Faults: dated failures that were triggered and recorded in the Garmin

The following files were identified as corresponding to the flight of the event:

- 1 Hz Log: *2019-06-06_064851_URMO_7.csv*

- CAS History: *2019-06-06_064851_URMO_8.csv*
- Faults: *2019-06-06_064849_URMO_2.csv*

The time used in the following paragraph is set to be the Garmin time, which is the UTC time. For the flight of the event, the continuous recording was recorded from 06:48:53 to 06:53:49. Relevant consistent parameters were plotted (see Appendix 2). Some failures and some CAS messages were also recorded for the event flight.

G1000H PFD

The G1000H PFD (right screen) was damaged. The screen was broken and the casing was bent. A SD card was present in the lower SD slot and the upper SD slot was empty. The casing was opened and the SD card from the lower slot was retrieved. It was plugged in a computer. The data was copied and sent to Garmin main office in USA for decoding.

No flight logs were recovered from this SD card, which is only a terrain and obstacle database, and no further work was performed on this computer.

At the end of the examinations, all the computers were given back to the IIC. All the data was provided to TAIIB.

Synchronization of the different times

It was decided to use the Garmin time, which is the UTC time, to ease the comparison between engine data and other factual information gathered by TAAIB. The EECU time was thus converted into the Garmin time.

To synchronize the two sets of data, the two increases in NR were taken in both sets of data, as shown in the figure below. These events were synchronized using the following time conversion:

EECU time (s) + 24380 (s) = Garmin UTC time.

This synchronization was validated with the start of the reduction in NR, N1 and N2, which is the third event displayed on the figure below. As the sample rate of the two sets of data is 1s, the synchronization of the two sets of data is precise within 2 seconds.

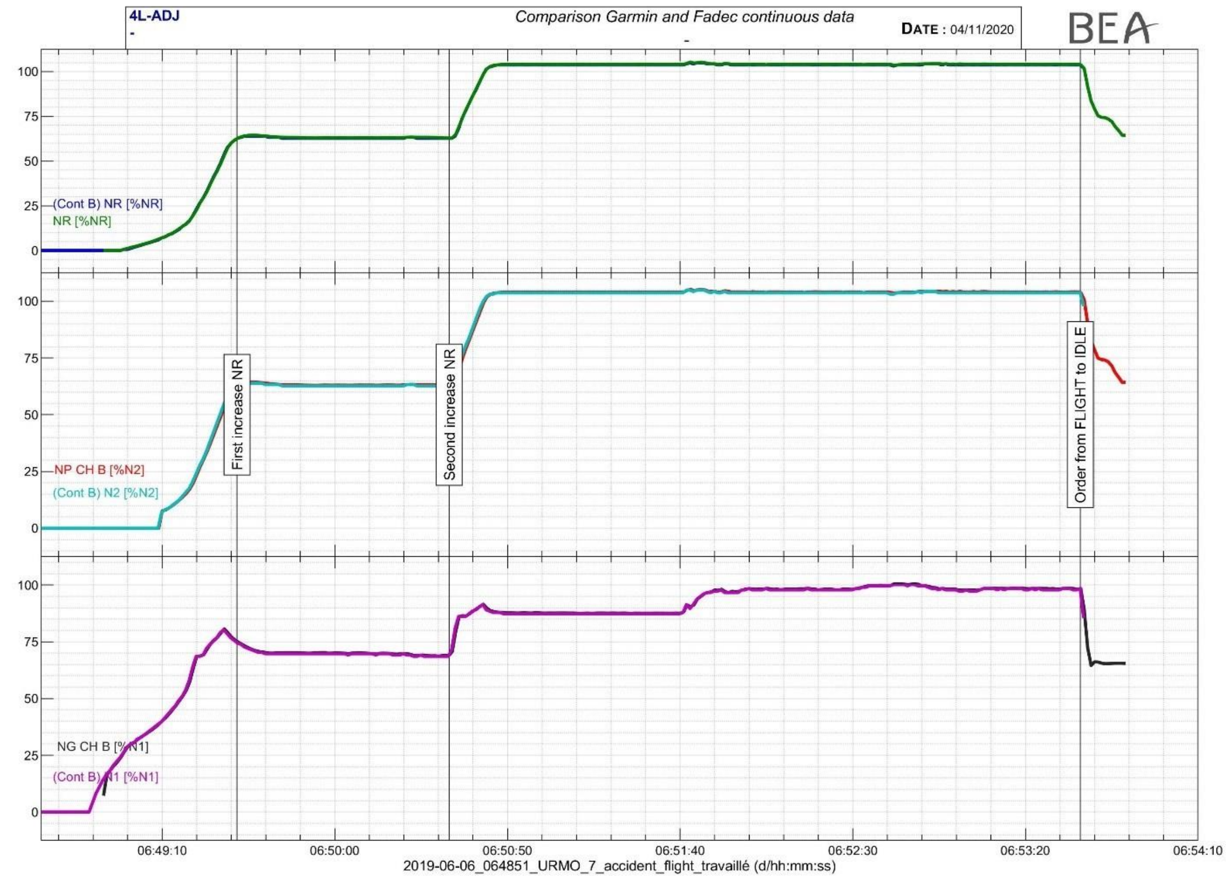


Figure 2: Synchronisation of EECU time and Garmin time

From now on until the end of the report, the time that will be used will be the Garmin UTC time.

ANALYSIS OF THE DATA

Comparison of the different sets of continuous data

After the time conversion, the different time intervals for the continuous sets of data are:

- EECU Channel A: [06:48:32.76 – 06:53:36.76] s
- EECU Channel B: [06:48:30.83 – 06:53:36.83] s
- Garmin 1Hz data: [06:48:53 - 06:53:49] s

It was noticed that the intervals of the different sources of data were not identical. This can be explained by the differences between the recording start and stop logics used by the computers, as described in the system presentation above.

A comparison between these two sets of data (EECU and Garmin) shown that, on common intervals, the data was consistent. Figure below shows some engine parameters data coming from the EECU channel

B and from the Garmin. The difference between NR coming from both sets of data is negligible until the impact with the ground.

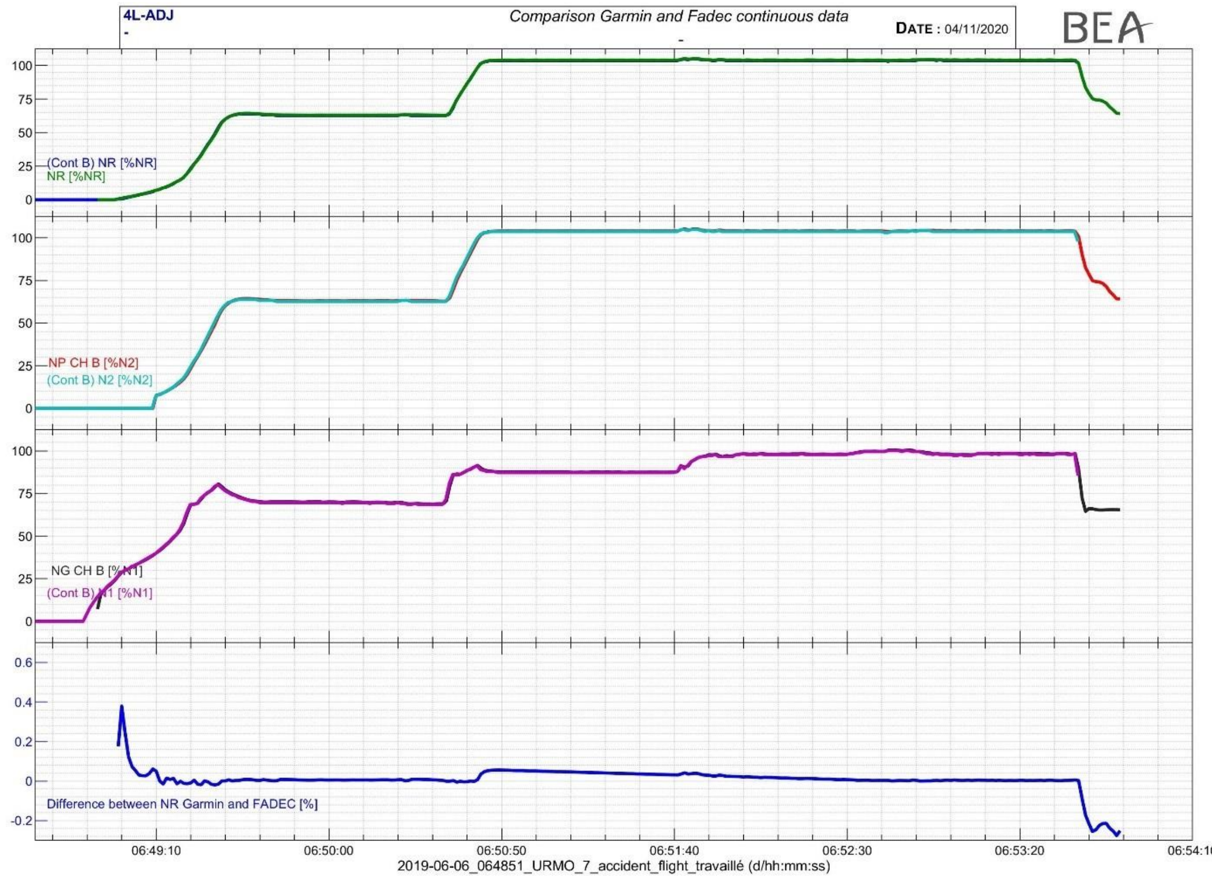


Figure 3: Example of comparison between the Garmin and the EECU recorded data

The comparison between EECU engine data and Garmin engine data is provided in Appendix 3.

From now on until the end of the report, the only continuous data that will be used will be the Garmin data, as the Garmin recorded values until the impact with the ground.

Failures and CAS messages

Failures recorded by the EECU

Following are the failures recorded by both channels of the EECU for the event flight. For most of these failures, several occurrences of the same failure were recorded by each channel. Only the first raise in each block is displayed in the below table.

	Time	Engine running	Channel triggered	Alert name
Block 1	06:46:34	False	A and B	T45 conformation
	06:46:34	False	A and B	TQ conformation
	06:46:34	False	A and B	Other Helicopter data erroneous
	06:46:34	False	A and B	Helicopter conformation data
	06:46:34	False	A	T45 conformation other channel
	06:46:34	False	A	TQ conformation other channel
	06:46:34	False	A	Other Helicopter data erroneous other channel
	06:46:34	False	A	Helicopter conformation data other channel
	06:46:34	False	A	Other Helicopter data erroneous other channel padding
	06:46:34	False	A	Helicopter conformation data other channel padding
Block 2	06:49:10	True	B	T4_1
	06:49:10	True	B	T4_2
Block 3 (from 06:53:50 to 06:54:07)	06:53:50	True	A and B	N2
	06:53:50	True	A and B	28V H/C
	06:53:50	True	A and B	ARINC 429
	06:53:50	True	A and B	XPC engine
	06:53:50	True	B	T4_1
	06:53:50	True	B	T4_2
	06:53:50	True	A and B	Main selector stop / idle / flight
	06:53:50	True	A and B	Inconsistency N2_A/ N2_B
	06:53:50	True	A and B	N2 other channel
	06:53:50	True	A and B	28V H/C other channel
	06:53:50	True	A and B	ARINC 429 other channel
	06:54:03	True	A and B	Fuel pressure / filter outlet
	06:54:07	True	A and B	Fuel pressure / filter inlet

Three blocks of failures can be distinguished:

- A first block of failures at 06:46:34. This block corresponds to initialization failures related to some discrepancies between the avionics and the EECU during their start-up process. Even though this phase design shows congruity issues, according to the manufacturer, these failure messages are to be expected in a normal boot-up
- A second failure block at 06:49:10. This failure block is related to a discrepancy on the T4 impedance measurement between the two EECU channels in stop/run-up phase. This failure is non-critical according to the manufacturer.
- A third block from 06:53:50 to 06:54:07. This block can be associated with the contact with the ground. The recorded failures may be associated with an ARINC cable being cut, the collective stick and the fuel circuit being destroyed, and the main aircraft power circuits being lost.

No significant failure was recorded by the EECU before the impact with the ground for the flight of the event.

Failures recorded by the Garmin

Time	Alert Name	Description	Helicopter status
06:48:49	GEN	Generator offline due to engine start	WOG
06:48:51	ENG OIL PSI LO	CH A Oil Pressure Min Active	WOG
06:48:51	ENG OIL PSI LO	CH B Oil Pressure Min Active	WOG
06:48:51	ENGINE OUT	NG < 50% or failed	WOG
06:48:51	FUEL PRESS	CH A Low Fuel Pressure Active	WOG
06:48:51	FUEL PRESS	CH B Low Fuel Pressure Active	WOG
06:48:51	XMSN OIL PRESS	GIA XOP Low Discrete active	WOG
06:48:51	XMSN OIL PRESS	ECU CH A XOP Low Active	WOG
06:48:51	XMSN OIL PRESS	ECU CH B XOP Low Active	WOG
06:48:57	FUEL PRESS	CH B Low Fuel Pressure Active	WOG
06:48:58	FUEL PRESS	CH A Low Fuel Pressure Active	WOG
06:49:11	ENG OIL PSI LO	CH A Oil Pressure Min Active	WOG
06:49:11	ENG OIL PSI LO	CH B Oil Pressure Min Active	WOG
06:49:11	ECU MAINT	CHA & CHB FAILURE Level 1 ARINC Status Active	WOG
06:49:18	GEN	Generator Fail input to GIA63H active	WOG
06:49:20	RPM	NR outside normal operating parameters	WOG
06:49:20	XMSN OIL PRESS	XSMN Oil Press < 24psi with NR >20% and <80%	WOG
06:49:21	XMSN OIL PRESS	GIA XOP Low Discrete active	WOG
06:49:21	XMSN OIL PRESS	XSMN Oil Press < 24psi with NR >20% and <80%	WOG
06:49:21	XMSN OIL PRESS	ECU CH A XOP Low Active	WOG
06:49:21	XMSN OIL PRESS	ECU CH B XOP Low Active	WOG
06:50:44	RPM	NR outside normal operating parameters	HOVER
06:53:39	RPM	NR outside normal operating parameters	Forward flight

The first block of failures was recorded from 06:48:49 to 06:49:21 when the aircraft was on the ground. These failures are usual warnings saved in the Garmin equipment, which trigger during the power up of the engine.

One failure was recorded in the Garmin during the hover phase, at 06:50:44. This failure is recorded at a height of 5m and is due to a main rotor speed being out of range.

Only one failure is recorded in the Garmin during the forward flight for the flight of the event. This failure is due to a main rotor speed lower than the minimum safe value. It triggered close to the estimated time of the impact with the ground. At 06:53:39, the NR value recorded in the Garmin was 84%.

CAS messages recorded by the Garmin

Time	Alert	Phase of flight
06:48:51	WOG	WOG
06:48:51	ENGINE OUT	WOG
06:48:51	ENGINE OIL PSI LO	WOG
06:48:51	FUEL PRESSURE	WOG
06:48:51	XMSN OIL PRESS	WOG
06:48:51	GEN	WOG
06:48:51	START	WOG
06:49:11	ECU MAINT	WOG
06:49:18	GEN	WOG
06:49:19	OK TO SHUTDOWN	WOG
06:49:20	RPM	WOG
06:49:45	BATT CHARGING	WOG
06:53:39	RPM	FORWARD FLIGHT

The first block of CAS messages was recorded from 06:48:51 to 06:49:45 when the aircraft was on the ground. These messages are usual warnings saved in the Garmin equipment.

Only one CAS message was displayed on the Garmin screen during forward flight for the flight of the event.

This failure is a low RPM failure, and triggered at 06:53:39, close to the estimated time of the impact with the ground.

This CAS message is consistent with the failure that was also recorded by the Garmin at the same time.

CAS messages and failure messages recorded in the Garmin are consistent for the flight of the event.

Conclusion on the analysis of failures and CAS messages

The analysis of the failures recorded in the Garmin and in the EECU showed that:

- No significant failure was recorded before the impact with the ground

- The low RPM alarm triggered at 06:53:39. Its associated CAS message was displayed on the Garmin screen.

Logical words analysis

Some logical words are exchanged between the Garmin and the EECU. These logical words allow the EECU and the avionics to communicate with each other on status reports. They are discrete values. For instance, the status of the engine is one of the information that is sent through this communication channel (STOP, START, FREEZE, INITIATION, RUN UP, IDLE, FLIGHT).

For the flight of the event, two logical words were identified in the contextual recordings of the EECU to gather the status of the system: logical word 1 and logical word 2.

- Logical word 2 represents the translation by the EECU of the orders coming from the avionics through an ARINC link (inputs of the system taken into account by Safran HE).
- Logical word 1 represents the state of the regulation sent back to the avionics (feedback from Safran HE system).

Relevant discrete parameters contained in those two words were plotted in Appendix 4 (Figure 4 in paragraph 3.3.3 is an example of the comparison between discrete parameters coming from the channel A and the channel B). The data recorded by both channels were consistent with each other.

Logical word 2

Logical word 2 represents the translation by the EECU of the order coming from the avionics through an ARINC link. It is the input of the system considered by Safran HE. The logical word is recorded every 20 millisecond.

Embedded in the logical word 2, some binary words are called Discrete Input Numbers (DIN). Two of them represent the requested state of the engine regulation (FLIGHT or IDLE), coming from the avionics. The two DINs are named DIN4 (IDLE order) and DIN10 (FLIGHT order).

The DINs associated with this logical word 2 showed the following facts:

- Before 06:53:35.88, the translation of the order given by the avionics was as follows:
  - FLIGHT mode (DIN10 value at 1)
  - The auxiliary metering unit was in neutral position (DIN14 value at 1)

These inputs from the Garmin are the expected inputs during a nominal forward flight.

- Between 06:53:35.88 and 06:53:35.9, the translation of the order sent by the avionics switched from FLIGHT to IDLE. This change is characterized by two discrete inputs changing within 20 milliseconds:
- o DIN10 switched from 1 to 0
- o DIN4, which characterizes IDLE mode, switched from 0 to 1
- o The auxiliary metering unit remained in a neutral position

At 06:53:50.02, another change was observed:

- DIN4 switched from 1 to 0
- DIN10 remained at 0
- The auxiliary metering unit remained in a neutral position

B	C	D	R	V	AB	AC	AD	AE	AY	AZ	BA	BB	BC
PON	Date ecriture message (s)	POT (s)	- DIN14 - Auxiliary metering unit neutral position	- DIN10 - Flight ()	- DIN4 - Idle ()	- DIN3 (spare) ()	- DIN2 (spare) ()	- DIN1 - Stop ()	- STOP ()	- START ()	- IDLE ()	- FLIGHT ()	- RUN-UP ()
348	435,9	435,8	1	1	0	0	0	0	0	0	0	1	0
348	435,9	435,82	1	1	0	0	0	0	0	0	0	1	0
348	435,9	435,84	1	1	0	0	0	0	0	0	0	1	0
348	435,9	435,86	1	1	0	0	0	0	0	0	0	1	0
348	435,9	435,88	1	1	0	0	0	0	0	0	0	1	0
348	435,9	435,9	1	0	1	0	0	0	0	0	1	0	0
348	449,94	445,94	1	0	1	0	0	0	0	0	1	0	0
348	449,94	445,96	1	0	1	0	0	0	0	0	1	0	0
348	449,94	445,98	1	0	1	0	0	0	0	0	1	0	0
348	449,94	446	1	0	1	0	0	0	0	0	1	0	0
348	449,94	446,02	1	0	1	0	0	0	0	0	1	0	0
348	449,94	446,04	1	0	1	0	0	0	0	0	1	0	0
348	449,94	446,06	1	0	1	0	0	0	0	0	1	0	0
348	449,94	446,08	1	0	1	0	0	0	0	0	1	0	0
348	449,94	446,1	1	0	1	0	0	0	0	0	1	0	0

Logical word 2: Flight and Idle discrete input

Logical word 1: EECU control modes

Logical word 1 represents the state of the regulation that is sent back to the avionics.
The logical word is recorded every 20 millisecond.
The DIN associated with this logical word 1 showed the following facts:

- Before 06:53:35.88, the control status returned to the avionics was as follows:
  - FLIGHT control status (BIT 17 value at 1)
  - The control was in automatic mode
  - Channel A of the EECU was in control
  - Variable N2 datum was active

These control status are consistent with a normal phase of forward flight.

- Between 06:53:35.88 and 06:53:35.9, the control status of the EECU switched from FLIGHT to IDLE. This change was recorded by two discrete inputs changing within 20 milliseconds:
  - BIT17 switched from 1 to 0
  - BIT16, which characterizes the IDLE control mode, switched from 0 to 1
  - Variable N2 datum was active

- Channel A of the EECU was in control
- The control was in automatic mode

- At 06:53:50,02, another change in the control status was observed:
  - BIT16 switched from 1 to 0 o BIT17 remained at 0
  - Engine shutdown authorization switched from 0 to 1

- At 06:53:50.08, on last change was recorded:
  - BIT18, which characterizes the RUN-UP control mode, switched from 0 to 1

B	C	D	R	V	AB	AC	AD	AE	AY	AZ	BA	BB	BC
PON	Date ecriture message (s)	POT (s)	- DIN14 - Auxiliary metering unit neutral position	- DIN10 - Flight ()	- DIN4 - Idle ()	- DIN3 (spare) ()	- DIN2 (spare) ()	- DIN1 - Stop ()	- STOP ()	- START ()	- IDLE ()	- FLIGHT ()	- RUN-UP ()
348	435,9	435,8	1	1	0	0	0	0	0	0	0	1	0
348	435,9	435,82	1	1	0	0	0	0	0	0	0	1	0
348	435,9	435,84	1	1	0	0	0	0	0	0	0	1	0
348	435,9	435,86	1	1	0	0	0	0	0	0	0	1	0
348	435,9	435,88	1	1	0	0	0	0	0	0	0	1	0
348	435,9	435,9	1	0	1	0	0	0	0	0	1	0	0
348	449,94	445,94	1	0	1	0	0	0	0	0	1	0	0
348	449,94	445,96	1	0	1	0	0	0	0	0	1	0	0
348	449,94	445,98	1	0	1	0	0	0	0	0	1	0	0
348	449,94	446	1	0	1	0	0	0	0	0	1	0	0
348	449,94	446,02	1	0	1	0	0	0	0	0	1	0	0
348	449,94	446,04	1	0	1	0	0	0	0	0	1	0	0
348	449,94	446,06	1	0	1	0	0	0	0	0	1	0	0
348	449,94	446,08	1	0	1	0	0	0	0	0	1	0	0
348	449,94	446,1	1	0	1	0	0	0	0	0	1	0	0

Logical word 1: Flight and Idle control mode

Analysis of the logical words

The analysis of the information retrieved in logical words 1 and 2 for channel A showed that:

- An order from FLIGHT to IDLE was taken into account by the EECU at 06:53:35.88
  - A control status from FLIGHT to IDLE was returned to the avionics at 06:53:35.9
  - The timeframe between the reception of the order and the update of the control status was 20 ms, which is within the manufacturer's tolerance.

At 06:53:50.02, logical word 1 switched from IDLE to RUN-UP, and an engine shut-off authorization was sent to the avionics. This authorization informs the pilot that conditions are met to shut the engine down.

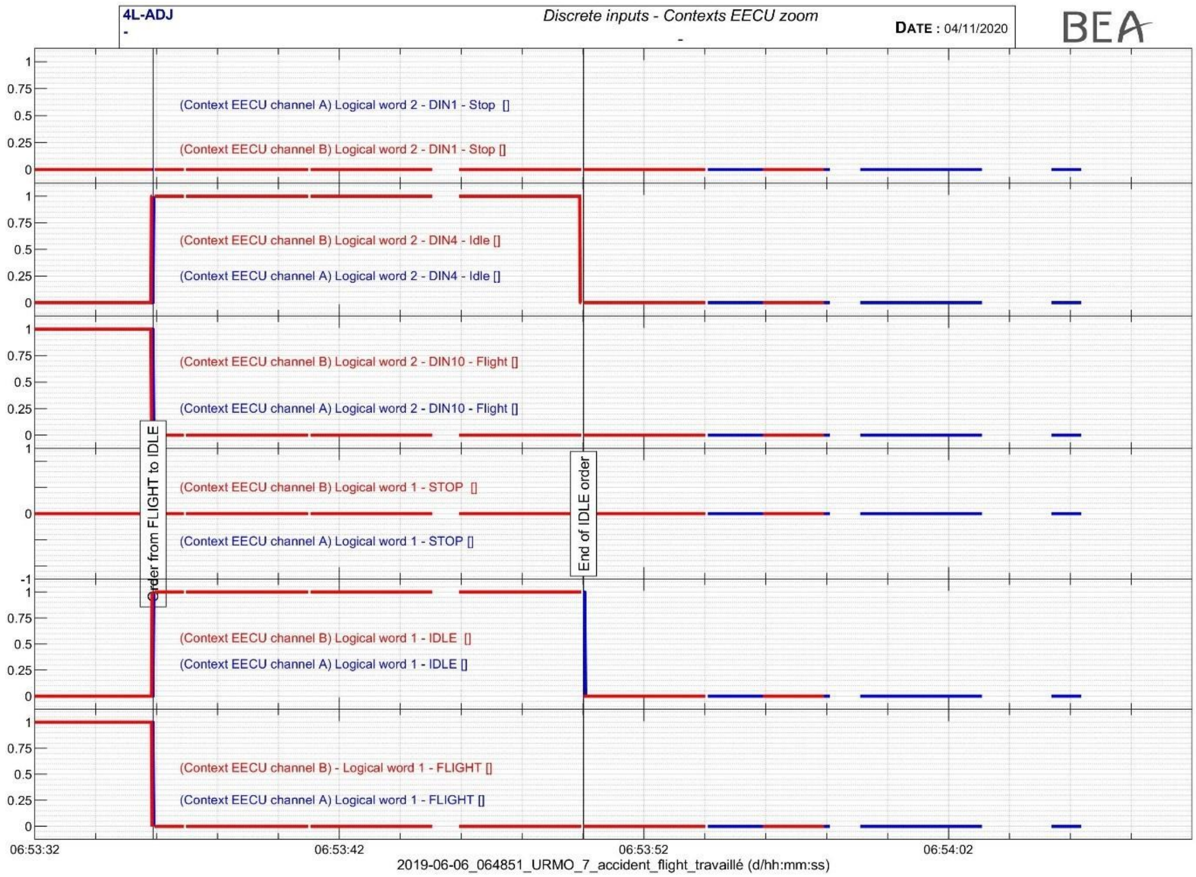


Figure 4: example of discrete inputs comparison

An internal verification is made by the EECU every 20 ms to check if the input and the output data are consistent. If a discrepancy is detected (for instance if two orders FLIGHT and IDLE are translated by the EECU at the same time in logical word 2), a flag is raised and a failure is recorded in the EECU. By default, if both orders are received within 30 ms by the EECU, the EECU regulation mode will stay in FLIGHT mode.

No failure of this type was recorded for the flight of the event.

Contextual recordings analysis

Contextual recordings analysis - Channel A

After the time conversion, the different time intervals for the contextual recordings recorded in the EECU were, for channel A:

- Context 1: [06:53:31.9 – 06:53:35.9] s
- Context 2: [06:53:45.94 – 06:53:49.94] s
- Context 3: [06:53:50.02 – 06:53:54.02] s
- Context 4: [06:53:54.1 - 06:53:58.1] s
- Context 5: [06:53:59.1 – 06:54:03.1] s

- Context 6: [06:54:05.38 – 06:54:06.36] s

Context number	Triggering cause	Description of the recording
Context 1	Switch from flight to idle	According to the manufacturer, this context is recorded anytime a switch is made from flight to idle, even without any failure
Context 2	May be associated to the group failures due to the impact with the ground	This context is related to some failures and has a duration of 4 seconds
Context 3	Associated to the group failures due to the impact with the ground	This context is related to some failures and has a duration of 4 seconds
Context 4	Associated to the group failures due to the impact with the ground	This context is related to some failures and has a duration of 4 seconds
Context 5	Associated to the group failures due to the impact with the ground	This context is related to some failures and has a duration of 4 seconds
Context 6	Associated to the group failures due to the impact with the ground	This context is related to some failures. It was noted that the context only had a duration of 98 ms

Contextual recordings analysis - Channel B

After the time conversion, the different time intervals for the contextual recordings recorded in the EECU were, for channel B:

- Context 1: [06:49:33.19 – 06:49:37.19] s
- Context 2: [06:49:37.27 – 06:49:41.27] s
- Context 3: [06:49:41.35 – 06:49:45.35] s
- Context 4: [06:49:53.19 - 06:49:57.19] s
- Context 5: [06:49:57.27 – 06:50:01.27] s
- Context 6: [06:52:37.09 – 06:52:41.09] s
- Context 7: [06:52:41.17 – 06:52:45.17] s
- Context 8: [06:52:45.25 – 06:52:49.25] s
- Context 9: [06:53:31.85 – 06:53:35.85] s
- Context 10: [06:53:35.93 - 06:53:36.89] s
- Context 11: [06:53:36.97 – 06:53:40.97] s
- Context 12: [06:53:41.05 – 06:53:45.05] s

- Context 13: [06:53:45.93 – 06:53:49.93] s
- Context 14: [06:53:49.97] s
- Context 15: [06:53:50.07] s
- Context 16: [06:53:50.37] s
- Context 17: [06:53:50.01 – 06:53:54.01] s
- Context 18: [06:53:55.91 – 06:53:57.91] s
- Context 19: [06:54:03.07] s
- Context 20: [06:54:07.47] s

Context number	**Triggering cause**	**Explanation of the recording**
Context 1	Unknown	This context wasn't associated to any failure recorded by the EECU.
Context 2	Unknown	This context wasn't associated to any failure recorded by the EECU.
Context 3	Unknown	This context wasn't associated to any failure recorded by the EECU.
Context 4	Unknown	This context wasn't associated to any failure recorded by the EECU.
Context 5	Unknown	This context wasn't associated to any failure recorded by the EECU.
Context 6	Unknown	This context wasn't associated to any failure recorded by the EECU.
Context 7	Unknown	This context wasn't associated to any failure recorded by the EECU.
Context 8	Unknown	This context wasn't associated to any failure recorded by the EECU.
Context 9	Switch from flight to idle	According to SafranHE engineers, this context is recorded anytime a switch is made from flight to idle, even without any failure

Context 10	Unknown	This context wasn't associated to any failure recorded by the EECU.
Context 11	Unknown	This context wasn't associated to any failure recorded by the EECU.
Context 12	Unknown	This context wasn't associated to any failure recorded by the EECU.
Context 13	Unknown	This context wasn't associated to any failure recorded by the EECU.
Context 14	Associated to the group failures due to the impact with the ground	This context is related to some failures. It was not determined why the context had a duration of less than 20 ms
Context 15	Associated to the group failures due to the impact with the ground	This context is related to some failures. It was noted that the context had a duration of less than 20 ms
Context 16	Associated to the group failures due to the impact with the ground	This context is related to some failures It was noted that the context had a duration of less than 20 ms
Context 17	Associated to the group failures due to the impact with the ground	This context is related to some failures and has a duration of 4 seconds
Context 18	Associated to the group failures due to the impact with the ground	This context is related to some failures. It was noted that the context only had a duration of 2 s
Context 19	Associated to the group failures due to the impact with the ground	This context is related to some failures. It was noted that the context had a duration of less than 20 ms
Context 20	Associated to the group failures due to the impact with the ground	This context is related to some failures. It was noted that the context had a duration of less than 20 ms

Conclusion on the contextual recordings

The following figure shows some relevant parameters for both channels A and B.

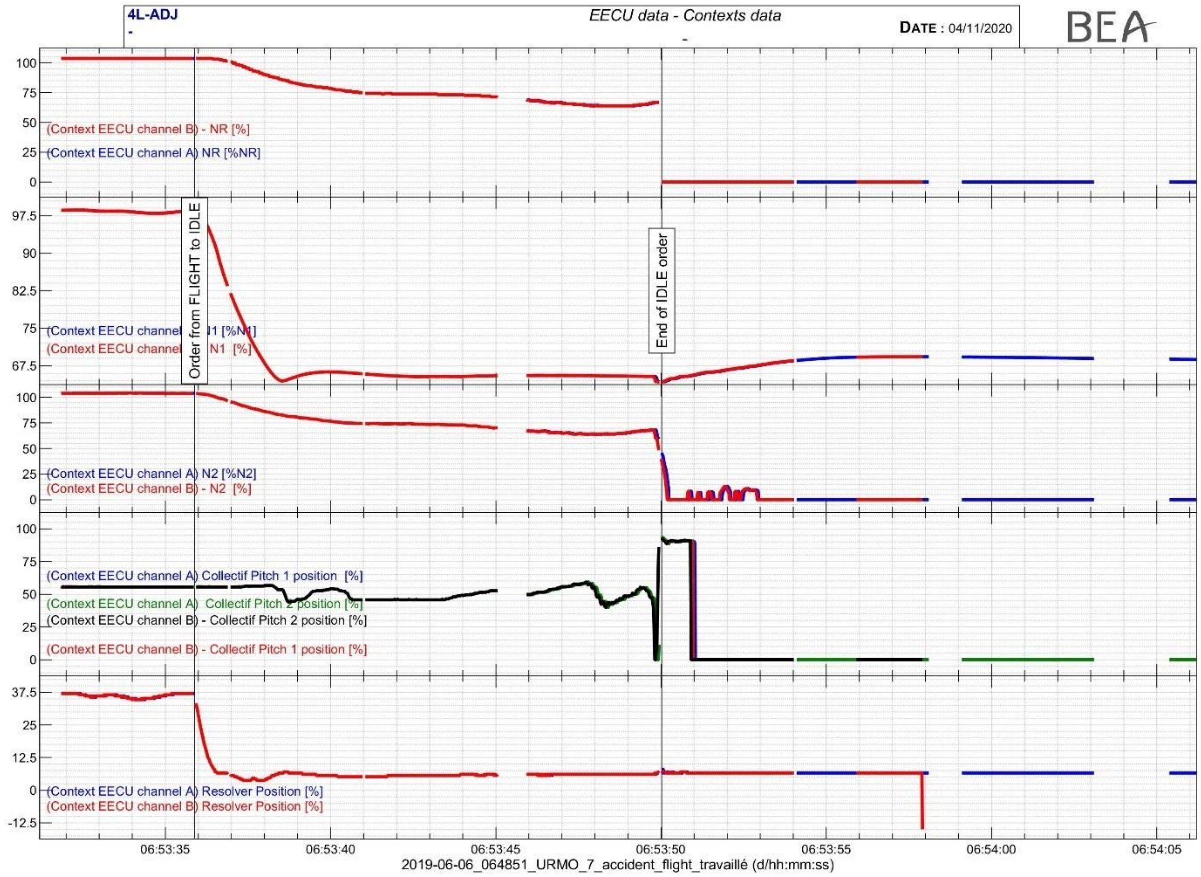


Figure 5 : Contextual data for both channels of the EECU

Reminder At around 06:53:35.9, the logical word 2 switched from FLIGHT to IDLE and the logical word 1 acknowledged this change within 20 ms.

On the figure above, it can be seen that directly following the order from FLIGHT to IDLE seen in the logical words, N1 recorded values decreased to reach the IDLE nominal value of 65%. This decrease corresponds to a standard N1 decrease in case of an IDLE order. This shows that the IDLE order was taken into account by the engine. The rest of the engine parameters followed the IDLE order (decrease in N2, T4, P3).

It was noticed that the contextual recordings for channel A and channel B were recorded on different time intervals. Only two contexts can be matched between the two channels:

- Context 1 of channel A can be matched to context 9 of channel B
- Context 3 of channel A can be matched to context 17 of channel B

It was also noticed that some contextual recordings were not associated with failures recorded in the EECU in channel B.

The following discrepancies were noticed in these recordings:

- Some contexts were recorded in channel B without any associated failure
- Some contexts, following the impact with the ground, had an abnormal duration
- Most parameters associated to the contexts were not recorded for channel B.

Relevant plots are available in Appendix 5 showing the different contextual recordings and the associated discrete information.

Conclusion on the engine

During the flight of the event, the EECU recorded an order to switch the regulation from FLIGHT to IDLE, at 06:53:35.88.

The EECU answered accordingly to the order (acknowledgment sent to the avionics 20 ms later).

The recorded engine parameters showed an engine responding to the IDLE order accordingly.
Despite some unexplained behavior in the contextual recordings of both channels of the EECU, the examination didn't show any evidence of engine failure.
No clue of any engine failure was detected inside the recorded engine parameters.

According to the engine parameters displayed in Chart 1:

- engine parameters were consistent with an engine operation from start to idle, and from idle to flight where the N2 rating is 104%.
- at 284 seconds after the engine started, the N1, N2, NR suddenly started to drop. This statement was in accordance with the change of state from flight to idle mode recorded in the logical word 2

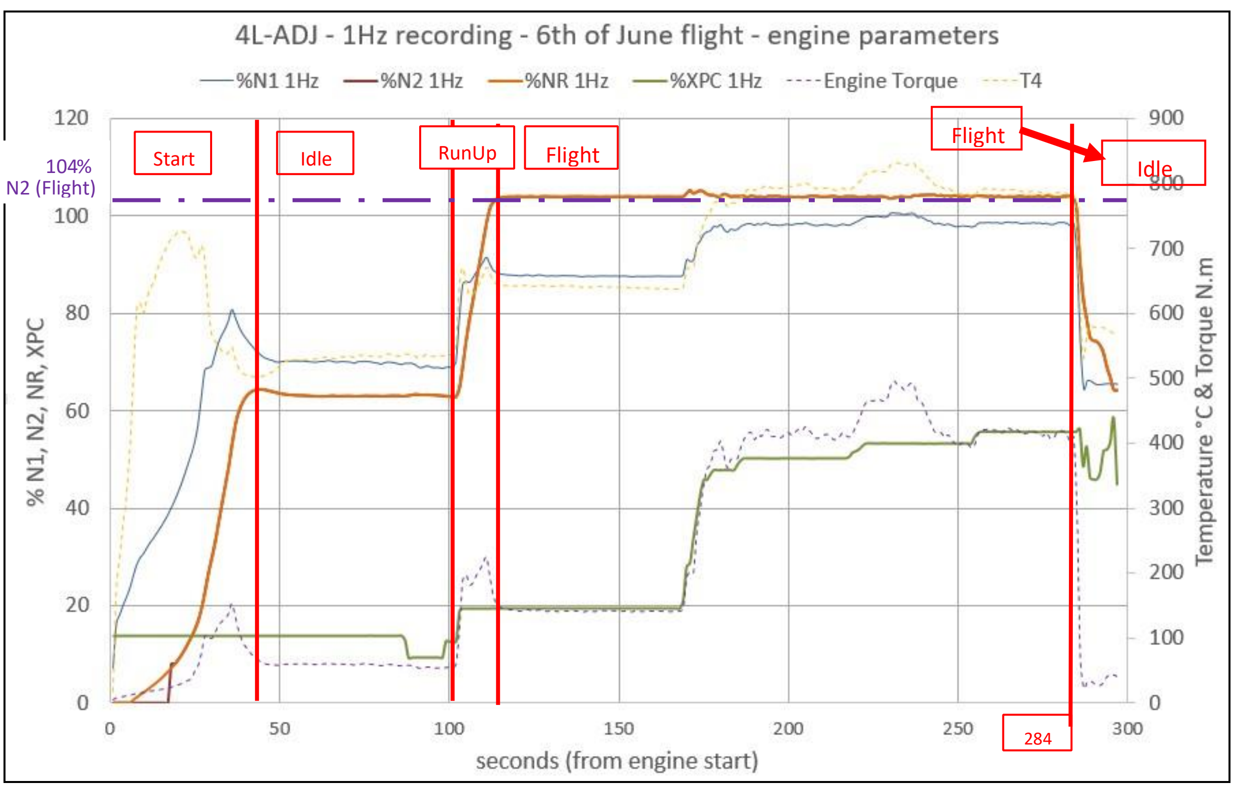
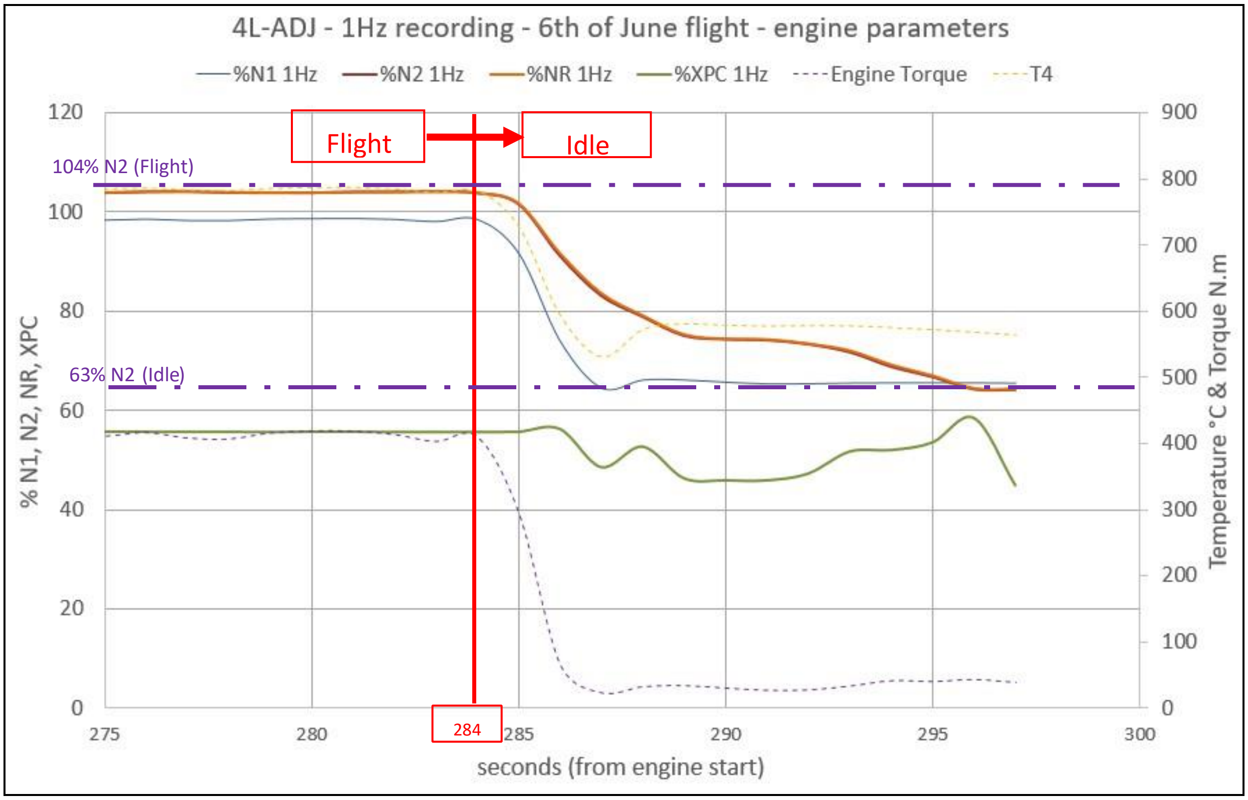


Chart 1 : **Engine parameters – 1Hz – from engine start to end of recording**

Following engine parameters displayed in chart 2, from 284s after engine start :

- all engines parameters were consistent with an idle mode rating (Chart 2; N2(idle) = 63%) ⇒ NR and N2 were synchronized.
- XPC (collective pitch position) was always higher than 46%.



<u>Chart 2 :</u> Engine parameters – 1Hz – zoom on the last 25 seconds of continuous recording

Conclusion on the helicopter debris

Examination of the helicopter debris, main rotor, tail rotor, transmission, landing gear, and engine modules at the scene of the accident revealed the nature of the damages:

- There was no collapse of the helicopter structure in the air during the flight.

- The structure of the helicopter collapsed as a result of a collision with the ground.

Flight analysis

As shown by the helicopter 4L-ADJ electronic engine control unit EECU and navigation control device GARMIN internal memory card data records, also the recordings from the CCTV cameras located on the base “Kazbegi Helicopters”, also the video footage, which was taken by the mobile phones of the eye-witnesses shows, that at 06:06:09 2019, the pilot of the helicopter, before the start of the flight, performs standard pre-flight inspection and preparation of the helicopter, after which to obtain a flight permition, he by mobile phone made comunication to the relevant service of “Sakaeronavigatsia” Ltd and obtains a flight permition. (Evidence - Extract -Pilot - “Sakaeronavigatsia” Ltd).

At 10:52:40 am, the commander of the helicopter with two passengers, took off by the helicopter 4L- ADJ from the concrete helipad, arranged in front of the hangar of the “Kazbegi helicopter” base.

The flight to an altitude of 170 meters was carried out in vertically take-off mode.

At an altitude of 170 meters, the helicopter enters the horizontal flight mode, starts developing speed, and heads in the direction of the pre-planned helipad of the hotel "Rooms".

06:53:36 AM, the flight altitude was 170 meters, the flight speed was 104 knt, the flight was without remarks.

At 06:53:36, the engine switches from "FLY" (flight) mode to "IDLE" (lowest engine operation) mode and begins a rapid decent, which at 06:53:50 ends with a collision with the ground .

As the decryption records of the above flight parameters have shown, at a given critical moment in the flight, time 06:53:36, when the collective position (XPC) was 56%, the engine was switched from FLY mode to IDLE mode, then in 3 seconds 06:53:39 the collective (XPC) was lowered to 46%, then in 2 seconds 6:53:41 the collective (XPC) was raised to 54%, Then in 2 seconds 6:53:43 the collective (XPC) was lowered to 47%, then rising in every second, in 06:53:45 collective (XPC) was 56% and 06:53:47 (XPC) was 60% and 06:53:47 (XPC) was 46% .

At this moment, the lowering of the collective pitch lever (XPC) all the way down, which is a necessary condition for maintaining autorotation, actually did not occur.

From this critical moment of time 06:53:36, the aerodynamic characteristics of the helicopter rotor deteriorated, therefore the helicopter main rotor rpm (Nr) began to decrease while remaining coupled to the power turbine, decrease the number of engine torque, starts decrease the lift therefore increase in the vertical speed of descent of the helicopter, after which it became impossible to control the flight, Maintain altitude and direction of the helicopter and perform a safe landing.

At the time of impact, the main rotor rpm had decreased to approximately 65%, which is not enough rotor rpm to arrest descent with full application of collective. (See Chart 1, which shows select data from the Garmin 1 Hz data log download file).

Because the flight was carried out in uncontrolled airspace as a rule, when flying in this zone, the pilot is not in contact with the air traffic controller. The pilot must send information to the central dispatcher only the flight region, the beginning and finishing of the flight.

Accordingly, during a particular flight (the time from the development of the critical situation to the collision with the ground was 13 seconds), the pilot had no opportunity to inform anyone of the reason for the forced decent.

As the dynamics of the flight show, In the present case, the pilot most likely performed the procedures for an immediate forced descent in autorotation mode. To enter the autorotation mode, the pilot must immediately lower the throttle lever all the way down. There is no evidence of this in the existing records.

1.18 safety measures taken by Safran Helicopter Engines since the accident

The 4L-ADJ EECU was set with software version V501. The software version V601 (modification Tf91) is already implemented in new production and available for retrofit, and includes a continuous recording of parameters into the EECU, from the engine start (exactly from N1>7%) to the engine stop (exactly to N1<5% or, if N1 is not valid, to N2<5%), whatever the engine mode that is selected (IDLE, FLIGHT, STOP). In January 2021, 91% of the fleet is equipped with V601."

3. Conclusions

Findings:

1. The Certificate of Registration, Certificate of Airworthiness & The air operator certificate was valid on the date of accident.

2. The helicopter and Engines were being maintained under continuous maintenance program approved by GCAA.

3. The PIC was having valid ATPL issued by GCAA, his medical license was valid.

4. The Pilot total flying experience was 1413 hrs out of which 85 hrs were on Bell 505 helicopter PIC.

5. The helicopter was controlled by the pilot from the pilot's right seat.

6. The controls installed on the left side of the helicopter (steering direction pedals, cyclic control stick, collective pitch lever) have been removed.

7. Based on the data and information available it was established that the pilot was familiar with the area as he had carried out many flights.

8. The helicopter Weight and balance were within prescribed limits.

9.The PIC had taken adequate rest prior to operating the flight.

10. Weather conditions was fine at the time of accident and hence weather was not a contributory factor to the accident

11. The handle of the main valve for emergency shutdown of the fuel supply to the helicopter engine in the cockpit was found in the closed position. It is possible that the pilot felt the inevitability of a collision with the ground, which could be followed by a fire, and just before the collision with the ground, that he closed the fuel supply to the engine to prevent a fire. (Constructively, after closing the valve, the engine is still supplied with fuel remaining in the fuel lines for 15 seconds, and continues to work, as evidenced by the decoded records.)

The fuel shut off valve may also have closed the fuel inlet during the crash, at impact. The cable that goes from the cockpit activator to the valve may have been put into tension due to structural warping of the airframe, and therefore closed the valve without manual action from the crew.”

12. A medical chemical-toxicological examination of the bodies of the pilot show, that “Methyl, ethyl, isopropyl alcohols were not found in the blood taken from the corpse of the helicopter pilot. No drugs or psychotropic substances were found in the blood and internal organs.

Cause of death was combined blunt trauma to the body with damage to skeletal bones and internal organs. The pilot died as a result of injuries sustained as a result of a helicopter crash.

13. A medical chemical-toxicological examination of the bodies of the passenger sitting to the left of a helicopter pilot show, that “Carbamazepine anticonvulsant was found in the blood and internal organs taken from the corpse of a passenger sitting to the left of a helicopter pilot. Methyl, ethyl, isopropyl alcohols were not found in the blood. No drugs or psychotropic substances were found in the blood and internal organs. The medical examination cannot explain whether the convulsant seizure with passenger took place in the flight.

Cause of death: combined blunt trauma to the body with damage to skeletal bones and internal organs.

14. A medical chemical-toxicological examination of the bodies of the passenger sitting in the back of the helicopter show, that “Methyl, ethyl, isopropyl alcohols were not found in the blood taken from the corpse. No drugs or psychotropic substances were found in the blood and internal organs.

Cause of death: combined blunt trauma to the body with damage to skeletal bones and internal organs.

15. DNA testing of the helicopter's engine collective lever shows, that on the broken parts of the collective lever's "fly-adle" switch was found only pilot's DNA.

16. The fly/idle switch on the collective stick that controls the engine throttle was tested for electrical continuity. The switch operated as required.

17. There was no evidence to indicate there were any issues with the LIVE mount assemblies or any maintenance discrepancies that would have limited its functionality.

18. Engines front bearing of the HMU’s intermediate gear was found damaged. By Safran explaining the loss of 7 out of 8 tabs is part of the known in-service experience of Arrius 2R, which allowed to establish that this damage did not alter the engine operation. (see attachment 1. „Safran Helicopter Engines“presentation)

19. According to the module M01 examination, no relation can be made between the accident sequence and the loss of front bearing tabs of the HMU intermediate gear.

20. EECU continuous recording: starts when the battery is switched on and stops when the collective switch is put in IDLE mode. This design was adopted by Safran HE's engineers to be able to record more flights in its memory.

21. Helicopter Engine, Arrius 2R S/N 50065, equipped with EECU P/N 70EMS01010 S/N 4495, revealed that during June 6th 2019 flight, the engine went from flight to idle following a dedicated input change of the EECU. The engine was operating as expected during the whole accident flight.

The aviation event, that occurred on June 6, 2019 in "Kazbegi" Municipality of Georgia, on the helicopter BELL-505, state registration number 4L-ADJ, classified as an aviation accident.

The work carried out during the investigation - examinations and studies showed, that the Garmin and the EECU recorded data can not explain the reason for switching the switch, on the collective pitch lever from the flight position to the idle position and the reason of the sudden start of the helicopter's descent, during which the lowering of the collective pitch lever (XPC) all the way down, which is a necessary condition for maintaining autorotation for performing a safe landing, practically does not occur and finished with collided of helicopter with ground.

Because the helicopter BELL-505, with state registration mark 4L-ADJ, the installation of flight data recorder (FDR) and cockpit voice recorder (CVR) on-board recording devices is not constructively considered, It is impossible to accurately describe the critical situation that developed in the cockpit during a given flight.

4. Safety recommendations

In order to improve flight safety and prevent future aviation accidents and reduce risks, the final aviation accident investigation report should be sent to:

- Georgian Civil Aviation Agency;
- Safety Investigation Bureau (BEA);
- Helicopter engine manufacturer "Safran Helicopter Engines"
- Canadian Transportation Safety Bureau (TSB)
- Helicopter manufacturer "Bell Helicopter"

addressers should ensure:

The Georgian Civil Aviation Agency :

1. Enhanced control over helicopter operators.

2. Consider adopting additional regulation in the country:

- Review requirements for helicopters operators to operate with two pilots in mountainous regions;

- Review the issue of pilots' admission to flights in mountainous areas (taking into account the flying hours on a specific type of helicopter as a commander and flight experience).

Helicopter engine manufacturer "Safran Helicopter Engines":

1. despite the fact, that during the investigation of the engine, the 4 metal tabs, from engines front bearing of the HMU's intermediate gear, found on the magnetic plug, are not a contributing factor to the accident, please consider improving the design of electrical magnetic plugs of the engine – "Arrius 2R", which will gives result – chip CAS message by the avionic, even in presence of small parts of metal shavings.

Helicopter manufacturer Bell Helicopter:

1. Consider installing a video recorder in the cockpit of a helicopter;

2. Consider issue of installing a transparent partition between the pilot seats of the helicopter cockpit, which will be used in the presence of a passenger in the seat next to the pilot.

Note: An accredited representative of the French "Bureau of Enquetes et D'Analyses", does not agree with the recommendation made by the Georgian Investigation Bureau, to the helicopter engine manufacturer "Safran Helicopter Engines" to improve the design of the Arrius 2R electromagnetic plugs of the engine.

According of ICAO Annex 13 ch. 6.3. the text submitted by (BEA) is attached to the final report as an addition. (See Appendix 1. Original in English).

Head of Civil Aviation and

Maritime Transport Accident

Incident Investigation Bureau

David Giunashvili

Appendix 1. The text submitted by (BEA)

Electrical Magnetic Plugs design

The Electrical Magnetic Plugs are designed to detect metal particles that are generated by the engine and that are drained into the oil system. The size of a particle that can reach the Electrical Magnetic Plug is limited because of the strainer which is located upstream in the oil circuit. Moreover, the functional specification for this kind of design is not tied to the metallic article size, but rather to the magnetic strength as well as as the air gap length.

The Arrius 2R Electrical Magnetic Plug is of the same technology as the whole Arrius 2 fleet electrical magnetic plugs, which global in-service experience show that no Uncommanded In Flight Shut Down ever occurred following an accessory gear bow bearing failure. All accessory gear bearings or gears failure have systematically led to chip detection.

In the case of a loss of a tab from an accessory gear box bearing, combined with no chip light early warning, it was demonstrated with engine bench tests and analysis (and shared with EASA) that :

- The loss of all tabs is necessary to initiate a bearing ball contact,
- This ball contact is generating a flaking wear which is very slow (more than 50 running hours),
- This flacking will lead to a scattering of metallic particles into the oil system that would eventually be detected by the Electrical Magnetic Plug.

In 2019, SafranHE worked and designed a modification “Tf90” (SB_319_72_4090) of the forward HMU shaft bearing which replaced the “stamped” design by a “riveted” design to avoid tabs to be lost by the separator cage (hence, that also avoids the tabs to go into the oil system).

Additionally, in 2021, a complementary modification “Tf96” will be implemented to replace all stamped type bearings into the accessory gear box of Arrius 2R by riveted design.

As a summary, the following paragraph of the draft Final Report shall be reviewed and amended taking into account the above explanations:

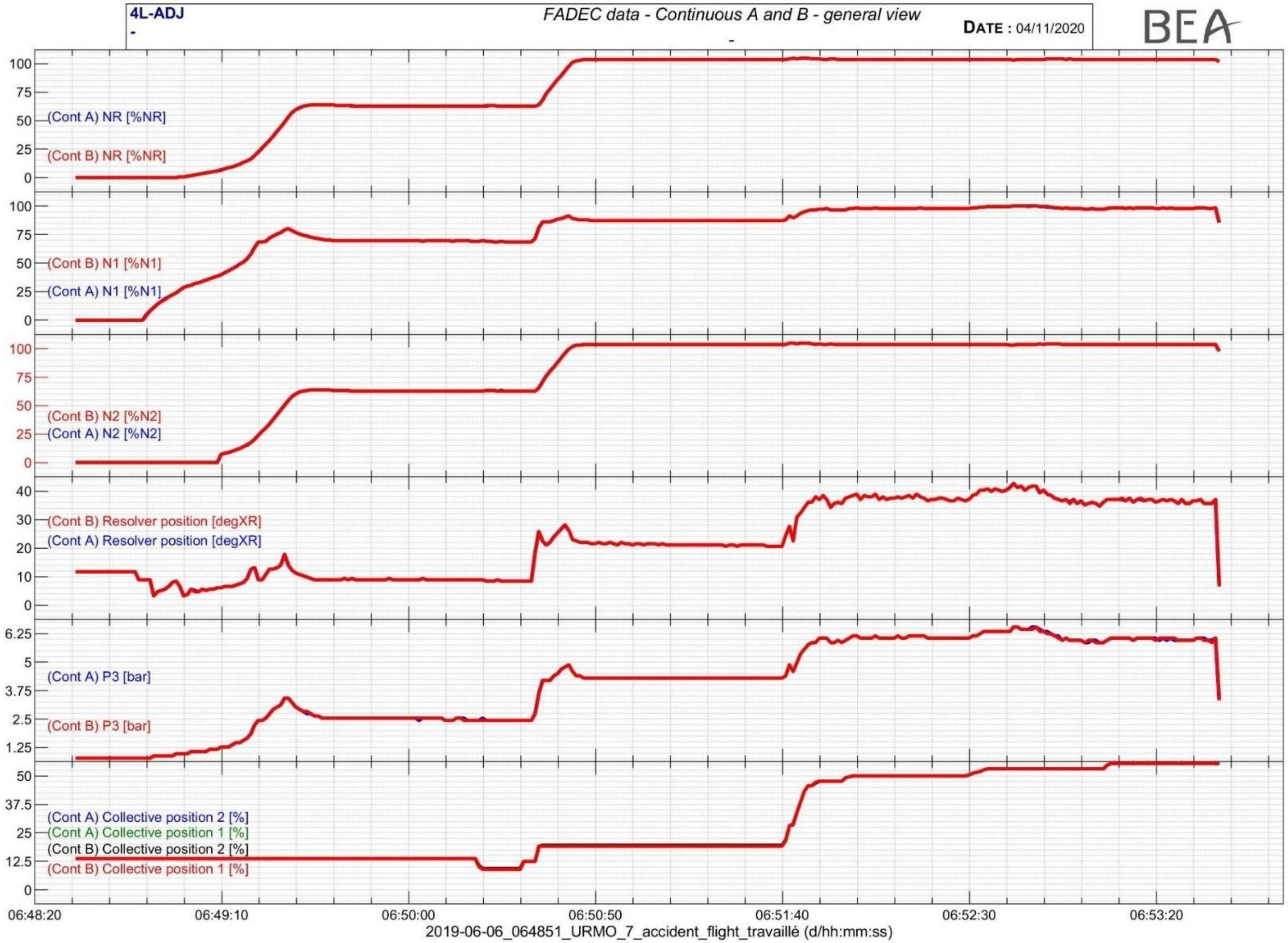
- Safety recommendations

“To change design of Electrical Magnetic Plugs of the engine, which will gives result - chip CAS message by the avionic even in the presence of small parts of metal shavings.”

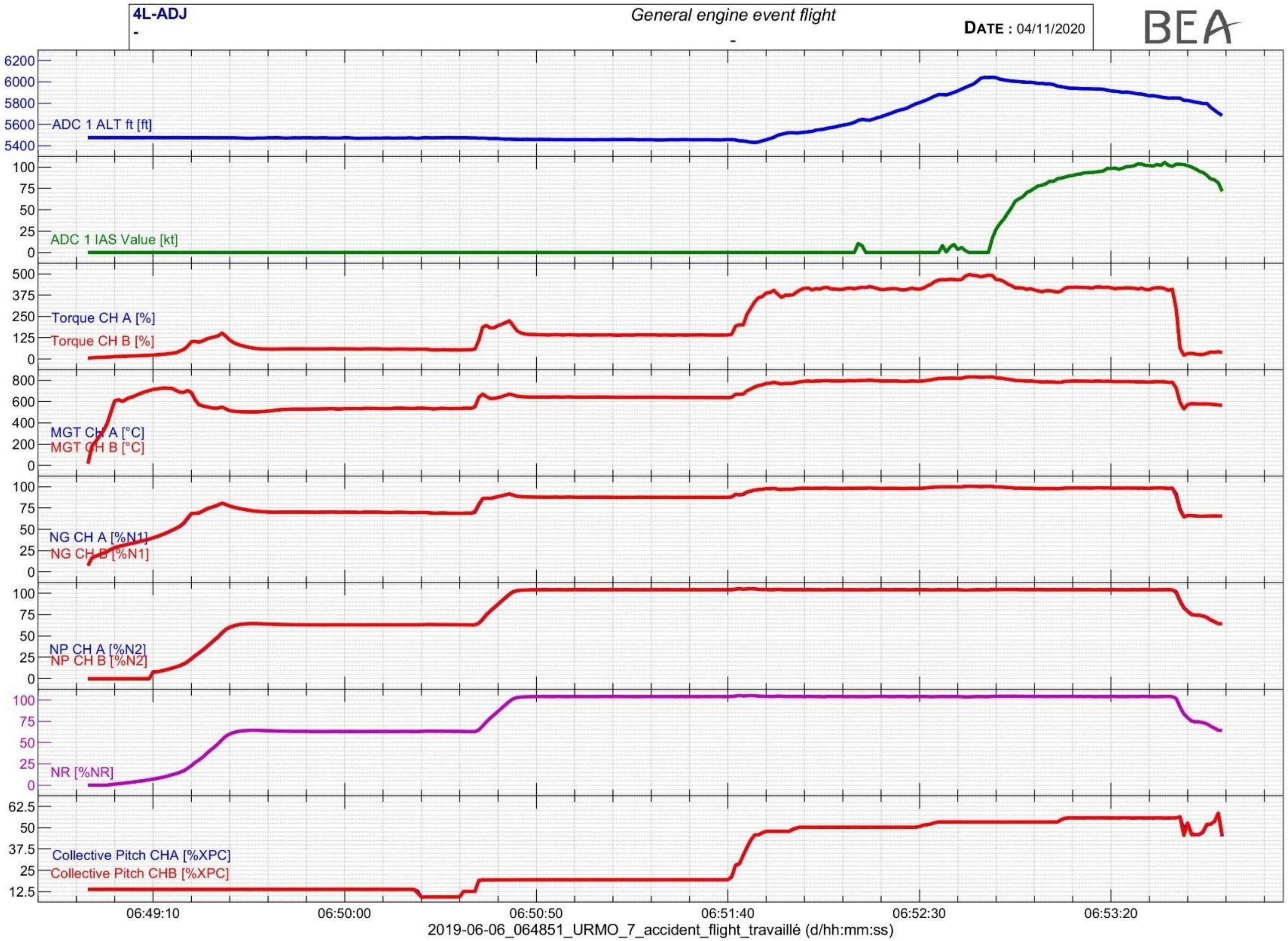
The BEA proposes to remove this recommendation since the present design of the magnetic plugs allows to detect small metallic particles and the loss of the tabs was not a contributing factor to the accident.

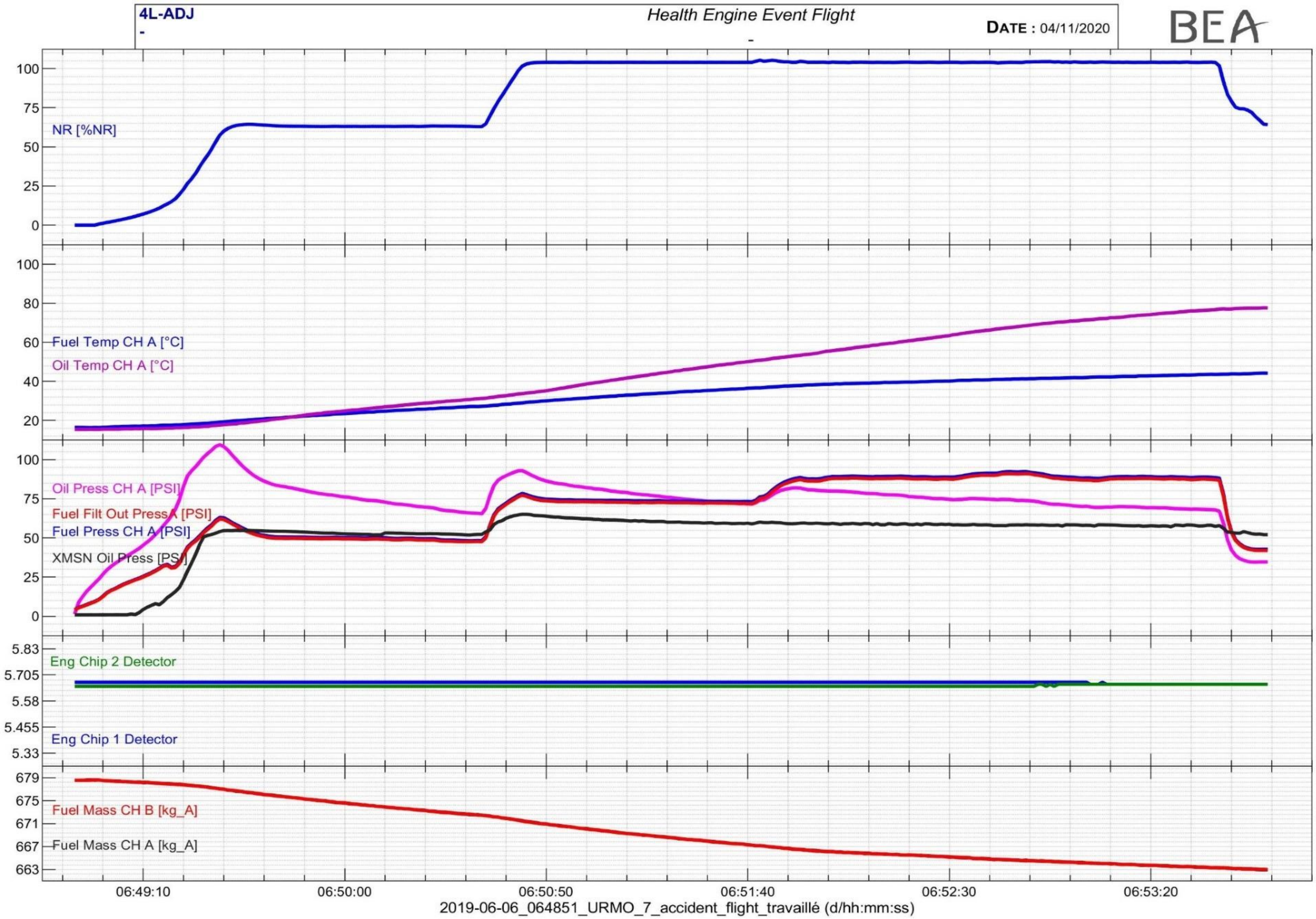
Appendices

Appendix 1 EECU DATA – CONTINUOUS A AND B – WHOLE FLIGHT

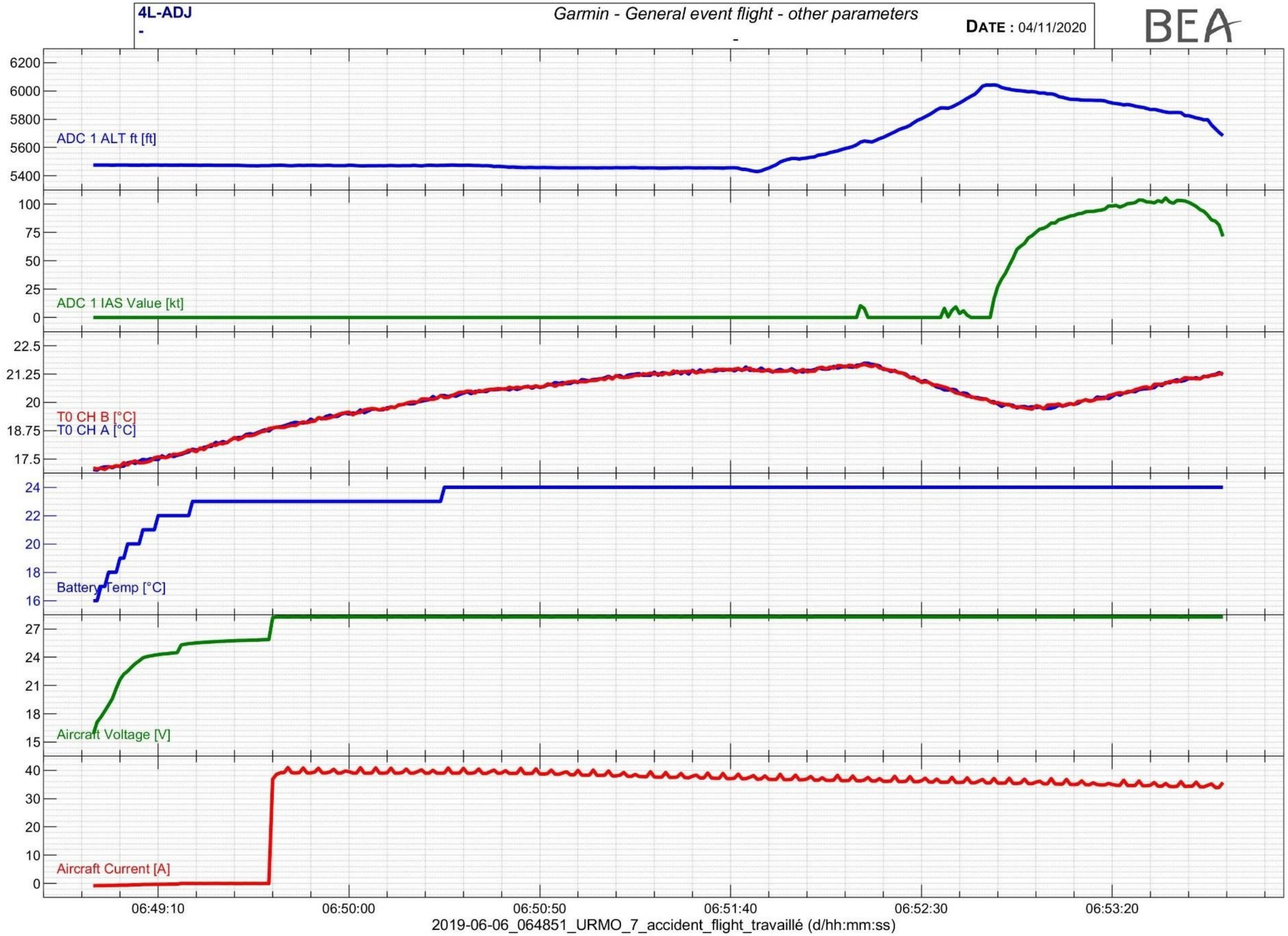


A 2.1 Engine related parameters in Garmin continuous recordings – event flight

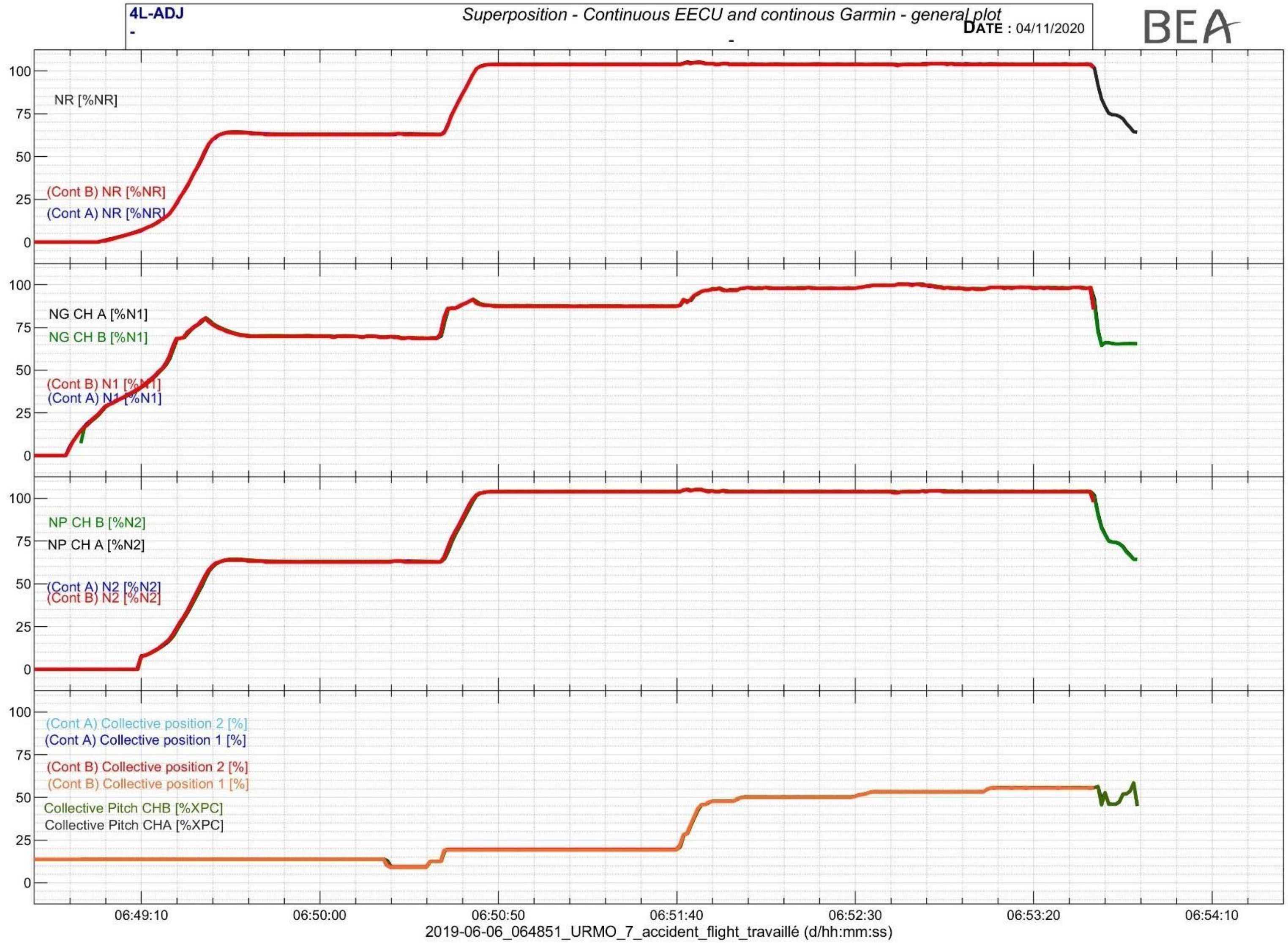



4L-ADJ
Health Engine Event Flight
DATE : 04/11/2020
BEA
NR [%NR]
Fuel Temp CH A [°C]
Oil Temp CH A [°C]
Oil Press CH A [PSI]
Fuel Filt Out PressA [PSI]
Fuel Press CH A [PSI]
XMSN Oil Press [PSI]
Eng Chip 2 Detector
Eng Chip 1 Detector
Fuel Mass CH B [kg_A]
Fuel Mass CH A [kg_A]
06:49:10
06:50:00
06:50:50
06:51:40
06:52:30
06:53:20
2019-06-06_064851_URMO_7_accident_flight_travaillé (d/hh:mm:ss)

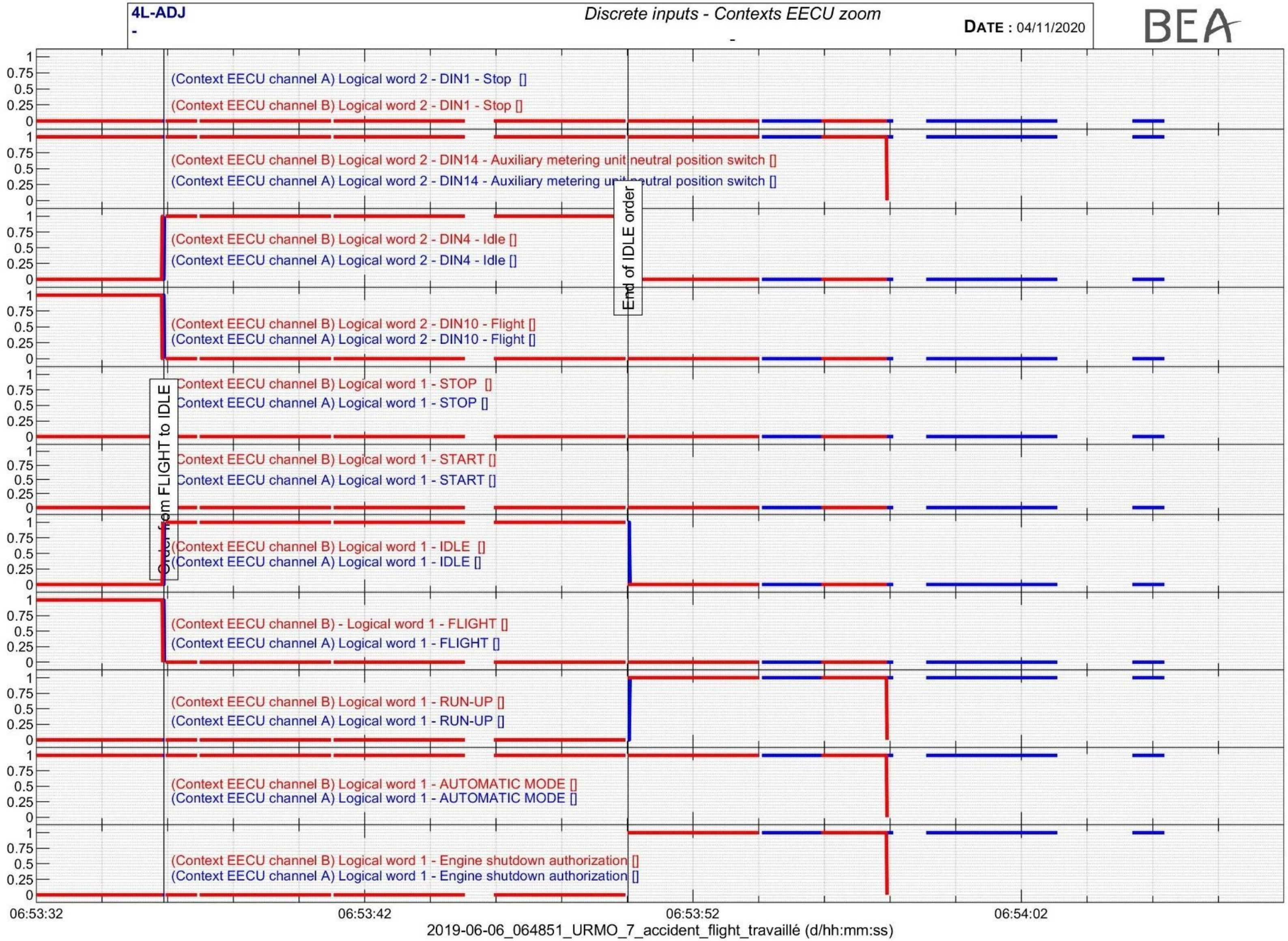
A 2.2 Other parameters in Garmin continuous recordings – event flight



Appendix 3 COMPARISON BETWEEN GARMIN DATA AND EECU DATA – CONTINUOUS B – WHOLE FLIGHT



Appendix 4 LOGICAL WORDS OF THE EECU



Appendix 5 CONTEXTUAL RECORDINGS OF THE EECU