ACCIDENT

Aircraft Type and Registration:	Bell 407, N120HH
No & Type of Engines:	1 Rolls-Royce M250-C47B turboshaft engine
Year of Manufacture:	2005 (Serial no: 53661)
Date & Time (UTC):	24 June 2020 at 1219 hrs
Location:	Long Marston, Stratford-upon-Avon, Warwickshire
Type of Flight:	Private
Persons on Board:	Crew - 1 Passengers - 1
Injuries	Crew - None Passengers - None
Nature of Damage:	Helicopter destroyed by post-impact fire
Commander's Licence:	Private Pilot's Licence (Helicopter)
Commander's Age:	45 years
Commander's Flying Experience:	16,900 hours (of which 475 were on type) Last 90 days - 7 hours Last 28 days - 2 hours
Information Source:	AAIB Field Investigation

Synopsis

The pilot and passenger were returning to Thruxton Aerodrome following a short flight over the Malvern hills when the engine failed. The pilot executed an autorotation landing in a field near Long Marston, after which they were both able to exit the helicopter without injury. However, the tail boom was severed during the landing and the helicopter was destroyed by fire.

The investigation found that the engine suffered an uncontained failure of the gas producing turbine disc due to insufficient oil reaching the bearings as a result of an oil leak. Due to the extensive damage to the helicopter it was not possible to determine the cause of the oil leak.

Two Safety Recommendations were made regarding the Failure Mode and Effects Analysis for the engine, and the fire resistance and crashworthiness of aircraft components. The helicopter manufacturer undertook two Safety Actions regarding advice to pilots, and one Safety Action regarding analysis of the airframe fuel filter.

History of the flight

The pilot collected the helicopter the day before the accident from Thruxton Aerodrome, Hampshire, where it had been undergoing an annual maintenance check. He was a friend of the owner and regularly flew N120HH. The helicopter needed to be flown for three to four hours before being returned to the maintenance facility for a torque check on the main rotor mast assembly and the pilot had agreed to fly these hours.

The pilot flew the helicopter from Thruxton to a private site near Peldon, Essex where the helicopter was normally kept; a total flight time of 56 minutes. The pilot described the flight as uneventful and reported that the helicopter flew well with no problems. The helicopter was parked in a hangar overnight and was fully refuelled.

On the day of the accident the pilot intended to fly to Wellsbourne Mountford Aerodrome (Wellsbourne), Warwickshire, to meet a friend and take them on a short sightseeing trip. He intended to return to Wellsbourne before flying the helicopter back to the maintenance facility at Thruxton later the same day. It was a clear day, the temperature was approximately 28°C and there was a light easterly breeze. Figure 1 shows N120HH before the first flight on the day of the accident; there is no visual evidence of an oil leak on the helicopter.

Figure 1

N120HH photographed before the first flight on the day of the accident
(image used with permission)

Before the first flight the pilot completed a walk-around, which included opening the engine cowlings and checking the fluid levels; he did not find any abnormalities. The engine oil level was showing as FULL on the sight glass[1]. The flight from Peldon to Wellsbourne, which took approximately 55 minutes, was uneventful and the pilot did not detect any problems with the helicopter. Prior to the second flight he completed another walk-around and checked the engine oil level via the oil tank sight glass, which was still showing FULL. The pilot recalled the helicopter had 620 lbs of fuel remaining. He took off from Wellsbourne with his passenger at 1039 hrs and the route flown is shown in Figure 2.

Footnote

[1] The pilot reported that the oil tank sight glass was clear and simple to read. By slightly nudging the helicopter he could clearly see the oil level moving.

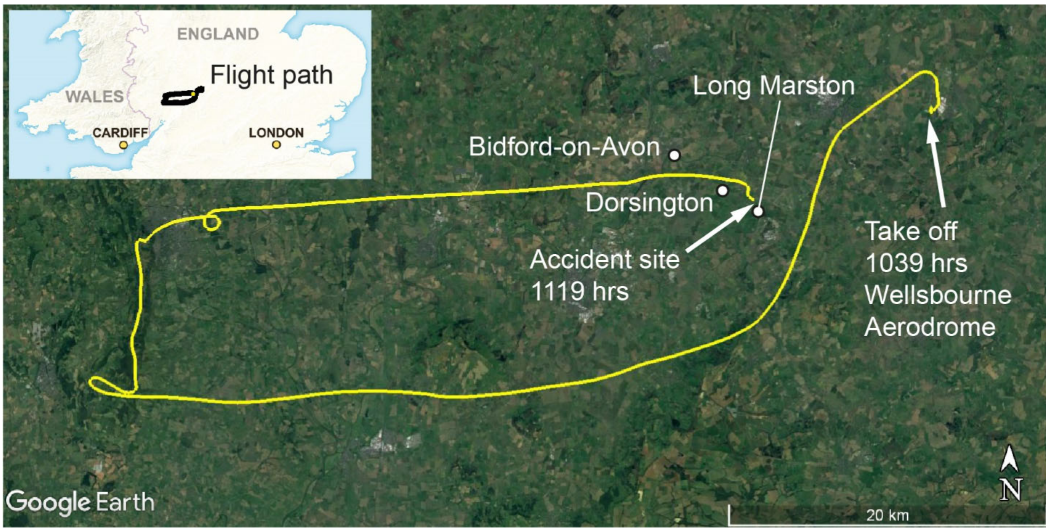

Figure 2

Route flown on the accident flight
(© 2021 Google, Image © Landsat / Copernicus)

Two witnesses noticed the helicopter as it was flying towards Wellsbourne (Witness 1 and 2). The first witness was just south of Bidford-on-Avon. He saw the helicopter flying towards him from the west and could see the helicopter clearly through his binoculars. He described seeing what looked like a "contrail" coming from above the cabin but below the rotor. He watched the helicopter as it passed overhead and flew to the east; the 'contrail' continued throughout this time. In the distance he saw it turn to the south but did not see the accident. The second witness, who was near Dorsington described seeing "white smoke" coming from the helicopter and took a couple of photographs (Figure 3). As it passed him, he heard a "popping noise", saw the smoke turn grey and saw the helicopter descend to the ground near Long Marston.

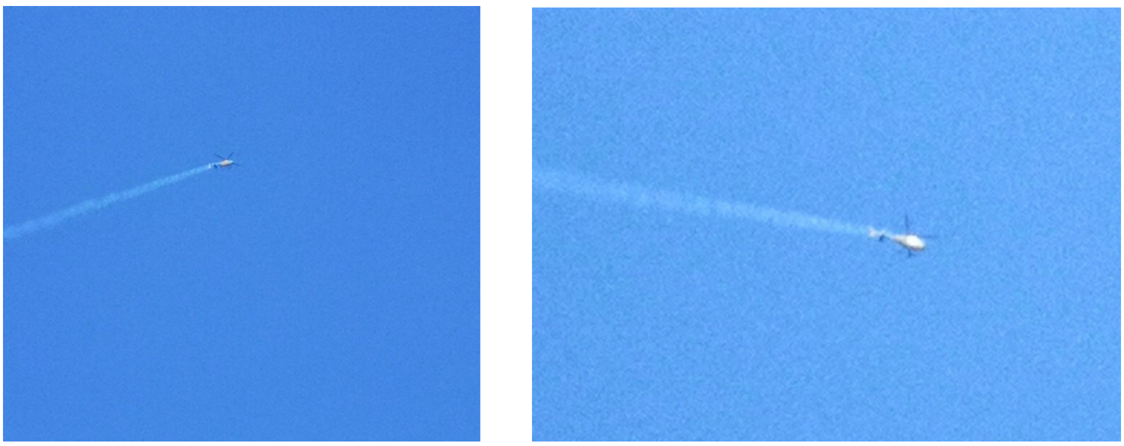

Figure 3

Witness photographs of helicopter in flight (images used with permission)

The pilot and passenger were not aware of the trail behind the helicopter. The helicopter was flying at approximately 1,800 ft amsl (approximately 1,700 ft agl) and 118 kt with the engine torque at 70 to 75%. The first indication in the cockpit of a problem was an amber FADEC FAULT light on the Caution and Warning Panel (CWP). When the pilot checked the instruments, he saw the N_R was at 100% and the N_P was at 90%[2]. He recalled looking at the torque and seeing a "5" but could not remember if it was fifty something percent or 5%, he did not recall the N_G[3]. The other engine instruments appeared normal.

Concerned by the apparent reduction in engine power the pilot wanted to check if he could still control the engine. He reduced the throttle slightly to match the throttle position to the N_G then selected the FADEC mode to manual[4]. He tried moving the throttle slightly and the engine responded as expected; he then selected the FADEC back to automatic. A few seconds after he did this an amber FADEC DEGRADED light illuminated on the CWP. He estimated that 45 seconds had elapsed since the initial warning. He recalled that the N_R was still at 100% and the N_P was at 90%. He increased the throttle slightly again to bring N_P to just below N_R to confirm he still had control of the engine. He then returned the throttle to the FLY detent. At this stage the pilot recalled there had been no audible warnings and he just had the two cautions on the CWP, the fuel gauges indicated 500 lbs of fuel onboard. The passenger also remembered just seeing the two lights.

The pilot estimated that approximately 27 seconds had elapsed after the second caution light when there was a "mechanical snapping noise", a violent yaw to the left, the red engine out warning illuminated, the torque dropped to 3% and a horn was heard. The pilot recalled a continuous horn, but the passenger recalled an intermittent horn. Other cautions and warnings also illuminated but the pilot did not recall which ones. He realised the engine had failed and immediately "fully" lowered the collective and moved the cyclic control rearwards to slow to 60 kt. Once established in autorotation he selected a field and made a MAYDAY call on the Wellsbourne frequency (Figure 4).

During the autorotation the pilot recalled that he just focused on N_R, airspeed and his selected field; he did not notice any further engine indications and did not switch anything off. As the helicopter approached the ground he flared and raised the collective bringing the helicopter to a nose high attitude. He levelled the helicopter and it touched down then skipped forward a few feet. As soon as the helicopter had stopped the pilot described "dumping the collective" as quickly as possible. The passenger immediately said she could smell smoke and that they needed to get out. The pilot recalled seeing a coloured liquid on the windscreen which he described as "off white coloured". They both exited the helicopter through their respective doors; as they looked back, they could see flames below the rotor. The pilot had no recollection of switching anything off prior to exiting the helicopter as his focus was on getting himself and his passenger out safely. The image at Figure 5 was taken by the pilot approximately 30 seconds after the helicopter landed and Figure 6 was taken 48 seconds after Figure 5. Neither occupant was injured.

Footnote

2 N_P is the power turbine rpm and N_R is the rotor rpm. In normal powered flight N_P is equal to N_R.

3 N_G is the rpm of the gas generator compressor and turbine.

4 FADEC is the Full Authority Digital Engine Control.

The flight lasted 39 minutes and N120HH had flown a total of 3 hours 30 minutes since leaving the maintenance facility.

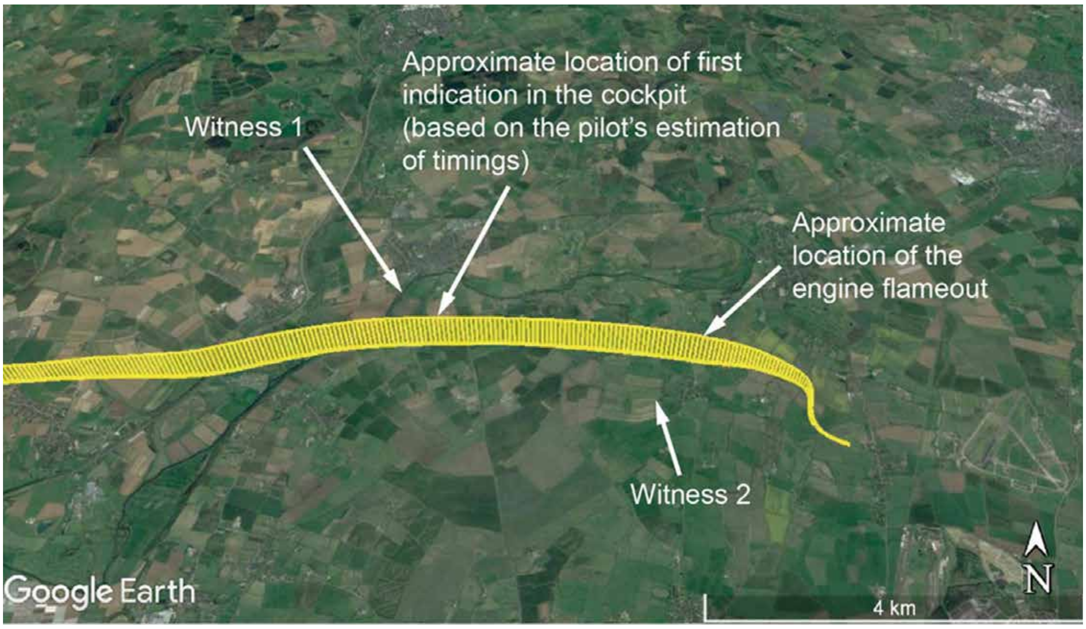

Figure 4

Witness positions and approximate location of cockpit indications
(© 2021 Google, Image © Landsat / Copernicus)

Figure 5

N120HH approximately 30 seconds after the landing (image used with permission)

Figure 6

N120HH 48 seconds after the image at Figure 5 was taken
(image used with permission)

Accident site

The accident site was contained in a relatively small area within a crop field (Figure 7). There were no indications on the ground that the helicopter had slid or struck the ground prior to its final position. The fire service advised that the wreckage had been covered in a precautionary layer of foam due to the magnesium fire in the gearbox casing.

Figure 7

Accident site (image used with permission)

The helicopter landed upright into wind and was destroyed by the ensuing fire (Figure 8). While the ground around the accident site was firm, the area around the wreckage was wet as a result of the fuel leaking from the helicopter and the foam applied by the fire service. The tail boom had separated from the helicopter and was lying in four sections distributed over an area to the right side of the helicopter. The tail boom was not fire damaged and was covered in oil.

Figure 8
Main wreckage

The main rotor blades had suffered fire damage along approximately a third of their inner span. One of the blades had witness marks consistent with it having struck the tail boom.

Type Certificate

The Bell 407 Type Certificate was issued by Canada as the State of Design. The design of the helicopter was initiated in the US to Title 14 Code of Federal Regulations (14 CFR); however, part way through the development of the helicopter the design was transferred to Canada, but 14 CFR was maintained as the basis of the design on the Type Certificate Data Sheet.

The guidance material in 14 CFR Part 27 and Part 33, which is applicable to this accident, is contained in US Advisory Circulars (AC). As ACs are not regulatory or mandatory, any change to guidance material, applicable to this accident, can be made by Transport Canada as State of Design who can then address it with other regulative authorities through their harmonisation processes.

Helicopter description

Introduction

The Bell 407 is a single engine, light helicopter fitted with a four-blade main rotor and two-blade tail rotor assembly. Standard configuration provides for one pilot and six passengers. N120HH was fitted with the optional high skid landing gear and had emergency flotation gear fitted to the skids.

Airframe

The airframe is constructed of aluminium honeycomb and carbon fibre composites and can be considered in two main sections, the main fuselage and the tailboom. The main fuselage provides the major load carrying capability. The tailboom supports the tail rotor and drive system, vertical fin and horizontal stabilizer.

Instrument panel

Figure 9 shows the instrument panel fitted to N120HH. The CWP is mounted just below the glareshield across the top of the instrument panel. As well as the visual indications provided by the CWP, the aircraft provides audible warning through three separate warning horns: ENGINE OUT (pulsating), LOW ROTOR (continuous) and FADEC FAIL (chime tone). The helicopter had been retrofitted, under a Supplemental Type Certificate, with a Primary Flight Display (PFD) and a Multi-Function Display (MFD).

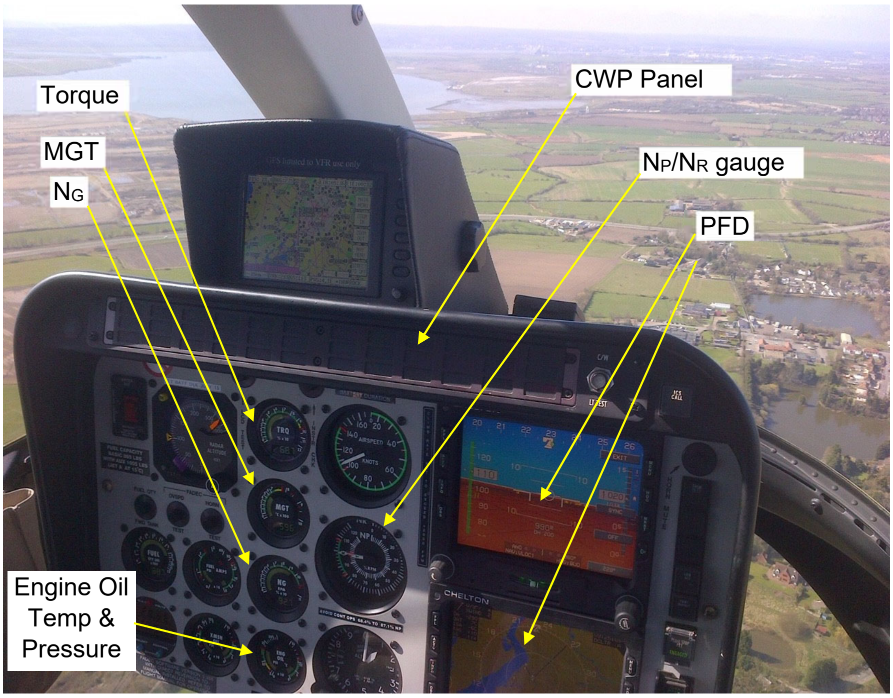

Figure 9

Cockpit instrument panel (image used with permission)

Main rotor

The main rotor is a four-bladed design with a composite hub. Pitch horns fitted to each rotor blade are connected via a pitch link to the swashplate, which enables the blade pitch angle to be varied.

Main transmission

The main transmission and rotor mast assembly transfers the engine torque to the main rotor system with a two-stage gear reduction. The translation of engine power to the transmission driveshaft is achieved via a freewheel unit.

Fuel System

The fuel system consists of two crash resistant, bladder type fuel cells. N120HH was also fitted with an optional auxiliary fuel tank. Fuel from the forward tank is transferred to the main tank by two transfer pumps located on the bottom of the forward fuel cell. Fuel from the main fuel cell is supplied to the engine through two boost pumps located at the base of the main fuel cell. The fuel from the two boost pumps joins into a common fuel line that passes through a fuel shutoff valve, then through an airframe mounted fuel filter before reaching the engine driven pump on the Hydro Mechanical Unit (HMU).

The fuel shutoff valve is controlled by a fuel valve switch located on the lower right side of the instrument panel. If the fuel valve switch is not switched OFF, then the left fuel boost and transfer pumps will continue to operate until the left fuel boost and transfer pump circuit breakers are selected OFF. These pumps operate directly from the battery and will not be deactivated when the battery switch is off.

In normal flight operations, fuel flow through the HMU to the fuel spray nozzle is controlled by a metering valve which the Electronic Control Unit (ECU) will command to meet the power demands required to maintain the commanded rotor speed. If the HMU fuel pump is no longer being driven (N_G is zero), then an internal pressurization check valve within the HMU prevents fuel leakage through the HMU.

Fire warning and protection

The requirements for fire protection[5], in 14 CFR Part 27, for structure, controls and other parts, require components essential to a controlled landing that would be affected by powerplant fires to be fireproof[6] or protected so they can perform their essential functions for at least five minutes under any foreseeable powerplant fire conditions. They also require[7] that the engine, including the combustor, turbine, and tailpipe sections of turbine engines must be isolated by a firewall, shroud, or equivalent means, from personnel compartments, structures, controls, rotor mechanisms, and other parts that are essential to a controlled landing. Additionally, each cowling and engine compartment covering must be at least fire resistant[8].

Regulations also stipulate the requirements on flammable fluid fire protection[9] to minimize the probability of ignition of the fluids and vapours, and the resultant hazards if ignition does occur.

Footnote

5 14 CFR Part 27 Airworthiness Standards: Normal Category Rotorcraft Subpart D--Design and Construction, Section 27.861 - Fire Protection.

6 With respect to materials and parts used to confine fire in a designated fire zone, means the capacity to withstand at least as well as steel in dimensions appropriate for the purpose for which they are used, the heat produced when there is a severe fire of extended duration in that zone.

7 14 CFR Part 27 Airworthiness Standards: Normal Category Rotorcraft, Subpart E, Sections 27.1191 and 27.1193 - Powerplant Fire Protection.

8 With respect to sheet or structural members means the capacity to withstand the heat associated with fire at least as well as aluminium alloy in dimensions appropriate for the purpose for which they are used.

9 14 CFR Part 27 Airworthiness Standards: Normal Category Rotorcraft Subpart D-Design and Construction, Section 27.863 – Flammable Fluid Fire Protection.

The Bell 407 engine bay has titanium firewalls forward, aft and below the engine. The firewalls are fireproof and can withstand a 2,000°F fire for a minimum of 15 minutes. The upper and lower engine cowls are fire resistant and can withstand a 2,000°F fire for a minimum of five minutes. The helicopter was not fitted with the optional[10] fire detection system which would provide warning to the pilot in the event of a fire.

The fuel system also contains crashworthiness features such as puncture resistant fuel bladders, flexible interconnects or breakaway valves and frangible structure.

Engine description

General description

The Bell 407 is powered by a Rolls-Royce 250-C47B turboshaft engine comprising a single stage centrifugal compressor driven by a two-stage Gas Producing (GP) turbine, which are connected by a coupling shaft using splined adaptors. A two-stage free Power Turbine (PT) provides power to the main transmission (Figure 10).

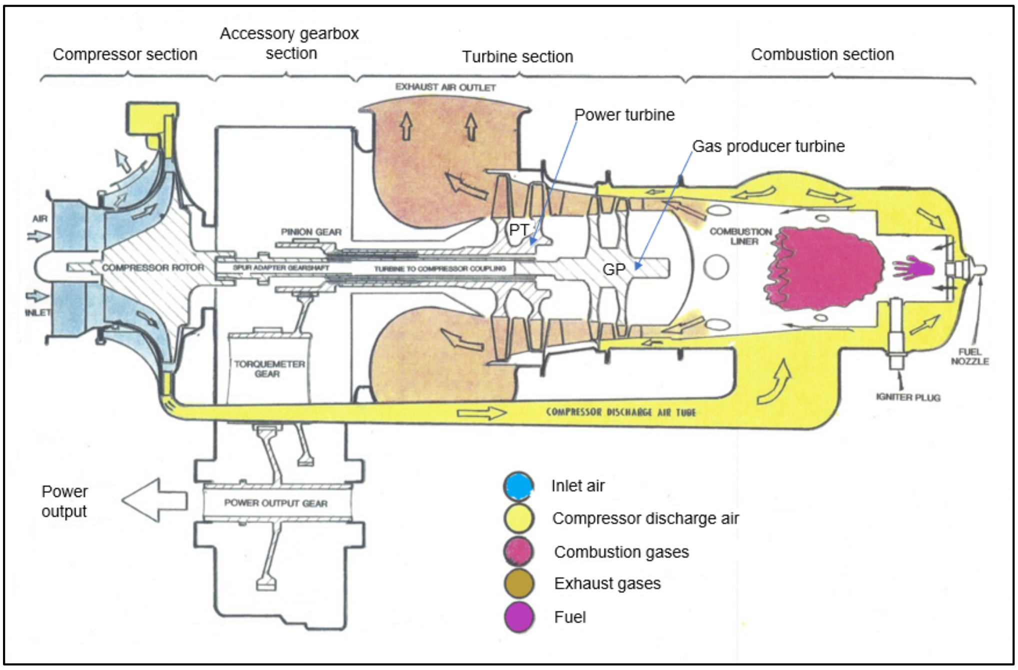

Figure 10

Rolls-Royce 250-C47B turboshaft engine
(Image used with permission of Rolls Royce Corporation)

Turbine assembly

The turbine assembly consists of a stationary set of nozzle guide vanes and rotor comprising a disc and aerofoil blades, which extract energy from the gas stream. The GP rotors are identified as GP1 and GP2, and the PT rotors as PT1 and PT2 (Figure 11).

Footnote

10 The aircraft does not include 14 CFR 27.1195 (Fire Detection Systems) as part of its certification basis.

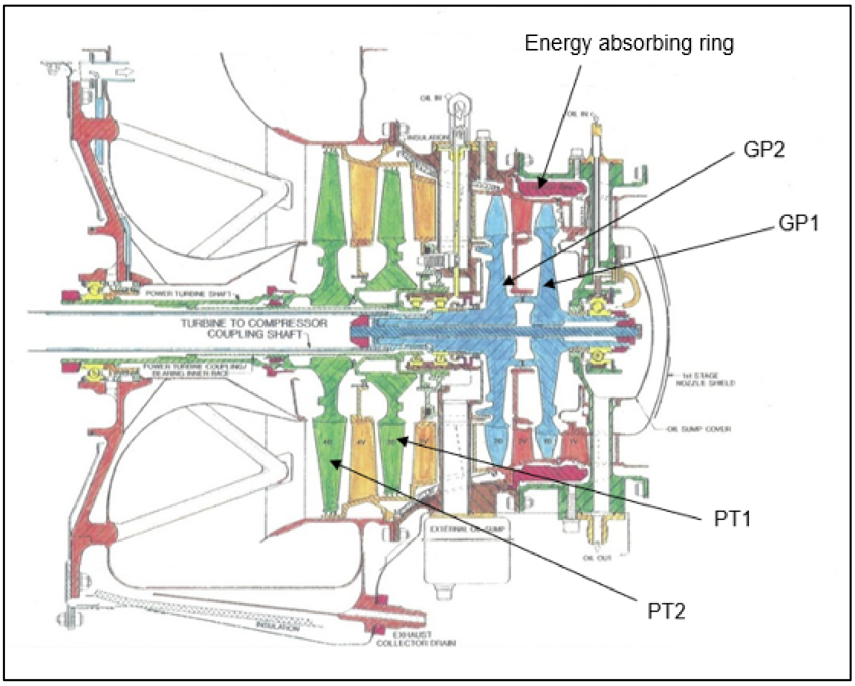

Figure 11

Rolls-Royce 250-C47B Turbine Section
(Image used with permission of Rolls Royce Corporation)

A curvic coupling[11] transfers torsional loads between GP1 and GP2. A tie-bolt, located through the centre of the rotors, provides a clamping load to maintain axial location of the rotor assemblies.

The PT runs at a different speed to the GP rotor and their speeds are each expressed as a percentage of their maximum design speed and referred to as N_P and N_G respectively.

Energy absorbing ring

The engine casing incorporates an energy absorbing ring, which is designed to attenuate the high energy debris from a GP1 disc burst.

Turbine rotors certification

The Rolls-Royce M250 Series 4 engines, as fitted to N120HH, demonstrated compliance against 14 CFR requirements for failure of turbine rotors[12].The critical condition for the rotors is 105% of the maximum speed that would occur from turbine loss of load[13]. The M250 Series 4 of engines do not have the ability to arrest a GP turbine loss of load event prior to burst. Therefore, the GP rotor was designed such that the GP1 will burst first and high energy debris will be attenuated by the energy absorbing ring. With GP1 no longer

Footnote

[11] A joint between driving and driven shaft systems which transmits the torque. In its simplest form it comprises two sets of meshing radial teeth of smooth curving profile.
[12] Originally certified in 1978 against 14 CFR Part 33 amendment 4 paragraph 33.27, the top level requirement is to minimize the probability of failure of turbine rotors.
[13] For example, if the GP turbine became decoupled from the compressor.

providing drive, the shaft will slow sufficiently to prevent the disc on GP2 bursting. The PT rotors were proven in certification testing to be resilient to loss of load events and therefore do not require energy absorbing rings.

Accessory gear trains

The accessory gearbox is the primary structural member of the engine. It provides mounting and support for the compressor and turbine assemblies. It contains most of the lubrication system components and incorporates two separate gear trains: the PT and GP.

The purpose of the PT gear train is to reduce the speed of the PT to the main engine output shaft. Therefore, when the engine is driving the main rotor, the main rotor speed (N_R) is directly proportionate to the PT speed (N_P). As N_R and N_P are expressed as a percentage, they should both be the same when the engine is driving the main rotor.

The GP gear train provides drive for the oil pumps, HMU and starter generator. The accessory gearbox housing and cover are magnesium alloy castings which house the bearings used to support the PT and GP gear trains.

Engine Indications

N_P is detected by a magnetic pickup sensor which is mounted on top of the gearbox cover. The pickup sensor extends into the area between the gearbox and the turbine assembly. It is positioned with a small gap between it and the teeth of the spanner nut, which is located on the PT shaft assembly just forward of the No 5 bearing (Figure 13). As the twenty teeth on the spanner nut rotate past the sensor an alternating voltage is induced, the amplitude and frequency of which is proportional to the speed of the shaft. This pickup has two channels designated N_{P1} and $N_{P2.}$ N_{P1} only supplies the ECU, N_{P2} supplies the N_P tachometer indicator and the ECU.

The N_G speed sensor is located in the accessory gearbox housing. It is a magnetic pickup sensor which produces a signal proportionate to the speed of the 86 teeth of the centrifugal breather gear. The breather gear is part of the gear train that is driven by the GP turbine via a spur adaptor gear shaft.

The PT gear train incorporates a torque meter to measure engine output torque. The torque meter is hydraulic and uses the engine lubrication system as its oil (hydraulic) pressure source. The system uses two helical reduction gears that produce a forward axial thrust on the helical torque meter gear shaft and provides a hydraulic pressure signal which is directly proportional to output torque. Therefore, engine torque indication is affected directly by changes in oil system pressure.

Engine control system

The engine incorporates a Full Authority Digital Electronic Control (FADEC) which comprises: the ECU, HMU, sensors and actuators. In automatic mode the ECU continuously monitors a number of variables and sends a signal to the HMU to adjust the fuel flow, which controls the speed, acceleration and temperature of the engine.

Some of the parameters monitored by the ECU include:

Measured Gas Temperature (MGT)[14],

Gas Producer speed (N_G),

Power Turbine speed (N_P),

Main Rotor speed (N_R),

Engine Torque Meter Oil Pressure (TMOP),

Collective Pitch (CP) and rate,

Power Lever Angle (PLA)/throttle position.

The ECU continuously monitors the system for faults and activates the appropriate warning on the CWP and horn. If the ECU detects a loss of both N_P signals it will use N_R as its input instead, and the ECU will continue to command the amount of power required to match the N_R demand. If any failure occurs in the ECU/HMU or in one of the input/output signals that significantly impacts the ECU or control of the HMU, the pilot will be alerted via the FADEC FAIL warning horn and the FADEC FAIL and FADEC MANUAL warning lights. If the detected failure does not significantly impair the functioning of the ECU, the pilot will be alerted via a FADEC DEGRADED or FADEC FAULT caution light, RESTART FAULT advisory light, or a combination of these, depending upon the nature of the fault.

A twist grip throttle is mounted on the collective lever to allow the pilot to control the engine. The throttle is marked with OFF, IDLE, FLY and MAX detents and an N_G scale (Figure 12). In normal flight, with the FADEC in AUTO, the throttle is left in the FLY position and the ECU automatically controls the engine. If the FADEC is switched to MAN (manual) the pilot can control the engine speed by rotating the throttle. With the throttle in the OFF detent, fuel will not flow from the HMU to the engine fuel spray nozzle; the OFF detent position corresponds to a PLA position of 5° or less.

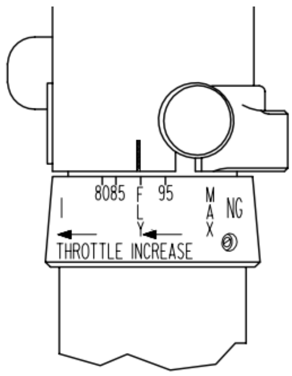

Figure 12

Diagram of the throttle control on the collective
(Image used with permission of Bell Helicopters Textron)

Footnote

[14] The measured gas temperature (MGT) gauge displays engine gas temperature of air between the GP and PT in degrees Celsius and is an indicator that the engine is operating within acceptable limits.

Bearings

Location and support of the rotor assemblies is achieved through the main bearings which are numbered 1 through to 8 in a front to rear direction (Figure 13). The main axial location of the engine is provided by No 2, 5 and 8 thrust (ball) bearings. The remaining (roller) bearings provide radial location.

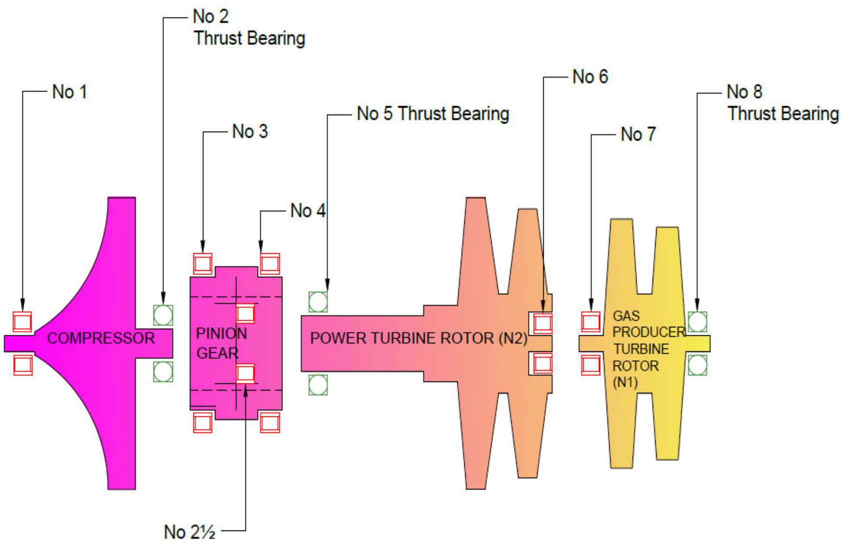

Figure 13
Main engine bearings

Engine oil system

The engine oil system includes an externally mounted oil tank (capacity 5.7 litre), a temperature bulb, and an oil cooler. The engine oil temperature input signal is provided by a temperature bulb located in the engine oil tank. The oil tank and oil cooler are installed on top of the fuselage, behind the aft firewall (Figure 14). The tail rotor drive shaft drives a fan that supplies and moves air through the cooler. The lubrication oil is supplied to and from the system through rigid lines and flexible hoses.

The oil tank supplies lubricating oil through the oil inlet tube to a gear-type pressure pump installed within the accessory gearbox. Oil under pressure (pressure oil) goes through the oil filter and a pressure regulating valve and is distributed to the four lubricating ports (which supply the bearings) and the torque meter pressure port. This oil lubricates the engine then becomes scavenge oil and returns to the four scavenge pumps (Figure 15).

The scavenge oil is returned to the oil tank via the two magnetic chip detectors[15] (MCD), oil return hose, external oil filter[16] and oil cooler. The engine oil pressure indication is supplied by the oil pressure sensor. As the oil system is regulated by pressure, depending on the

Footnote

[15] An indicating type, quick disconnect, self-sealing MCD is installed at the bottom of the gearbox. A second detector is in the engine oil outlet connector. It is not self-sealing.

[16] The scavenge oil filter is mounted externally together with the fuel filter in the Combined Engine Filter Assembly..

operating conditions there is typically more oil in the system than required to maintain the steady state pressure of between 115 – 130 psi. This relatively high pressure is required because the torque meter in the engine gearbox is hydraulic and uses the engine lubrication system as its oil. Excess oil is circulated via the bypass valve.

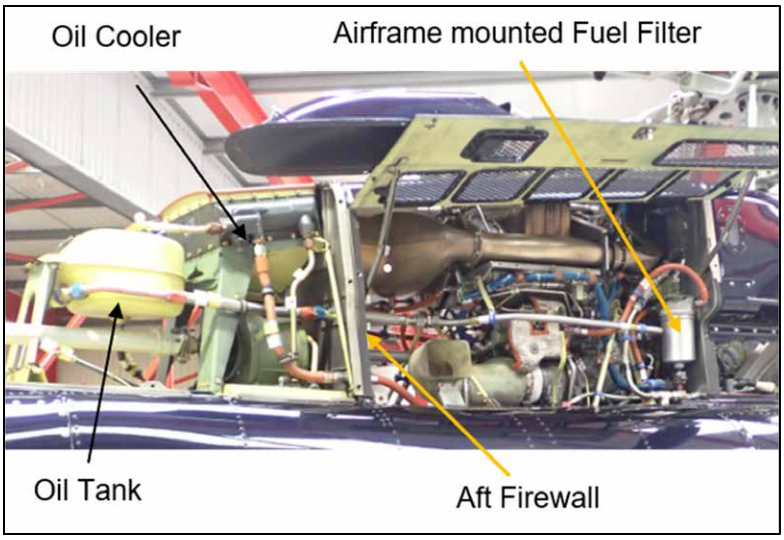

Figure 14
Bell 407 engine oil system compartment

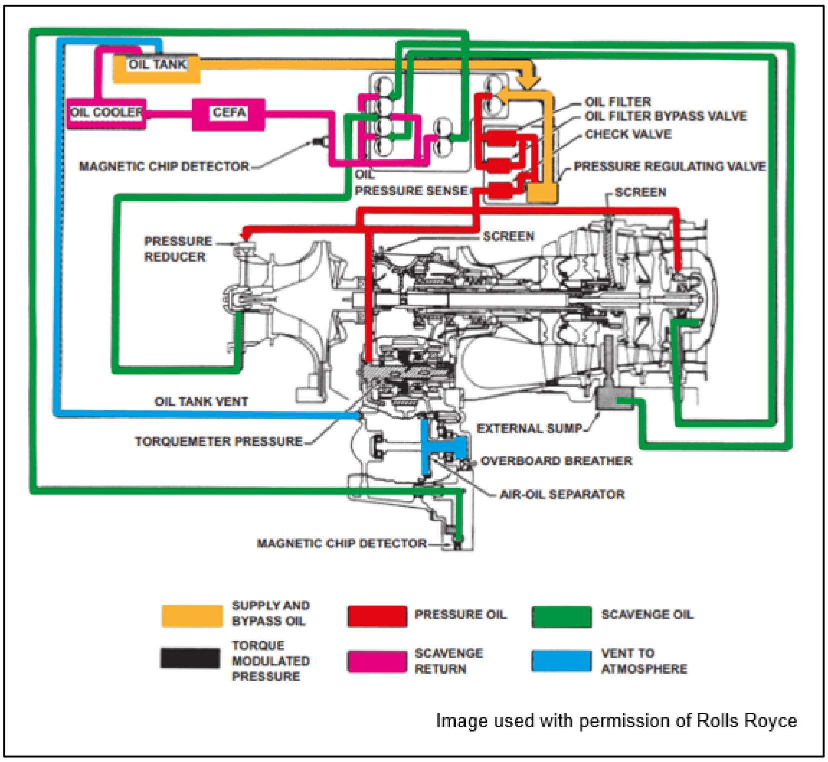

Figure 15
Engine oil system

Engine flame-out

An engine flame-out occurs when combustion fails in the engine, it is associated with causes other than intentional fuel shut-off. The Bell 407 Rotorcraft Flight Manual[17] (RFM) for an engine failure in flight provides the following indications:

'3-3-A-2. ENGINE FAILURE IN-FLIGHT

- *INDICATIONS:*

  1. *Left yaw.*
  2. *ENGINE OUT and RPM warning lights illuminated.*
  3. *Engine instruments indicate power loss.*
  4. *Engine out audio activated when NG drops below 55%.*
  5. *NR decreasing with RPM warning light and audio on when NR drops below 95%.'*

Figure 16

RFM guidance on engine failure Indications

Recorded information

The helicopter was not fitted with a flight recorder, nor was it required to be. Despite severe thermal damage, data was recovered from the PFD and the ECU. Other potential sources of data, such as the individual engine instruments, were found to have suffered too much heat damage to be recoverable.

PFD and MFD

The PFD and MFD store flight parameter data, once per second, on an internal memory card, with both units sharing and storing the same data. The system configuration on N120HH did not record engine related data. Warnings and cautions are not recorded.

Data for the last five flights was recovered and the flight path data was used to create Figures 2 and 4. Other flight parameters are combined with the ECU data and are shown in Figure 17.

ECU

The data was recovered from the ECU non-volatile memory chip and converted to usable files with assistance from the ECU and engine manufacturers.

Footnote

17 Bell Flight Manual, Section 3, Emergency/Malfunction Procedures.

The data included a time-stamped list of faults triggered during the accident flight. When a fault becomes active, it is given a time-stamp and added to the list. If the FADEC is switched from manual to auto, any active faults are re-detected, with a time-stamp reflecting the time of checking, regardless of whether the fault was already recorded in the list. A maximum of 30 faults can be recorded in the time-stamped fault category, after which the oldest faults are overwritten with the latest faults. The maximum of 30 faults were recorded during the accident flight.

In addition, the ECU records a list of all the faults triggered during the last engine run and a list of all the faults accumulated since the last maintenance reset. The time of triggering and the number of times the faults triggered are not recorded. The two lists recovered from the ECU were identical and did not record any faults not recorded in the time-stamped fault list from the accident flight.

Analysis of the faults indicates that some of the 30 time-stamped recorded faults had likely been previously triggered during the accident flight and then been overwritten. Though faults are likely to have occurred on the accident flight prior to the recorded time-stamped faults, they can only have been previously triggered occurrences of the same faults.

The ECU also records snapshots of parameters triggered by one of a set of predefined trigger conditions including exceedances of torque, torque rate, MGT, N_P, $N_{G,}$ and N_R. The log can record ten such events, and five were triggered during the accident flight. None of the events were triggered until the engine flameout. These are shown in Table 1.

Snapshot	ECU time stamp hhhh:mm:ss	Seconds after flameout	Snapshot Trigger
1	2154:34:32.376	0	Flameout
2	2154:34:33.096	0.720	N_G Under Low Limit and N_R Droop in Auto or Manual
3	2154:34:39.288	6.912	Hard Fault
4	2154:34:39.312	6.936	ECU hard fault - [Reversionary]
5	2154:35:02.472	30.096	N_R Droop in Auto or Manual

Table 1

Snapshots triggered during the accident flight

The ECU records additional data from before and after these snapshots. These consist of parameters recorded every 1.2 seconds from 12 seconds before the first snapshot event to 48 seconds after the last snapshot event, or until power is lost.

Combined data

Figure 17 shows data extracted from the PFD and ECU from the later part of the accident flight. The two sources of data were combined by aligning the ends of both recordings.

The helicopter took off at 1039 hrs and climbed away shortly after.

The earliest recorded faults were at 1117:17 hrs, both relating to torque sensing. As torque is sensed using oil pressure, this may be the first recorded indication of a loss of oil rather than a fault with the electronic sensing system. The loss of torque sensing means that the reversionary governor[18] would not have been capable of taking control; however, there would have been no effect on engine operation if the primary governor was in control. While the loss of torque sensing would have affected the cockpit torque indication, it would not have triggered a CWP alert.

The FADEC was switched from MAN back to AUTO at about 1117:18 hrs, enabling previous faults latched in the ECU memory to re-trigger. This resulted in ten further faults being recorded in quick succession, indicating they were already active. These related to the torque sensing faults, both N_P sensors and the secondary N_G sensor. Because of the limit on the number of faults recorded, it is not possible to assess when these first activated during the flight. The data did not indicate the time at which the FADEC had been switched from AUTO to MAN control.

Either N_P faults can trigger the FADEC FAULT caution on their own, but when both faults are active together the FADEC DEGRADED caution and the RESTART FAULT advisory alerts are triggered. The secondary N_P and N_G sensors supply data to the respective cockpit engine instruments. Given that the faults were likely to have originally been triggered before the FADEC was switched back to auto, the cockpit N_P and N_G indications were likely to have been affected earlier in the flight.

ECU buffered data, triggered by a later snapshot event, was recorded from about 1117:48. At this time the helicopter was flying at 118 kt with a GPS altitude of 1,866 ft amsl, a magnetic heading of 102° and a vertical speed of -250 ft/min. The ECU recorded an N_G of 96%, an N_R of 100% and an MGT of 1,340°F. It also recorded an N_P of 0% and a torque of 4%.

About a second later a power supply fault was recorded that had no immediate operational effect given the processes being used to govern the engine, but would have become an issue had a reversionary mode been required.

Footnote

[18] A reversionary governor is a backup method of governing the engine based on alternative sensor inputs that takes control if the primary governor or its sensor inputs are compromised.

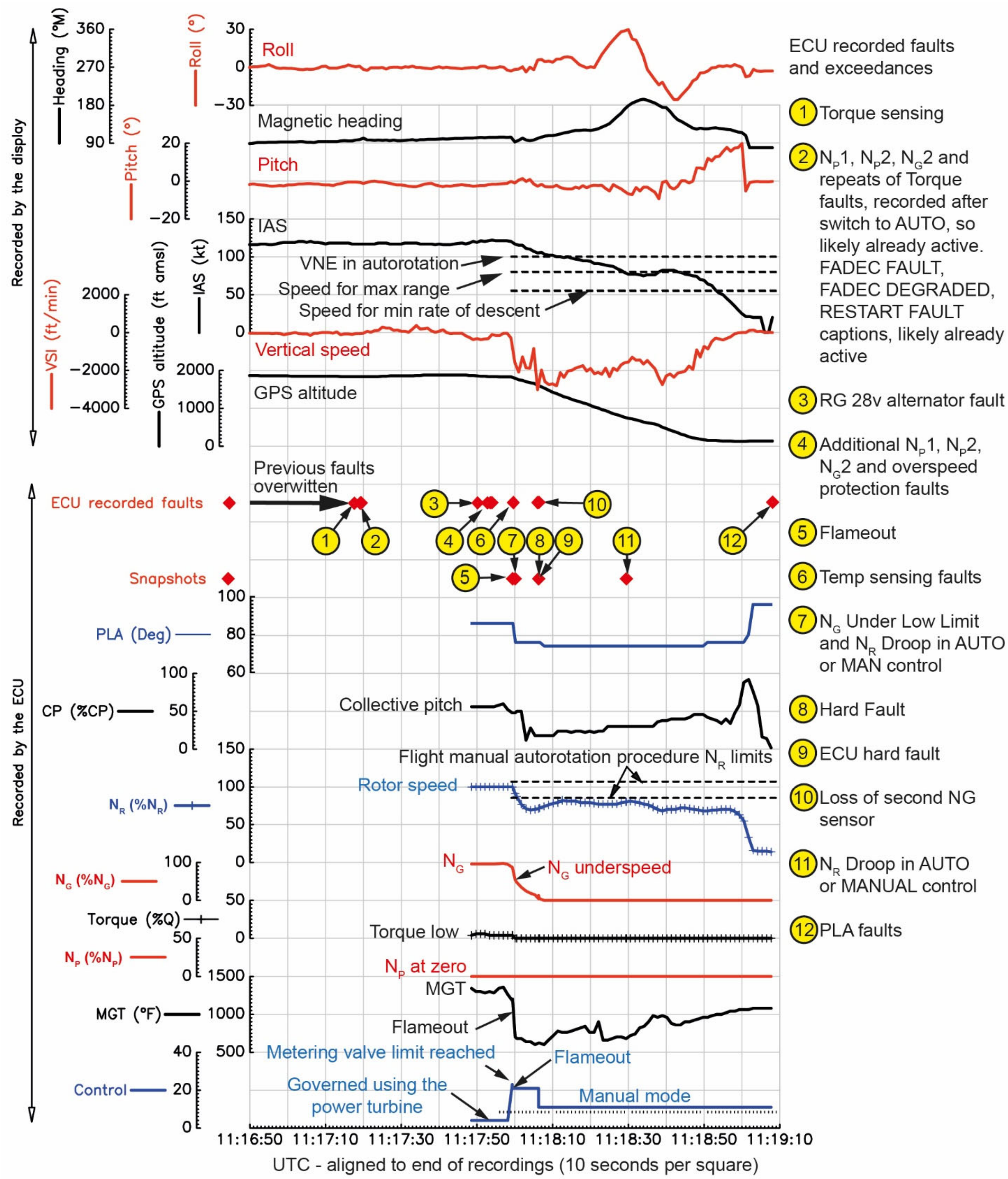

Figure 17

Extracts from the data recorded by the PFD and ECU

This was shortly followed by additional faults associated with both N_P sensors, the primary N_G sensor and overspeed protection. At about 1117:58 hrs, the ECU control status switched from being governed by the power turbine, to reaching a control limit and then being in a flameout situation. It recorded a temperature sensing fault and triggered a snapshot event for a flameout. This was closely followed by snapshots triggered by the N_G dropping below a trigger threshold and an N_R dropping below 92% of desired level.

The pilot reduced the collective within about four seconds, the N_R dropped to 69% before increasing again to 82%. The recorded N_R then stayed between 68% and 82% until the collective was raised during the landing, approximately 60 seconds after the flameout. The recorded N_R during the autorotation was below the 85% lower N_R limit in the RFM. The recorded collective pitch did not capture the collective at the lowest value as indicated by the data at the end of the flight. However, the data at the start of the autorotation indicates that the sample rate of this parameter was insufficient to accurately capture the dynamic movement of the collective therefore it is possible that the collective was momentarily fully lowered at the start of the autorotation. The collective pitch started increasing during the descent. The throttle angle was initially reduced along with the collective pitch and then increased to a higher setting as the collective was finally lowered after the flare; it remained within the flight range throughout the autorotation.

After the flameout, the IAS reduced over about 30 seconds to stabilise between 75 kt and 82 kt before reducing as the helicopter approached the landing flare. The descent rate varied around -2,000 ft/min until approaching the ground when it was smoothly reduced. The pitch up for the flare started at approximately 70 ft, with an IAS of 74 kt. A 'g-force' parameter showed little activity during the landing. The limited sample rate of the parameter means it is unlikely to have recorded the peak force experienced during the landing.

The MGT dropped sharply to 700°F at the point of flameout and then with some initial variation climbed to 1,080°F by the time the helicopter landed.

Throughout the ECU recorded period of continuous data, the N_P stayed at 0%. The torque was 4% at the start and dropped to 0% at the point of flameout. N_G dropped to zero within about seven seconds at which point the ECU recorded a fault with the secondary N_G sensors.

Maintenance history

The helicopter had recently undergone scheduled maintenance at a manufacturer approved maintenance repair organisation. The maintenance work package included an Annual Inspection[19] for the airframe and the 300-hour maintenance inspection on the engine.

Airframe

Unscheduled work arising from this maintenance included replacement of the main transmission freewheel unit and pitch horns. The latter resulted in a requirement for a torque check on the main rotor mast assembly after the helicopter had flown 3 to 4 hours.

Engine oil system

Maintenance of the engine oil system was carried out in accordance with the 300-hour schedule[20]. This required oil flow checks to ensure that sufficient oil was being circulated round the system and not being restricted by the build-up of carbon on components. The oil

Footnote

[19] In accordance with Bell Helicopter Textron (BHT)-407 DMC 407-A-05-40-00-00/01A-281A-A Rev 1.
[20] Rolls Royce M250-C47B Operation and Maintenance Manual, Chapter 72-00-00, Page 613, Table 606.

flow checks exceeded the minimum requirement. The oil filter and No 1 bearing pressure reducer were also cleaned and both MCDs were checked. The only other component in the oil system that was disturbed was the oil cooler fan, this required re-balancing on refit following removal to enable lubrication of the tail rotor drive shaft splines. The combined engine and fuel assembly filter did not require maintenance.

The last time that the oil system main supply and return pipes were removed was 130 flying hours prior to the accident flight when the rear engine firewall was replaced due to cracking.

Helicopter examination

Initial wreckage examination at accident site

Apart from the tail section, the helicopter had been badly damaged by fire. There were distinct impact marks on the tail boom indicating contact with a main rotor blade and the tail rotor drive shaft, which had sheared. There was no indication that the vertical fin and tail skid had struck the ground on landing, the landing skids did not appear to be splayed, they were not cracked and there was no other structural indication of a hard landing.

The engine casing was ruptured and inspection of the interior of the engine showed that the GP turbine shaft had sheared and that both GP rotor discs had burst (Figure 18). Parts of GP1 had been retained but other parts had escaped the energy absorbing ring and were located both within the engine bay and on the ground near the wreckage.

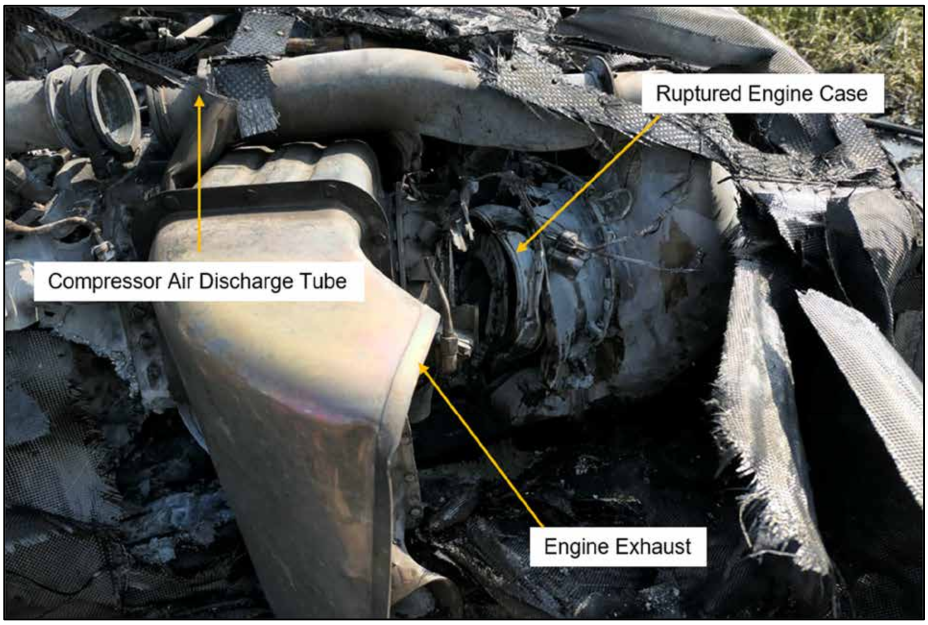

Figure 18

Engine at the accident site

Detailed examination

The main structure had lost structural integrity and the weight of the main rotor mast and head had collapsed the roof of the helicopter onto itself. The flight controls were found to be intact.

The engine oil system compartment, which sits aft of the engine rear fire wall, had been destroyed by the fire. Such was the extent of the damage that inspection of the wreckage could not identify the tank, oil cooler or fan assembly. Analysis of the oil on the tail boom determined it to be a close match to engine oil[21].

The engine bay was removed from the wreckage and assessed prior to removal of the engine itself. Damage in this area could be attributed to the fire and the release of the turbine rotor discs. There was no evidence that the fire walls had been breached. The main fuel supply line to the HMU was intact. The magnesium gearbox had been partly destroyed although many of the internal components such as gears remained in place. There was evidence that some interconnections had been severed by the released turbine rotor discs. These connections are mapped against the projectile analysis at Figure 19.

Key	System	Type of Damage
1	Oil feed tube to No 6, 7 & 8 bearings	Severed in line with GP rotor release
2	Oil scavenge tube from No 8 bearing	Severed in line with GP rotor release
3	Main engine oil feed from oil tank	Severed in line with GP rotor release
4	HMU fuel supply to fuel spray nozzle	Severed in line with GP rotor release
5	R/H discharge air tube	Impact damage and punctured wall in line with GP rotor release
6	Oil hose from freewheel unit	Severed in line with GP rotor release
7	No 5 bearing scavenge tube	Severed in line with GP rotor release
8	Exhaust nozzle	Impact and penetration damage in line with GP rotor release

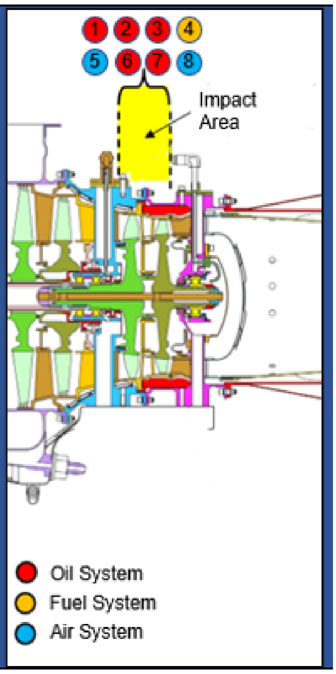

Figure 19

Projectile analysis

The exhaust duct was significantly deformed, with evidence of impact holes (Figure 20). This damage was consistent with the release of blades from the GP turbine rotor disc which escaped via the exhaust duct.

Footnote

21 The transmission oil is similar, but differentiation can be made by identifying additives not present in both.

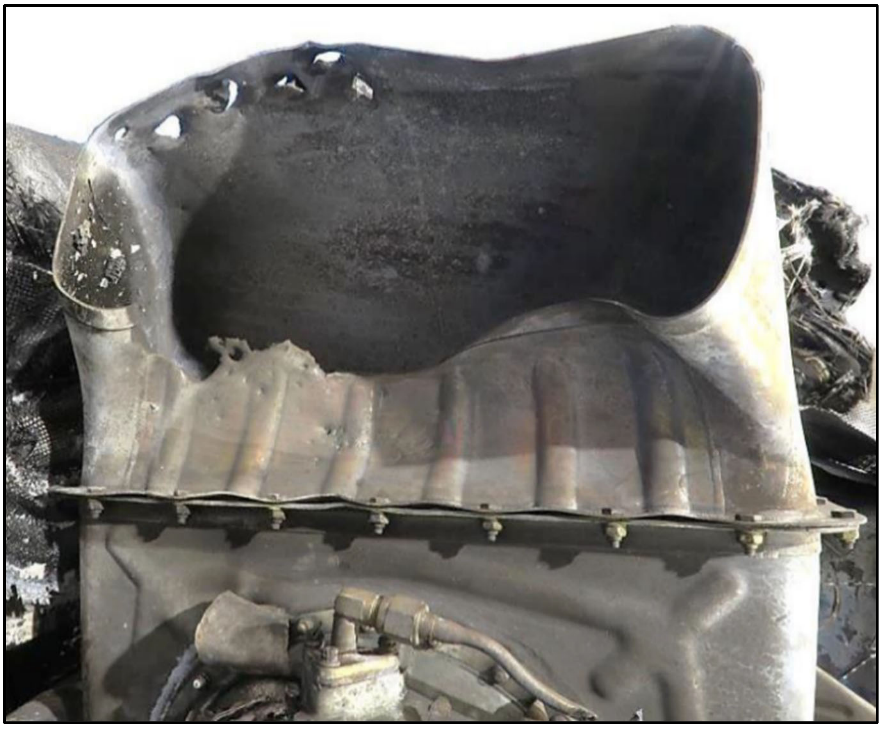

Figure 20
Exhaust damage

The airframe mounted fuel filter had fractured, which metallurgy showed was not caused by impact from the disc burst fragments but was more consistent with a failure due to heat and overload (Figure 21).

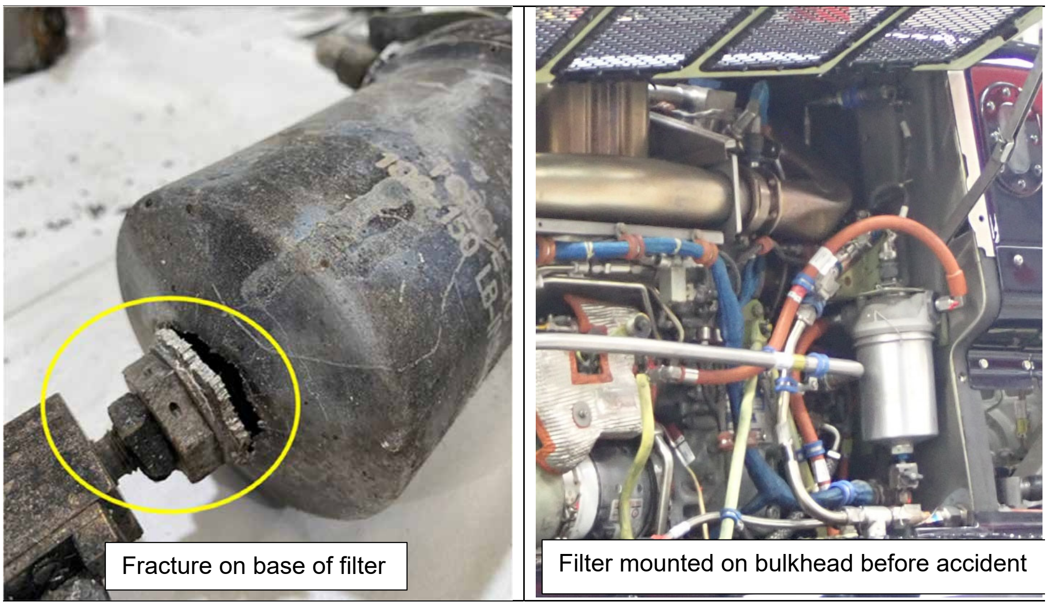

Figure 21
Damage to airframe mounted fuel filter

Detailed engine inspection

The engine was taken to an engine overhaul organisation, where it was disassembled under the supervision of the AAIB and a representative from the engine manufacturer's sister company in the UK. Figure 22 shows the key components on which observations were made during the disassembly.

The energy absorbing ring around the GP1 rotor was significantly deformed (Figure 23). There was no deformation to the casing around the PT turbine.

The GP1 rotor shaft had fractured and the tie-bolt which locates the GP turbine rotors had sheared (Figure 24) in overload in line with the GP1 disc and shaft, separating the curvic couplings in the process. A small middle section of the tie-bolt was not found.

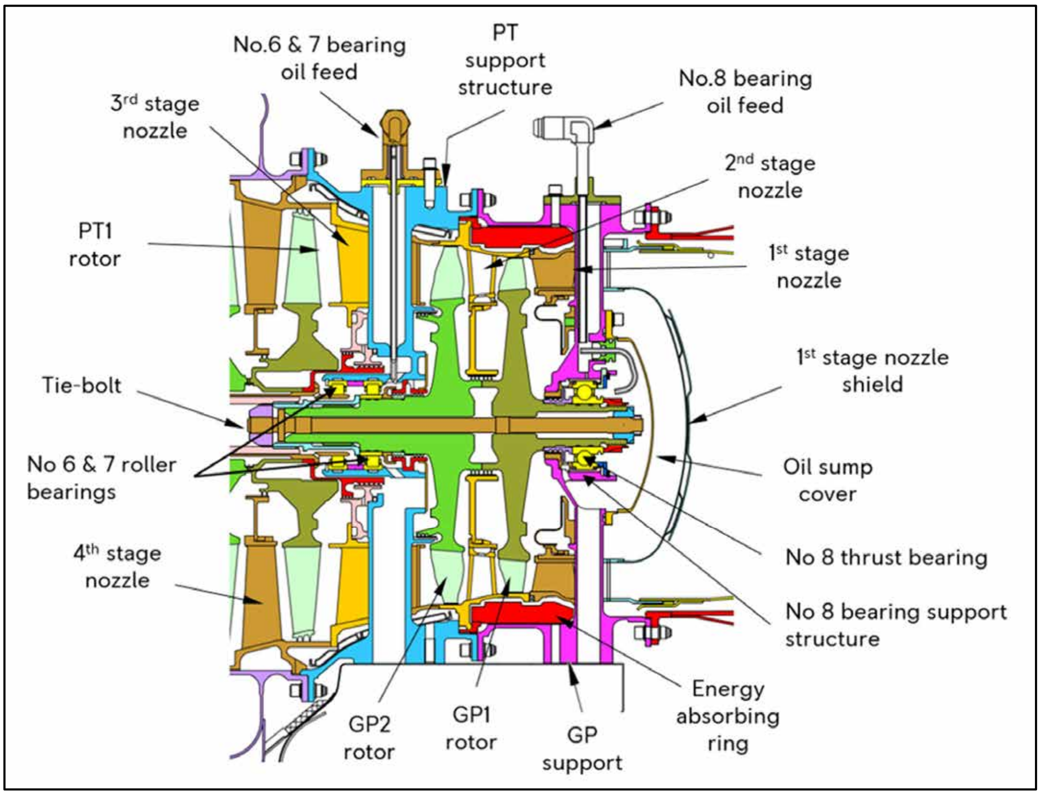

Figure 22

Rolls-Royce 250 C47B turbine section
(Image used with permission of Rolls Royce Corporation)

Figure 23

Energy absorbing ring (Front view)
(Image used with permission of Rolls-Royce PLC)

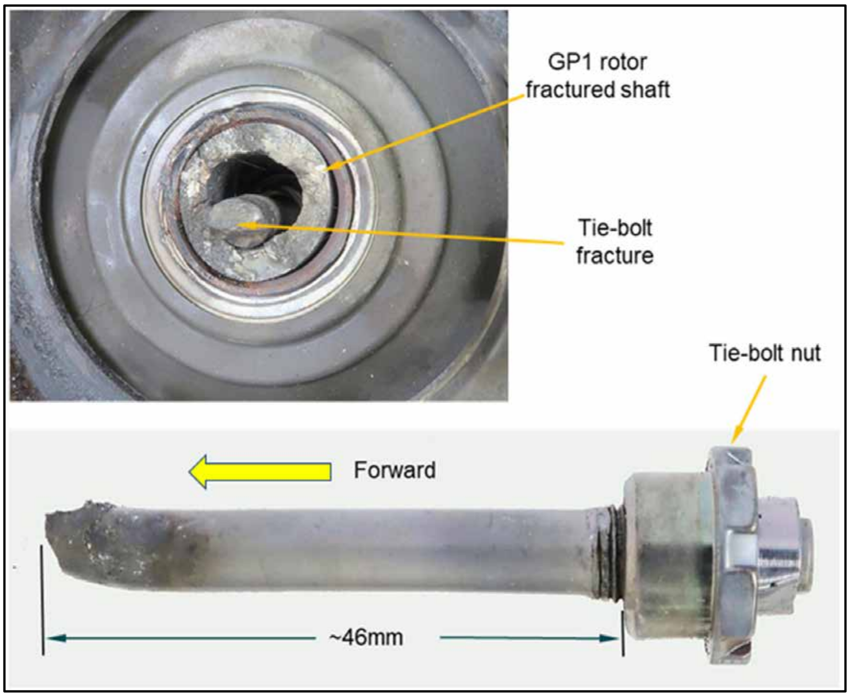

Figure 24

GP turbine tie-bolt fracture
(Image used with permission of Rolls-Royce PLC)

Assessment of the No 8 bearing support structure showed that four of the radial struts had fractured adjacent to the outer casing (Figure 25). The oil feed tube into the No 8 bearing chamber exhibited significant deformation caused by displacement between the hub and the outer case. The strut fractures were assessed as secondary damage as a result of high rotor loads during the engine breakup sequence with no evidence of the oil pipe having failed beforehand.

The No 8 bearing chamber appeared dry, with no evidence of oil or oil breakdown residues such as soot or carbon accumulation. To determine if the absence of oil was caused by a blockage in the oil feed tube, the assembly was removed for further examination and inspected using a CT scan[22]. This showed the oil nozzle, which directs oil from the strut chamber directly into the bearing assembly, was free from damage and was not blocked.

The No 8 bearing was found to be dry with no evidence of oil or carbon residues (Figure 26)[23]. The bearing outer race is designed with an axial split to enable the loading of the balls during assembly. The split in the race had opened indicating that the assembly is likely to have been subjected to significant loads during the event. The rear shoulder of the inner race also exhibited a large burr, which had formed from redistributed material from the raceway.

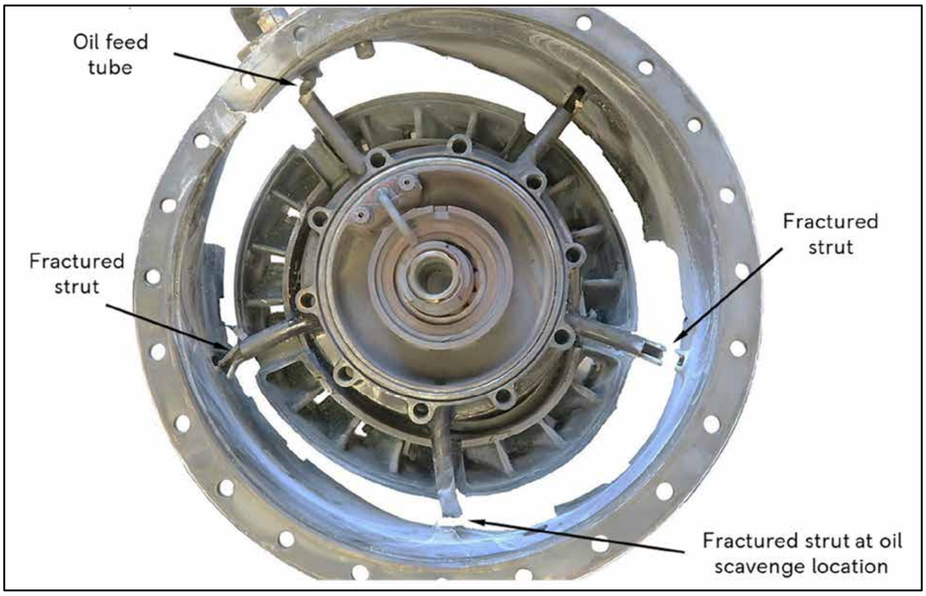

Figure 25

No 8 bearing support structure (rear view)
(Image used with permission of Rolls-Royce PLC)

Footnote

22 Computerised Tomography (CT) Scan.
23 Release fluid was used to remove the retaining nut from the GP1 shaft, which may give a wet appearance to the bearing surfaces in the images.

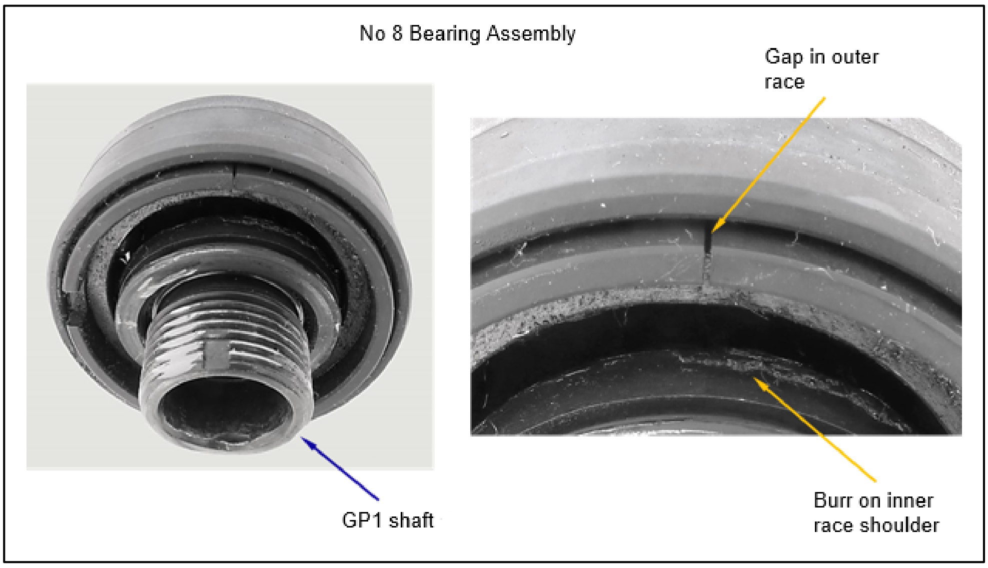

Figure 26

Damage to No 8 bearing assembly
(Image used with permission of Rolls-Royce PLC)

The bearing was removed with the remaining GP1 rotor shaft section seized in place. Detailed inspection found the cage had seized onto the outer race due to overheating. This type of damage typically occurs when the cage expands into the race while it is rotating. The designed split in the outer race was noted to gape, and likely to indicate excessive hoop stresses from the thermal expansion of the cage. The assembly was stripped to inspect the ball bearings. All the balls were misshapen with none of the original contact surface remaining. The inner race exhibited smearing on the rear lip indicating the balls were sliding rather than rolling over the circumference (Figure 27). Inspection of the outer race also noted a circumferential burr to the front lip. This would also correlate with the condition of the bearing degrading prior to the disc fragmentation.

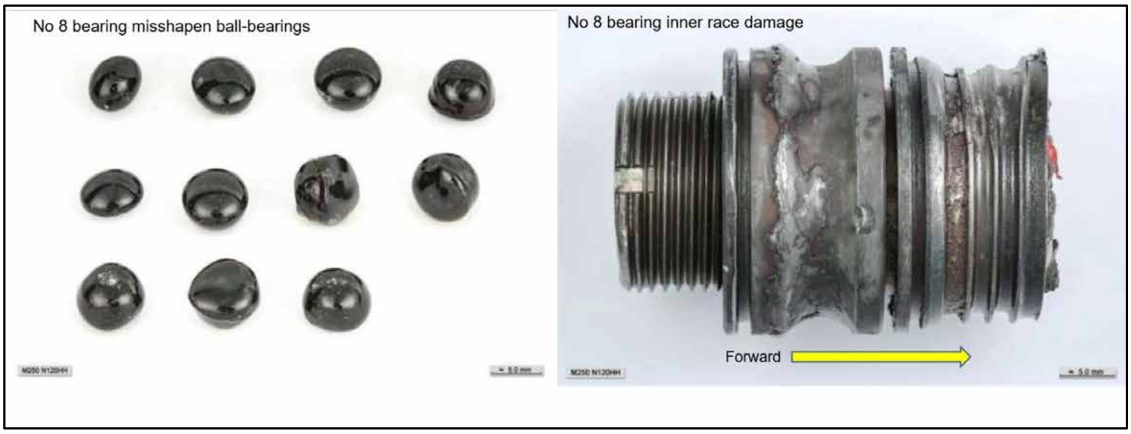

Figure 27

No 8 ball bearings and inner race
(Image used with permission of Rolls-Royce PLC)

The PT bearing support structure showed contact damage on its hub face. The axial aligned labyrinth air seal was displaced radially (Figure 28) from axial movement and orbiting of the rotating GP rotors. Both the No 6 and No 7 roller bearings had disintegrated, with no distinguishable material remaining other than remnants of the inner and outer race.

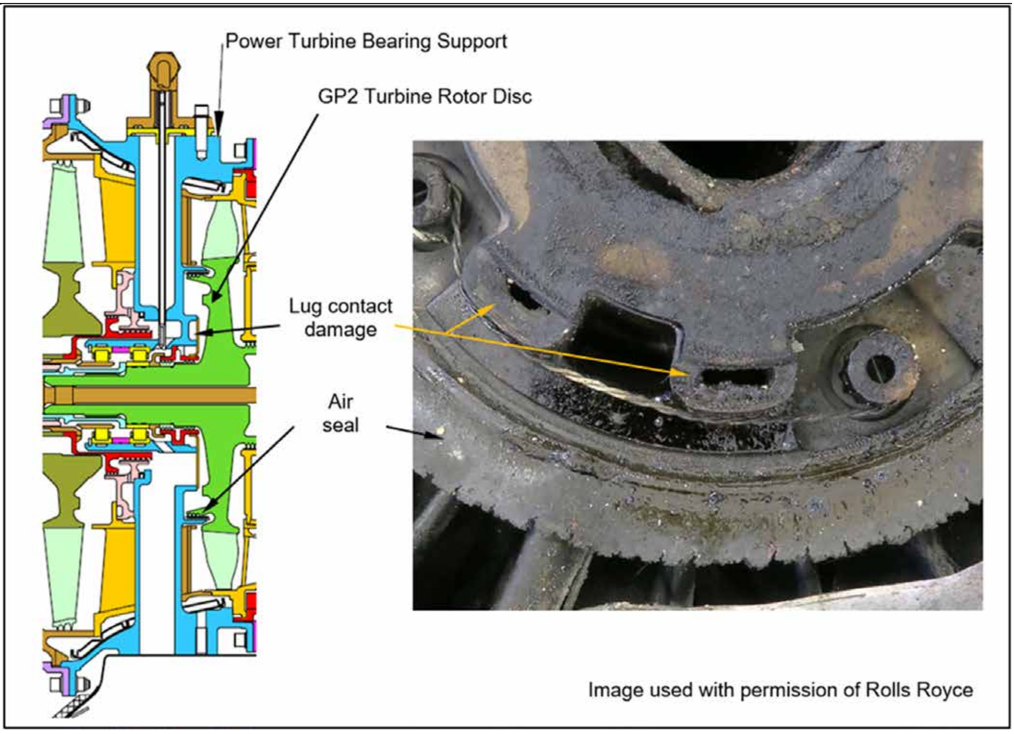

Figure 28

Power turbine support contact damage

The spline adaptor which transmits torque from the GP turbine assembly into the GP turbine to compressor coupling shaft had splayed radially indicating high temperature and forward movement of the GP turbine assembly (Figure 29).

The PT 1st stage rotor assembly had significant nicks and dents from impacts to its blades leading edges. The 2nd stage rotor assembly had no significant damage and the engine was in better condition moving forward towards the compressor. The No 5 bearing, on which the power turbine rotates, showed dark carbonated residues from the increased thermal environment during the event. There were no obvious signs of wear and the bearing assembly rotated freely.

There was no significant damage to the compressor impellor vanes and no evidence of impellor contact with the shroud. The No 1 roller bearing, which provides radial location of the compressor, was wet with oil and had no significant wear or distress to the bearing rollers. The No 2 location bearing, positioned at the rear of the impeller, was darkened with oil residues. The bearing rotated freely, with no obvious spalling or wear to the nine balls. The No 5 bearing assembly was free to rotate with no evidence of significant wear. The bearing outer surfaces were coated in lacquered oil and carbon, consistent with elevated working temperatures.

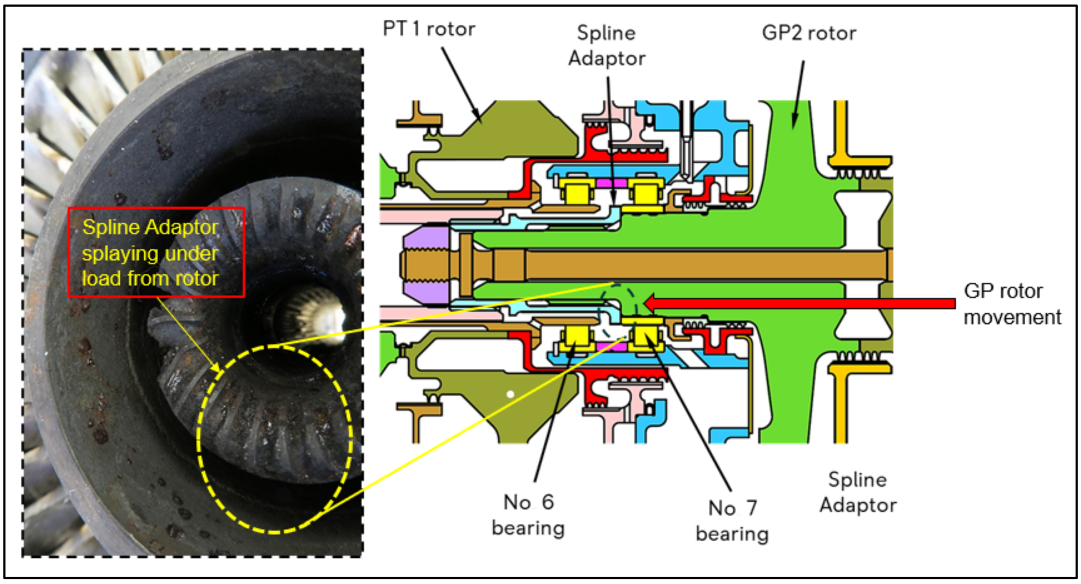

Figure 29

Spline adaptor damage
(Image used with permission of Rolls Royce)

Debris was found in the outer combustion case and compression discharge air tubes, analysis of this debris confirmed it was consistent with magnesium oxide, a bi-product of the burned magnesium engine accessory gearbox case which would have been generated in the post landing fire.

The engine accessory gearbox casing was significantly damaged by fire, exposing gear trains and the oil pump. The lower MCD assembly was recovered from the engine bay as the gearbox lug to which it attaches had fractured. The MCD itself was correctly located in the bayonet fitting. The end of the MCD probe was, however, missing so no chip material could be collected for analysis.

The upper MCD and its housing remained in place and the MCD was correctly seated. On removal the probe end was observed to be damaged so no chip material could be collected for analysis. The torque meter oil pressure transducer was recovered from the engine bay and while it had suffered fire damage, there was no evidence of leakage or loose connections. The gearbox structure to which the torque meter was attached had been destroyed in the fire. The N_P sensor probe was heat damaged and the internal wiring was protruding from the sensor tip.

Inspection of the remaining oil and fuel tube B-nut connections found no obvious pre-existing loose or leaking fittings, although, because of the heat damage, torque checks proved to be ambiguous.

Metallurgy

Approximately half of the GP1 disc was recovered from the engine bay, which comprised of eight separate fragments (not including the shaft) of varying sizes (Figure 30).

Figure 30

GP1 rotor disc fragments
(Image used with permission of Rolls Royce PLC)

A part of the disc extending over five blade sections was examined, although little of the blades remained and the surfaces were heavily smeared from contact with the surrounding static structures (Figure 31). Detailed inspection of the disc showed the features of failure to be those associated with overload only with no evidence of structural degradation due to overheating. Further analysis of the turbine blades showed evidence of grain structure degradation due to overheating induced by friction which is indicative of a heavy rub against another structure. The heat affected structure indicated frictional temperatures in the rub region were in excess of 1,150°C which was high enough to cause plastic deformation in the aerofoil structure.

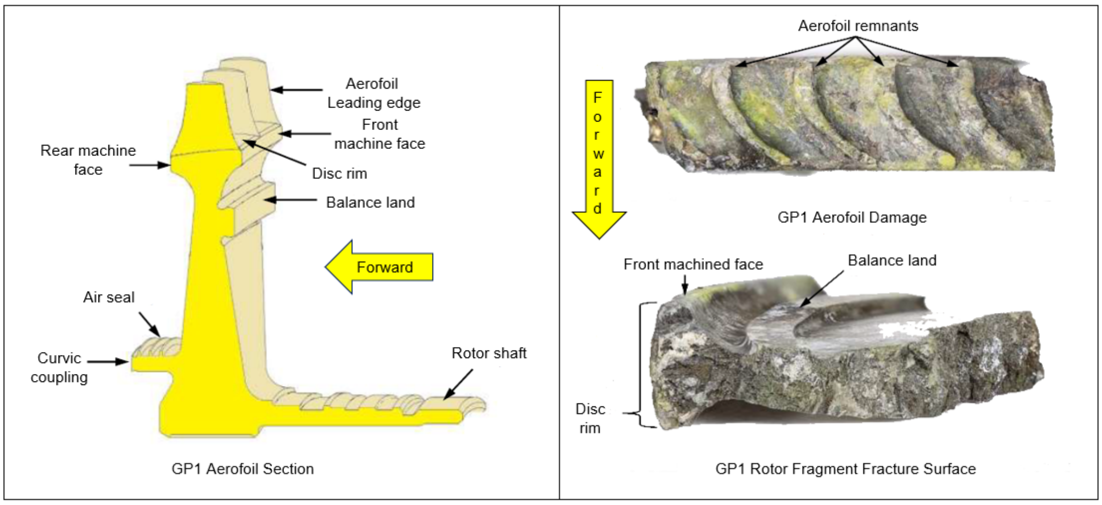

Figure 31

GP1 rotor disc fragment
(Image used with permission of Rolls Royce PLC)

Approximately one eighth of the GP2 disc was recovered (Figure 32). The largest fragment of rotor was examined and found to have failed in overload, with multiple radial cracks in the machined face of the rotor rim. The remnants of the turbine blade were found to have experienced similar heating effects to the blades on GP1.

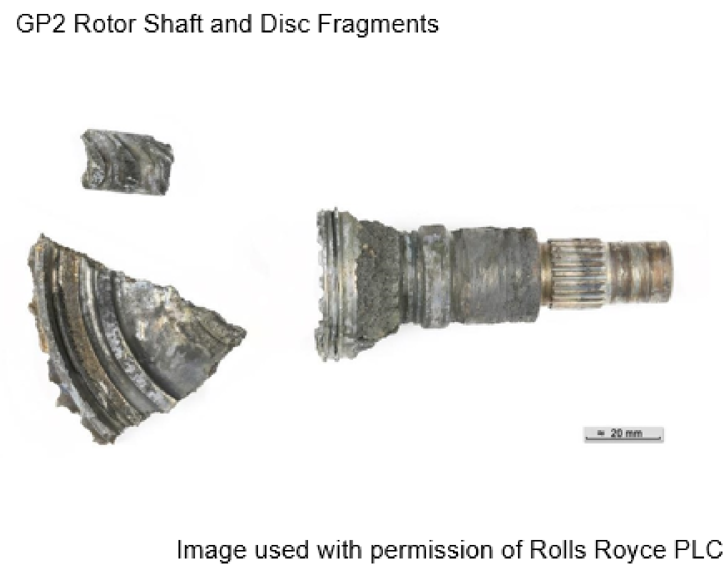

Figure 32

GP2 rotor fragments

Previous Events

A review of the Rolls-Royce 250-C47B service history identified seven previous oil loss events dating back to 2002 (Table 3), two of which, N407KH and 5N-BIC, resulted in uncontained turbine disc failures. In each event listed below, the following similar observations were made:

- Torque meter indication faults observed prior to the engine power loss.
- Oil deposits found externally on the aircraft fuselage or ground.
- Engine disassembly and inspection confirmed that an oil loss had resulted in significant damage to the GP rotor thrust bearings and caused associated secondary damage.

Date	Registration	Cause
2002	XA-RRV	Oil tank split
2005	N75SP	Upper MCD incorrect installation
2006	N407KH	Loose oil pressure transducer line – incorrectly installed
2009	5N-BIC	Loss of oil pressure
2015	C-FNOB	Upper MCD incorrect installation
2015	TG-JJV	Upper MCD incorrect installation
2019	DQ-HFH	Loose connection on No 6,7,8 bearing oil feed – incorrectly installed.

Table 2

Previous oil loss events

Pilot information

The pilot held an Airline Transport Pilot's Licence and a helicopter Private Pilot's Licence. He initially qualified in September 2012 on the Robinson R22 then completed a type rating for the Bell 407. The last revalidation of his type rating was carried out on 24 May 2017 and was valid until 31 May 2018. The pilot's logbook did not record a flight with an instructor, or that he had demonstrated an autorotation with an instructor since the revalidation flight on 24 May 2017 that was flown 155 hours prior to the accident flight. At the time of the accident the pilot had accumulated 516 hours in helicopters, 475 hours were on the Bell 407.

Autorotation

When drive to the main rotor is lost, the rotor rpm can be maintained by descending the helicopter in autorotation. This is where the relative airflow through the rotor system is used to drive the rotor rather than the engine. By maintaining the rotor rpm and sufficient forward airspeed, as the helicopter approaches the ground it will have enough energy for the pilot to arrest the rate of descent and land safely. If the rotor rpm drops below the minimum recommended rpm, there is a danger of a significant increase in the rate of descent and a lack of energy in the rotor system to arrest the descent before landing.

Once the helicopter has landed the collective can be lowered. The FAA Helicopter Instructor's Handbook[24] contains the following note:

> *'Numerous tail boom strikes have occurred due to improper collective pitch response on ground contact and completion of the manoeuvre. Particular attention must be emphasized as to the proper rate and timing of lowering collective to avoid potential damage to aircraft components.'*

Bell 407 Rotorcraft Flight Manual

Engine failure and autorotation

The Bell 407 RFM, *'Emergency /Malfunction Procedures'* states that the rotor rpm should be maintained between 85% and 107% during the autorotation; the maximum airspeed for steady state autorotation is 100 kt, the minimum rate of descent airspeed is 55 kt and the maximum glide distance airspeed is 80 kt.

Once established in the descent, if a restart is not attempted the procedure states the fuel valve switch should be switched to OFF and once at a low altitude the throttle should be moved to OFF. Once on the ground it suggests the pilot should complete the normal helicopter shutdown procedures. However, a note in the introduction of the Emergency / Malfunction procedures states:

> *'All corrective action procedures listed herein assume pilot gives first priority to helicopter control and a safe flight plan.'*

Engine fire

In the event of an engine fire in flight, the RFM recommends immediately entering autorotation. Once established in autorotation the pilot should move the throttle to the OFF detent, the fuel valve switch to OFF, the fuel boost pump and transfer pump circuit breaker switches to OFF and once landed, select the battery switch to OFF.

For an engine fire on the ground it states the pilot should select the throttle to the OFF position, the fuel valve switch, generator and battery switch to OFF; engage the rotor brake and exit the helicopter.

FADEC CWP lights

The pilot reported initially seeing an amber FADEC FAULT light on the CWP combined with a split between the N_R and N_P. Approximately 45 seconds later he saw an amber FADEC

Footnote

[24] https://www.faa.gov/regulations_policies/handbooks_manuals/aviation/media/FAA-H-8083-4.pdf (accessed 3 November 2020).

DEGRADED light. The guidance given in the Bell 407 RFM for the FADEC FAULT and FADEC DEGRADED lights is shown in Table 4. The RFM defines *'land as soon as practical'* as:

> *'Landing site and duration of flight are at the discretion of* [the] *pilot. Extended flight beyond [the] nearest approved landing area is not recommended'*

CWP Caption	Fault Condition	Corrective Action
FADEC FAULT	PMA and or MGT, N_P or N_G automatic limiting circuit(s) not functional.[25]	Remain in AUTO mode. Land as soon as practical.
FADEC DEGRADED (in flight)	FADEC ECU operation is degraded which may result in N_R droop, N_R lag, or reduced maximum power capability.	Remain in AUTO mode. Fly helicopter smoothly and non-aggressively. Land as soon as practical. Note It may be necessary to use FUEL VALVE switch to shut down engine after landing.

Table 3

RFM guidance for FADEC CWP lights

Engine underspeed

The indications of an engine underspeed are a decrease in N_G, a subsequent decrease in N_P, a possible decrease in N_R and a decrease in torque. Therefore, when the pilot saw the N_P had decreased to 90% with a drop in torque, he could have interpreted this as an engine underspeed. The guidance in the RFM[26] for an engine underspeed is shown in Figure 33.

'3-3-C. ENGINE UNDERSPEED *No caution/warning/advisory lights illuminated* • *INDICATIONS* 1. *Decrease in NG.* 2. *Subsequent decrease in NP.* 3. *Possible decrease in NR.* 4. *Decrease in TRQ.*	• *PROCEDURE* 1. *Collective – Adjust as required to maintain 85 to 107% NR.* 2. *Throttle – Confirm in FLY detent position.* 3. *Throttle – Position throttle to the approximate bezel position that coincides with the gauge indicated NG.* 4. *FADEC MODE switch -MAN.* 5. *NR – Maintain 95 to 100% with throttle and collective.* 6. *Land as soon as practical.'*

Figure 33

RFM guidance for engine underspeed

Footnote

25 PMA – Permanent Magnetic Alternator, MGT – Measured Gas Temperature.
26 Bell Helicopters Textron RFM, Section 3 Emergency/Malfunction Procedures.

Analysis

Overview

The helicopter was in level flight when a white smoke trail was seen coming from the area below the rotor. A few moments later the engine failed and the pilot flew an autorotation into a field. After landing, the main rotor blades severed the tail section and a fire took hold destroying the helicopter. Inspection of the engine established that there had been an uncontained failure of the GP turbine rotor discs.

There was no evidence that the landing was heavy or that the helicopter, other than the tail boom, was damaged during the touchdown.

Uncontained failure of the GP turbine rotor discs

Inspection of the engine revealed that there was no evidence of pre-impact damage to the centrifugal compressor; therefore, ingestion of a foreign object as an initiating factor for the failure of the discs was discounted.

While the GP1 rotor shaft and tie-bolt had fractured, and the curvic coupling had separated, this was assessed as secondary damage resulting from the breakup of the GP turbine discs. From the data recorded by the ECU, there was no evidence of an engine overspeed having occurred prior to the engine flame-out.

Metallurgy showed that both GP discs failed as a result of overload with no indication of fatigue or pre-existing damage. It is likely that the overload initiated in the rotor rims, where the highest loads would be felt during overspeed. This was corroborated by the multiple radial cracks in the machined face of the rotor rim. It is also likely that areas with heat affected microstructures due to frictional rubbing, such as the aerofoil stubs, would have assisted the initiation of cracking in the rim during the event.

Initiating factor for the release of the discs

There was physical evidence that the aerofoil sections on the GP rotors had rubbed on the inside of the engine casing, generating sufficient heat to cause plastic deformation. The GP2 rotor also made contact with the PT support structure resulting in damage to the labyrinth air seal and the spline adaptor, which transmits torque from the GP turbine to the compressor, and made contact with the power turbine inner shaft. It was assessed that this damage resulted from the forward and axial movement of the GP rotors and was not in itself an initiating factor.

Examination of the eight bearings, which support and locate the main rotating assemblies, revealed that the No 6, 7 and 8 bearings had been subject to high temperatures. The rollers on the No 6 and 7 bearings had disintegrated. The No 8 GP thrust bearing chamber was dry with no evidence of carbon deposits. The outer race had been distressed and there was evidence of the balls, which were badly deformed, sliding rather than rolling along the inner race. There was no evidence of an oil fire having occurred in or around the bearing cavities. From the condition of the bearings it was concluded that a bearing failure was the most likely initiating factor for the failure of the discs.

Cause of bearing failure

The No 8 bearing operates in the hottest part of the engine and the damage it sustained could be accounted for by either a mechanical failure or extreme overheating. However, the nature of the damage to the bearing was more typical of overheating.

The bearing relies on oil for lubrication and cooling, and a disruption to this supply could have caused the No 8 bearing to overheat and fail. When the FADEC FAULT light illuminated on the CWP, the pilot noted that the torque indication had dropped from the expected 70 to 75% range with no change in the N_R or helicopter performance. The torque meter relies on the engine oil pressure and a restriction or reduction in engine oil pressure would affect the reading.

The helicopter had recently completed its 300-hour maintenance schedule during which the oil flow checks were found to be satisfactory. Inspection of the No 8 bearing cavity after the accident did not identify any evidence of a carbon build up, which might restrict the oil flow. It is, therefore, unlikely that an oil restriction occurred.

Smoke trail

Witness reports and photographs of the helicopter showed white smoke coming from below the main rotor prior to the pilot seeing the FADEC FAULT caution light. The possibility of the smoke being caused by a fuel or oil leak were considered. The pilot did not report any abnormal fuel consumption and the engine was running normally up to the point of failure, so a fuel leak was considered unlikely. However, a significant amount of unburnt engine oil was present on the tail boom. The pilot confirmed that the oil level was showing as FULL during his pre-flight checks and there was no evidence of oil leakage on the outside of the helicopter. It is therefore probable that during the accident flight the helicopter developed an oil leak which caused the smoke trail and led to the under reading of the torque meter. The fact that the oil on the tail boom was not thermally degraded indicates that the leak most likely occurred prior to the uncontained engine failure and consequent fire.

The aircraft had neither CWP or audible warnings to alert the pilot to the decreasing oil pressure and rising temperature. Therefore, he would only have noticed a change in these parameters when he scanned the oil temperature and pressure gauge, which was mounted at the bottom of the instrument panel. The pilot did not report seeing any abnormal oil temperature or pressure indications prior to the engine failure. Without knowing the rate of the oil leak, it is not possible to determine how quickly these parameters might have changed.

Source of oil leak

The major components in the oil system bay, most of which are of aluminium construction, were destroyed in the fire. The magnesium fire destroyed many of the interfaces between the oil pipes and the pumps, valves and filters in the system. The connections that remained were found to be intact and retained integrity despite the accident.

The main evidence of engine oil loss was found on the tailboom. Due to the turbulent airflow round the helicopter and effect of the main rotors, it was not possible to determine the source of this oil from inspection of the airframe. The oil could have been lost from the engine bay and flowed out of the drains and been picked up in the airflow; or could have been lost in the oil system bay and been distributed by the oil cooler fan through the upper vents into the air flow and deposited onto the rear of the helicopter. Similarly, a failure in the oil cooler matrix would have caused oil to have been blown into the path of the airflow and onto the tailboom.

Other than routine oil flow checks, there was no other maintenance activity that indicated significant disturbance of the oil system, such as major component replacement. Post-maintenance checks revealed no abnormal oil system indication and the helicopter flew five uneventful flights prior to the accident flight. Because the oil system operates at a relatively high pressure, a leak from a small diameter hole could distribute significant amounts of oil in a short period of time.

Given the extent of the fire which destroyed most of the oil system components, it was not possible to determine the source of the oil leak.

Failure sequence

With insufficient oil to lubricate and cool the No 8 bearing it would have overheated and started to fail, eventually causing the bearing to lose its ability to provide axial and radial location of the GP turbine shaft. This axial movement of the shaft would have caused contact between the GP turbine splined adaptor and the PT drive shaft resulting in frictional heating. The spline adaptor was splayed radially by this contact indicating that the GP rotor was still intact and generating load, which is further evidence that a failure in the GP assembly was not the initiating event.

As the spline adaptor splayed, the GP rotor would have decoupled from the compressor allowing the turbine to momentarily overspeed and orbit off-centre. The increasing temperature, lack of lubrication and change in load on the No 6 and 7 bearings would have resulted in their destruction as they countered the high-speed orbiting of the GP turbine shaft. While the No 6 bearing supports the PT, it is located in very close proximity to the No 7 bearing and would have been similarly affected by the high temperatures and loads.

Damage to the PT bearing support structure and aerodynamic blades show that the GP rotors made contact with the static structure. The heating effects from the friction resulted in plastic deformation of the blade material. This would have initiated cracks on the rims of the discs where the loads were highest. The overspeed would have accelerated rapidly until the stresses in the turbine rotors were too great and resulted in failure of the discs in overload. While the GP2 rotor is designed to have a higher burst speed than GP1 (such that once GP1 bursts GP2 would normally decelerate), the contact with the PT support structure due to the axial movement of the GP rotor had the effect of reducing the GP2 burst speed. The effect was a near simultaneous release of both GP rotor discs.

Containment

14 CFR Part 33 requires the containment of ≥80% of a single aerofoil only, with no specific requirements for the rotor disc itself. The engine failure on N120HH involved the loss of multiple aerofoils, along with the fragmentation and release of significant areas of disc material, which was beyond the scope of the containment requirements. It was considered that the absorbing ring had fulfilled its intended purpose of acting to attenuate much of the energy from the GP1 rotor, with considerably more GP1 rotor fragments recovered during the investigation than from the adjacent GP2 rotor.

During certification it was demonstrated that in a pure overspeed scenario there is a margin between the GP1 and GP2 rotor burst speeds. The N120HH event was outside of the Rolls Royce design intent in that the loss of load event occurred in conjunction with the axial translation of the GP rotor. The release of the rotor fragments most likely contributed to the fire starting as there was no evidence to suggest the onset of a fire in the engine compartment or airframe prior to the engine failure.

The accident sequence has shown that a loss of oil can lead to a burst of the GP2 rotor, the debris of which would not have been attenuated by the energy absorbing ring. Because the axial movement of the GP turbine is outside of the Rolls Royce design intent and certification criteria, the Failure Modes and Effects Analysis for the engine may not fully capture the linkage between loss of engine oil and uncontained bursting of the GP2 rotor. Therefore, the following Safety Recommendation is made:

Safety Recommendation 2021-047

It is recommended that Rolls-Royce Corporation includes the scenario of a loss of engine oil leading to the uncontained failure of both Gas Producer Turbine Discs in the Failure Mode and Effects Analysis for the Rolls-Royce M250 Series 4 engines.

Helicopter fire

The helicopter had not been fitted with the optional fire detection and warning system, so the pilot and passenger where not aware of the fire until after the helicopter had landed. The time from the engine failure to an extensive fire taking hold on the ground (Figure 6) was approximately 2.5 minutes, which is within the 5 minute minimum requirement specified in 14 CFR Part 27. On this occasion the time was sufficient to complete the autorotation and for the pilot and passenger to safely vacate the helicopter.

The precise source of the fire and when it started could not be positively determined. The most likely source was from the released GP rotor fragments severing oil and or fuel pipes located adjacent to the casing breach. There was no evidence that the firewalls or puncture resistant fuel bladders were breached. The opening in the engine casing would have enabled hot combustion gases and released material to enter the engine bay, providing an ignition source for fluid leaking from the severed pipes.

The data recorded by the ECU shows that following the flameout the PLA / throttle twist grip remained in the flight range. This would have made little difference in sustaining the fire as the engine quickly wound down and the HMU check valve would have operated preventing fuel flowing to the engine spray nozzle.

However, the fuel pipe between the HMU and nozzle had been severed and high pressure fuel would have initially continued to flow out of the HMU and into the engine bay until engine rotation ceased.

As the fuel valve switch was most likely still selected ON after landing, and as the fuel pumps were connected directly to the aircraft batteries, fuel would have continued to be fed to the HMU. While the HMU check valve should have prevented fuel flowing out of the severed pipe to the spray nozzle, other fuel system components in the engine bay, potentially vulnerable to damage from the disc burst or the initial fire, could have been leaking fuel into the fire. This leakage of fuel is consistent with the pilot's report of a fluid running down the windscreen after he landed.

The manufacturer highlighted that the requirement to shutoff the fuel valve is already captured in the emergency procedures published in Section 3 of the Bell 407 RFM and covers the safety measures necessary for the pilot to safely land the helicopter in the event of an engine failure or in-flight fire. Both emergency conditions require the throttle to be rolled to the OFF position, followed by switching the fuel shutoff valve OFF. However, to provide more information to pilots on the importance of switching off the fuel valve and moving the throttle to the OFF dent, following an engine failure, the manufacturer has taken the following safety action:

> The manufacturer is revising the data section of the Bell 407 Rotary Flight Manual to add the following clarification in section 1-25-D for the Fuel Shutoff Valve: '*Shutoff valves isolate fuel from the engine compartment during shutdown or in the event of an engine fire*'.

The requirements for flammable fluid protection in 14 CFR Part 27.863 require rotorcraft components that are critical to safety of flight to withstand fire and heat, and to minimize the hazard of fuel fires to occupants following an otherwise survivable impact or crash landing. There are also specific fire resistance requirements specified for the powerplant in 14 CFR Part E. Because of the extensive fire damage, it was not possible to determine the cause of the fluid leak; however, it is likely the fluid running down the windscreen was fuel. A possible source could have been from the damaged airframe mounted fuel filter located in the engine bay which was found to have fractured at its base. As the investigation was unable to determine why the filter fractured, the manufacturer undertook the following safety action and was satisfied that the filter met the applicable regulations:

> Bell Helicopters Textron undertook an analysis of the failure of the Bell 407 airframe mounted fuel filter to show that it retains sufficient crashworthiness properties whilst meeting the applicable fire resistance requirements laid out in 14 CFR Part 27.1183 for the component.

The guidance for 14 CFR Part 27 does not appear to clearly articulate the need for components to be able to meet their crashworthiness requirements after exposure to fire and heat. Given the potential for emergencies and crash landings involving fire the following Safety Recommendation is made:

Safety Recommendation 2021-048

It is recommended that Transport Canada assess its guidance material for 14 CFR Part 27 and Part 33 requirements on fire resistance and crashworthiness such that fuel system components certified to be fire resistant also retain sufficient residual material integrity to meet their crashworthiness requirements.

Cockpit indications & pilot actions

The first indication the pilot saw in the cockpit was a FADEC FAULT light on the CWP. This probably corresponds with the secondary N_P fault. The pilot recalled that this occurred approximately 70 seconds before the engine failed. Because the ECU recorded list of time-stamped faults was full, and faults from this period were overwritten, it was not possible to correlate the recalled events with a recorded fault. The pilot also recalled looking at the torque gauge and seeing a "5" but he could not remember if it was fifty-something or just five. The ECU snapshot data shows the torque was reading 4 - 6% just prior to the failure so this could correspond with the 5% the pilot recalled.

The pilot recalled that the N_P had dropped to 90% while the N_R remained constant at 100%. As N_P cannot be different to N_R (while the engine is driving the rotor normally) the constant difference seen on the cockpit gauge and the N_P faults on the ECU data were likely the result of misalignment or damage to the N_P probe sensor. The damage observed in the coils of the probe have the potential to alter the accuracy of the sensor. It was not possible to determine what had caused the damage to the sensor, but it is possible that the increased thermal build up in the bearing cavity caused the breakdown of the probe sensor elements.

The RFM response to a FADEC FAULT light is to remain in AUTO and '*land as soon as practical*'. However, due to the drop in N_P the pilot thought he was seeing an engine underspeed. The manual states the indications of an underspeed are '*decrease in N_G, subsequent decrease in N_P, possible decrease in N_R and a decrease in torque*'. The actions for the underspeed require the pilot to switch to MAN and to manually control the N_R with the twist grip throttle. Given the conflicting indications the pilot decided to select MAN and confirm he could still control the engine then return to AUTO. There was no evidence that these actions had any adverse effect on the engine. The ECU did not record any exceedances during this time.

Shortly after returning to AUTO the pilot recalled that the FADEC DEGRADED light illuminated in addition to the FADEC FAULT light. Both the pilot and passenger had a clear memory of having just two lights illuminated at this stage. It is possible that this corresponds with the primary and secondary N_P fault recorded in the ECU. The primary and secondary N_P fault would cause the FADEC DEGRADED light to illuminate but it should also illuminate the RESTART FAULT light. The investigation did not determine a reason why this third caution light was not seen.

The helicopter has an engine oil temperature and pressure gauge located centrally at the bottom of the instrument panel. It does not have any caution or warning lights associated with engine oil. When the initial indications occurred, the pilot checked the other engine instruments and did not recall seeing anything else abnormal other than an unusual torque indication. This accident is the third occurrence where a loss of oil has been identified as leading to an engine rotor disc failure on the Bell 407 helicopter. Therefore, to increase pilot awareness that torque indication faults might be a symptom of an oil leak, the manufacturer has taken the following Safety Action:

> The helicopter manufacturer added information to the supplementary material to the Bell 407 Rotary Flight Manual to inform pilots that unusual torque indications might be the result of a loss of engine oil.

Autorotation

Following the engine flameout, the pilot executed an autorotation into a field. During the descent his focus was on flying the helicopter to a safe landing, he was not aware of the fire and said he did not have time to switch the fuel valve OFF.

The pilot reported that he “fully” lowered the collective when he entered the autorotation’. The data showed that the collective was lowered within four seconds of the flameout and the N_R dropped to 69% before recovering to 82%. During the autorotation the N_R fluctuated between 68% and 82% and the airspeed remained between 75 and 82 kt. While the airspeed was within the range specified in the RFM, the N_R was below the specified range of 85 to 107%. The collective was not in the fully lowered position throughout the autorotation; lowering the collective would have increased the N_R. The risk in the N_R dropping below the minimum speed is that it reaches a point where the rpm becomes unrecoverable, the rate of descent increases and there is insufficient kinetic energy in the rotor system to perform the flare and landing.

The pilot recalled hearing a continuous horn during the autorotation whereas the passenger recalled an intermittent horn. The helicopter generates a pulsing horn to indicate an engine flameout and a continuous horn to indicate low rotor rpm when the N_R is below 95%. It is likely that the intermittent horn was audible when the engine initially failed and that this was quickly followed by the continuous low rpm horn when the N_R dropped. It is also likely that the low rpm horn remained on throughout the autorotation. A possible explanation for the pilot’s and passenger’s different recollection is that they remembered different parts of the failure sequence.

Once the helicopter had landed the pilot recalled “dumping” the collective lever as quickly as possible. It is likely that this rapid action resulted in the rotor blades contacting the tail boom and causing it to separate. There was no evidence that the tail had contacted the ground during the flare or landing.

The pilot was not aware of the fire during the descent and said he did not have time to switch the fuel valve OFF. Once on the ground, the passenger alerted him to the smell of smoke and his priority was then to ensure that he and his passenger got out quickly and safely. He did not have time to select the fuel or any electrics OFF prior to exiting the helicopter.

The time from the engine failing and the successful landing was approximately 60 seconds, during which the pilot's focus was on flying the helicopter and selecting a field. While the pilot had not flown with an instructor or practiced an engine failure and autorotation for just over three years prior to the accident flight, it was not possible to establish if this was a factor in the handling of the helicopter during the descent.

Conclusion

The investigation concluded that the uncontained failure of the gas producing turbine rotors occurred due to insufficient oil being supplied to the No 8 main engine bearings. The lack of oil was the result of a leak that occurred during the accident flight. Due to the extent of the fire damage to the helicopter it was not possible to establish the source of the leak.

The investigation established that the gas producing rotors were not contained because the engine was not required to be designed or certified to contain such a catastrophic failure.

The uncontained failure resulted in damage to the fuel and oil pipes in the engine compartment, and the leaking fluids ignited in the presence of the hot components. As the helicopter fuel valve remained open, and the booster pumps continued to run after the occupants had left the helicopter, fuel continued to be pumped into the engine compartment through the damaged component further feeding the fire.

As there was no fire detection system fitted to the helicopter, and because of the extensive fire damage, it was not possible to establish if the helicopter met the fire protection certification requirements. However, the protection offered by the engine bay fire walls enabled the pilot to make a controlled landing and the occupants to safely vacate the helicopter.

Although, the N_R was below the speed specified in the RFM during the autorotation, the pilot was able to land safely. The rapid lowering of the collective after landing resulted in a main rotor blade striking and severing the tail boom.

Safety Recommendations

The following Safety Recommendations were made:

Safety Recommendation 2021-047

It is recommended that Rolls-Royce Corporation includes the scenario of a loss of engine oil leading to the uncontained failure of both Gas Producer Turbine Discs in the Failure Mode and Effects Analysis for the Rolls-Royce M250 Series 4 engines.

Safety Recommendation 2021-048

It is recommended that Transport Canada assess its guidance material for 14 CFR Part 27 and Part 33 requirements on fire resistance and crashworthiness such that fuel system components certified to be fire resistant also retain sufficient residual material integrity to meet their crashworthiness requirements.

As a result of this accident the manufacturer undertook the following safety actions:

- A revision of the data section of the Bell 407 Rotary Flight Manual to add the following clarification in section 1-25-D for the Fuel Shutoff Valve: *'Shutoff valves isolate fuel from the engine compartment during shutdown or in the event of an engine fire'*.

- An analysis of the failure of the Bell 407 airframe mounted fuel filter to show that it retains sufficient crashworthiness properties whilst meeting the applicable fire resistance requirements laid out in 14 CFR Part 27.1183 for the component.

- Added information to the supplementary material to the Bell 407 Rotorcraft Flight Manual to inform pilots that unusual torque indications might be the result of a loss of engine oil.

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