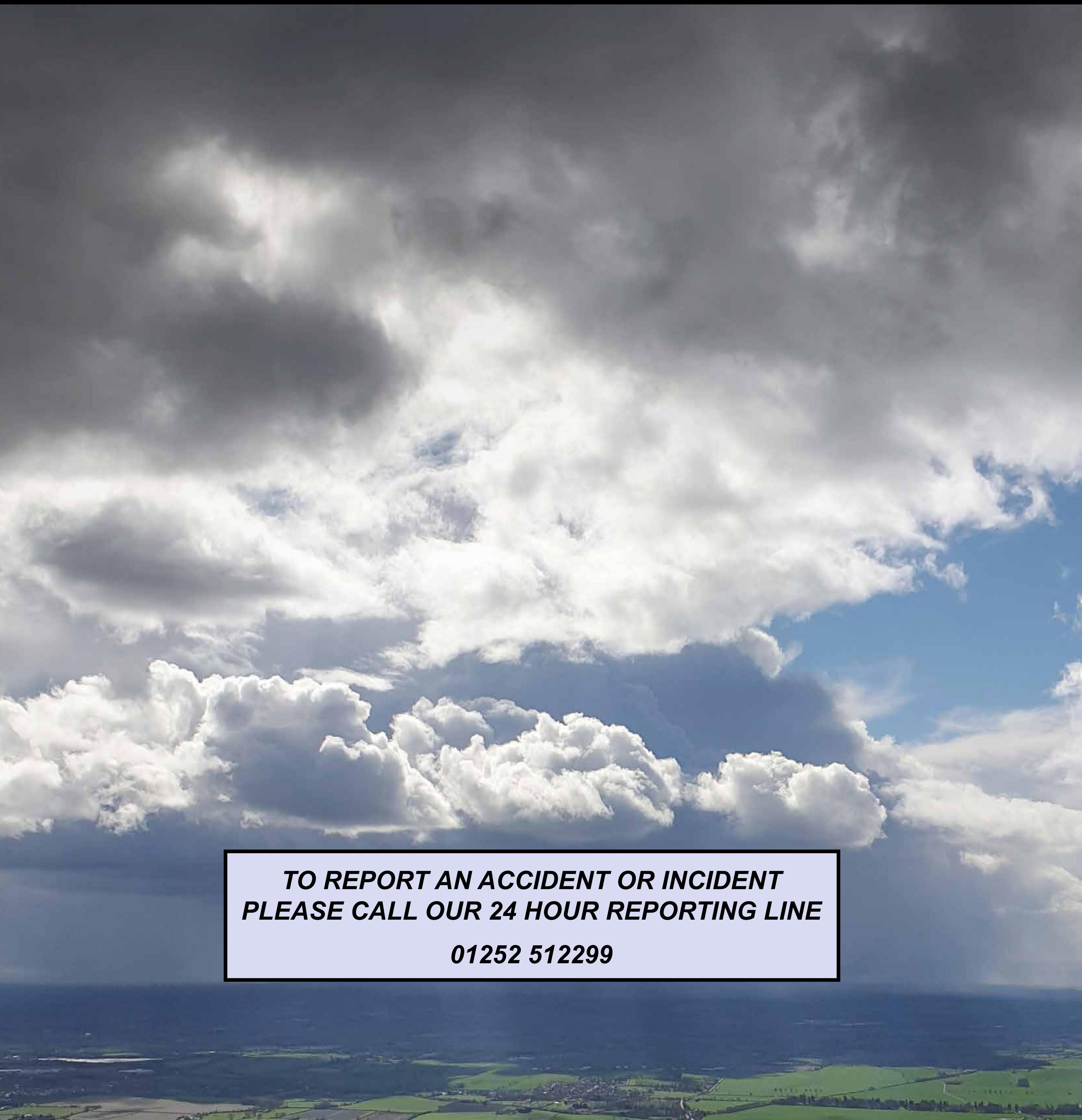
AAIB
Air Accidents Investigation Branch

AAIB Bulletin 11/2021



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01252 512299

Air Accidents Investigation Branch
Farnborough House
Berkshire Copse Road
Aldershot
Hants GU11 2HH

Tel: 01252 510300
Fax: 01252 376999
Press enquiries: 0207 944 3118/4292
http://www.aaib.gov.uk

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CONTENTS

SPECIAL BULLETINS / INTERIM REPORTS

None

SUMMARIES OF AIRCRAFT ACCIDENT ('FORMAL') REPORTS

None

AAIB FIELD INVESTIGATIONS

COMMERCIAL AIR TRANSPORT

FIXED WING

None

ROTORCRAFT

Bell 407	N120HH	24-Jun-20	3

GENERAL AVIATION

FIXED WING

Cirrus SR22	G-CTAM	31-May-20	46

ROTORCRAFT

None

SPORT AVIATION / BALLOONS

None

UNMANNED AIRCRAFT SYSTEMS

Tekever AR5 Evolution Mk 2	G-TEKV	29-Dec-20	68

AAIB CORRESPONDENCE INVESTIGATIONS

COMMERCIAL AIR TRANSPORT

None

GENERAL AVIATION

Corby CJ-1 Starlet	G-CBHP	02-Jul-21	87
Grob G109	G-CLIA	28-May-21	89
Robinson R44 II	G-OPDG	22-Jul-21	95
Rotorway Executive 162F	G-OHOV	31-Mar-21	98

SPORT AVIATION / BALLOONS

None

CONTENTS Cont

AAIB CORRESPONDENCE INVESTIGATIONS Cont

UNMANNED AIRCRAFT SYSTEMS

Brian Taylor AT6	N/A	19-Apr-21	101

RECORD-ONLY INVESTIGATIONS

Record-Only UAS Investigations reviewed:	August / September 2021	105

MISCELLANEOUS

ADDENDA and CORRECTIONS

None

List of recent aircraft accident reports issued by the AAIB 109

(ALL TIMES IN THIS BULLETIN ARE UTC)

AAIB Field Investigation Reports

A Field Investigation is an independent investigation in which AAIB investigators collect, record and analyse evidence.

The process may include, attending the scene of the accident or serious incident; interviewing witnesses; reviewing documents, procedures and practices; examining aircraft wreckage or components; and analysing recorded data.

The investigation, which can take a number of months to complete, will conclude with a published report.

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.

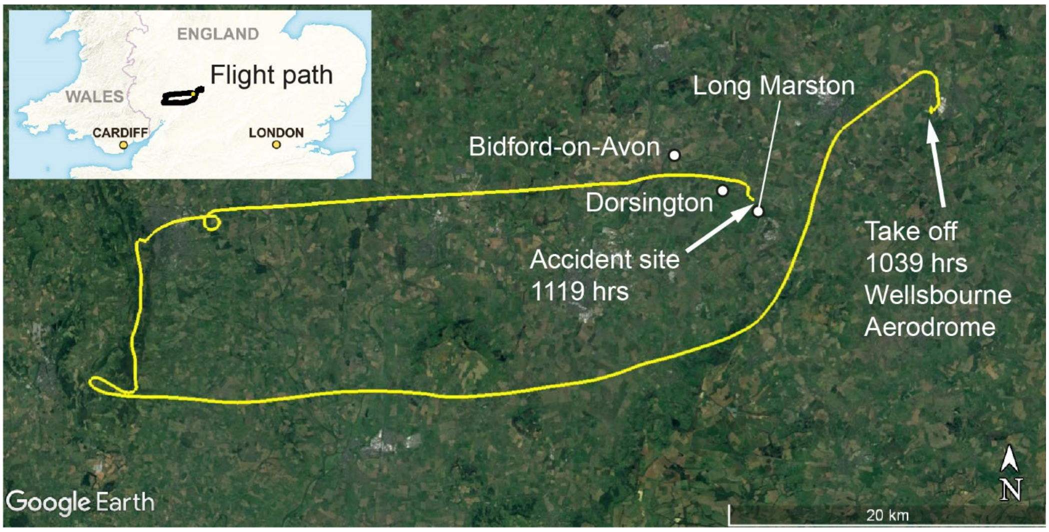


Figure 2

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.

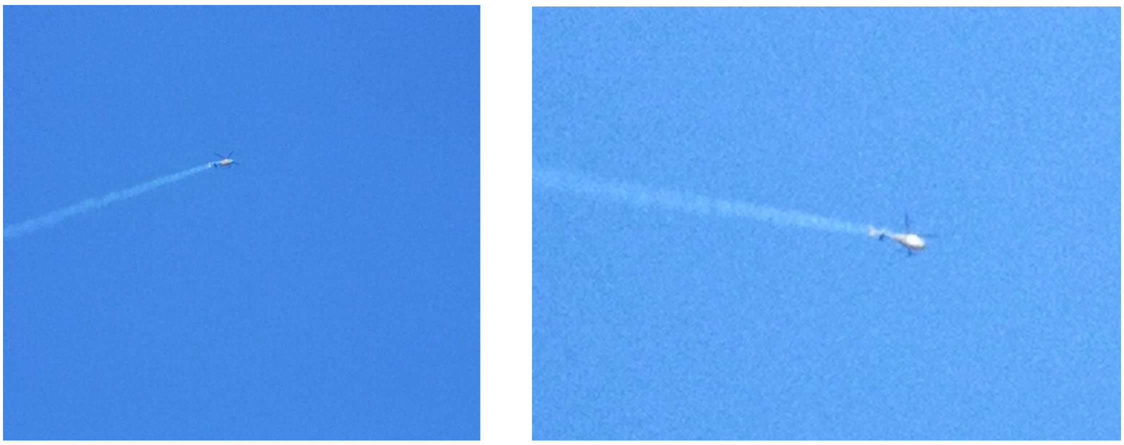


Figure 3

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.

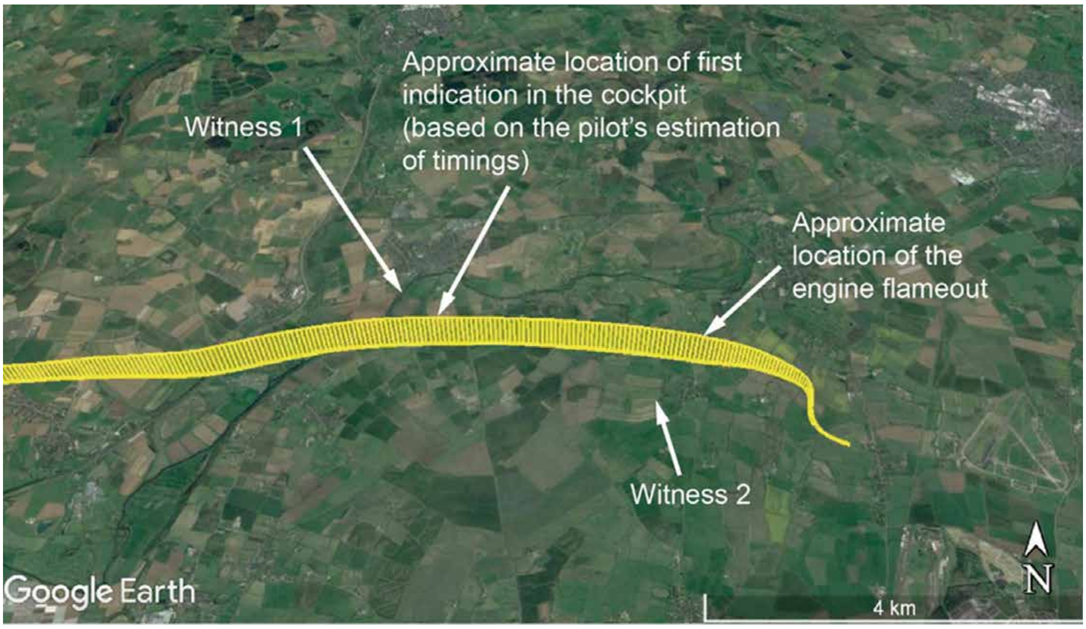


Figure 4

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).

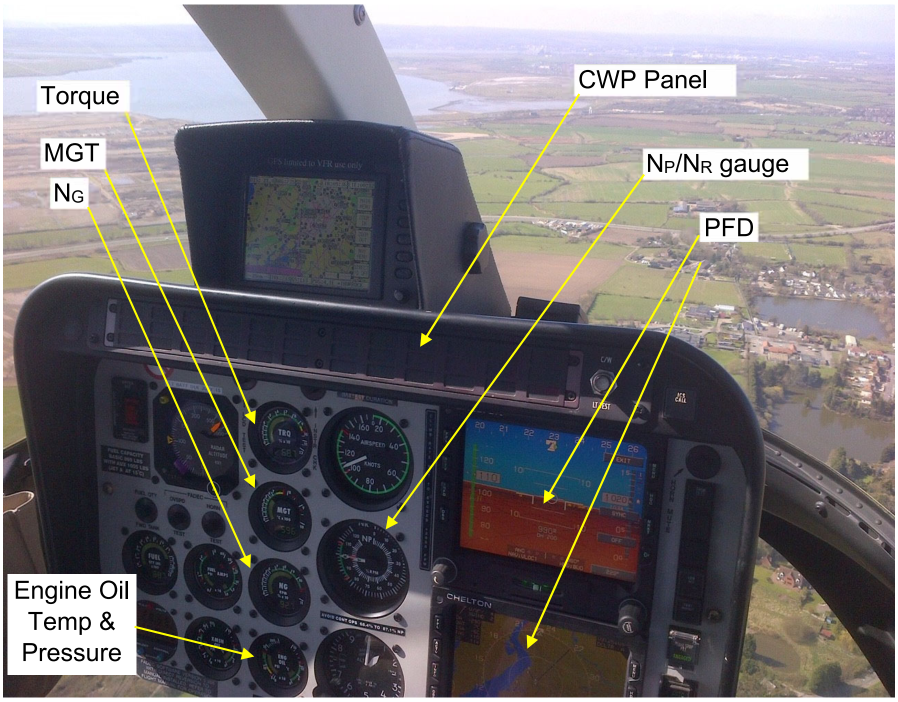


Figure 9

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).

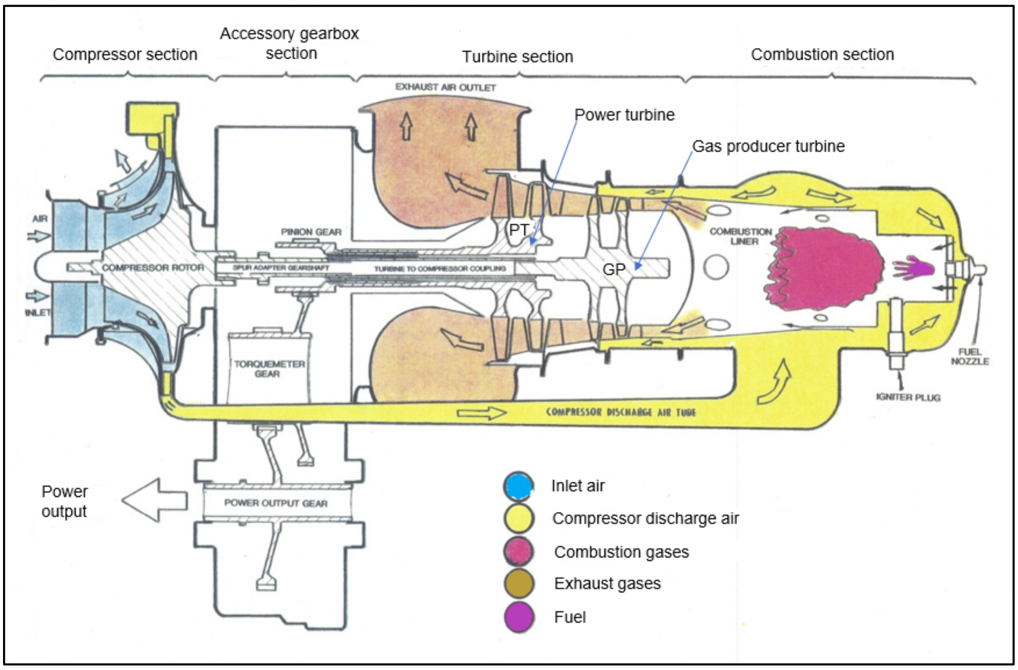


Figure 10

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.

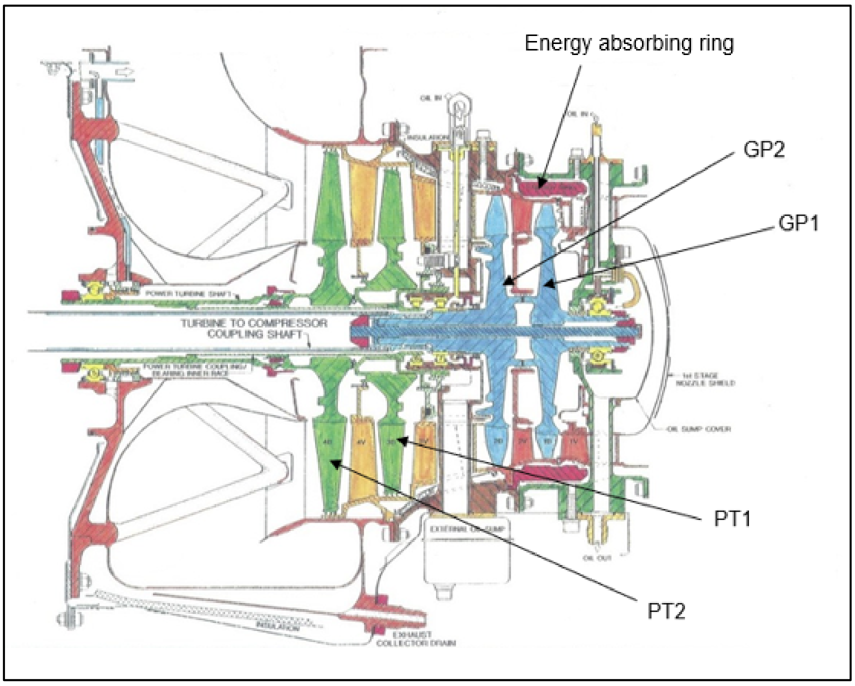


Figure 11

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.

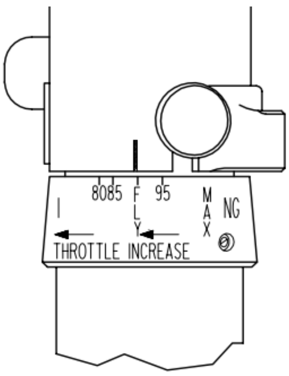


Figure 12

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.

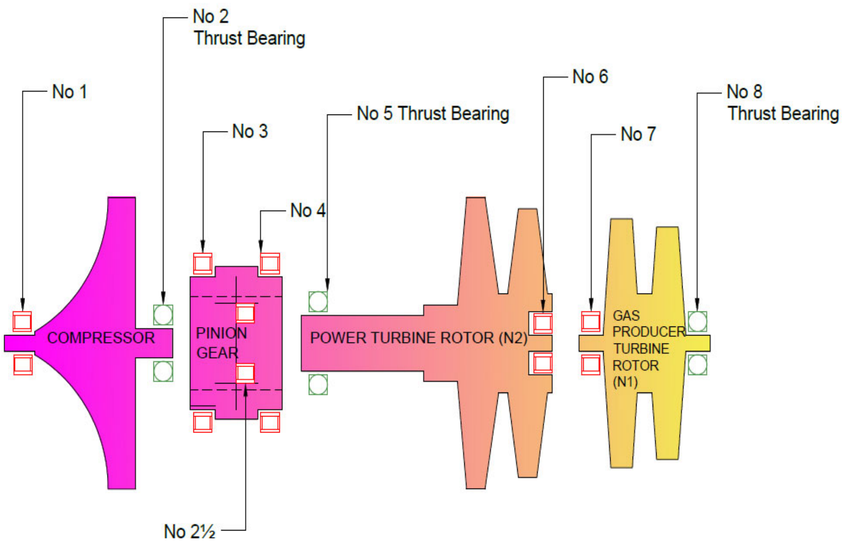


Figure 13

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.

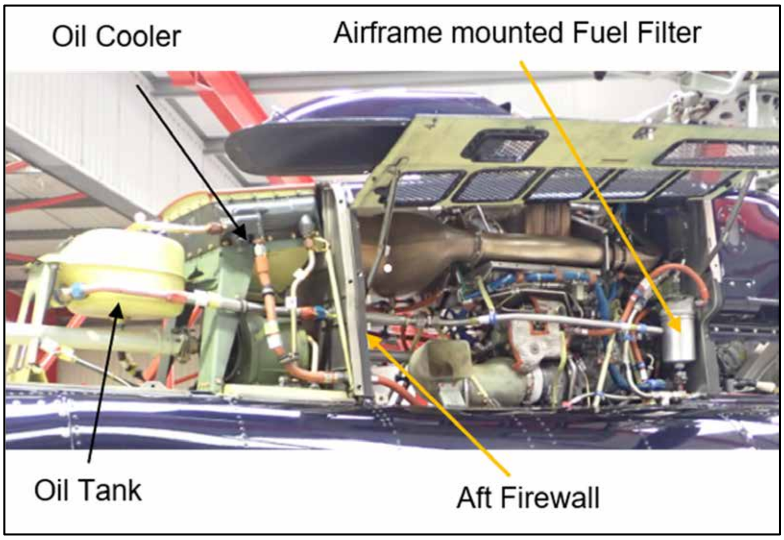


Figure 14

Bell 407 engine oil system compartment

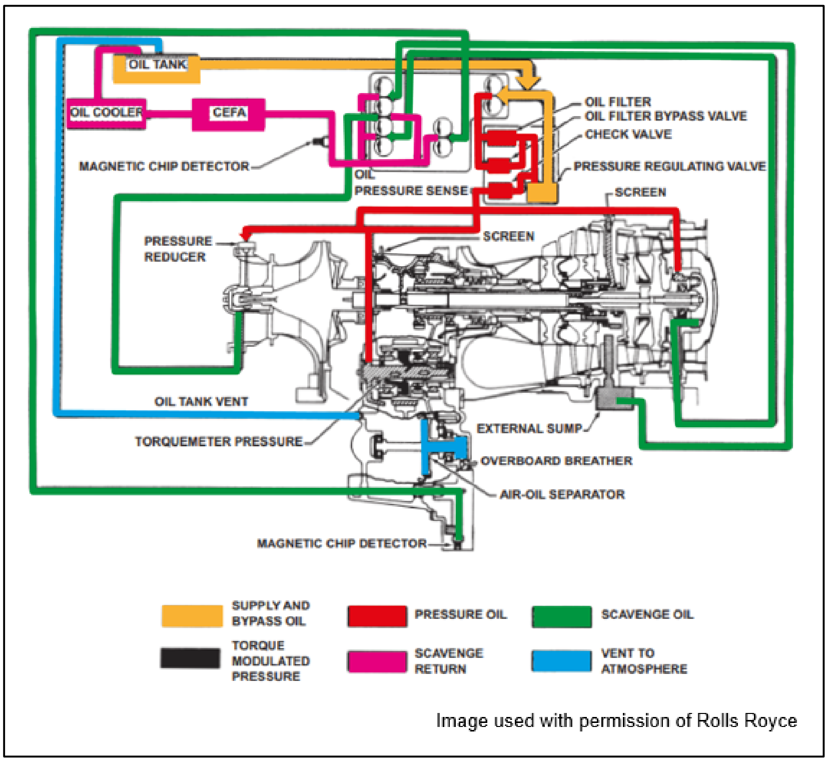


Figure 15

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.

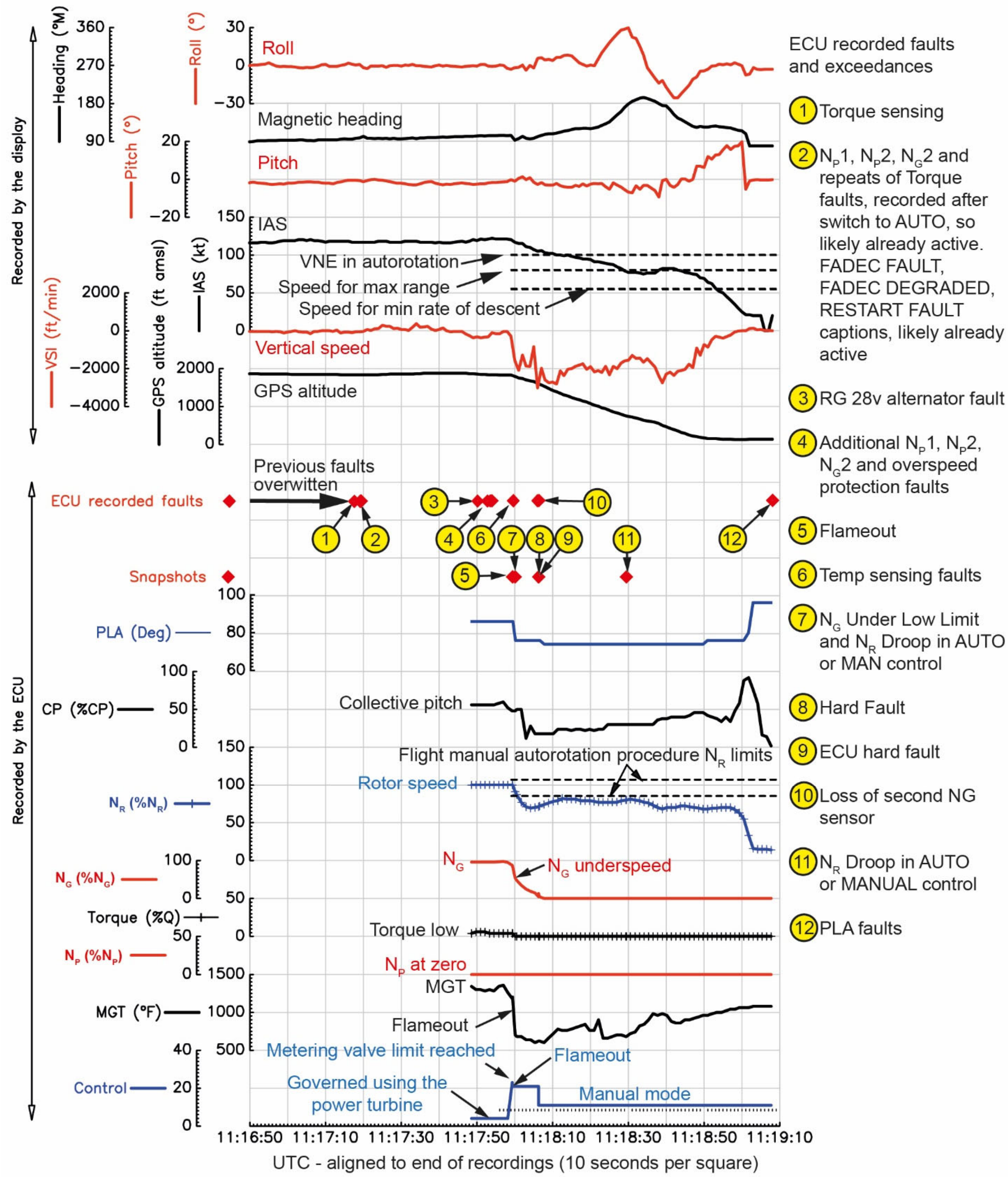


Figure 17

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.

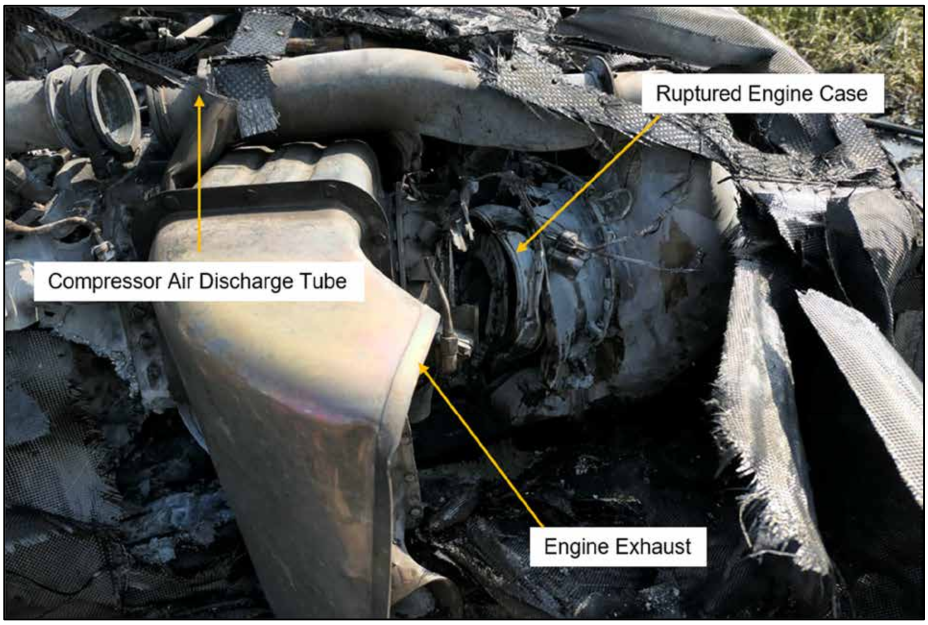


Figure 18
Engine at the accident site

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

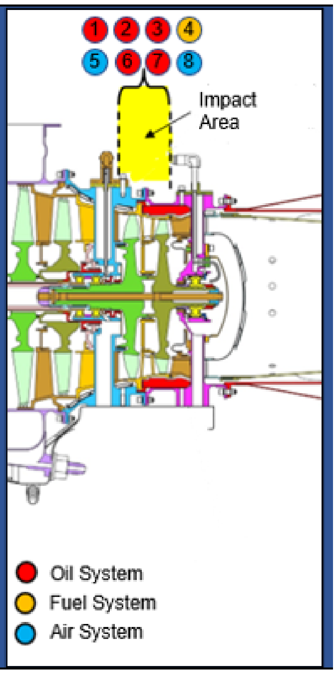


Figure 19

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.

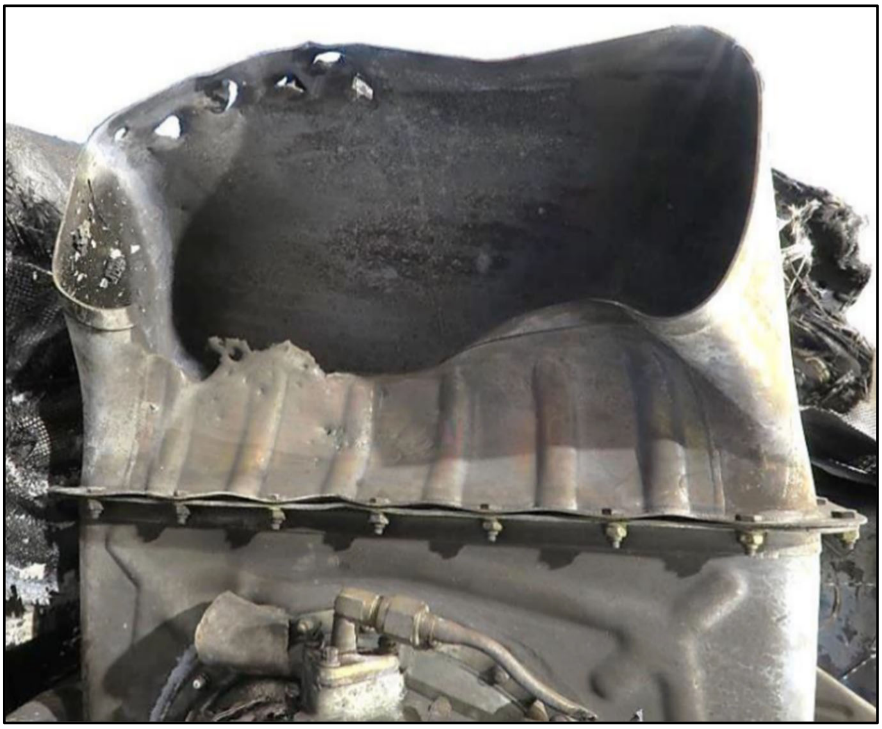


Figure 20
Exhaust damage

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).

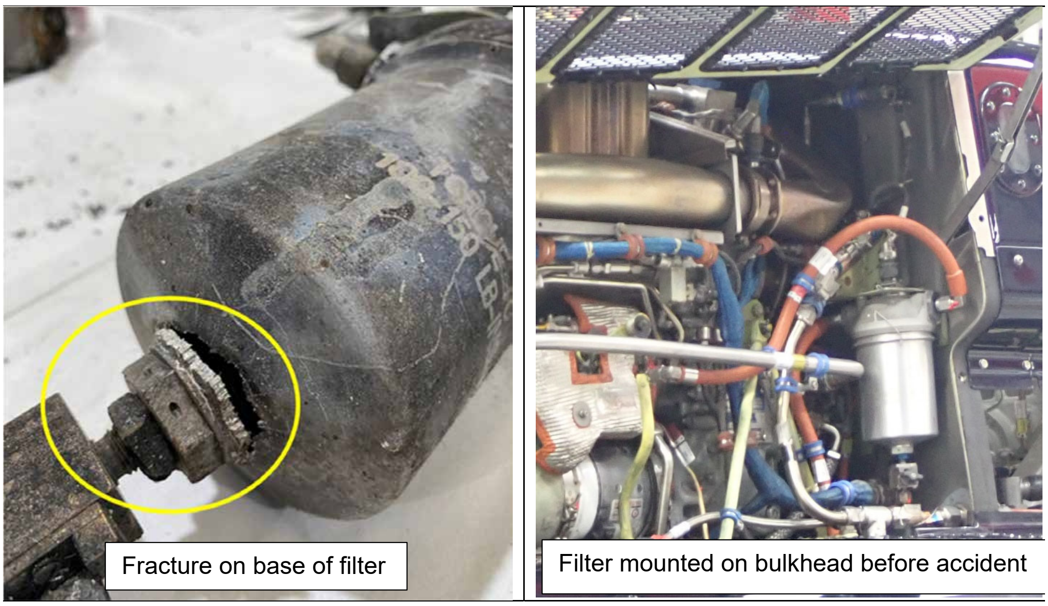


Figure 21
Damage to airframe mounted fuel filter

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.

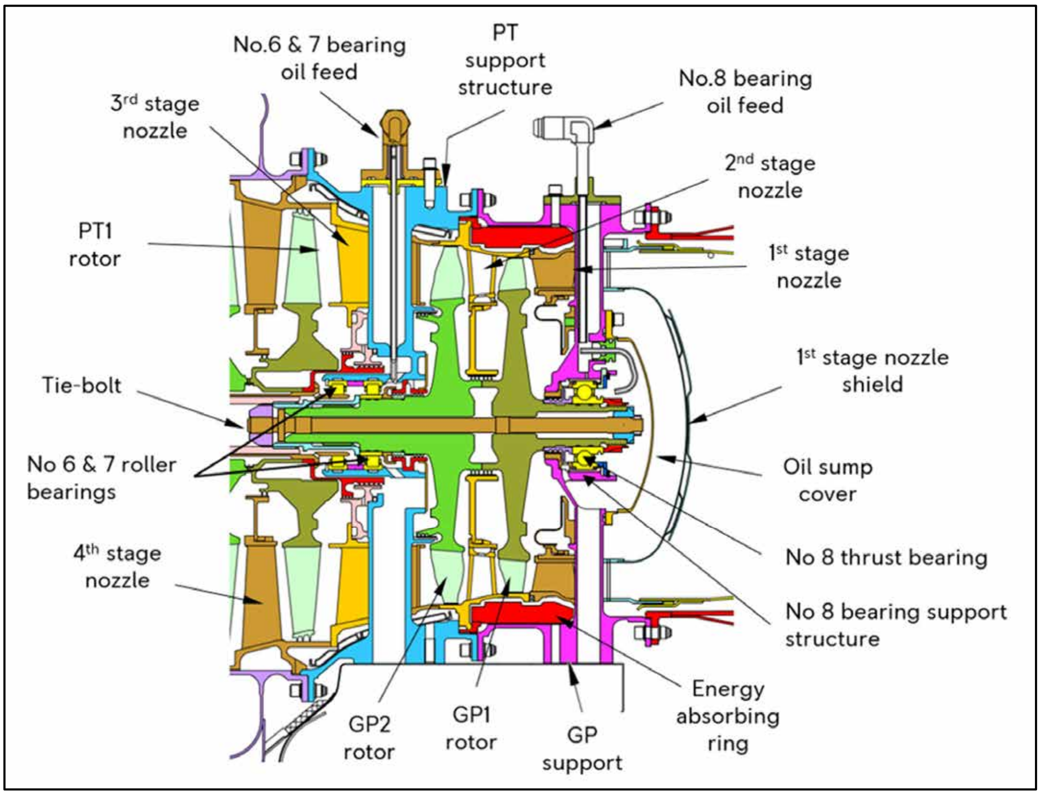


Figure 22

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)

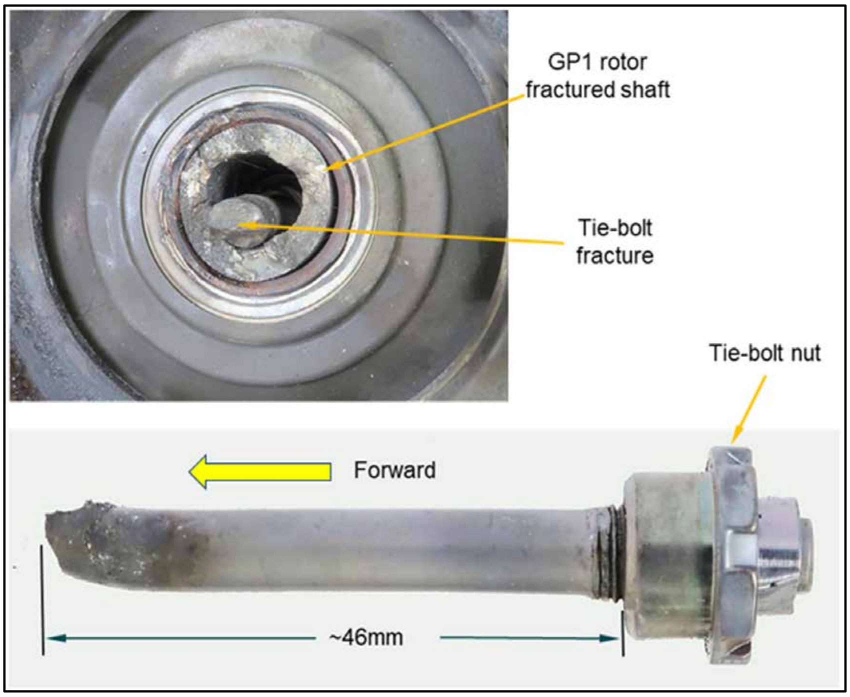


Figure 24

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.

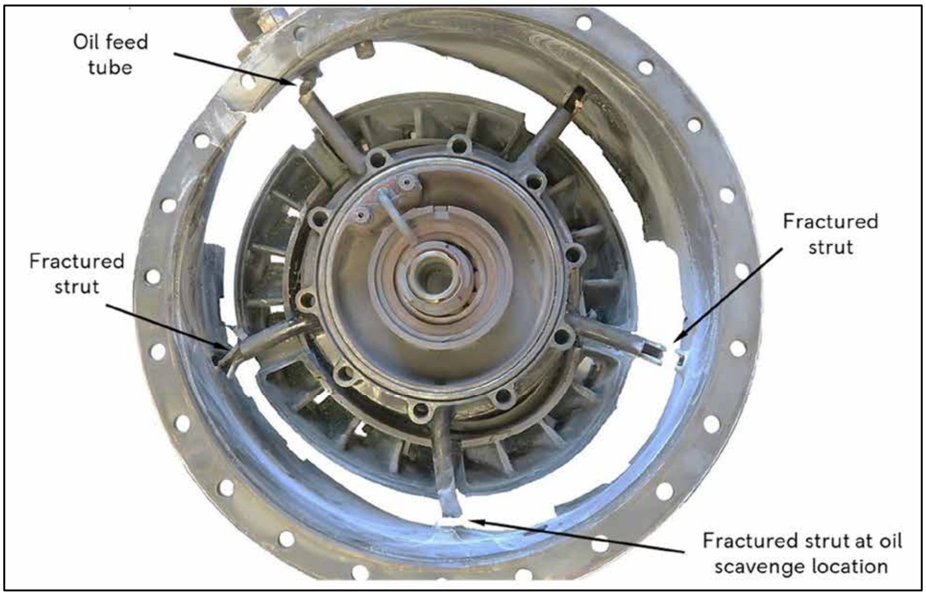


Figure 25

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.

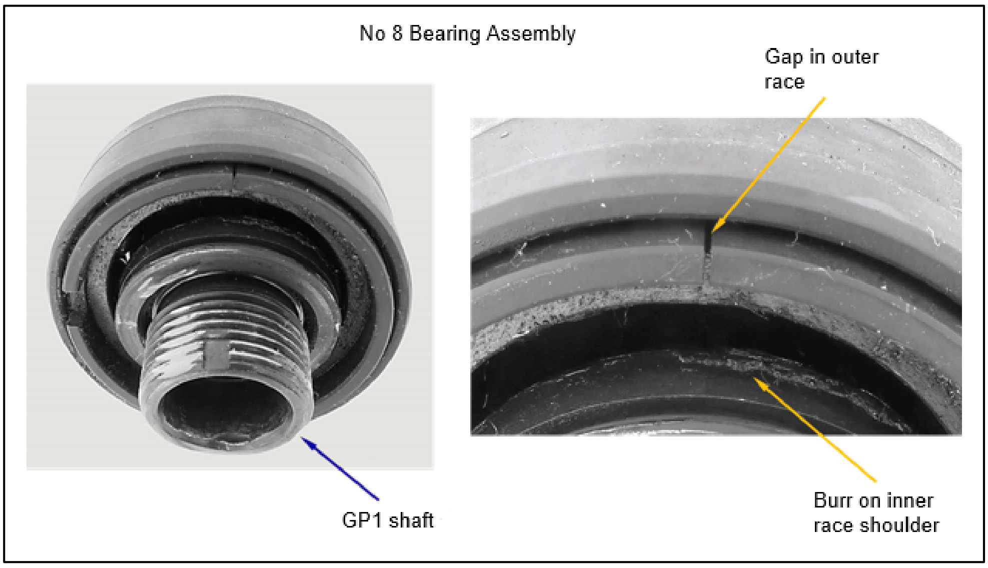


Figure 26

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.

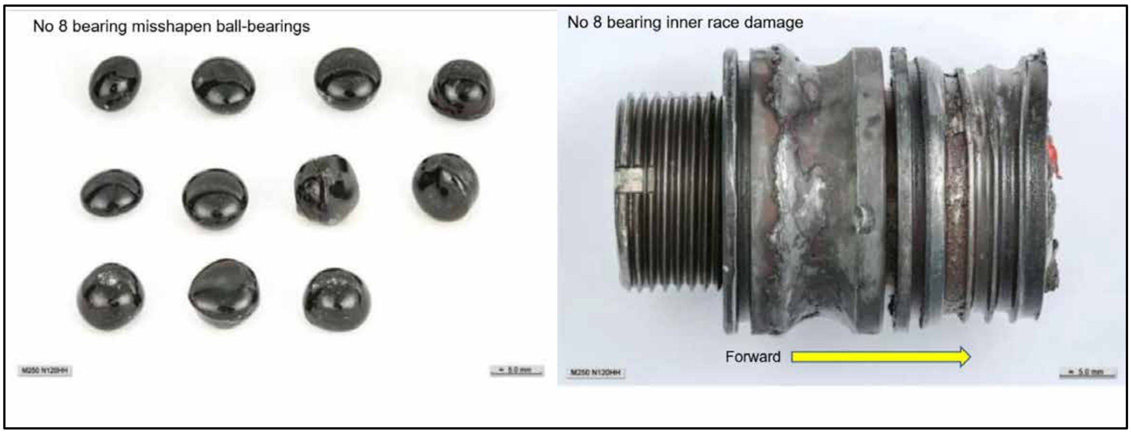


Figure 27

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.

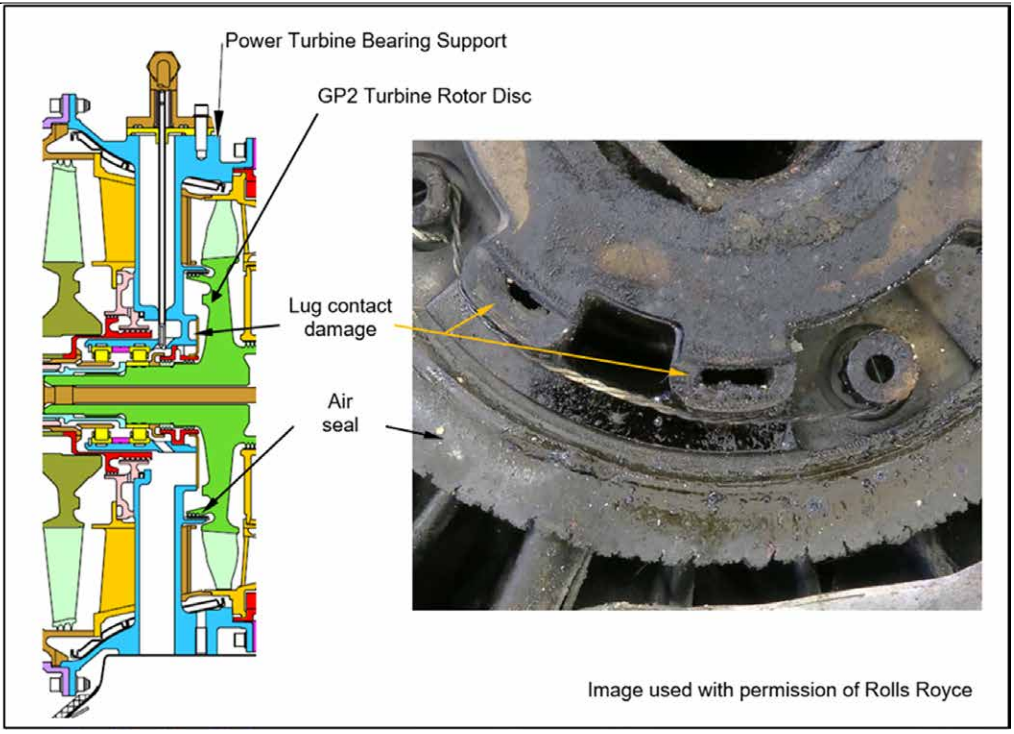


Figure 28

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.

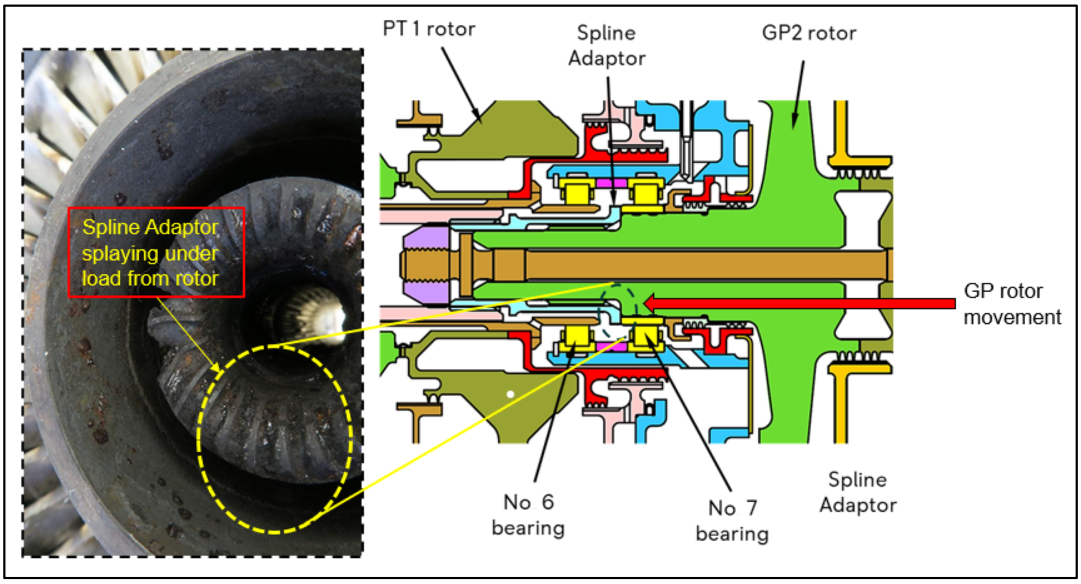


Figure 29

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.

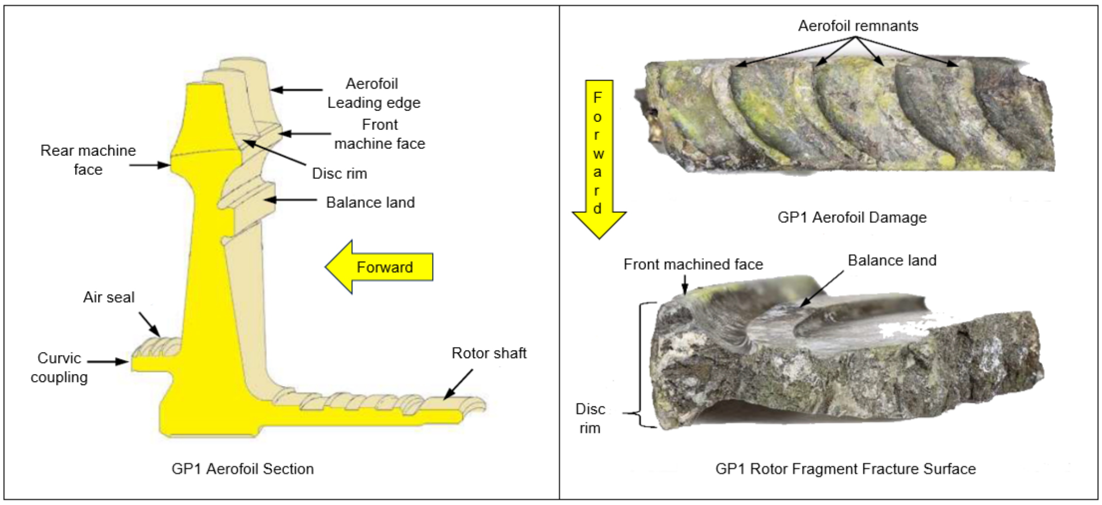


Figure 31

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.

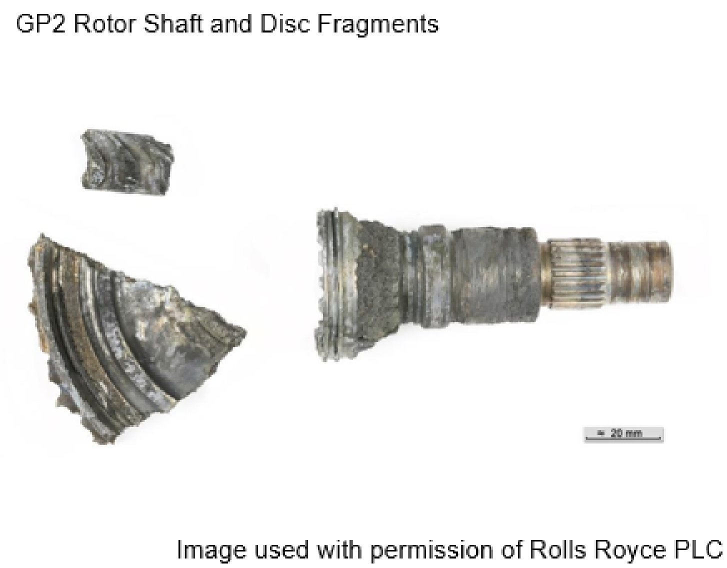


Figure 32

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.

Published: 30 September 2021.

ACCIDENT

Aircraft Type and Registration:	Cirrus SR22, G-CTAM
No & Type of Engines:	1 Teledyne Continental IO-550-N piston engine
Year of Manufacture:	2007 (Serial no: 2740)
Date & Time (UTC):	31 May 2020 at 1345 hrs
Location:	Calshot Spit, Hampshire
Type of Flight:	Private
Persons on Board:	Crew - 1 Passengers - 1
Injuries:	Crew - None Passengers - None
Nature of Damage:	Damaged beyond economical repair
Commander's Licence:	Private Pilot's Licence
Commander's Age:	56 years
Commander's Flying Experience:	587 hours (of which 89 were on type) Last 90 days - 11 hours Last 28 days - 6 hours
Information Source:	AAIB Field Investigation

Synopsis

Passing 1,400 ft in a descent towards an airfield the engine started to run roughly and subsequently lost power. The pilot turned the aircraft parallel to the shore and deployed the aircraft's Ballistic Parachute Recovery System. The parachute descent was successful and both occupants escaped from the aircraft uninjured. The loss of power was probably caused by fuel starvation to the engine, but the cause of the starvation could not be determined.

History of the flight

The aircraft was refuelled on the morning of 28 May 2020 at Lee-on-Solent (EGHF), and it was fuelled to tabs that are visible in the tank filler necks. The tab level gives a fuel load of 227 litres and 126 litres were added to the aircraft tanks to achieve this. The pilot selected 'tabs' on the Multi-Function Display (MFD) and reduced the displayed total to 220 litres to provide a contingency margin. The aircraft flew a return trip from Lee-on-Solent to Turweston Aerodrome (EGBT), and the pilot recorded 1 hour 45 minutes in his logbook for these two sectors and stated that the airborne time was 1 hour 13 minutes.

On 31 May the pilot took the aircraft out of its hangar at Lee-on-Solent and conducted pre-flight checks. During these he noted that the aircraft's MFD showed a usable fuel load of 127 litres. The aircraft departed Lee-on-Solent at 1054 hrs and flew to Dunkeswell (EGTU), landing at 1130 hrs. The pilot stated that he used a timer in the aircraft cockpit to remind him to change the fuel tank selection every 20 minutes. Both the occupants had lunch and returned to the aircraft at 1245 hrs to prepare for the return flight. During the checks the pilot noted that the MFD showed a usable fuel of 84 litres. As 43 litres had been used for the flight to Dunkeswell, the pilot assessed that sufficient fuel remained for the return leg.

The aircraft took off at 1305 hrs and climbed to 3,400 ft amsl for the return leg. Throughout these sectors, the pilot stated that he used the fuel leaning functions of the aircraft's engine management controls to achieve a cruise power of approximately 75%. During the flight the pilot altered the planned route to fly close to the Bournemouth Airport overhead. After Bournemouth the pilot descended to 1,700 ft at reporting point OLGUD[1]. The pilot stated that during the descent the fuel remaining display on the MFD[2] showed that there would be 12 USG at OLGUD and 11 USG at Lee-on-Solent. He preferred to use the fuel remaining figure on the MFD as his primary fuel reference during flight, cross-referring to the aircraft fuel gauges as required, although he did not trust the accuracy of the fuel gauges.

After OLGUD, the pilot commenced a slow descent toward Lee-on-Solent and called the airfield on RTF to obtain traffic information. At around 1,400 ft he recalled the engine "coughed a little." His first thought was that he had retarded the fuel mixture rather than the power lever, but on checking he found the lever in the correct position. He did not consider a low fuel situation because of his recent fuel checks. Then, the pilot recalled, the FUEL caution light on the instrument panel illuminated, and the engine began to run very roughly. The pilot did not recall seeing any fuel cautions or warnings on the MFD.

The pilot called Lee-on-Solent and informed them he was running out of fuel. He changed the fuel tank selection and the engine recovered briefly but then faltered again. He did not select the fuel pump switch to BOOST. The aircraft had now descended to 800 ft, and the pilot changed the fuel tank selection again but with no improvement, and he called Lee-on-Solent to inform them he was preparing to ditch. The last altitude the pilot recalled seeing was 600 ft before he turned the aircraft parallel to the shore, briefed the passenger and deployed the ballistic parachute recovery system (BPRS). The routes of the flights detailed in this section are shown in Figures 1 and 8.

On touchdown the aircraft nose went briefly underwater before bobbing back to the surface. The aircraft floated upright with the wings just slightly below the surface and only a small amount of water entering the cockpit. Both occupants left the aircraft through the right door onto the wing. The parachute remained airborne and caused the aircraft to invert as the occupants swam clear. They were assisted by a nearby windsurfer before being rescued by the Hamble Lifeboat. Both were uninjured. The parachute remained inflated and dragged the ditched aircraft to the shore.

Footntoe

1 OLGUD is a reporting point at the west end of The Solent, approximately 2 nm south-east of Lymington.
2 While on the ground the MFD Initial Fuel Page shows fuel quantity in litres. In flight the displays show fuel quantity in USG.

Accident site

The aircraft ditched in the sea 3.4 nm west of Runway 05 at Lee-on-Solent Airport and 0.5 nm south-east of the beach at Calshot Spit (Figure 1). The inverted aircraft washed up on the beach at Calshot.

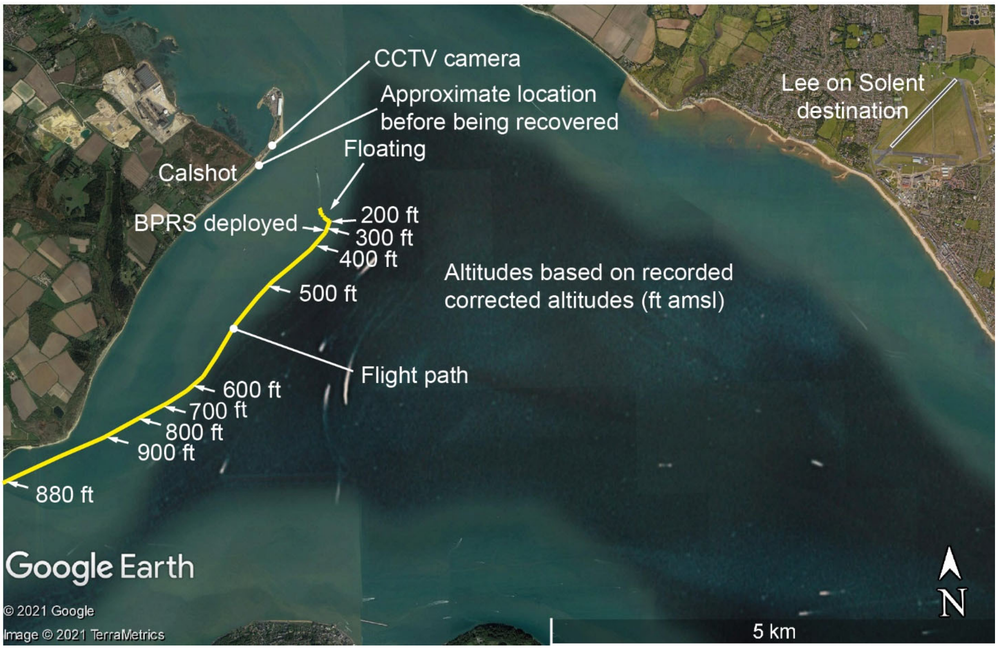


Figure 1

Final flight path and ditching location
Mapping © 2021 Google, Image © 2021 TerraMetrics

A rescuer of the Royal National Lifeboat Institution (RNLI), who was also a pilot, boarded the aircraft and turned off the fuel selector, battery master switch and magnetos. He did not move the fuel pump switch or the mixture lever. The police were also on scene and did not report any fuel leaks or fuel spills. The aircraft was guarded overnight and was then examined the next day by an insurance loss adjuster. Photos taken by the loss adjuster showed the throttle lever fully forward, the mixture lever near fully forward, the fuel pump switch in the OFF position and the fuel selector in the RIGHT-OFF position. An aircraft recovery team opened the inverted fuel tanks via the lower access panels and the loss adjuster reported that both the left and right main tanks contained no more in total than two or three pints of fuel and a small amount of sea water. The inboard collector tanks contained a trace of fuel and some sea water. There was no visible fuel leakage around the aircraft.

The aircraft's wings were cut off and the aircraft was transported to a secure facility. The aircraft had suffered significant damage to its upper fuselage structure and windscreen from being dragged up the beach inverted (Figure 2).

Figure 2

Aircraft after it was righted, the engine cowling removed, and the wings cut off

Aircraft information

The Cirrus SR22 is a four-seat single-engine composite aircraft that was first certified in 2000 and more than 6,000 have since been produced. There are different variants and G-CTAM was a Cirrus SR22 G3 with a 310-horsepower Continental IO-550-N engine and an after-market Tornado Alley turbocharger conversion. The G3 variant has a modified wing with a higher fuel capacity than the G2 and with fuel caps further outboard on the wing. The cockpit features a large Primary Flight Display (PFD) and a MFD. The aircraft is equipped with a whole-plane BPRS.

Maintenance history

G-CTAM, had accumulated 708 airframe and engine hours. The aircraft's last maintenance was a 50-hour check completed on 10 January 2000. During the aircraft's last annual inspection in June 2019, at 668 hours, work on the engine including cylinder removals was carried out to address low compression on four of the cylinders.

Fuel system description

Two main wing tanks and two collector tanks provide for a total capacity of 94.5 USG (358 litres), of which 92.0 USG (348 litres) are usable. When filled to the tabs, below each filler neck, the total usable fuel is 60 USG (227 litres). Fuel is gravity-fed from each tank to the associated 2.8 USG collector tank where the engine-driven fuel pump draws fuel through a filter and selector valve to pressure feed the engine fuel injection system (Figure 3). An electric fuel pump is provided for engine priming when the fuel pump switch is set to PRIME and vapour suppression when set to BOOST. Each wing tank and collector tank has a drain valve to collect water and contaminants. Each wing tank has an air vent at the top of the tank which is connected to an air inlet on each lower outboard wing surface. Each collector tank has a vent line connected to its main tank.

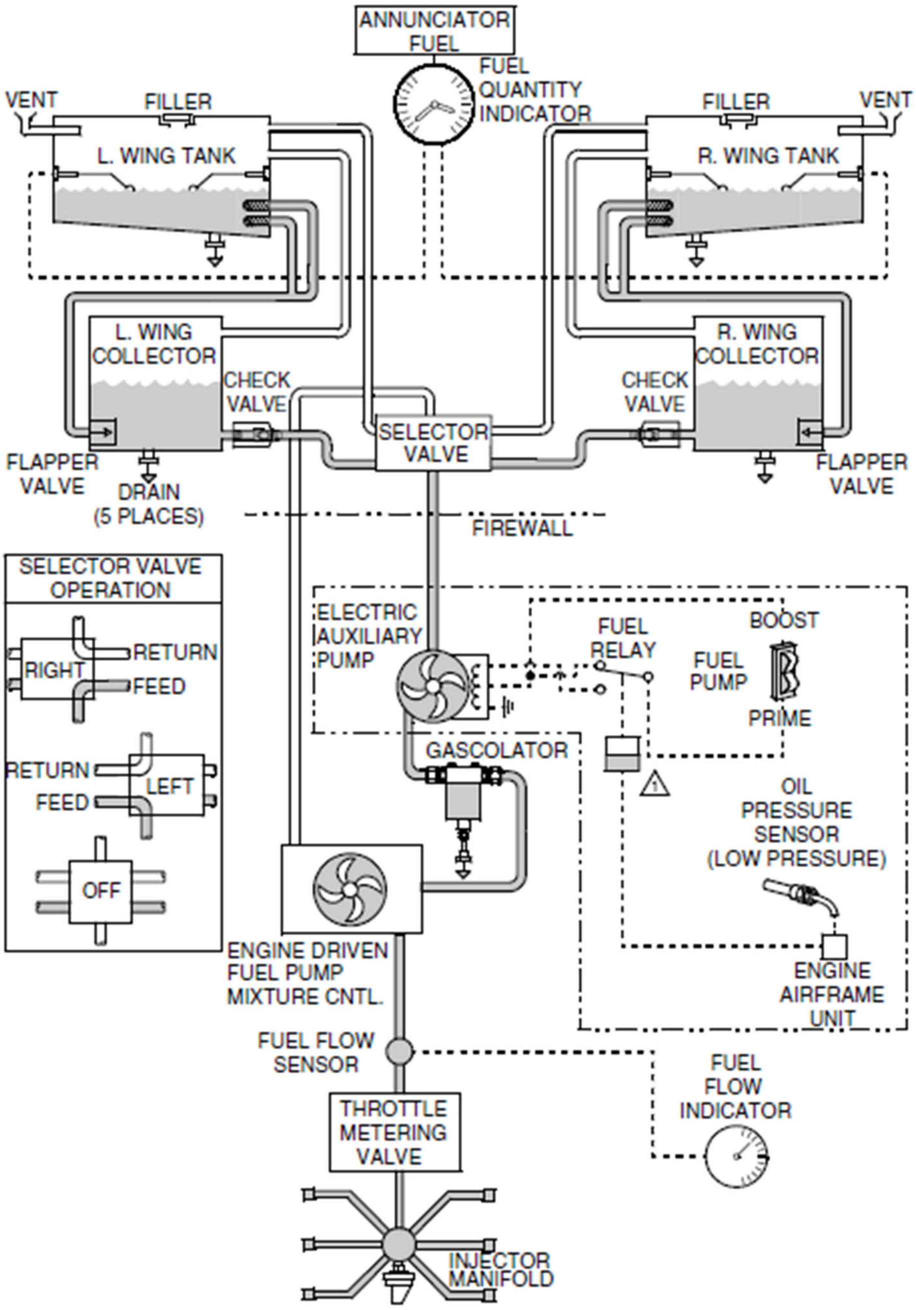


Figure 3

Fuel system diagram for Cirrus SR22 with G3 wing

The fuel tank selector valve is mounted in the centre console (Figure 4). It has four positions: LEFT, RIGHT, RIGHT-OFF and LEFT-OFF. To move the selector to one of the OFF positions requires lifting the circular plunger in the centre of the selector; this design is intended to prevent an inadvertent OFF selection. Fuel is balanced throughout the flight by the pilot alternating the selection between left and right tank. The maximum allowable fuel imbalance is 10 USG.

Figure 4

Fuel selector valve, fuel gauge and fuel pump switch

Fuel quantity indication and alert system

Each wing tank contains two analogue float-type sensors, one outboard and one inboard. These sensors provide fuel level information to the analogue fuel quantity indicator mounted forward of the fuel selector. An amber FUEL caution light located to the left of the PFD (Figure 5) comes on to indicate a low fuel condition. It is triggered by switches in the fuel quantity indicator when both tanks are indicating below approximately 14 USG. If both tanks are equally balanced at about 14 USG, then the light will illuminate at an indicated usable fuel quantity of about 28 USG. However, if one tank is run dry when the other tank is above about 14 USG, then the light will illuminate at a usable quantity of about 14 USG.

The fuel filler cap is located far outboard on the wing, further outboard than on earlier versions of the Cirrus SR22 which had smaller fuel tanks. Because of the wing dihedral, there can be fuel in the main wing tank without it being visible through the filler neck. The aircraft manufacturer estimated that the minimum usable fuel level that would still be visible below the filler neck in a wing is 15 USG with an aircraft that is perfectly level. So, it is possible for the aircraft to have just below 30 USG of usable fuel remaining without it being visible to the pilot.

Figure 5

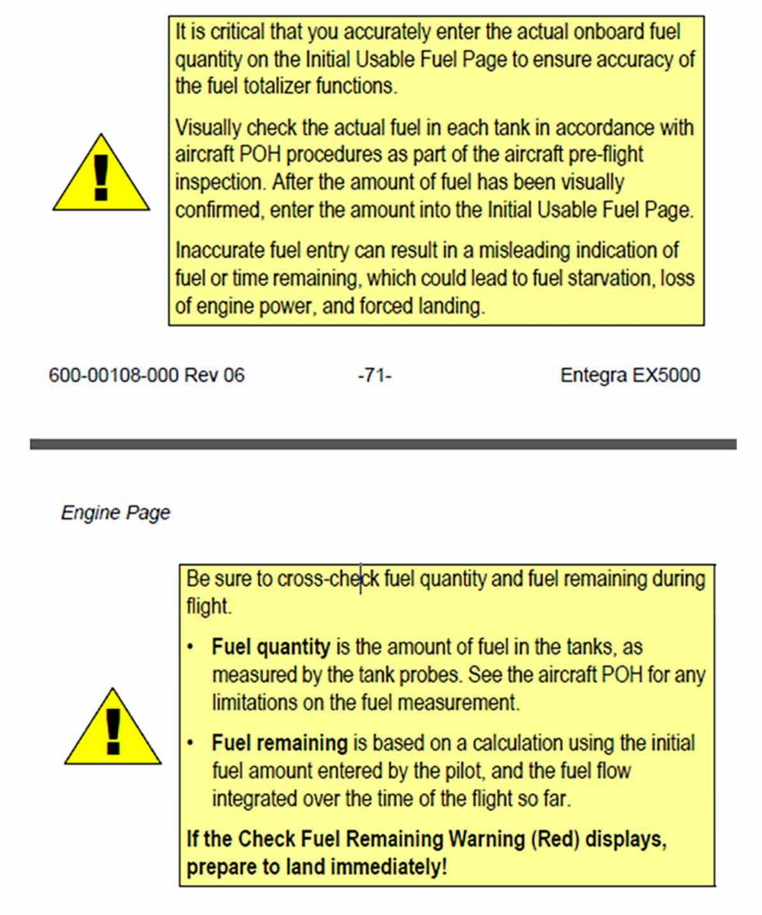
PFD, MFD and location of FUEL caution light

The aircraft manufacturer stated that the fuel indication system on this aircraft is accurate to within a few gallons when new, but over time the resistive elements in the analogue float sensors can suffer from wear which can result in inaccurate readings. Cirrus SR models manufactured after 2012 have a digital fuel quantity indication system, which uses a more reliable digital magnetoresistive float sensor; it is also available for retrofit on older models.

Multi-Function Display

The aircraft is fitted with an Avidyne Entegra EX5000C MFD. This provides engine management, navigation and fuel management capabilities. The system calculates fuel remaining at upcoming waypoints in the flight plan and displays a time remaining to fuel exhaustion. This information is portrayed in data blocks which are shown to the top left and right of the MFD map. The MFD has an input of fuel flow rate from a Data Acquisition Unit (DAU) which supplies all other engine parameters as well. The pilot enters the fuel quantity via a fuel initialisation page on the MFD and then the system calculations are based on that figure and measured fuel flow. Neither the DAU nor the MFD have any interface to the float-based fuel quantity indication system. The MFD calculates total fuel remaining but cannot calculate or display the distribution of that fuel across the aircraft's tanks as only the total fuel flow to the engine is measured.

Due to the reliance of the MFD on accurate input of fuel quantity, the Pilots Guide for the MFD contains the cautionary note shown at Figure 6.



It is critical that you accurately enter the actual onboard fuel quantity on the Initial Usable Fuel Page to ensure accuracy of the fuel totalizer functions.

Visually check the actual fuel in each tank in accordance with aircraft POH procedures as part of the aircraft pre-flight inspection. After the amount of fuel has been visually confirmed, enter the amount into the Initial Usable Fuel Page.

Inaccurate fuel entry can result in a misleading indication of fuel or time remaining, which could lead to fuel starvation, loss of engine power, and forced landing.

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Engine Page

Be sure to cross-check fuel quantity and fuel remaining during flight.

- **Fuel quantity** is the amount of fuel in the tanks, as measured by the tank probes. See the aircraft POH for any limitations on the fuel measurement.
- **Fuel remaining** is based on a calculation using the initial fuel amount entered by the pilot, and the fuel flow integrated over the time of the flight so far.

If the Check Fuel Remaining Warning (Red) displays, prepare to land immediately!

Figure 6

Caution note from MFD Pilots Guide

The MFD has a message bar which appears at the bottom of the screen whenever a message is posted by any of the system functions. The messages do not trigger any aural warning tone. Messages are displayed in yellow for cautions and red for warnings. For total fuel remaining, a yellow caution is displayed with 28 USG remaining and a red warning with 10 USG remaining. The note in Figure 6 states that pilots should prepare to land immediately should the red fuel warning appear.

Aircraft examination

This investigation began as an AAIB Correspondence Investigation which is why the AAIB did not attend the scene or initially recover the wreckage. However, as there was recorded data on board it was decided to recover the PFD and MFD. Following data analysis, it was decided to recover the aircraft wreckage to the AAIB's facility for further examination. This occurred after the loss adjuster had already arranged for the engine to be removed and stripped for storage. No faults, apart from corrosion, were found during the engine strip, and the engineer who removed the engine reported that he did not find any fuel in the engine fuel hoses. The level of corrosion was as expected from the aircraft's exposure to saltwater following the ditching. No loose connections were reported during the engine removal and engine strip.

The lack of usable fuel found in the fuel tanks did not necessarily indicate that the aircraft had run out of fuel. Sea water was found in the fuel tanks and sand was found inside the fuel selector which indicated that the fuel system had been compromised after immersion in the sea. It was possible that while the aircraft was inverted in the sea, fuel had drained away via the fuel tank vents. As the external part of the vents are located on the lower surface of the outboard part of the wing, the wing's dihedra l could have resulted in an inverted wing's wingtip being lower than its vent pick-up point inside the tank, resulting in a siphoning effect. The aircraft manufacturer stated that there was nothing to prevent the collector tank from draining via its vent tubing into the main tank and then through the main tank vent line to the exterior if the aircraft was inverted. The recorded data also provided evidence suggesting there was usable fuel remaining, so further airframe and engine examinations were carried out.

The fuel selector was removed at the AAIB's facility, and although its rotating mechanism had seized due to corrosion as a result of sea water ingress, once the corrosion was removed the fuel selector operated normally and the central knob functioned correctly to reduce the chance of a pilot inadvertently turning it off while changing tanks. A test fluid was passed through the left and right inlets and there was no restriction to flow in either the left or right position.

The fuel tank drain valves were tested for leaks and no leaks were found. The inboard floats inside the main fuel tanks operated freely. One of the outboard floats provided some resistance to movement, most likely due to corrosion, but this outboard part of the tank would have been dry at the time of the loss of power.

Because the engine had already been removed, the AAIB could not determine if there had been any loose connections forward of the engine firewall, but none were reported by the engineer who removed it.

The recorded data indicated an issue with fuel flow so the engine-driven fuel pump, fuel control unit and fuel flow divider were tested and strip-examined by an approved engine overhaul organisation. Some corrosion was found but no faults were found that would explain a loss of fuel flow.

Recorded information

There are no requirements for this aircraft to be fitted with a flight recorder. Unlike many other aircraft of this type, an optional Recoverable Data Module (RDM) was not installed. Data was recovered from the Avidyne PFD and MFD. The end of the flight was also captured by a CCTV camera.

The avionics had been submersed in sea water and were not operational when recovered. The MFD stored data was on an internal Compact Flash (CF) memory card. This had suffered from corrosion damage but was repaired and the data was recovered using a working MFD. The PFD stored its data on non-volatile memory chips mounted on the unit's circuit boards. These were removed and downloaded. The manufacturer converted the data to a usable format.

The PFD recorded general flight parameters but was not to the latest software standard so did not record engine parameters. The MFD recorded engine and fuel flow parameters as well as the cumulative fuel used during the flight derived from the fuel flow data. The MFD also retained the last calculated figure for total fuel remaining, as displayed to the pilot, based on the last pilot input of total fuel quantity after refuelling and the subsequent sensed fuel flow values. This figure was 11.4 USG (43 litres). The recordings do not include fuel quantities sensed from the fuel tanks. Low fuel warnings generated by either the MFD or onboard systems are not recorded.

The combined PFD and MFD data for the final part of the accident flight is shown in Figure 7. There was a sensor issue with one of the engine cylinder EGTs which is not believed to be relevant to the investigation. The fuel flow data shows an initial dip in fuel flow followed by a transition to a rich fuel mixture (fuel flow of 25 USG/hr) prior to the descent towards Lee-on-Solent. In the descent, there was a decrease to 4 USG/hr, which would have been insufficient to sustain combustion, followed by a recovery of the fuel flow. The decrease occurred again 20 seconds later and this time the EGTs reduced to zero indicating that no combustion was taking place. The fuel flow recovered again for about another 20 seconds before it dropped again and the EGTs reduced to zero again. There was a final recovery and drop-off before the BPRS was activated below 400 ft amsl.

The data was analysed and discussed with the aircraft manufacturer, the engine manufacturer and the turbocharger manufacturer. The consensus view was that the engine's performance was following the fuel flow. When the fuel flow was sufficiently high combustion was occurring and power was being produced, and when the fuel flow dropped sufficiently combustion ceased. The engine-driven fuel pump had been tested and functioned normally and therefore the consensus was that the loss of fuel flow was indicative of fuel starvation, ie the demanded fuel was not always reaching the fuel pump.

The data does not show when the pilot switched fuel tanks and therefore a conclusive analysis of each fuel flow reduction and increase could not be performed. The manufacturers stated that when a fuel tank is almost 'dry' the fuel outlet port can become uncovered and a mixture of air and fuel can be sucked up to the fuel pump, which can cause engine surging. When only air is sucked, combustion ceases. Due to fuel slosh, the port might alternately cover and uncover with fuel. At a fuel flow rate of 20 USG/hr (0.021 litres/second), a supply of 0.42 litres of fuel would be sufficient to run the engine for 20 seconds. It is possible that one or more of the fuel flow reductions and increases observed in the data was caused by fuel slosh. It is likely that at least one of the fuel flow reductions and increases was a consequence of the pilot switching tanks. When a pilot switches tanks after the system has sucked in air from a dry tank, the engine can fail to pick up or can run roughly.

The turbocharger manufacturer stated that the fuel flow measurement is averaged over two or four seconds, so momentary drops in fuel flow due to air entrainment would not show up in the data. Fluctuations in longitudinal acceleration indicate that the engine was probably not running smoothly for very long in between fuel flow drops. The engine-driven fuel pump cannot suck fuel through a large amount of air, whereas the electric fuel pump, which is located below the collector tank and is gravity fed by it, can push fresh fuel up the fuel lines to clear any air. This is why activating the fuel pump is important if a tank has run dry.

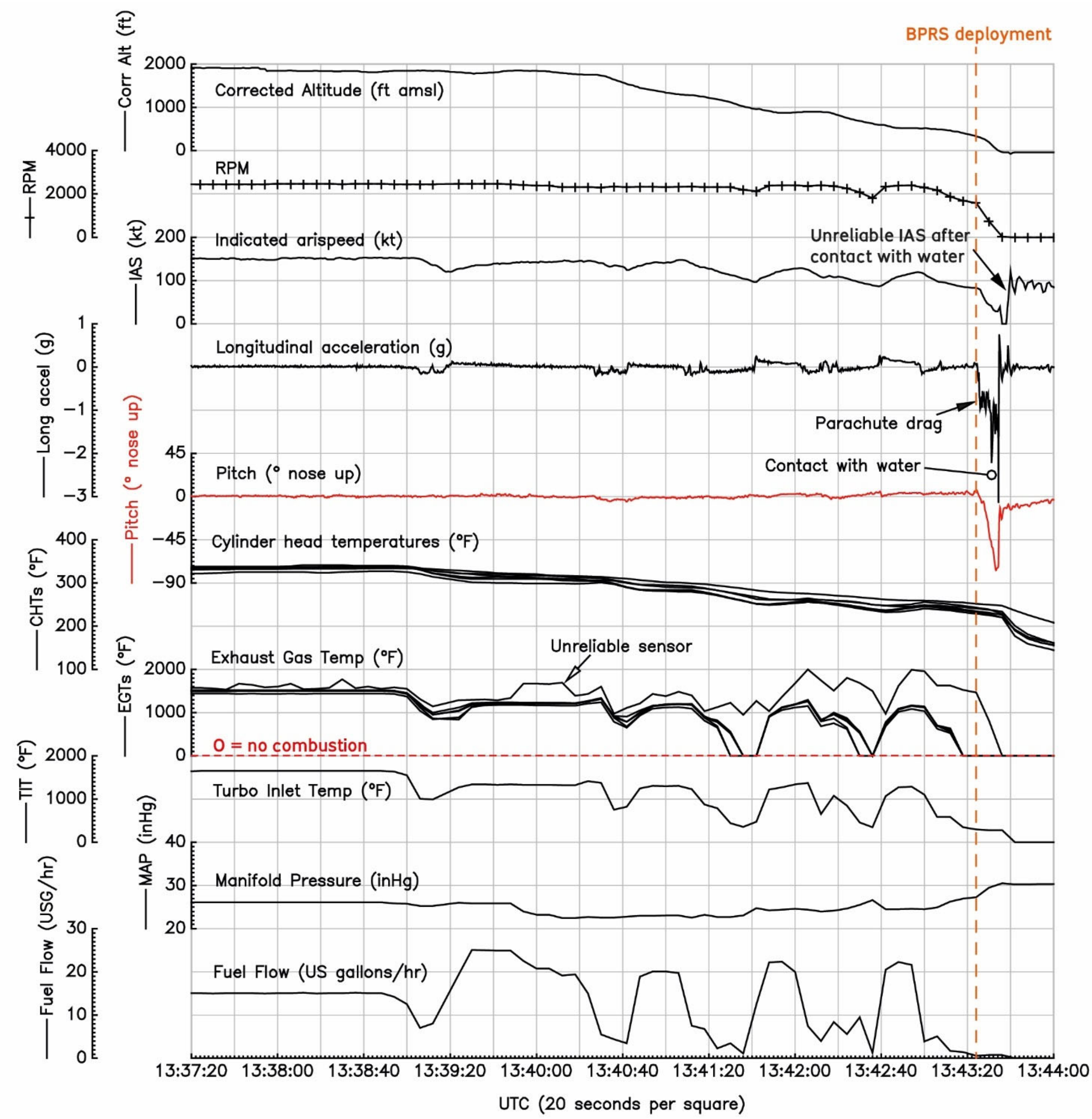
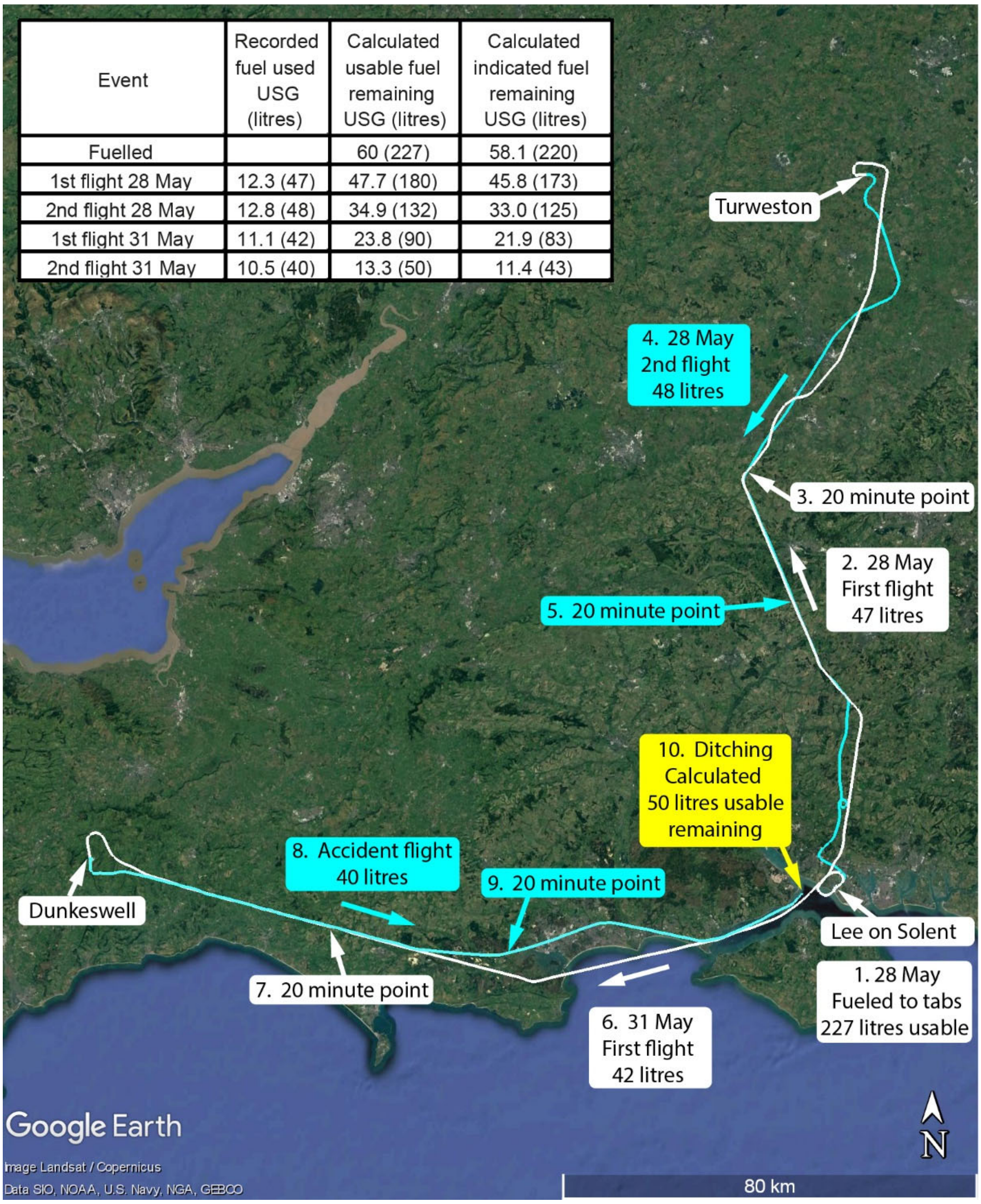


Figure 7

Extracts of PFD and MFD data at the end of the accident flight.

Analysis of the CCTV imagery correlated with the recorded data indicated a time lag in the recorded altitude and altitude rate parameters compared to the accelerometers. The combined data indicated that the BPRS was deployed descending through 340 ±20 ft above sea level. Taking the parameter timings into account, the average descent rate over the last 1.5 seconds before impact with the water was approximately 1,500 ft/min. At the time of contact with the water the pitch of the aircraft was recorded as 74° nose down. The recorded longitudinal acceleration increased by 2 g, the normal acceleration increased by 3 g and the lateral acceleration peaked at 1.5 g, although it was unclear how accurate these values were as the system was unlikely to have been designed or tested for accuracy during dynamic impact situations such as this.

The recorded fuel usage for the four flights flown since refuelling are shown in Figure 8.



Event	Recorded fuel used USG (litres)	Calculated usable fuel remaining USG (litres)	Calculated indicated fuel remaining USG (litres)
Fuelled		60 (227)	58.1 (220)
1st flight 28 May	12.3 (47)	47.7 (180)	45.8 (173)
2nd flight 28 May	12.8 (48)	34.9 (132)	33.0 (125)
1st flight 31 May	11.1 (42)	23.8 (90)	21.9 (83)
2nd flight 31 May	10.5 (40)	13.3 (50)	11.4 (43)

Figure 8

Fuel usage since refuelling

Spot checks of engine rpm, manifold pressure, altitude, airspeed and fuel flow from the recorded data compared favourably to the data in the performance tables in the POH, which indicated that the recorded fuel flow data and derived fuel usage data was reliable, assuming no leaks. If fuelling to tabs was accurate (60 USG / 227 litres) and the recorded fuel flow and fuel used values were accurate, there should have been 13.3 USG (50 litres) of usable fuel available at the time the aircraft ditched.

The pilot stated that he started a timer on takeoff and that he switched tanks every 20 minutes. The four flights since refuelling were all more than 20 minutes long and less than 40 minutes so should have triggered one tank switch each. The tanks were not switched on the ground between flights.

Applying the fuel flow figures to the tank switching method described meant that at the time of the engine problems there should have been approximately 6.6 USG (25 litres) of usable fuel in each tank. The pilot did not believe that he had missed an in-flight switch. However, if one of the in-flight switches had been missed during one of the flights on 28 May, then calculations showed that the tank in use at the time of the engine problems would have run dry and the other tank would have been close to the nominal 14 USG alert level in the location of the engine problems (plus or minus less than 2 USG). 20 minutes into the first flight on 28 May coincided with the transition from a longer stable flight leg to a turn onto a new heading, with multiple corrections afterwards. 20 minutes into the second flight on 28 May was approximately halfway along a stable flight leg. There are potential sources of error with these calculations, such as assuming the nominal 'tabs' fuel quantity was accurately achieved and imprecision in the timing for the remaining fuel tank switch-overs.

POH Checklist

The POH[3] procedure for changing the selected tank states:

> *'Fuel BOOST must be used for switching from one tank to another. Failures to activate the Fuel Pump before transfer could result in delayed restart if the engine should quit due to fuel starvation'.*

On the centre console, in front of the fuel pump switch, there is a label which states:

> *'TURN BOOST PUMP ON DURING TAKE OFF, CLIMB, LANDING, AND SWITCHING FUEL TANKS.'*

The aircraft POH contains checklists for actions in the event of an engine failure in flight. The checklist drill is shown at Figure 9.

The POH states that items which are underlined '*should be memorized for accomplishment without reference to the procedure*'.

Footnote

[3] Pilot's Operating Handbook and EASA Approved Airplane Flight Manual for the Cirrus Design SR22 covering aircraft serials 0002 thru 2978 and some others. P/N 13772-001E Revision A10.

Cirrus Design
SR22

Section 3
Emergency Procedures

Engine Failure In Flight

If the engine fails at altitude, pitch as necessary to establish best glide speed. While gliding toward a suitable landing area, attempt to identify the cause of the failure and correct it. If altitude or terrain does not permit a safe landing, CAPS deployment may be required. *Refer to Section 10, Safety Information*, for CAPS deployment scenarios and landing considerations.

• WARNING •

If engine failure is accompanied by fuel fumes in the cockpit, or if internal engine damage is suspected, move Mixture Control to CUTOFF and do not attempt a restart.

1. Best Glide Speed.. ESTABLISH
2. Mixture .... AS REQUIRED
3. Fuel Selector.. SWITCH TANKS
4. Fuel Pump .... BOOST
5. Alternate Induction Air ....ON
6. Air Conditioner (if installed)..........................OFF
7. Ignition Switch..CHECK, BOTH
8. If engine does not start, proceed to *Engine Airstart* or *Forced Landing* checklist, as required.

Figure 9

Engine Failure checklist

The CAA Skyway Code

The CAA Skyway Code[4] is intended to provide General Aviation pilots with practical guidance on issues relevant to their flying. The section dealing with fuel states the following about fuel gauges:

> *'Fuel gauges in most GA aircraft are not sufficiently reliable for the purposes of flight planning or pre-flight checking. Physically examining the fuel levels with a method appropriate to the aircraft type (such as a dipstick) is the best way of assessing the aircraft's fuel state. However, in flight, low fuel gauge readings should never be ignored.'*

It goes on to say the following about fuel totalisers:

> *'A fuel totaliser, if fitted, is a good indicator of fuel burn. However, for the purpose of counting fuel remaining it is completely dependent on the initial fuel level being correct. It only measures fuel consumed by the engine rather than the content of the fuel tanks and would not detect a leak.'*

Footnote

[4] https://publicapps.caa.co.uk/docs/33/CAP1535S%20Skyway%20Code%20Version%203.pdf [Accessed September 2021]

Manufacturer's flight test involving switching fuel tanks

According to Federal Aviation Regulation FAR 23.955(e), for aircraft with multiple fuel tanks and reciprocating engines, if engine power loss results from fuel depletion of the selected tank, it must be possible after switching to any full tank to obtain 75% maximum continuous power in not more than 10 seconds for naturally aspirated single-engine aeroplanes. The figure is 20 seconds for turbocharged single-engine aeroplanes provided that 75% of normally aspirated power is regained in 10 seconds. A fuel boost pump can be used to meet this requirement.

The aircraft manufacturer conducted a flight test against this requirement in a Cirrus SR22 G1 with a naturally aspirated engine. The engine was run on one tank until the fuel was exhausted and then the tank was switched to the full tank. This was done three separate times with the boost pump on and three times with the boost pump off. This was then repeated on a subsequent flight with the other tank empty. The results are shown in Table 1.

			Time to regain 75% power (seconds)	
Flight No.	**Tank run dry**	**Tank turned on**	**Using boost pump**	**Without boost pump**
1	Right	Left	8.5	14
1	Right	Left	6.8	8.4
1	Right	Left	6.1	17.5
2	Left	Right	6.6	11.9
2	Left	Right	5.5	45
2	Left	Right	6.1	No start in 1 minute

Table 1

Results from Cirrus SR22 G1 fuel tank switching flight test

As it was not possible to consistently restart the engine to 75% power within 10 seconds with the boost pump off, the POH contains a limitation to engage the boost pump prior to switching tanks, and to engage it as part of the emergency procedures following loss of power.

These tests were not repeated in later models of the Cirrus SR22. A later flight test involving an SR22 G3 only involved switching tanks without running one dry. Another flight test involved an SR20 G3 (which has a lower power engine), which also revealed that the fuel pump was needed to consistently regain power within 10 seconds. The turbo manufacturer stated that the addition of a turbocharger would not affect these performance figures at low altitude.

The aircraft manufacturer had not retained the time-trace data from these test flights, so it was not possible to see the effect on fuel flow when running a tank dry. However, the manufacturer's engineer involved with the flight tests stated that "the engine did not lose power suddenly but coughed and spluttered for maybe up to a 1 minute or maybe 30 seconds".

Ballistic parachute recovery system

The Cirrus SR22 is equipped with a whole-aircraft BPRS, proprietarily called the Cirrus Airframe Parachute System (CAPS). The following are extracts from the POH description of the system:

> *'The CAPS consists of a parachute, a solid-propellant rocket to deploy the parachute, a rocket activation handle, and a harness imbedded within the fuselage structure.*
>
> *When the rocket launches, the parachute assembly is extracted outward due to rocket thrust and rearward due to relative wind. In approximately two seconds the parachute will begin to inflate.*
>
> *Following any nose-up pitching, the nose will gradually drop until the aircraft is hanging nose-low beneath the canopy. Eight seconds after deployment, the rear riser snub line will be cut and the aircraft tail will drop down into its final approximately level attitude. Once stabilized in this attitude, the aircraft may yaw slowly back and forth or oscillate slightly as it hangs from the parachute. Descent rate is expected to be less than 1700 feet per minute with a lateral speed equal to the velocity of the surface wind.*
>
> *The Cirrus Airframe Parachute System (CAPS) is designed to lower the aircraft and its passengers to the ground in the event of a life threatening emergency. However, because CAPS deployment is expected to result in damage to the airframe and, depending upon adverse external factors such as high deployment speed, low altitude, rough terrain or high wind conditions, may result in severe injury or death to the aircraft occupants, its use should not be taken lightly. Instead, possible CAPS activation scenarios should be well thought out and mentally practiced by every SR22 pilot. The following discussion is meant to guide your thinking about CAPS activation. It is intended to be informative, not directive. It is the responsibility of you, the pilot, to determine when and how the CAPS will be used.'*

The POH offers a range of scenarios in which the use of the BPRS is suggested. One of those is as follows:

> ***'Landing Required in Terrain not Permitting a Safe Landing***
>
> *If a forced landing is required because of engine failure, fuel exhaustion, excessive structural icing, or any other condition CAPS activation is only warranted if a landing cannot be made that ensures little or no risk to the aircraft occupants. However, if the condition occurs over terrain thought not to permit such a landing, such as: over extremely rough or mountainous terrain, over water out of gliding distance to land, over widespread ground fog or at night, CAPS activation should be considered.'*

While there is no absolute limit on altitude for deployment of the BPRS the POH gives the following advice on its use:

> ***'Deployment Altitude***
>
> *No minimum altitude for deployment has been set. This is because the actual altitude loss during a particular deployment depends upon the airplane's airspeed, altitude and attitude at deployment as well as other environmental factors. In all cases, however, the chances of a successful deployment increase with altitude. As a guideline, the demonstrated altitude loss from entry into a one-turn spin until under a stabilized parachute is 920 feet. Altitude loss from level flight deployments has been demonstrated at less than 400 feet. With these numbers in mind it might be useful to keep 2,000 feet AGL in mind as a cut-off decision altitude. Above 2,000 feet, there would normally be time to systematically assess and address the aircraft emergency. Below 2,000 feet, the decision to activate the CAPS has to come almost immediately in order to maximize the possibility of successful deployment. At any altitude, once the CAPS is determined to be the only alternative available for saving the aircraft occupants, deploy the system without delay.'*

The pilot stated that he had been taught during training on the aircraft to deploy the BPRS by 400 ft above the surface and to deploy it in preference to attempting a forced landing.

According to the aircraft manufacturer there have been 10 CAPS activations over water in Cirrus SR aircraft (including G-CTAM). In all but one of these accidents all occupants survived. One accident resulted in one fatality and three serious injuries following deployment in a spin at a height of 528 ft where the aircraft hit the water 4 seconds after activation. One of the accidents resulted in a serious injury and the remaining eight did not result in any serious injuries.

Manufacturer's online training programme

The aircraft manufacturer has produced a CAPS training programme which discusses many issues related to the system. This is free of charge to all pilots and can be accessed here: https://learning.cirrusapproach.com/learning-catalog. In the training the manufacturer suggests a CAPS briefing format to be used for every flight and suggests pilots adopt a "CAPS available" call when the aircraft climbs through 500 ft agl. It also addresses pilot decision making and examines a variety of scenarios for use of the CAPS. For the engine failure case the training package recommends immediate deployment of the CAPS should the failure occur between 500 ft agl and 2,000 ft agl (600 ft agl and 2,000 ft agl for SR22T Generation 5 or later), regardless of the terrain.

The aircraft manufacturer stated that the guidance in the online training programme is based on their research of many successful CAPS activations over the years since the POH guidance was first published, and research of unsuccessful forced landings which resulted in injuries or fatalities.

Survivability

The BPRS, although deployed at low altitude, performed as expected and the aircraft hit the sea with a rate of descent of approximately 1,500 ft/min, and a nose-down pitch attitude of 74° (enough time did not pass from activation of CAPS until impact with the sea for the rear riser snub line to be cut and for the aircraft to adopt a more level attitude). While still a significant descent rate, this was preferable to a ditching at the normal approach speed of approximately 80 kt. After touchdown, the aircraft remained upright long enough for both occupants to escape onto the wing, but the parachute remained inflated and caused the aircraft to invert as they swam clear. The inversion of the aircraft would have significantly impeded their escape had it occurred immediately after touchdown.

Aircraft performance

Using the engine management function of the MFD to achieve best fuel performance in the cruise gives a power setting of approximately 75%. In ISA conditions at 4,000 ft, this equates to a calculated burn rate of 17.5 USG (66 litres) per hour. The recorded data indicated the aircraft had burned 15 USG (57 litres) per hour in the cruise. Time to climb to 4,000 ft would be approximately three minutes and burn 1.3 USG (5 litres). The aircraft had completed four takeoff and climb events and a total of 143 minutes of airborne time since the refuel to 226 litres described by the pilot. Based on these figures, the total in-flight fuel burn would have been approximately 43.4 USG (164 litres). Recorded data showed a total fuel burn of 177 litres.

Analysis

Cause of the loss of power

The final loss of power was preceded by large fluctuations in fuel flow from 20 USG/ hour down to near zero. Both times that the fuel flow dropped to 1 USG/hour, the EGT's dropped to zero, indicating that combustion briefly ceased, before recovering as the fuel flow recovered. During the earlier fuel flow reduction to 3 USG/hour the EGT's did not drop to zero but at this low fuel flow rate combustion would have also ceased. The consensus view after discussion with the representatives of the aircraft, engine and turbocharger manufacturers was that the engine was producing power when the fuel flow was normal. The engine-driven fuel pump was tested after the accident and functioned normally and therefore it was most likely that the demanded fuel was not always reaching the fuel pump. This was consistent with the evidence from the engineer who removed the engine stating that there was no fuel in the fuel hoses in the engine. It is therefore probable that the loss of power was caused by fuel starvation, due to either a fuel leak or another mechanism.

Possible cause of fuel starvation if there was no fuel leak

If a selected fuel tank is used until there is no more usable fuel remaining, a loss of power will ensue due to fuel starvation. This may be preceded by engine power fluctuations as the engine-driven fuel pump alternately picks up fuel or sucks in air as the fuel pick-up point in the tank becomes exposed. The aircraft manufacturer's engineer stated that the engine coughed and spluttered for a while when they ran a tank 'dry' during flight tests. The flight tests also revealed that if a pilot changes fuel tanks after this has happened, the engine can take a long time to recover if the electric fuel pump is not set to BOOST; and in one test the engine failed to recover.

If the fuel tanks were filled to tabs as described, and if there had been no fuel leaks, then according to the recorded data there should have been about 13 USG (49 litres) remaining. Only about 0.5 USG of fuel was recovered from all the fuel tanks, but because the aircraft had been inverted the remaining fuel could have escaped via the air vents. According to the POH 2.5 USG of fuel is unusable, so if both tanks had been run 'dry' and no fuel had escaped post-impact, there should have been closer to 2.5 USG of fuel remaining. This was another indication that fuel probably escaped post impact.

The pilot did not recall any fuel cautions or warnings on the MFD, and it was not possible to confirm from retrieved data whether or not they appeared. The MFD low fuel caution comes on at 28 USG remaining so should have been illuminated prior to departure from Dunkeswell. Regardless of whether it illuminated, the pilot is likely to have been aware that he had less than 28 USG prior to departure because the 84 litres he noted before departure is 22 USG. The MFD red fuel warning would not yet have appeared as it illuminates when the system has calculated there are 10 USG remaining.

The pilot recalled the separate FUEL caution light illuminating just as the engine began to run roughly. Since the light illuminates when the fuel quantity in each tank is below about 14 USG, it would be possible for one tank to be empty and the caution light to illuminate when the second tank reached about 14 USG. The pilot stated that he changed tanks every 20 minutes to minimise fuel imbalance and avoid such a situation.

In theory, a larger tank imbalance could have been achieved by missing one of the timed tank switches during one of the four flights since the aircraft was last fuelled. If one of the in-flight switches had been missed, then calculations showed that the tank in use at the time of the engine problems would have run dry and the other tank would have been close to the nominal 14 USG alert level in the location of the engine problems, plus or minus less than 2 USG. In this scenario it would be more likely that the low fuel caution illuminated after the pilot switched tanks to address the rough running, whereas the pilot recalled it illuminated before he switched tanks. The only fuel indicators that could have differentiated between the contents of the left and right wing tanks were of an older, less reliable design, and not trusted by the pilot. As they linked into the low fuel caution system, this could explain the light not illuminating when it should have.

If one tank was empty with all the remaining fuel in the other tank, then it would be expected that the engine would recover when the tank was changed. However, this did not always occur in flight tests with the electric fuel pump off because air gets sucked into the fuel system when a tank is run dry. The pilot chose to switch tanks before deciding to deploy the BPRS, and the POH states that the fuel pump must be set to BOOST whenever the tank is changed. The pilot did not select the pump on, however, and this would likely have had an impact on the engine's capacity to recover had there been fuel in the tank to which the change was made.

Although it is not clear which of the reductions in fuel flow corresponded with the first fuel tank switch, the engine recovered for a short period despite the electric pump not being selected to BOOST. However, the pilot reported that he switched tanks again because the engine faltered again. If fuel remained in the tank he first switched to, it is possible that he did not leave enough time for the engine recovery to stabilise, or that it would have stabilised more quickly with the electric pump set to BOOST. Had he waited longer with the pump on, it is possible he might not have felt the need to switch back to the original tank.

Possible cause of fuel starvation if there was a fuel leak

Another possibility for the cause of fuel starvation is that both tanks ran dry due to a fuel leak upstream of the fuel flow sensor (between the sensor and the fuel tanks). The MFD only shows fuel used as measured by the fuel flow sensor so any fuel lost upstream of the sensor would not be detected by the MFD. For this to have happened the last fuel tank switch before the engine started faltering (about 11 minutes earlier), would have needed to have occurred just as that tank was about to run dry. Also, in the fuel leak scenario the FUEL caution light should have come on much earlier in the flight, in which case it was either not noticed or one of the gauges was indicating at least 14 USG while the tank was actually empty.

The fuel leak scenario demonstrates the importance of not relying on the MFD figures. The only way to detect a fuel leak in flight is by direct reference to the fuel gauges and comparing their indications to expected fuel figures along the route or MFD figures. The risk of running both tanks dry as a result of a fuel leak can be mitigated in the following ways:

1. Cross check the MFD figures with the fuel gauges to look for discrepancies between fuel remaining and fuel quantity.
2. Replace faulty fuel flow sensors if the fuel gauges are not reading accurately.
3. Calibrate the fuel gauges so you know the indications which correspond with actual quantities.
4. Carry more fuel so that there is always visible fuel in both tanks before departure, thereby increasing confidence in the reliability of the starting quantity.

Decision to deploy the BPRS

The event began at a relatively low altitude of 1,400 ft amsl, so little time was available to the pilot to action drills and attempt to resolve the engine issue. By the time the pilot had informed Lee-on-Solent of his situation and attempted a fuel tank change, the aircraft had descended to 800 ft amsl. The pilot attempted another fuel tank change, but this still had no effect. Believing that he would be unable to glide to land, the pilot deployed the BPRS in accordance with the training he received for engine failures in Cirrus aircraft, and the last altitude he recalled seeing was approximately 600 ft amsl. Data retrieved from the aircraft indicated that the BPRS was deployed descending through 340 ±20 ft above sea level.

The aircraft hit the sea in a nose-down attitude of 74° because there had not been time for the rear riser snub line to be cut, which would have pitched the aircraft back up to an approximately level attitude. This indicated that the system was deployed below a height that would have enabled the deployment sequence to complete.

The aircraft hit the sea at about 1,500 ft/min (15 kt) vertical speed, and despite the nose-low attitude the occupants were not injured and were able to egress successfully.

Guidance in the POH on use of BPRS

The guidance on the use of BPRS/CAPS in the POH says that, following an engine failure, CAPS activation should be considered if the forced landing would be on extremely rough or mountainous terrain, over water, over widespread ground fog or at night. There is a different emphasis in the manufacturer's online training programme which advises pilots to activate CAPS immediately following an engine failure between 500ft/600ft and 2,000 ft regardless of the terrain.

The POH states that below 2,000 ft CAPS should be activated almost immediately, but it does not provide a minimum safe altitude to deploy CAPS following an engine failure. It states that the height loss from level flight deployments has been demonstrated as less than 400 ft, but it does not provide a figure of the height loss when the aircraft is already descending in a glide descent following an engine failure. However, the online training programme provides a clear minimum safe height for activating CAPS after an engine failure.

The aircraft manufacturer stated that it would convene a panel to discuss updating the CAPS guidance in the POH to reflect the guidance in the CAPS online training programme, gained through in-service experience of CAPS deployments.

Conclusion

The aircraft suffered a loss of power, probably due to fuel starvation, but the cause of the fuel starvation could not be determined. It is possible that one fuel tank ran dry, and that the engine did not fully recover when the fuel tank was switched because the fuel pump had not been set to BOOST and the pilot had not kept this tank selected for long enough. However, the possibility of a fuel leak causing both tanks to run dry could not be ruled out. The pilot, believing that it would not be possible to glide to land, deployed the BPRS and the aircraft descended by parachute into the sea. Both those on board escaped uninjured.

The investigation highlighted the importance of setting the fuel pump switch to BOOST when changing fuel tanks, the importance of checking the fuel gauges for fuel quantity and any imbalance, and ensuring the serviceability of the fuel indication system. This is because neither the fuel totaliser system of the MFD nor the low fuel indication system can warn the pilot that a tank is about to run dry. Furthermore, if the fuel level in each tank is below about 15 USG, then due to the wing dihedral the pilot cannot visually check the fuel tank contents.

Safety action

The aircraft manufacturer stated that it would convene a panel to discuss updating the CAPS guidance in the POH to reflect the guidance in the CAPS online training programme.

Published: 30 September 2021.

SERIOUS INCIDENT

Aircraft Type and Registration:	Tekever AR5 Evolution Mk 2, G-TEKV	
No & Type of Engines:	2 3W 2-stroke piston engines	
Year of Manufacture:	2019 (Serial no: E505)	
Date & Time (UTC):	29 December 2020 at 1446 hrs	
Location:	Lydd Airport, Kent	
Type of Flight:	Commercial operations (UAS)	
Persons on Board:	Crew - None	Passengers - None
Injuries:	Crew - None	Passengers - N/A
Nature of Damage:	None	
Commander's Licence:	Other	
Commander's Age:	33 years	
Commander's Flying Experience:	1,514 hours (of which 665 were on type) Last 90 days - 105 hours Last 28 days - 34 hours	
Information Source:	Field Investigation	

Synopsis

While orbiting south of the runway in preparation for landing, both the unmanned aircraft's engines shut down unexpectedly. The External Pilot on the ground, who was visual with the aircraft, took control and landed it without further incident. The dual engine shutdown was likely to have been caused by an on-aircraft data error. Various safety actions, including improvements to the aircraft's hardware and software, and the Ground Control Station software, have been taken to reduce the risk of a reoccurrence.

History of the flight

The unmanned aircraft, G-TEKV, was returning to Lydd Airport from a flight over the English Channel. Flight operations were conducted from a Ground Control Station (GCS) where the crew control the aircraft from takeoff to landing and operate the payload to fulfil the mission objectives. The GCS contained two stations, the flight GCS (fGCS) and the mission GCS (mGCS). The fGCS focused on all aspects of the control of the aircraft platform, whereas the mGCS focused on the mission goals and operation of the payload.

The GCS was manned by the Mission Commander (MC), the oncoming Internal Pilot (IP), the off-going IP, and the Payload Operator (PO). An External Pilot (EP)[1] and a Maintenance Technician (MT) were positioned at the side of the runway abeam the intended touchdown position for the aircraft and both could communicate with the IP through air band radios.

Footnote

[1] The EP may also be referred to as a Safety Pilot (SP).

While the aircraft was orbiting off the coast prior to transiting back to the airfield, the two IPs conducted a handover; the off-going IP remained to act as a second pilot to assist with the conduct of the remainder of the flight. Meanwhile the EP advised that the wind favoured a landing on Runway 03 with a light crosswind.

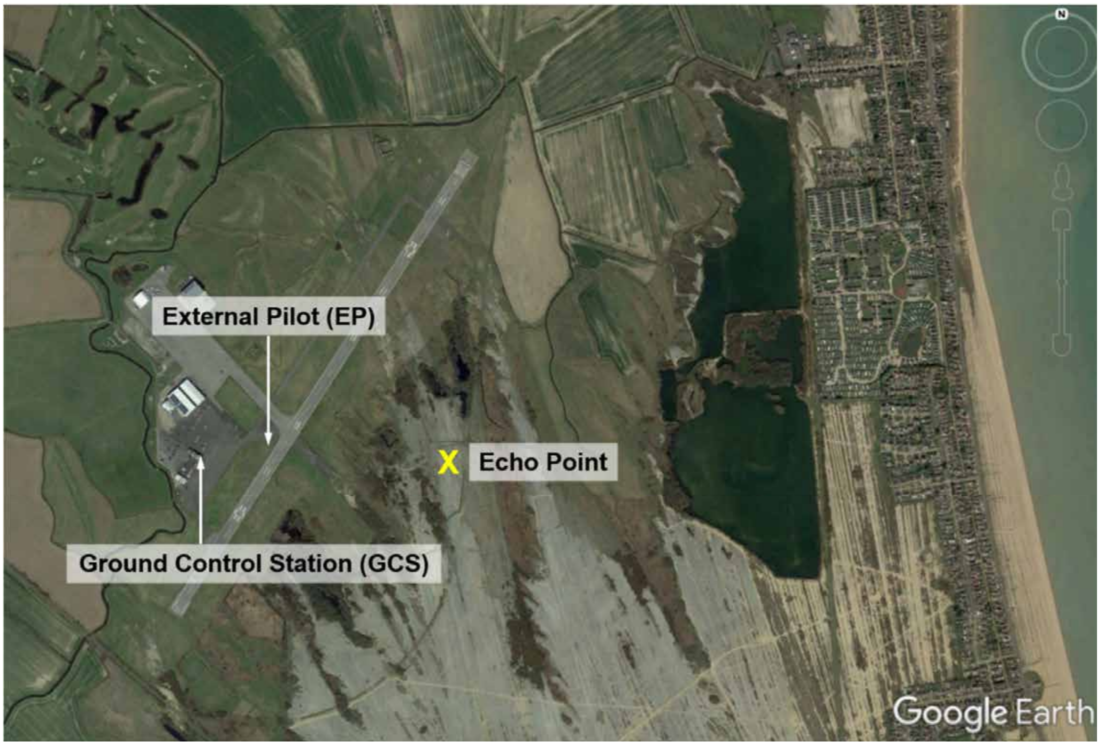


Figure 1

Lydd Airport and Echo Point

The aircraft transited towards the airfield at 700 ft amsl to remain clear of the cloud and icing. On reaching Echo Point, overhead the airfield (Figure 1), the aircraft entered an orbit while the IP, assisted by the off-going IP, proceeded to load the mission waypoints for a landing on Runway 03. Meanwhile, the EP reported to the GCS that he could hear the aircraft but was not visual with it. The MC instructed the IP to descend the aircraft to 600 ft at which point the EP confirmed that the aircraft was visual and clear of cloud.

With the aircraft established at 600 ft in the orbit at Echo Point and the mission points uploaded, the IP informed the EP that the aircraft was set up for the landing. The EP acknowledged and the IP switched the aircraft to ROUTE mode[2] to proceed with the approach and landing on Runway 03. After the aircraft completed two more orbits, the crew in the GCS noticed that it did not appear to leave the orbit at the expected point to establish itself downwind.

As the aircraft flew the final orbit, the EP outside was expecting the call 'downwind' from the GCS team. He noticed the aircraft level its wings, as expected when departing the orbit,

Footnote

[2] This is an automatic mode where the aircraft follows a specified route defined of sequential waypoints defined by location and altitude.

but observed the nose drop more than normal. At this point the EP became aware that he was not able to hear the aircraft's engines. He operated the throttles and confirmed that there was no engine response. The EP switched to FLY-BY-WIRE (FBW)[3] mode, took control of the aircraft, confirmed control response, and instructed the MT to inform the GCS about the complete loss of engine power.

While this was happening, the flight team in the GCS was first alerted that something was amiss when they observed the aircraft fly on a westerly heading towards the runway and not along the expected track to establish itself downwind parallel to the runway. None of the team reported seeing or hearing any alarms or warnings. The MC noticed that the height of the aircraft appeared low, and the off-going IP then noticed that the displayed parameters for both engines indicated zero rpm.

The MC, unaware that the EP had already taken control of the aircraft, gave instruction to the IP to advise the EP to do so and went outside the GCS to observe the aircraft. The MT advised the IP that the EP had already taken control and so, from that point on, the IP provided speed information to the EP until the aircraft had landed.

The EP assessed the conditions and positioned the aircraft on final approach; it landed without further incident.

June 2020 event

This event followed a related one that occurred in June 2020 where, during an integration ground test of equipment onto a new AR5 aircraft at the manufacturing and development site in Portugal, both engines shut down, uncommanded by either the GCS or the EP.

Lydd Airport

Lydd Airport has an elevation of 13 ft amsl and is situated about 1 nm inland from the south coast. At the time of this event the airfield was closed, and ATC was not manned. The operator had authorisation to operate the aircraft from the airfield when it was closed.

Personnel

Crewing

The MC leads the team in the execution of the flight and its mission.

The IP is the fGCS operator and is responsible for the control and navigation of the aircraft.

The IP communicates with ATC, when available, and other aircraft to ensure deconfliction. The PO is the mission GCS (mGCS) operator and is responsible for the management of the payload.

Footnote

[3] The EP controls the aircraft's roll and pitch angles and throttle (using his radio controller) and the aircraft is stabilized within control limits. The rudder is either controlled manually or automatically with aileron movement. In this mode of operation, the IP is required to constantly update the EP with speed information.

The MC, IP and PO are located in the GCS for the mission.

The EP is responsible for ground manoeuvring, takeoffs and landings, and provides redundancy while the aircraft remains within Radio Line of Sight (RLOS) of him in the event of signal loss between the aircraft and the GCS. For departure and arrival, the EP stands at the side of the runway abeam the intended departure or landing point. The EP has an air band radio to listen to the information passed by the IP in the GCS but relies on the MT to transmit any information back to the GCS.

The MT is responsible for ensuring the aircraft is serviceable and operational. During takeoffs and landings, the MT accompanies the EP to act as a communications relay between the EP and the IP in the GCS.

Training and experience

All the pilots satisfied the qualification and competency requirements specified in CAP 722[4]. Each of the crew members had successfully completed the operator's own competency training programme for the aircraft in their specific roles. The MC, IP and PO all had previous experience operating UAS and were trained for both the IP and PO roles. The EP had satisfied the required competency training for his role and it was noted that he had extensive experience in flying radio-controlled models, including 40% scale aircraft.

Recorded information

Recorded data was available from both the aircraft and GCS. Flight data was stored in a log file on the aircraft which included: flight control data (such as received control commands and control surface deflections); GPS-sourced data; engine data, aircraft attitudes and aircraft modes (such as 'Fly by wire'); and messages (such as 'Ignition OFF'). The aircraft was also fitted with cameras, but these were not active during the event. The GCS recorded data into three log files. One of these included telemetry data from the aircraft that was GPS timestamped. The GCS had no facility to record alarms and warning triggers, either in a log file, visually or audibly.

Figure 2 shows the flight track of the aircraft from the last of six orbits around Echo Point through to the landing. The aircraft came into radio range of the EP's controller at 1437:30 hrs near Echo Point and commenced the orbits at about 750 ft. From the flight log it was determined that during the last orbit an engine 'Ignition OFF' command was detected by the autopilot after which the engines shut down. The EP started making roll control inputs to level the wings 17 seconds after the 'Ignition OFF' command (overriding the autopilot) before switching to FBW mode 11 seconds later.

Footnote

[4] CAA, CAP 722, 'Unmanned Aircraft System Operations in UK Airspace – Guidance' 7th Edition, amendment 2019/03, 4 September 2019.

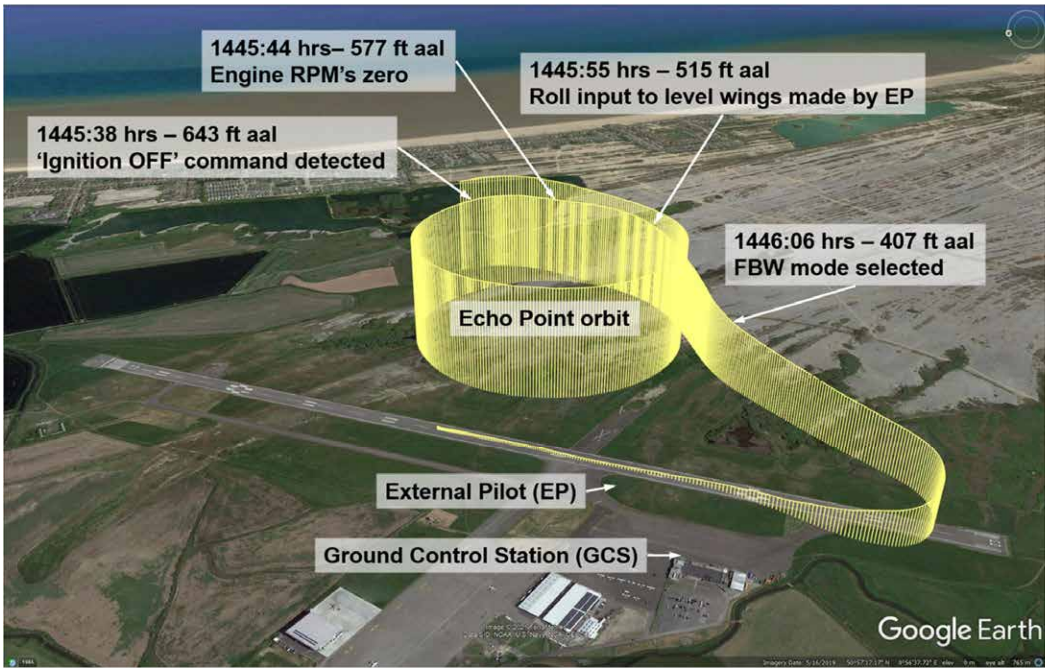


Figure 2
Aircraft flight path during the event

Aircraft information

System description

The Tekever AR5 Evolution UAS (AR5) consists of a manned GCS and an unmanned aircraft (Figure 3). G-TEKV was manufactured and operated by the same organisation and, for this report, is referred to as the operator.

Command and control of the aircraft is achieved through the use of RLOS and Beyond Radio Line of Sight (BRLOS) communication data links. The system has six data links in total, five of which carry primary and secondary RLOS and BRLOS capability. The sixth link is a radio link used for control of the aircraft by the EP. The RLOS control system operates in one frequency band and has a maximum range of 3 km, the backup RLOS system operates at a different frequency and has a range of 4 km. For BRLOS the system uses 4G and Satcom channels to maintain communication with the aircraft. In the event of a loss of the link, the aircraft can remain in a holding pattern to try and re-establish communications. If, after a defined period, the aircraft has not established communications, it can return home and perform an automatic landing.

Figure 3
Tekever AR5 Evolution

Aircraft description

The aircraft has a maximum takeoff mass of 180 kg, a wingspan of 7.29 m, a length of 4.03 m and a fuel tank capacity of 60 litres. It is powered by two 170 cc two-stroke boxer engines manufactured by 3W which generate 9.43 kW at 5,600 rpm. An external electrical engine starter is manually connected to each engine before start-up. Once the engines are running the starters are disconnected. The AR5 does not possess a built-in engine start capability. The aircraft fuel system consists of one central tank in the fuselage and two electric fuel pumps.

The avionics systems can be considered as falling into two main categories. Firstly, all the critical systems that affect safe control and navigation of the aircraft. These comprise the Flight Management System (FMS), Autopilot, Actuator Control Electronics (ACE), Electronic Fuel Injection (EFI) and the sensors and actuators to control the flying control surfaces. Secondly, the avionics and systems that enable exploitation of the aircraft's mission systems. These include video processing and thermal imaging cameras.

In normal flight conditions the aircraft is controlled via the RLOS and BRLOS communication channel from the GCS. However, in an emergency the EP can take direct control of the aircraft via a separate controller with its own radio link. Although there is only one transmitter (used by the EP), at the time of the event the AR5 had a dual receiver system which provided redundancy should one of the receivers fail. The dual receiver system also marginally extended the range at which the EP could control the aircraft. The outputs from the dual receivers were routed through a multiplexer which merged the signals and passed them via the ACE to the Autopilot (Figure 4). When out of range of the EP's controller, receivers are programmed to send a signal down one of the control channels to the autopilot. The autopilot then ignores all the other control channel signals until a receiver detects that the aircraft is back in range.

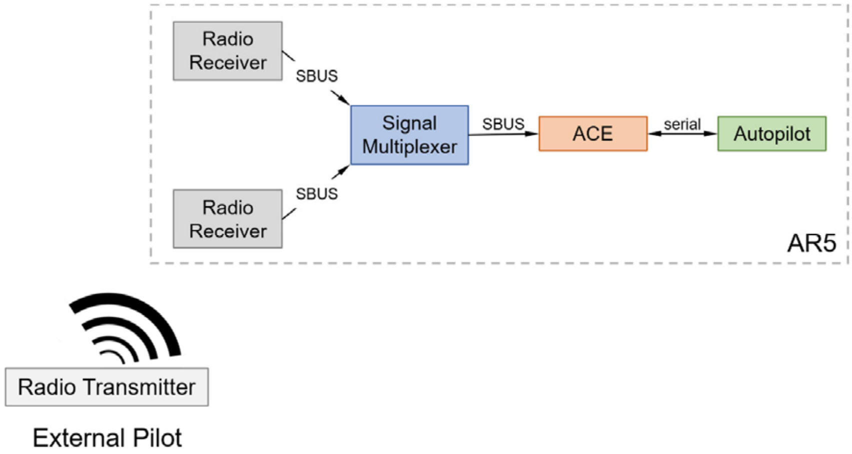


Figure 4
EP control path

Generator power unit (GPU)

The GPU is responsible for the management of the aircraft's electrical power systems, swapping from generator energy to batteries, charging batteries and managing external power sources. Within the GPU, switching regulators were used and the circuitry design followed the recommendations of the regulators' manufacturer. The regulators were certified to meet IPC 9592A, Category 2, Class II.[5]

EP radio link

The EP radio link is used to control the aircraft's ailerons, elevator, throttles, rudder, flight modes and the ignition switch. The autopilot receives command signals via the serial data bus from the multiplexer at a rate of 50 Hz but are only recorded in the flight log at 10 Hz (every fifth value received by the autopilot).

Each of the channel data on the serial data bus was in the form of an 11-bit value (0 to 2,047) for use by the autopilot. For the ignition switch channel, the 'Ignition ON' command was made by sending a pre-defined non-zero value, and the 'Ignition OFF' command by sending a value which is larger than a set threshold. Only one instance of a value exceeding the set threshold was required (of the 50 per second sent) to command 'Ignition OFF' and cut both engines. As such, a single bit error would be sufficient to modify the 'Ignition ON' signal value so that it exceeded the 'Ignition OFF' threshold.[6] Bit errors on the control surface channels, if infrequent, would have little noticeable effect as the control surfaces would not be able to react quickly enough before being commanded back to a 'good' value 0.02 seconds later.

Footnote

[5] IPC-9592A specifies the requirements for design, qualification testing, conformance testing and manufacturing quality/reliability processes of power conversion devices for the computer and telecommunications industries. Category 2 devices are board-mounted dc to dc convertors.

[6] For bit errors in the SBUS data stream to modify the 11-bit default value so that it exceeded the threshold, two combinations existed for a single-bit change, 15 combinations for a two-bit change, 56 for a three-bit change etc. of the 2,048 combinations of an 11-bit word. The highest number (but least likely) of bit-error combinations was 290 for changes to six bits.

GCS stations and displays

General

The fGCS consisted of a dual screen display, of which the primary screen displayed the Mission Station (MiSt) screen (Figure 5). The mGCS consisted of a three-screen display, one of which mirrored the MiSt display.

Figure 5

fGCS Primary display – MiSt screen

Flight Information Bar

The flight information bar is situated on the left of the MiSt screen. A pictorial display of an artificial horizon with speed, heading and altitude information is displayed at the bottom. In the middle of the bar, the signal integrity for each of the communication links is displayed through icons which are coloured according to a traffic light system. Above this, aircraft parameters are displayed and include information on the signals, battery levels, and engine parameters (which include engine rpm and throttle status for each engine). An enlarged view of that part of the flight information bar is shown in Figure 6. None of the aircraft parameters were colour coded according to status.

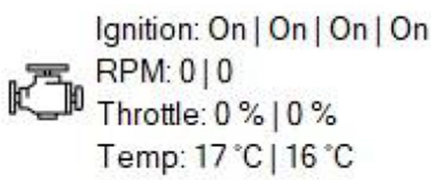


Figure 6

Flight Information Bar – Engine Parameters

Notifications and warnings

The conditions to trigger alerts and warnings could be set and enabled or disabled individually and were specific to the fGCS and mGCS stations. The conditions for engine alerts, including the LOW ENGINE RPM warnings were set on the fGCS station. An audible alert could be either enabled or disabled but applied to all warnings and alerts and could not be configured individually. There was also a configurable snooze function, which allowed an alert to be made temporarily inactive for a specified period of time.

The notification toolbar was situated in the bottom right corner of the screen and displayed any new event notifications, alerts or warnings to the IP. A new alert or warning was brought to the attention of the IP through an ALERTS/WARNING button at the bottom right of the screen which flashed red if a new warning had been activated.

By clicking the ALERTS/WARNING button on the notification toolbar, the IP could bring up a dialogue box with more detailed information on the warning or alert which allowed the IP to choose to either SNOOZE the alarm, IGNORE it, or LOCATE it. Once an action had been selected the ALERTS/WARNING button would turn black with white writing.

Organisational information

Regulation

At the time of the event, the aircraft was operating under the ANO (2016); the CAA had granted an Exemption for BVLOS operations. The original authorisations were granted under CAP 722 edition 6, but at the time of the event edition 7 was extant and stated:

> *'Operators of unmanned aircraft over 20 kg are required to apply for an exemption from the CAA. Any commercial operations aspects will also be covered within this exemption. The application must include a safety case including a risk assessment which demonstrates that the operation can be conducted in a safe manner.'*

The operator had provided an Operational Safety Case (OSC) to support its application to the CAA for an Exemption and, since that time, had updated it where appropriate.

Double Engine Failure

Volume 2 of the OSC outlined the emergency procedures in the event of a double engine failure on takeoff or en route; the routing to and from the airfield was designed to reduce the subsequent risk to third parties or property. The procedures included the designation, prior to departure, of emergency landing locations in case of loss of communications or propulsion, as well as the use of an External Pilot as a safety pilot.

Volume 3 of the OSC outlined the risk assessment. It identified three scenarios of relevance to this event together with measures to control the risk (Table 1).

Scenario	Extant Control Measures
A9 – SP [EP] *takes control of UAV* [Unmanned Aerial Vehicle] *with degraded performance*	1. Only qualified crew and in good health conditions operate Tekever AR5 Evolution 2. Ensure constant communications among crew members during the take-off/landing manoeuvres 3. IP is aware of degraded performance using the fGCS instruments 4. IP will inform EP about degraded performance
O2 – Non-EMI [Electromagnetic Interference] *related Double engine failure*	1. UAV maintenance is done by qualified crew and in good health conditions. Maintenance schedules reduce the risk of engine failure 2. IP monitors engine readings during the flight to detect potential problems 3. The UAV will attempt to land in a pre-planned landing area 4. The crew can use Flight termination system (FTS) to prevent further flight
O15 – Erroneous control actions by the autopilot	*1. UAV maintenance is done by qualified crew and in good health conditions to operate Tekever AR5 Evolution* *2. IP will detect erroneous actions by autopilot and perform the corrections needed* *3. FTS available* *4. Operating in restricted airspace within safety areas'*

Table 1

OSC risk assessment scenarios of relevance to the event

CAP 722 and airworthiness standards

Section 4 of CAP 722 (Edition 6) titled '*Certification & Airworthiness*' contained guidance on certification and the suggested approach for aircraft which do not require certification to formal standards. For this operation, the CAA used the OSC process, where applications would be scrutinised in a proportionate way to the risk its design and usage posed to the general public and property and, where there was a lack of demonstrable airworthiness, risks could be mitigated by operational limitations.

Appendix C of CAP 722 provided a template and guidance on how to complete an OSC. Volume 2, Section 1.1 '*Details of design and manufacturing organisation(s) and any recognised standards to which the equipment has been designed, built and tested*' stated that:

> *'Details of any standards that may or may not be aviation related and may add to the safety argument. Where known this must include test and evaluation evidence.'*

In Edition 7 of CAP 722, the following text was added to this section:

> *'Provide details of any recognised standards to which equipment relevant to the application has been designed, built and tested, for example, aeronautical standards such as EUROCAE and RTCA, or product standards such as ISO, ASTM, STANAG and BSI.'*

The OSC for this operation, developed under Edition 6 of Cap 722, did not contain details of any standards used. However, the operator advised that aircraft had been designed and built using STANAG 4703[7] as guidance, along with '*industry best practice*' but was '*not fully compliant to the standards.*' It also advised that software is designed using DO-178B[8] as guidance during the development process to '*ensure that there were no catastrophic errors.*' No formal testing against any of these standards was made, nor was any required for the Operational Authorisation.

In January 2021, the operator published internal document '*EMC* [Electromagnetic Compatibility] *Integration Considerations*' which detailed '*the procedures and best practices*' to be used by the operator '*to ensure that the integration of new equipment maintains the baseline EMC values*.'

The OSC contained information on how modifications were embodied and how configuration control was maintained. It did not, however, describe the process for developing the modification, but this was later clarified in the operator's document TAS-CMN-QAP-008_00 *Development Procedures in Electronics, Version 1* which was issued on 11 February 2021.

Operator's investigation of June 2020 event

Following the double engine shutdown event on the ground in Portugal in June 2020, the operator analysed the recorded data available and determined that, although the autopilot had received an 'Ignition OFF' command, the command had not been sent from either the GCS or EP.

After further testing, the operator identified erroneous values, in the form of bit errors, within the data passed from the multiplexer via the ACE to the autopilot (Figure 7) and began an investigation to understand how and where these signal errors were being introduced.

Footnote

[7] STANAG 4703 – Allied Engineering Publication AEP-83 '*Light Unmanned Aircraft Systems Airworthiness Requirements*' is a set of technical airworthiness requirements intended for the airworthiness certification of fixed-wing light military UAS with a maximum takeoff weight not greater than 150 kg that intend to regularly operate in non-segregated airspace over all population densities.

[8] DO-178B – Software Considerations in Airborne Systems and Equipment Certification – is a guideline dealing with the safety of safety-critical software used in airborne systems. The Software Level, ranging from Catastrophic (failure may cause a crash) to No Effect (failure has no impact of safety) is determined by examining the effects of a failure condition in the system.

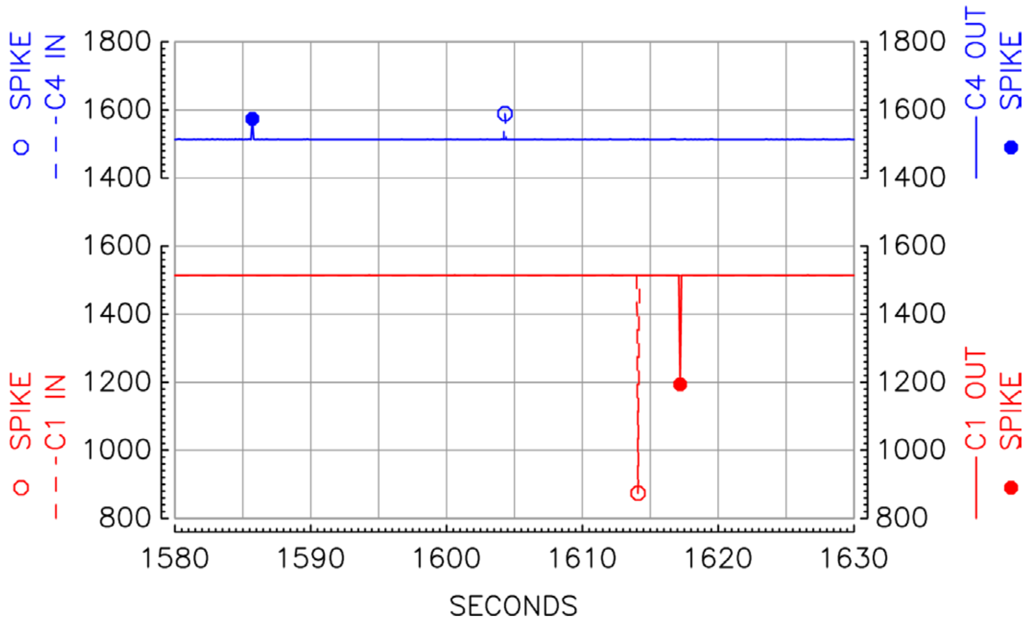


Figure 7

Data from the aircraft log file showing spikes on the control channels in and out of the autopilot for Aileron (C1) and Rudder (C4) for the June 2020 event

The operator's investigation followed several lines of inquiry:

1. ***EP radio link system robustness*** – since it appeared that there were wrong values being sent to the autopilot system.
2. ***Source of noise*** – since it appeared that higher system current loads increased the probability of the event happening.
3. ***Autopilot system*** – study the possibility of the autopilot incorrectly interpreting the control inputs and why the data logged for the ignition switch control contained no erroneous values.

The operator concluded that:

- The event could be replicated only if the multiplexer was in the system and that noise was detected on the output signal from the multiplexer.
- The noise level increased when the current loading from the aircraft's electrical systems increased implying that the GPU was the source of the electrical noise.
- The autopilot recorded the 50 Hz signal data at a rate of 10 Hz (to limit the quantity of data recorded) so some erroneous values might not be logged.
- Checking for a continuous 'Ignition OFF' command over a defined period of time (rather than a single value) would reduce the likelihood of the engines being shut down due to transient erroneous commands.

As a result of the operator's investigation into the June 2020 event the following safety actions were taken:

1. A new SBUS multiplexer hardware revision was released in June 2020 that fine-tuned the internal PCB design and some components to minimise the impact of external interference. It was noted that it was '*not fully protected against noise*'; however, the inadvertent engine shutdown event could not be reproduced during tests. This revision was embodied on the entire AR5 fleet, including G-TEKV which was modified in June 2020.
2. The GPU was redesigned to reduce Electromagnetic Interference (EMI) emissions. The GPU redesign was based on the standards of STANAG 4703, and tests were performed to comply with CISPR25 – Class 1.[9] This work was completed in the second quarter of 2021 and the new GPUs are being installed on new build aircraft and on those returned to the factory for maintenance.

The operator also recommended that the autopilot firmware be updated as soon as possible with a change to the detection of an 'Ignition OFF' command from a single instance to a requirement to have 10 consecutive valid commands. This change would prevent single erroneous 'Ignition OFF' commands from shutting down both engines. However, the operator decided that this was '*not a critical update'* given the apparent effectiveness of the multiplexer redesign. As a consequence, the firmware update was only applied to aircraft returned to the factory, or to new build aircraft. G-TEKV was built prior to June 2020, had not been returned to the factory since that date, and so did not have this firmware update embodied.

Following the serious incident in December 2020, the operator has now embodied this autopilot firmware update on the entire AR5 fleet of aircraft. It has also disconnected one of the EP channel radio receivers and bypassed the multiplexer completely. The operator noted that this decreased the radio range but '*continues to be in the distance needed for safe operation as described in the OSC*'. The operator also introduced an additional procedural control measure to disable the 'Ignition OFF' command unless the aircraft was within 500 m of the EP.

The operator reviewed the implementation of warnings and alerts to users of the GCS. The review noted that alert thresholds could be changed by users but that '*the warnings offer a high level of verbosity'* and that *'multiple warnings can be triggered when in a 'normal' state'*. It identified that this saturation of warnings could result in an alert or warning being made inactive by users to remove the distraction or that warnings risked being ignored.

Footnote

[9] Comité International Spécial des Perturbations Radioélectriques (CISPR) sets standards for controlling electromagnetic interference in electrical and electronic devices. CISPR25 details limits and procedures for the measurement of on-board radio disturbances in the range of 150 kHz to 2500 MHz. The standard applies to any electronic/electrical component intended for use in vehicles as well as boats powered by internal combustion engines, and devices also powered by internal combustion engines. Class 1 is the least stringent of the five classes within the standard.

The review recommended the implementation of a revised warning system with a notification/warning message bar which would identify the alert without user action or intervention, and which would ensure that the information remained visible. In addition, the corresponding parameter on the flight information bar would be highlighted in red. From April 2021, the operator started to implement the revised system status reporting within the GCS software. This includes two alert levels (caution and warning) and a revised flight information bar with parameters colour coded according to status (an extract is shown in Figure 8).

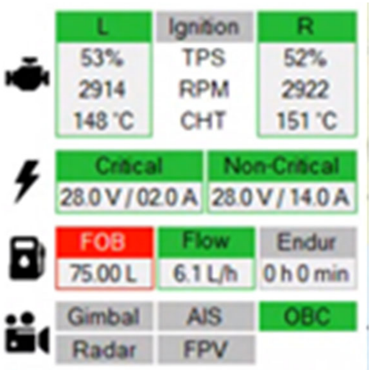


Figure 8
Revised flight information bar extract

Analysis

The event

Whilst the aircraft was returning to the airfield to land, both engines shut down at the same time and there was no means to restart them in the air.

The event occurred over the airfield at a time when the EP was prepared and expecting to take over flight control from the GCS team for the landing. Although the cause of this event was such that it could only have occurred whilst the aircraft was in radio range of the EP, it was fortuitous that it happened at a time when he had good visibility of the aircraft. He saw that there was a problem, took control in accordance with the actions stipulated in the operator's OSC, and his training and experience enabled him to glide the aircraft to an otherwise uneventful 'dead stick' landing on the runway.

There was a short period during which the crew in the GCS were unaware of the aircraft's degraded performance. None of them saw or heard any warnings or alerts associated with the loss of engine power and only became aware of an issue when they observed the aircraft deviating from the planned trajectory because the EP had taken control. The GCS crew noticed the aircraft descend and this then drew their attention to the engine indications on the fGCS primary display. Following an internal review, the operator has implemented a revision to the system status reporting within the GCS software.

The reason for the simultaneous shut down of both engines was likely to be a spurious 'Ignition OFF' command generated on the aircraft's EP controlling link through corruption of data provided to the autopilot from two radio receivers via a multiplexer. The presence of a single erroneous command was sufficient to cause the simultaneous engine shutdowns. This data corruption most likely originated from electrical noise as a result of electromagnetic interference originating from the GPU, which affected the operation of the multiplexer. Data corruptions so caused were difficult to identify as only one out of five samples of data were recorded and the level of noise generated varied with the load on the aircraft's electrical system. Such data corruptions could only have an effect when the aircraft was within range of the EP's transmitter as the autopilot was programmed to ignore any commands received via this radio link at all other times.

Design, build and test standards

Following an earlier similar event in June 2020, the operator did identify and implement improvements to the multiplexer to make it more resistant to electromagnetic interference. Subsequent testing could not reproduce the issue and so the operator considered that the problem had been resolved and this improvement was embodied on all AR5 aircraft. However, believing that the multiplexer redesign had solved the problem, an associated update to the autopilot firmware to improve resilience to spurious commands (which included the agreed change to the 'Ignition OFF' logic in the autopilot software) was considered not to be critical. It was decided that existing aircraft would only be updated on their next return to the factory for maintenance. G-TEKV was equipped with the improved multiplexer but not the updated autopilot firmware at the time of the event in December 2020.

It is unclear exactly what EMC tests were carried out after the June 2020 event and if these were the same as those carried out during the development and testing phase of the aircraft. However, with the modified multiplexer installed, these tests did show that noise could still be detected on the control signals. The fact that the noise did not trigger an 'Ignition OFF' command during these tests and would have had negligible impact on the other control inputs to the autopilot, resulted in a false assumption that the issue had been resolved.

The standards to which the aircraft was built and tested to were not described in the OSC and, although the OSC template in CAP 722 had provision for this information, there was no requirement for it to be included.

Various aeronautical and transport standards were used as guides for the design and testing of the aircraft, representing best practice at the time. No formal testing against any of these standards was conducted so it is unknown if the aircraft design would have passed these. The OSC template referred to in later additions to CAP 722 now include examples of aeronautical standards that could be applied but there are still no requirements for these standards to be met. In March 2021, in a report of an accident to a Alauda Airspeeder Mk II UA[10], the AAIB made recommendations to the CAA and EASA to: '*adopt appropriate design, production, maintenance and reliability standards for all Unmanned Aircraft Systems with aircraft capable of imparting over 80 joules of energy*[11]*.*' In response

Footnote

[10] AAIB Bulletin 03-2021.indd (publishing.service.gov.uk) [accessed 9 August 2021]

[11] 80 joules of kinetic energy (equivalent to a 1 kg mass falling to the ground from about 8 m) is the limit specified in EU Commission Implementing Regulation 2019/947 for UAs in the Open category that can be operated intentionally over 'uninvolved people'.

to these recommendations, both organisations have indicated the intent to develop, and adopt into regulation, such standards as are deemed appropriate and proportionate for the various types of UAS design and operation.

Conclusion

While the unmanned aircraft was orbiting south of the runway in preparation for landing, both engines were commanded to shut down due to a spurious 'Ignition OFF' signal being detected by the autopilot. It was likely to have been caused by electromagnetic interference from the aircraft's generator power unit corrupting some data on the radio channel used by the EP to control the aircraft. The EP on the ground, who was visual with the aircraft, took control and landed it without further incident.

This event followed a similar erroneous 'Ignition OFF' signal six months earlier during a ground test. As a result of the operator's investigation into the first event, design changes were made to the aircraft to limit the effect of interference, not all of which had been implemented by the time of the second event. Following this second event, a number of additional safety actions have been taken by the operator.

Safety actions

Since the June 2020 and December 2020 events, the operator has:

Redesigned the GPU to reduce emissions and is installing these on the AR5 aircraft fleet.

Changed the design of the radio command channel used by the EP and bypassed the multiplexer.

Embodied an autopilot firmware update on the AR5 fleet of aircraft so that 'Ignition OFF' must be asserted for 10 consecutive commands to trigger engine shutdown.

Introduced an additional procedural control measure to disable the 'Ignition OFF' command unless the aircraft is within 500 m of the EP.

Implemented a revision to the alerting system within the GCS software to improve its effectiveness.

Revised its internal documentation to include modification design and embodiment processes and to ensure that consideration is given to EMC for the integration of new equipment.

Published: 21 October 2021.

AAIB Correspondence Reports

These are reports on accidents and incidents which were not subject to a Field Investigation.

They are wholly, or largely, based on information provided by the aircraft commander in an Aircraft Accident Report Form (AARF) and in some cases additional information from other sources.

The accuracy of the information provided cannot be assured.

ACCIDENT

Aircraft Type and Registration:	Corby CJ-1 Starlet, G-CBHP
No & Type of Engines:	1 Jabiru 2200A piston engine
Year of Manufacture:	2005 (Serial no: PFA 134-12498)
Date & Time (UTC):	2 July 2021 at 1605 hrs
Location:	Bluebutts Farm, Slaidburn, Lancashire
Type of Flight:	Private
Persons on Board:	Crew - 1 Passengers - None
Injuries:	Crew - 1 (Serious) Passengers - N/A
Nature of Damage:	Beyond economic repair
Commander's Licence:	Private Pilot's Licence
Commander's Age:	81 years
Commander's Flying Experience:	1,141 hours (of which 332 were on type) Last 90 days - 2 hours Last 28 days - 2 hours
Information Source:	Aircraft Accident Report Form submitted by the pilot

Synopsis

The pilot reported that the aircraft encountered sinking turbulent air very late on finals. The pilot added full power but the aircraft struck a boundary wall causing significant damage to the aircraft and serious injuries to the pilot.

History of the flight

The Corby CJ-1 is a home-built single-seat fixed-wing aircraft with a 340 kg maximum takeoff weight. The pilot was operating from his home strip at Bluebutts Farm. The pilot estimated the strip length at 278 m. The strip was surrounded by higher ground and was sloped so Runway 02 was only used for takeoff and Runway 20 was only used for landing. The wind was from 200° at 6 to 9 kt. The pilot had been flying for an hour which had included a practice stall and some 'touch-and-go' landings. He was flying an approach for a final landing on Runway 20 when he reported encountering sinking air very late on finals. He applied full power and he believed that he would clear the boundary wall. However, the aircraft struck the boundary wall, knocking over a section of it, and causing significant damage to the airframe (Figure 1). The pilot was seriously injured. A witness and the pilot's wife assisted the pilot and drove him to hospital.

The pilot assessed the cause of the accident as sinking turbulent air encountered very late on finals.

Figure 1

Accident site and damaged boundary wall shown on the right

ACCIDENT

Aircraft Type and Registration:	Grob G109, G-CLIA
No & Type of Engines:	1 Limbach L 2000-EB1AA piston engine
Year of Manufacture:	1982 (Serial no: 6108)
Date & Time (UTC):	28 May 2021 at 1630 hrs
Location:	Husbands Bosworth, Leicester
Type of Flight:	Private
Persons on Board:	Crew - 1 Passengers - 1
Injuries:	Crew - None Passengers - None
Nature of Damage:	Extensive damage. The starboard wing and fuselage broke in two, the left wing broke but remained attached. Propeller damaged
Commander's Licence:	Private Pilot's Licence
Commander's Age:	59 years
Commander's Flying Experience:	169 hours (of which 38 were on type) Last 90 days - 5 hours Last 28 days - 5 hours
Information Source:	Aircraft Accident Report Form submitted by the pilot

Synopsis

The pilot seated in the left seat was nominated as pilot in command, and the co-owner, a qualified pilot, instructor and examiner was seated in the right seat and nominated as a passenger for the flight. During the takeoff, aircraft acceleration was sluggish, and the pilot called out his intention to stop. The passenger called out that he had control, and shortly afterwards the aircraft became airborne. It quickly became apparent that the aircraft was not climbing and, after reaching a maximum height of 100 ft agl, it began to descend. It was turned towards a field, but the right wing struck a tree as it descended, causing the aircraft to rotate through 180° and pitch down before striking the ground in a nose-down attitude. The canopy flew open and forward, and both occupants were able to leave the aircraft without assistance. Neither occupant was injured.

The takeoff had been attempted with the propeller in coarse pitch leading to a significant loss of performance.

History of the flight

The pilot arrived at Sulby Airstrip with the intention of flying the aircraft solo for a local flight before landing at the longer Husbands Bosworth Airfield to collect the co-owner. Both pilots had flown the aircraft from Sulby together on previous occasions but had noted that the performance was marginal for the 410 m of runway available if the wind was not favourable.

Despite there being a headwind at Sulby on the day, the pilot decided that he would still fly the aircraft solo and arranged the pickup at Husbands Bosworth. Having completed the pre-flight actions, the pilot took off from Sulby, climbing to 1,200 ft aal before completing a 15-minute local flight during which the propeller was set to coarse pitch. Positioning to join the circuit at Husbands Bosworth, the pilot recalls reselecting fine pitch, before completing a normal landing on Runway 27. After taxiing in, the aircraft was shut down and the pilot exited the aircraft.

Later, having pushed the aircraft into position for starting, both occupants boarded the aircraft and the pilot commenced the pre-start procedures using the aircraft checklist. After the engine was started, the aircraft was taxied for takeoff, backtracking the runway. Once the pilot was happy with the position, both pilots noted that there was a glider on finals so the takeoff was commenced without delay. The pilot had missed the final power check, which required the engine to be run at full power and a check of the maximum rpm to be performed to confirm that the propeller was in fine pitch.

The pilot described the aircraft as sluggish on the takeoff roll and, as the aircraft approached the half-way point of the runway, he checked that the airbrakes were stowed, and the rpm was within normal limits as he was expecting to be airborne by this point. Seeing the yellow winch caravan at the end of the runway approaching, the pilot called out his intentions to abort the takeoff and stop. The passenger in the right seat, then called out that he had control and the pilot let go of the controls. Shortly afterwards the aircraft became airborne, although it was immediately clear that it was not climbing away as expected. The aircraft reached approximately 100 ft agl before beginning to descend. The pilot called to the passenger, who was now flying the aircraft, that there was a suitable field to their left. The aircraft banked left but struck a tree with the right wing as it descended. It turned through 180° before pitching down and striking the ground nose first. The canopy flew open and forward, and both occupants were able to vacate the aircraft without injury. The aircraft was extensively damaged. Figure 1 shows it in its final position.

Figure 1

G-CLIA after the accident (used with permission)

Aircraft information

The Grob G109 is a single engine motor glider with a manually controlled variable pitch propeller.

Propeller pitch

The pitch of the propeller is the angular setting of the blades. This setting affects the efficiency, fuel economy, rate of acceleration and maximum speed of an aeroplane. A propeller with a low blade angle (fine pitch) will rotate easily, taking less of a 'bite' of air each rotation. This allows the engine to spin the propeller at high speed (rpm). Fine pitch is preferable for takeoff and landing, which occur at lower speeds, with the propeller able to rotate at maximum rpm and produce the greatest thrust.

With a higher blade angle (coarse pitch) the propeller will take a larger 'bite' of air with each rotation, but it will limit the maximum speed at which the engine can operate. Cruise speeds are relatively higher and coarse pitch offers a more efficient configuration for this phase of flight. Using the propeller in coarse pitch for takeoff reduces aircraft performance significantly as the propeller rotates at a reduced maximum rpm compared to a finer pitch.

A variable pitch propeller can be compared to the gears in a car in that pulling away or driving slowly is most effective in a low gear (fine pitch on the propeller) but that as the car accelerates, a higher gear (coarse pitch on the propeller) is selected to increase efficiency. Trying to pull away in a high gear in a car also results in poor performance.

Grob G109

The propeller can be set to two pitch settings, fine (known as the climb or start setting) and coarse (known as cruise setting). The propeller can also be feathered completely with the engine off and the propeller windmilling for gliding.

The pitch of the propeller is adjusted using a pull-out propeller control knob in the cockpit, with the feathering accomplished by a separate and lockable feathering handle. For takeoff and climb the propeller is set to fine pitch. In the cruise the propeller can be adjusted to coarse pitch by ensuring the rpm is within limits specified in the pilot handbook, then pulling out the propeller control knob around 7 cm momentarily. There will be an approximately 500 rpm drop if the pitch change has been successful. The maximum rpm will also be lower in coarse pitch than with the propeller in fine pitch. There are no other indications in the cockpit of the position the propeller is in. The propeller can be returned to fine pitch using the same technique.

The pilot of G-CLIA indicated that changing the propeller from fine to coarse pitch was more difficult, and often required more than one attempt. Changing back to fine pitch was more straightforward, almost always changing on the first attempt.

The pilot also commented that after a recent change of propeller, the maximum rpm in fine pitch was lower than normal by around 200 rpm. This meant that the difference between fine and coarse pitch was also less than normal at 300 rpm.

Aircraft examination

The propeller and hub were removed from the aircraft and examined by the AAIB. The examination of the pitch control guides within the hub showed damage that was consistent with the propeller being in coarse pitch when it struck the ground.

Aircraft performance

With the aircraft propeller in fine pitch and at maximum takeoff weight, the performance charts for the aircraft indicate that it requires around 360 m of dry grass to become airborne. Taking into consideration that the grass may have been damp, the expected takeoff distance should be increased by 30%[1] giving a maximum expected distance of 470 m. Information supplied by the pilot showed that the aircraft took over 900 m before it became airborne.

The aircraft performance section of the pilot handbook indicates that the landing distance for an aircraft at maximum takeoff weight is 205 m. With the 30% factored for damp grass this gives 267 m. The runway at Husbands Bosworth is 1,200 m long although G-CLIA did not start its takeoff roll from the beginning of the runway but from a point approximately 25 m in. The distances show that it is likely that almost until the aircraft was airborne there was sufficient distance to abort the takeoff and stop. CAA Safety Sense Leaflet 7b[2] suggests that you should nominate a decision point which it describes as:

> *'Decision point: you should work out the runway point at which you can stop the aeroplane in the event of engine or other malfunctions, e.g. low engine rpm, loss of ASI, lack of acceleration or dragging brakes. Do NOT mentally programme yourself in a GO-mode to the exclusion of all else.'*

Personnel

The pilot, who was sitting in the left seat, had less experience and had been converted onto the aircraft type by the passenger sitting in the right seat who was a qualified instructor and examiner. Although the left seat pilot was nominated as pilot in command, it was the passenger who took control at the critical point on the takeoff roll. There had been no discussion between the parties before the flight about their roles should there be an emergency or unexpected occurrence.

Analysis

Examination of the propeller and hub showed that the propeller was in coarse pitch when it struck the ground. Given the deficiency in takeoff performance demonstrated by the long takeoff roll and the inability of the aircraft to climb away, it was considered that the takeoff was attempted with the propeller in coarse pitch. Although the pilot in command recalled completing the engine start and post start checks from the checklist, the position of the propeller was not picked up. Due to a glider on finals, the pilot did not perform the

Footnote

1 CAA The Skyway Code Version 3 (March 2021) Page 49 available from https://publicapps.caa.co.uk/modalapplication.aspx?appid=11&mode=detail&id=7920 [accessed August 2021]

2 CAA Safety Sense Leaflet 7b Aircraft performance available from http://publicapps.caa.co.uk/modalapplication.aspx?catid=1&pagetype=65&appid=11&mode=detail&id=1913 [accesses August 2021]

engine power checks prior to takeoff, which would have required the engine to be run up to full power to check the rpm was correct for fine pitch. This was the last chance for the pilot to identify that the propeller was not in the correct pitch for takeoff. It was not possible to determine when the propeller was placed in coarse pitch or whether the landing at Husbands Bosworth and the subsequent start-up had occurred in coarse pitch despite the use of the checklist.

The reduction in maximum rpm in fine pitch after the change of propeller may also have contributed to the occupants not recognising the incorrect pitch setting, because they were used to seeing a narrower margin of change than the normal 500 rpm.

It is likely, given the runway remaining when the pilot in command called that he was going to abort the takeoff and stop, that the aircraft would have successfully stopped in the distance remaining, but the passenger either did not hear the pilot in command or did not consider there was enough distance to stop. In continuing the takeoff roll, the aircraft became airborne without sufficient power to climb away. Despite attempting to position the aircraft for a field landing, the right wing struck a tree leading to the aircraft striking the ground causing severe damage.

Flying together can bring great benefits through the exchange of knowledge, greater lookout, and additional mental capacity, but it can also bring the possibility of confusion over who has control of the aircraft especially when unexpected events happen. It is best to discuss these issues with the other pilot before commencing the flight. It may be that, should there be an emergency, the more experienced pilot will assume control, but this should be clearly discussed and agreed beforehand so that both parties are clear about their roles and responsibilities during the flight.

Conclusion

The accident flight takeoff was attempted with the propeller in coarse pitch. Despite using the checklist for the engine start and after start procedures the incorrect position of the propeller was not picked up. The last chance to check the propeller position during the final power checks was omitted probably due to the presence of a glider on finals and a perceived need to be airborne and out of the way before it landed. Whilst this is an understandable action on behalf of a pilot not wanting to be in the way of another aircraft, doing things at haste risks forgetting or missing vital actions that could compromise the safety of the aircraft and its occupants.

The pilot realised that the aircraft was not accelerating as normal and announced his intention to stop. It is likely that there was sufficient runway to abort the takeoff and stop the aircraft until shortly before the aircraft became airborne. The passenger, who was also a qualified pilot, either did not hear the call made by the pilot of his intention to stop or did not believe there was sufficient room to stop and took control of the aircraft. The aircraft became airborne with insufficient performance to climb away and so began to descend. Although the aircraft was positioned towards a field, the right wing struck a tree and the aircraft struck the ground.

Using a decision point on the runway would have given both occupants a good understanding of where it was possible to abort the takeoff rather than continue with insufficient performance. Discussions amongst pilots before the flight can also ensure that should an emergency or unexpected event occur, both pilots know what their roles are to be and what responsibilities they have in controlling the aircraft.

ACCIDENT

Aircraft Type and Registration:	Robinson R44 II, G-OPDG
No & Type of Engines:	1 Lycoming IO-540-AE1A5 piston engine
Year of Manufacture:	2007 (Serial no: 11815)
Date & Time (UTC):	22 July 2021 at 2025 hrs
Location:	Maghera, County Londonderry
Type of Flight:	Private
Persons on Board:	Crew - 1 Passengers - None
Injuries:	Crew - None Passengers - N/A
Nature of Damage:	Damaged beyond economic repair
Commander's Licence:	Private Pilot's Licence
Commander's Age:	34 years
Commander's Flying Experience:	79 hours (of which 18 were on type) Last 90 days - 10 hours Last 28 days - 4 hours
Information Source:	Aircraft Accident Report Form submitted by the pilot

Synopsis

Whilst reducing the rate of descent to land, the low rotor rpm warning light illuminated, and the warning horn sounded. The pilot attempted to recover rotor rpm whilst also manoeuvring the helicopter to avoid obstacles but, on touching down, the helicopter struck a fence and building.

History of the flight

The pilot was considering purchasing G-OPDG and had arranged to fly it from its owner's private site near Dromore, to another private site at Maghera, where he intended to further evaluate the helicopter. The landing area was a grassed area measuring about 30 m x 30 m and was surrounded by a wooden fence of about 1.5 m in height, with an adjacent low rise building. Near one corner of the grassed area was a large tree stump. The landing area is shown in Figure 1. The pilot had previously landed at the site in a Robinson R22.

G-OPDG had flown six hours since its 50 hour maintenance check in April 2021, with the owner reporting that there were no defects noted during this period of operation.

The helicopter's total takeoff weight was 905 kg, which included 155 kg of fuel. The weather conditions were dry with visibility in excess of 9 km, no reported cloud, a temperature of 21°C and a 2 kt wind from 220°. The pilot was using a GPS navigation application installed on a tablet computer, which recorded the helicopter's flight path.

The takeoff at 2003 hrs, the 20-minute VFR flight to Maghera and initial approach were uneventful, with the pilot completing an orbit of the landing site to check it was clear. At a height of 150 ft agl and airspeed of about 40 kt, the helicopter was established onto the final approach to land, having turned onto an into wind heading of 230°. The helicopter's descent rate was about 500 fpm, and its airspeed was gradually reducing as it descended.

When the helicopter was at about 30 ft agl with an airspeed of approximately 15 kt, the pilot started to raise the collective lever and increase aft cyclic, to further slow the helicopter and arrest the descent. The data showed that during the next four seconds the helicopter's descent was arrested at about 10 ft agl and its airspeed was 10 kt, with the helicopter now within a few metres of the landing site. However, the pilot reported that during this manoeuvre, the rotor low rpm warning horn sounded, the low rpm warning light illuminated[1], and the helicopter started to yaw right and descend. The pilot stated this had happened very quickly. He was aware that he was close to the tree stump and recalled that he "overrode the governor to increase the throttle" and may also have lowered the collective, whilst manoeuvring the helicopter to touch down on the grassed area.

As the helicopter touched down, its tail boom hit the adjacent fence, and the helicopter yawed right, turning it towards the adjacent building, which the main rotors subsequently struck before the helicopter came to a stop. The pilot, who was uninjured, shutdown the helicopter. The helicopter was damaged beyond economic repair.

Figure 1
The accident site

Footnote

[1] Low rpm warnings are provided when the rotor rpm is below 97%.

Comments

The pilot attempted to recover the helicopter's low rotor rpm when close to the ground but lost control of the helicopter whilst avoiding a tree stump. It was not established why the rotor rpm had decayed.

Although the pilot had landed uneventfully at the same location before, the site had a number of obstacles which increased the risk of contact with the helicopter if a problem was encountered when close to the ground. The pilot stated that that he would look to reduce this risk in the future by reviewing his landing site selection criteria.

ACCIDENT

Aircraft Type and Registration:	Rotorway Executive 162F, G-OHOV
No & Type of Engines:	1 Rotorway RI 162F piston engine
Year of Manufacture:	2004 (Serial no: 6885)
Date & Time (UTC):	30 March 2021 at 1340 hrs
Location:	Street Farm, Takeley, Essex
Type of Flight:	Private
Persons on Board:	Crew - 1 Passengers - None
Injuries:	Crew - None Passengers - N/A
Nature of Damage:	Extensive damage to main rotor blades, tail boom and tail rotor
Commander's Licence:	Private Pilot's Licence
Commander's Age:	65 years
Commander's Flying Experience:	238 hours (of which 155 were on type) Last 90 days - 2 hours Last 28 days - 2 hours
Information Source:	Aircraft Accident Report Form submitted by the pilot and further enquiries made by the AAIB

Synopsis

On the fifth and final of a series of test hovers to 'bed-in' recently replaced drive belts, the helicopter appeared to lose power. The helicopter was at a height of about 10 ft when this occurred, and the pilot carried out a run-on landing. The helicopter touched down on soft rutted earth which prevented the skids from sliding and they dug-in. The helicopter tipped forward and the rotating main rotor blades contacted the ground. As a result, the helicopter was 'violently' tipped onto its side causing disruption to the main rotor blades and tail boom. No fault or malfunction was found with the helicopter. An analysis of data held within the engine control system could not positively determine the cause for the loss of power although a fault code associated with a Central Processing Unit (CPU) error was registered by one of the engine controllers.

History of the flight

The helicopter's transmission drive belts had been replaced and a series of low level hover test manoeuvres had been carried out to 'bed-in' the drive belts. Approximately two hours flight time had been accrued. During the fifth and final test flight, the helicopter was hover taxiing at 10 kt, at a height of approximately 10 ft, when there appeared to be a loss of engine power.

The pilot initiated a run-on landing, bled off some forward speed and settled the aircraft onto its skids. The helicopter landed at "a fast walking pace". The ground where the skids touched down, was soft and rutted which prevented the skids from sliding and they dug in. This caused the helicopter to tip forward, at which point the rotating main rotor blades contacted the ground. As result the helicopter was 'violently' tipped onto its side causing disruption to the main rotor blades and tail boom.

The pilot was uninjured, made the helicopter safe, released his safety belts and exited through the cabin door.

Discussion

The pilot described how, on the start-up for the fifth flight, there was a problem with one of the fuel pumps. After a brief examination, repositioning and tightening of its electrical connectors, it functioned correctly and so the flight went ahead. All was normal until the pilot became aware of the engine "rolling back" and despite checking and ensuring that the hand throttle on the collective was open, the rotor rpm (N_R) started to decay. The pilot was not aware of any warnings or indications of a malfunction prior to or during the event.

Investigation

Engine control system description

The engine is controlled by a Fully Automated Digital Electronic Control (FADEC) system which incorporate two separate Engine Control Units (ECU)s, referred to as FADEC 1 and FADEC 2. These were removed and downloaded using software and information supplied by the manufacturer.

The ECUs are configured to operate as primary and secondary control devices. The primary ECU (FADEC 1) controls and monitors the engine performance and parameters. The secondary ECU (FADEC 2) monitors in the background and is brought into operation by the FADEC system should a fault develop in the primary ECU or any of its sensors.

The start up procedure requires the engine to be started with the No 1 fuel pump and FADEC 1 selected ON. When the engine has started and stabilised, fuel pump No 2 and FADEC 2 should be set to ON. The system should then be tested by selecting FADEC 1 and the No1 fuel pump to OFF to ensure the engine continues to run on the No 2 pump and FADEC 2. Fuel pump No 1 and FADEC 1 are then reselected to ON. There is a note in the system description in the maintenance manual, that the helicopter can start, hover and fly on the secondary system but it will not control the fuel air ratio as accurately as the primary system. Under certain conditions this will result in a sluggish throttle response.

Analysis

The data shows that FADEC 1 was within its correct parameters with no faults or warnings. In contrast, FADEC 2 shows a discrete indication of a CPU fault during the last few seconds of the data set whilst, at the same time, the throttle position data appears erratic.

Considering the pilot's observations of engine roll back, the erratic throttle position data and the CPU fault recorded, it is possible FADEC 2 was having an influence on engine output rather than the primary. However, there was no indication that FADEC 1 had developed a fault requiring a switchover to FADEC 2. The download also showed that there was a significant difference in the 'data time stamp' between FADEC 1 and 2. It is not clear whether they contained the data from the same flight. An explanation for this difference could not be found.

The problem with the fuel pump was also considered but despite the accident damage to the helicopter, it was found to operate correctly.

Conclusion

The damage to the helicopter's main and tail rotor and associated structure was directly attributable to the roll over after the main rotor blades contacted the ground. Although FADEC 2 recorded a CPU fault and erratic throttle position data and there was a mismatch in the data time stamps between FADECs, no specific fault or malfunction was identified that would have led to the uncommanded power reduction.

SERIOUS INCIDENT

Aircraft Type and Registration:	Brian Taylor AT6
No & Type of Engines:	1 Single cylinder 4 stroke engine
Year of Manufacture:	2020 (Serial no: N/A)
Date & Time (UTC):	19 April 2021 at 1700 hrs
Location:	Fradley, Lichfield, Staffordshire
Type of Flight:	Private
Persons on Board:	Crew - None Passengers - None
Injuries:	Crew - N/A Passengers - N/A
Nature of Damage:	Destroyed
Commander's Licence:	BMFA-B certificate
Commander's Age:	58 years
Commander's Flying Experience:	Unknown hours (of which unknown were on type) Last 90 days - unknown hours Last 28 days - unknown hours
Information Source:	Aircraft Accident Report Form submitted by the pilot

Synopsis

A large model aircraft was being flown at a club site when communication between the pilot and with the aircraft was lost. The aircraft struck the side of a parked lorry trailer in a distribution park. The cause of the communication loss could not be determined.

History of the flight

The Brian Taylor AT6 is a 1:5 scale flying model of the North American T-6 Texan (Harvard). It has a wingspan of 2.3 m (92"), a takeoff mass of 5.4 kg (12 lb) and is powered by a 29.35 cc (1.79 cu in) four-stroke engine using a fuel mixture of methanol, nitromethane and oil.

The aircraft was flown at a model flying club airfield near a distribution centre and the pilot had completed multiple circuits of the field when he lost communication with the UA. The unresponsive aircraft maintained a straight and level attitude in a north-easterly direction, at approximately half throttle, until it struck the side of a parked curtain-sided lorry trailer. A witness nearby heard the impact and reported that the engine continued to run after the aircraft had been destroyed.

The function of the aircraft systems had been checked by the pilot prior to take off, the battery was charged and its voltage verified. Examination by the pilot and other club members after the accident, could not identify any faults which would explain the loss of communication.

It was suspected by the pilot that 2.4 GHz jamming devices were being operated by some of the companies at the distribution centre to prevent staff from using mobile telephones. It was confirmed to the AAIB that no such devices were in operation. Furthermore it is an offence under section 68 of the Wireless Telegraphy Act 2006[1] to "*use any apparatus, including jammers, for the purposes of deliberately interfering with wireless telegraphy (radio communications) in the UK*"[2].

Footnote

[1] Wireless Telegraphy Act 2006 (legislation.gov.uk) - accessed 26 August 2021.
[2] Radio frequency jammers - Ofcom – accessed 26 August 2021.

AAIB Record-Only Investigations

This section provides details of accidents and incidents which were not subject to a Field or full Correspondence Investigation.

They are wholly, or largely, based on information provided by the aircraft commander at the time of reporting and in some cases additional information from other sources.

The accuracy of the information provided cannot be assured.

Record-only UAS investigations reviewed: August - September 2021

11-Jun-21 **DJI Inspire 2** **Shrewsbury, Shropshire**

The UAS was being used to film two motocross riders whilst the track was closed to the public. However, during the flight the UA clipped a flag pole and fell about 4 m to the ground causing a cut to the clothing of a person taking part in the filming.

23-Jun-21 **MA Acrowot** **Near Swindon, Wiltshire**

During the approach to land, the pilot inadvertently applied incorrect roll control which caused the model aircraft to impact the ground. It hit another model aircraft and there were reports of minor injuries from flying debris. The pilot considered it likely that a long period with no flying contributed to him making an incorrect control input.

15-Jul-21 **DJI Mavic Enterprise** **Edgbaston, Birmingham**

Whilst conducting a 'non-GPS' mode training flight at approximately 20 m agl, an incorrect input caused the UA to manoeuvre towards and collide with a tree. Two propellers were damaged.

23-Jul-21 **DJI Mavic 2 Pro** **Thurso, Caithness**

The UA span out of control whilst trying to avoid bird activity. It struck a building roof and was extensively damaged.

03-Aug-21 **Iflight Chimera 4** **Hythe, Kent**

Whilst carrying out a test flight the connection between the controller and UA was lost. The pilot was unable to reconnect with the UA and it entered its fail safe mode. It fell uncontrolled, landed heavily on a shop and damaged roof tiles.

08-Sep-21 **DJI Mavic 2 Pro** **Wandsworth, London**

During a video orbit, the observer misjudged the distance and the UA either struck or flew behind a nearby building. The signal was lost and the UA struck the ground.

09-Sep-21 **Clogworks Dark Matter Hex** **East Ilsley, Berkshire**

The UAS was being operated on a check flight following storage. The operator was losing manual control link. He turned off the link to trigger a return to home; the UA lost control and struck the ground.

14-Sep-21 **DJI Mavic II Pro** **Firth of Clyde**

While operating over water there was a power loss and the UA dropped into the water.

Record-only UAS investigations reviewed: August - September 2021 cont

16-Sep-21 **DJI Mavic Zoom** **Ealing, London**

The UA changed direction without input from the pilot. The controller screen was blank and a yellow 'IMU' (Inertial Measurement Unit) error was displayed. The UAS descended and was lost.

18-Sep-21 **MA Lysander** **Old Warden Airfield, Bedfordshire**

The model aircraft was being displayed at a public event. After turning onto final the pilot believes the model was caught by a gust of wind, stalled and struck the ground. The accident site was beyond the spectator line and away from any third parties.

19-Sep-21 **DJI Mavic 2** **Tingon, North Shetland**

The battery indicated a charge of 96% when the motors were started, but 75 secs into a flight over the sea the operator received a warning that the battery charge was critically low. The UAS initiated an automated landing and the UA began to descend. It was too far away from land and was lost after it descended into the sea.

21-Sep-21 **M1 Mavic Pro** **Bridgend, Glamorgan**

Whilst operating over open ground, control was lost and the UA was substantially damaged when it landed in brambles. The pilot, having seen something detach from the aircraft, believes that the battery fell out during flight.

Miscellaneous

This section contains Addenda, Corrections and a list of the ten most recent Aircraft Accident ('Formal') Reports published by the AAIB.

The complete reports can be downloaded from the AAIB website (www.aaib.gov.uk).

TEN MOST RECENTLY PUBLISHED FORMAL REPORTS ISSUED BY THE AIR ACCIDENTS INVESTIGATION BRANCH

1/2015 Airbus A319-131, G-EUOE
London Heathrow Airport
on 24 May 2013.

Published July 2015.

2/2015 Boeing B787-8, ET-AOP
London Heathrow Airport
on 12 July 2013.

Published August 2015.

3/2015 Eurocopter (Deutschland) EC135 T2+, G-SPAO
Glasgow City Centre, Scotland
on 29 November 2013.

Published October 2015.

1/2016 AS332 L2 Super Puma, G-WNSB
on approach to Sumburgh Airport
on 23 August 2013.

Published March 2016.

2/2016 Saab 2000, G-LGNO
approximately 7 nm east of Sumburgh Airport, Shetland
on 15 December 2014.

Published September 2016.

1/2017 Hawker Hunter T7, G-BXFI
near Shoreham Airport
on 22 August 2015.

Published March 2017.

1/2018 Sikorsky S-92A, G-WNSR
West Franklin wellhead platform, North Sea
on 28 December 2016.

Published March 2018.

2/2018 Boeing 737-86J, C-FWGH
Belfast International Airport
on 21 July 2017.

Published November 2018.

1/2020 Piper PA-46-310P Malibu, N264DB
22 nm north-north-west of Guernsey
on 21 January 2019.

Published March 2020.

1/2021 Airbus A321-211, G-POWN
London Gatwick Airport
on 26 February 2020.

Published May 2021.

Unabridged versions of all AAIB Formal Reports, published back to and including 1971, are available in full on the AAIB Website

http://www.aaib.gov.uk

GLOSSARY OF ABBREVIATIONS

aal	above airfield level
ACAS	Airborne Collision Avoidance System
ACARS	Automatic Communications And Reporting System
ADF	Automatic Direction Finding equipment
AFIS(O)	Aerodrome Flight Information Service (Officer)
agl	above ground level
AIC	Aeronautical Information Circular
amsl	above mean sea level
AOM	Aerodrome Operating Minima
APU	Auxiliary Power Unit
ASI	airspeed indicator
ATC(C)(O)	Air Traffic Control (Centre)(Officer)
ATIS	Automatic Terminal Information Service
ATPL	Airline Transport Pilot's Licence
BMAA	British Microlight Aircraft Association
BGA	British Gliding Association
BBAC	British Balloon and Airship Club
BHPA	British Hang Gliding & Paragliding Association
CAA	Civil Aviation Authority
CAVOK	Ceiling And Visibility OK (for VFR flight)
CAS	calibrated airspeed
cc	cubic centimetres
CG	Centre of Gravity
cm	centimetre(s)
CPL	Commercial Pilot's Licence
°C,F,M,T	Celsius, Fahrenheit, magnetic, true
CVR	Cockpit Voice Recorder
DFDR	Digital Flight Data Recorder
DME	Distance Measuring Equipment
EAS	equivalent airspeed
EASA	European Union Aviation Safety Agency
ECAM	Electronic Centralised Aircraft Monitoring
EGPWS	Enhanced GPWS
EGT	Exhaust Gas Temperature
EICAS	Engine Indication and Crew Alerting System
EPR	Engine Pressure Ratio
ETA	Estimated Time of Arrival
ETD	Estimated Time of Departure
FAA	Federal Aviation Administration (USA)
FIR	Flight Information Region
FL	Flight Level
ft	feet
ft/min	feet per minute
g	acceleration due to Earth's gravity
GPS	Global Positioning System
GPWS	Ground Proximity Warning System
hrs	hours (clock time as in 1200 hrs)
HP	high pressure
hPa	hectopascal (equivalent unit to mb)
IAS	indicated airspeed
IFR	Instrument Flight Rules
ILS	Instrument Landing System
IMC	Instrument Meteorological Conditions
IP	Intermediate Pressure
IR	Instrument Rating
ISA	International Standard Atmosphere
kg	kilogram(s)
KCAS	knots calibrated airspeed
KIAS	knots indicated airspeed
KTAS	knots true airspeed
km	kilometre(s)
kt	knot(s)
lb	pound(s)
LP	low pressure
LAA	Light Aircraft Association
LDA	Landing Distance Available
LPC	Licence Proficiency Check
m	metre(s)
mb	millibar(s)
MDA	Minimum Descent Altitude
METAR	a timed aerodrome meteorological report
min	minutes
mm	millimetre(s)
mph	miles per hour
MTWA	Maximum Total Weight Authorised
N	Newtons
N_R	Main rotor rotation speed (rotorcraft)
N_g	Gas generator rotation speed (rotorcraft)
N_1	engine fan or LP compressor speed
NDB	Non-Directional radio Beacon
nm	nautical mile(s)
NOTAM	Notice to Airmen
OAT	Outside Air Temperature
OPC	Operator Proficiency Check
PAPI	Precision Approach Path Indicator
PF	Pilot Flying
PIC	Pilot in Command
PM	Pilot Monitoring
POH	Pilot's Operating Handbook
PPL	Private Pilot's Licence
psi	pounds per square inch
QFE	altimeter pressure setting to indicate height above aerodrome
QNH	altimeter pressure setting to indicate elevation amsl
RA	Resolution Advisory
RFFS	Rescue and Fire Fighting Service
rpm	revolutions per minute
RTF	radiotelephony
RVR	Runway Visual Range
SAR	Search and Rescue
SB	Service Bulletin
SSR	Secondary Surveillance Radar
TA	Traffic Advisory
TAF	Terminal Aerodrome Forecast
TAS	true airspeed
TAWS	Terrain Awareness and Warning System
TCAS	Traffic Collision Avoidance System
TODA	Takeoff Distance Available
UA	Unmanned Aircraft
UAS	Unmanned Aircraft System
USG	US gallons
UTC	Co-ordinated Universal Time (GMT)
V	Volt(s)
V_1	Takeoff decision speed
V_2	Takeoff safety speed
V_R	Rotation speed
V_{REF}	Reference airspeed (approach)
V_{NE}	Never Exceed airspeed
VASI	Visual Approach Slope Indicator
VFR	Visual Flight Rules
VHF	Very High Frequency
VMC	Visual Meteorological Conditions
VOR	VHF Omnidirectional radio Range

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