Accidents Investigation Branch

Department of Transport

# Report on the accident to Aerospatiale Puma 330J G-BJWS at Aberdeen Airport on 10 October 1982

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**List of Aircraft Accident Reports issued by AIB in 1985**

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| 5/84 | Cessna Citation 500 G–UESS<br>Isle of Lewis<br>December 1983 | February 1985 |
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**Department of Transport**
**Accidents Investigation Branch**
**Royal Aircraft Establishment**
**Farnborough**
**Hants GU14 6TD**

10 June 1985

*The Rt Honourable Nicholas Ridley*
*Secretary of State for Transport*

Sir,

I have the honour to submit the report by Mr D A Cooper an Inspector of Accidents, on the circumstances of the accident to Aerospatiale Puma 330J G–BJWS at Aberdeen Airport, on 10 October 1982.

I have the honour to be
Sir
Your obedient Servant

**G C WILKINSON**
*Chief Inspector of Accidents*

# Contents

Accidents Investigation Branch

Aircraft Accident Report No: 2/85
(EW/C803)

| | | |
|---|---|---|
| *Operator:* | | Bristow Helicopters Ltd |
| *Aircraft:* | *Type:* | Aerospatiale Puma |
| | *Model:* | 330J |
| | *Nationality:* | British |
| | *Registration:* | G–BJWS |
| *Place of Accident:* | | Aberdeen Airport, Scotland |
| *Date and time:* | | 10 October 1982 at 0856 hrs |
| | | All times in this report are GMT |

## Synopsis


The Accidents Investigation Branch was notified of the accident by Aberdeen Airport Air Traffic Control at 0940 hrs and the investigation began the same day.

The accident occurred on a take-off during a co-pilot training flight after the No 1 engine caught fire and failed. The helicopter entered a steep descent from about 225 feet, flared, and touched down heavily in a nose-up attitude. It was rapidly consumed by an intense fire. Both crew members, the only occupants, escaped serious impact injury but died from asphyxiation.

The report concludes that the accident was caused by the failure of the No 1 engine when the crew were carrying out a practice single-engine continued take-off with the No 2 engine retarded, and when the helicopter was at a height and airspeed too low to permit either a recovery using the No 2 engine or an autorotative landing.

The reason for the failure of the No 1 engine could not be established but a possible cause is discussed. Three safety recommendations are made.

# 1. Factual Information

## 1.1 History of the flight

At 0832 hrs on 10 October 1982 Puma G–BJWS taxied out at Aberdeen Airport for a familiarisation training flight with two pilots aboard. The commander, who occupied the port (P2) seat, was a company training captain of considerable experience. The co-pilot, in the starboard (P1) seat, was being converted from the Sikorsky S61 to the Aerospatiale Puma and this was the third flight of his conversion course. The training schedule, amongst other items, called for an engine fire exercise to be carried out on this flight and Bristow Helicopters Ltd (BHL) considered that it was likely that the commander intended to conduct exercises in which the failure of one engine is simulated during the take-off. These exercises comprised rejected take-offs where the helicopter is immediately landed, and continued take-offs where it climbs away. WS took off at 0836 hrs, on a Special Visual Flight Rules clearance, in conditions of a low cloud base and 4,000 metres visibility. After completing two circuits WS was cleared by Air Traffic Control (ATC) to perform rejected engine failure exercises on runway 17. Having made either two or three practice rejected take-offs along the runway, at 0855 hrs WS obtained clearance to take-off and rejoin the circuit.

The helicopter lifted off from a point about 630 metres short of the runway threshold (see appendix 1) and was seen to climb away. During the climb eyewitnesses on the ground saw a 'torching' flame coming from the port engine jet pipe. One witness (an engineer) reported that his attention was drawn to the helicopter when he heard a screaming metallic sound. Five seconds after flames were first seen the helicopter began a steep descent in a nose-down attitude, having achieved an estimated maximum height of about 225 feet. At 0855:56 hrs the commander made a truncated 'Mayday' call, only 'MAYD' being transmitted. The aircraft crash landed in a nose-up attitude on the extended runway centreline 163 metres beyond the end of the runway, and came to rest in an upright position. The airframe was rapidly consumed by an intense fire, which distorted it to the extent of causing the helicopter to roll onto its starboard side. Both pilots died of asphyxiation.

## 1.2 Injuries to persons

| Injuries | Crew | Passengers | Others |
|---|---|---|---|
| Fatal | 2 | – | – |
| Serious | – | – | – |
| None | – | – | |

## 1.3 Damage to aircraft

The helicopter was destroyed by impact damage and fire. A photograph is at appendix 2.

## 1.4 Other damage

There was no significant other damage.

## 1.5 Personnel information

| | | |
|---|---|---|
| *1.5.1* | *Commander:* | Male |
| | *Age:* | 43 years |
| | *Licence:* | Airline Transport Pilot's Licence (Helicopters and Gyroplanes). Valid until 10 January 1987. |
| | *Aircraft Rating (A) Part 1 Pilot in Command:* | Wessex 60, Sikorsky S58T, Bell 212, SA330J Puma, Sikorsky S61N. The last Certificate of Test on the SA330 was dated 9 July 1982. |
| | *Instrument Rating (Helicopters):* | Last renewal test on SA330J was on 12 August 1982. Valid until 30 September 1983. |
| | *Flying Instructor's Rating:* | Valid until 7 August 1984 for conversion and instrument flying training on S61N and SA330J types. The last Certificate of Test on the SA330J was dated 8 July 1982. |
| | *Authorised Examiner* | |
| | *Qualification:* | Valid until 31 May 1985 for S61N and SA330J types. |
| | *Medical Certificate:* | Last examination 11 August 1982, valid until 28 February 1983. No limitations. |
| | *Flying experience* | |
| | *Total:* | 6,700 hours |
| | *Total on type:* | 1,700 hours |
| | *Total in last 24 hours:* | 2:05 hours |
| | *Rest period prior to accident flight:* | 18:30 hours |
| *1.5.2* | *Co-pilot:* | Male |
| | *Age:* | 20 years |
| | *Licence:* | Commercial Pilot's Licence (Helicopters and Gyroplanes). Valid until 20 December 1991. |
| | *Aircraft Rating (A) Part 1 Pilot in Command:* | Bell 47, Sikorsky S61N. Last Certificate of Test was on S61N on 25 July 1982. |

| | |
|---|---|
| *Instrument Rating:* | None |
| *Medical Certificate:* | Last examination 15 October 1981, valid until 31 October 1982. No limitations. |

*Flying experience*

| | |
|---|---|
| *Total:* | 617 hours |
| *Total on type:* | 3:05 hours |
| *Total in last 24 hours:* | 2:05 hours |
| *Rest period prior to accident flight:* | 18:30 hours |

BHL assessed the co-pilot's performance during the conversion training flights he had accomplished so far as above average.

## 1.6 Aircraft information

### *1.6.1 Main particulars*

| | |
|---|---|
| *Manufacturer:* | Aerospatiale, Societe Nationale Industrielle, France. |
| *Type:* | SA330J Puma |
| *Year of manufacture:* | 1980 |
| *Aircraft serial number:* | 1517 |
| *Registered owner:* | Bristow Helicopters Ltd, Redhill Aerodrome, Redhill, Surrey. |
| *Certificate of Airworthiness: (C of A)* | No 10925–1. Transport Category (Passenger). Last renewed 26 February 1982 and valid until 25 February 1983. |
| *Total airframe hours:* | 1592:15 |
| *Engines:* | Two Turbomeca Turmo IVC. Time between overhauls (TBO) is 2,750 hours. The gas generator turbines are renewed at overhaul. |
| *No 1 engine:* | Serial number: 509. Constructed September 1971 as Turmo IVB and converted to model IVC in January 1975.<br>Hours since last overhaul: 2,368<br>Hours since manufacture as model IVB: 3,349 |
| *No 2 engine:* | Serial number: 1338. Constructed March 1977.<br>Hours since last overhaul: 204<br>Hours since manufacture: 4,210 |

1.6.2 *Maintenance*

The maintenance schedule in use was BHL 330J Puma, Issue 2. The schedule consisted essentially of six 50 hour inspections phased over a 300 hour cycle, together with daily inspections, major inspections, and a number of out-of-phase inspections. A Certificate of Maintenance, reference WS003, was issued by Bristow Helicopters Ltd on 10 August 1982. It was valid until 1,677 total airframe hours, or until 11 November 1982, whichever occurred the sooner.

1.6.3 *Weight and balance*

| | |
|---|---|
| *Maximum weight authorised:* | 16,300 lb |
| *Take-off weight:* | 15,853 lb |
| *Estimated accident weight:* | 15,353 lb |
| *C of G Limits:* | 177 to 194.7 ins aft of datum |
| *Estimated accident C of G:* | 188.7 ins aft of datum |
| *Fuel on board at take-off:* | 3,200 lb, divided between tanks as follows: |

| | |
|---|---|
| Group 1 internal | 1,350 lb |
| Group 2 internal | 950 lb |
| Port external | 300 lb |
| Starboard external | 600 lb |

| | |
|---|---|
| *Estimated fuel on board at time of accident:* | 2,700 lb, divided between tanks as follows: |

| | |
|---|---|
| Group 1 internal | 1,100 lb |
| Group 2 internal | 700 lb |
| Port external | 300 lb |
| Starboard external | 600 lb |

1.6.4 *No 1 engine history*

Following its last overhaul by Turbomeca in December 1979 engine S/N 509 was installed in BHL Puma G–BJFY which was operating from Aberdeen. This helicopter was transferred to Malaysia in July 1980. In April 1981 engine S/N 509 was fitted to Puma 9M–SSI which was transferred to Western Australia in April 1982. The engine was removed in June 1982 and returned to Turbomeca for a modification (TU 169) to cure an oil dilution problem.

Subsequent to this work, the following parameters, relative to a new engine were recorded on the test bench. The turbines were not inspected.

| | |
|---|---|
| Power | : 2.5% reduction |
| Turbine inlet temperature ($T_3$) | : +23°C |
| Turbine outlet temperature ($T_4$) | : +5°C |

The engine was then installed in G–BJWS at Aberdeen in September 1982, at 2,250 hours since overhaul, as the No 1 engine. This was the occasion on which the engine's maximum contingency power rating was last checked. Since then it had run

118 hours of which 12:20 hours had been flown during 9 training flights. It was not possible to establish the number of practice engine failure on take-off procedures that the engine had experienced as the 'live' power unit during its life. BHL estimated that of the 12:20 hours, some 12 continued and 12 rejected take-offs would have been practised in G–BJWS, probably shared equally between the two engines. However the previous flight to the accident, conducted on 9 October, was the only one of the 9 training flights during which the temperature at aerodrome level was less than 10°C and visible moisture was present ie in potential engine icing conditions. This flight included engine failure on take-off exercises, comprising both continued and rejected take-offs.

### 1.6.5 *Engines*

#### 1.6.5.1 *Description*

The Turmo IVC is a turboshaft engine comprising a gas generator and a free turbine (see diagram at appendix 3). The output shaft from the latter is at the rear and drives the main rotor gearbox. The gas generator is a single spool device consisting of one axial and one centrifugal compressor, connected to a two stage turbine section. Fuel is delivered to the hollow central shaft, whence it is injected into the annular combustion chamber via small holes drilled radially into the shaft. An exhaust diffuser makes up the connection between the gas generator turbine section and the free turbine. The gas temperature in this region is denoted T4, and the sensing for a gauge in the cockpit is by means of two thermocouple probes mounted in the diffuser. Both the engines in G–BJWS were pre-Modification TU 162; this modification fixes the gas generator speed adjustment of the maximum contingency stop before the engine leaves the manufacturer.

The engine instruments include two T4 indicators, two gas generator speed (Ng) indicators, and two rotor speed indicators. The T4 indicators are two inches in diameter (see appendix 4). They are graduated in intervals of 20°C up to 800°C only, although the end stop is at an extrapolated value of 830°C. A red arc extends over the interval 790°C to 830°C.

During the investigation BHL stated that in their experience the small size of the T4 indicator together with parallax error resulting from its position makes it difficult for pilots to assess precise T4 values quickly. To overcome the problem BHL are carrying out trials with a new type of indicator which has an improved analogue scale, a digital read out, and a green/amber/red set of warning lights which illuminate at pre-selected values.

#### 1.6.5.2 *Engine anti-icing*

Engine anti-icing is provided by means of hot air bled from the gas generator compressor, and fed into the front bearing supports and the intake area. The Flight Manual in Section 1, Limitations, stated in block capitals: "The SA330J 'Puma' helicopter must be operated in compliance with the limitations prescribed in this section", and also "These engines are provided with an anti-icing system which must be operated when the outside air temperature is less than +10°C (see Section 2.3, page 5) in conditions of precipitation and visible moisture". The instructions in Section 2.3 page 5 made no reference to take-off, being concerned with minimum gas generator speed during en-route flight. However, Section 2.2.2, Normal Procedures, Pre Take-Off checks listed as an item "Engine anti-icing system . . . . . . . . . . off"; and Section 2.2.3, Take-off, stated: "On take-off cabin air conditioning and anti-icing systems should be switched off. They can be switched on after take-off at an altitude of about 30m (100 feet) if outside air temperature is below +10°C". BHL stated that their check lists for the Puma called for the engine anti-icing switches to be set "as required" prior to take-off because they believed that, to comply with the limitations section of the Flight Manual, the engine anti-icing would have to be on for take-off and landing in conditions of visible moisture at ambient temperatures of +10°C or below.

Additional temperature rises are imposed by the selection of engine anti-icing whereby high pressure air is bled from the compressor, some of which is re-ingested at the intake. The increased temperature of the inlet air is thus propagated through the engine, resulting in a higher T3 turbine entry temperature. There is a further T3 rise due to an increase in the fuel flow needed to prevent any reduction in Ng caused by air being bled from the compressor. Depending on the age of the engine, the T4 increase can be 30 to 40°C at take-off power, however the T4 limitations stated in the Flight Manual would still apply.

Aerospatiale and Turbomeca thought that BHL was conforming with Flight Manual, Section 2, Normal Operations, where the Pre Take-Off Checks called for engine anti-icing to be off. It seems, therefore, that the engines in BHL Pumas had on occasion been exposed to more severe conditions than those envisaged by Turbomeca when the Turmo IVC's overhaul life was established.

### *1.6.5.3 Limitations*

The Flight Manual (Section 1) gave the power plant limitations as follows:

(a) Both engines operative:

(i) Maximum take-off rating: 99.8% Ng (33,450 rpm) and 780°C T4, 5 minutes maximum duration

(ii) Maximum continuous rating: 97% Ng (32,500 rpm) and 710°C T4

(b) One engine inoperative:

(i) Maximum contingency rating: 100.9% Ng (33,800 rpm) and 790°C T4, 2½ mins maximum duration

(ii) Intermediate contingency rating: 99.5% Ng (33,350 rpm) and 770°C T4, no time limit.

Following a Flight Manual amendment which increased the one engine inoperative intermediate contingency rating from 98.2% to 99.5% Ng, BHL was concerned lest the use of high power in single engine training might be detrimental to engine life. They therefore, with Civil Aviation Authority (CAA) approval, issued Change Sheet No 1 to the Flight Manual. This read as follows:

"*Training and Testing of Pilots – Reduced Power Settings*

To reduce the possible accelerated engine deterioration brought about by the continued use of single engine high power settings during the training and testing of pilots, the following simulated one-engine inoperative reduced power ratings shall be observed:

a. Maximum contingency power (2½ minute rating) shall be limited to 99.5% Ng

b. Intermediate contingency power shall be limited to 97% Ng

To offset these reduced ratings all performance graphs shall be entered with the actual OAT increased by 8°C. This will ensure that the same effective performance margins, using standard procedures, are preserved. For information Group 'A' WAT limiting performance at 97% Ng shall not be restricted below 16,300 lbs max certificated weight below 27°C at sea level."

### 1.6.6 *Fuel control units*

The fuel control unit for the Turmo is a hydromechanical device comprising a fuel pump, a fuel flow control for starting, a free turbine governor, and a gas generator governor. In addition there is an acceleration control unit, a barostatic compensator, and an overpower stop which is solenoid operated. The free turbine speed is directly proportional to main rotor speed as the two are linked via the main rotor gearbox. Thus, if a rotor speed droop occurs (due, for example, to a loss of power from one engine) then the free turbine governor of the other engine will act on its metering valve to allow it to increase power. The gas generator speed will increase (and therefore the free turbine speed) until the rotor speed is restored or the Ng limit is reached, at which point the gas generator governor acts to bleed fuel away to return. The gas generator governor setting can be altered slightly by means of the overpower stop, the solenoid of which is controlled by a pilot operated push-button switch. The solenoid is normally de-energised (stop disengaged) for take-off, so permitting gas generator speed to increase to 33,800 rpm – the maximum contingency rating of 100.9% Ng. The solenoid is then energised (stop engaged) during the after take-off checks, limiting gas generator speed to a maximum of 33,450 rpm – the maximum take-off rating of 99.8% Ng.

BHL practice on training flights involving engine failure practices was to operate with the overpower stop engaged. However they stated that it was usual for a training captain to demonstrate, at the beginning of a conversion course, a continued take-off exercise with the overpower stop dis-engaged so as to give the trainee experience of the conditions he would encounter following an actual engine failure. BHL stated that this was in accordance with the practice of Aerospatiale's own instructors.

### 1.6.7 *Aircraft thermometers*

G–BJWS was equipped with a direct reading outside air temperature (OAT) probe and indicator, mounted on the windshield. Also fitted was a second OAT instrument which was electrically operated. The tolerance of each system was stated to be ± 3°, and the CAA stated that this is taken into account in the appropriate sections of the Flight manual ie the crew do not have to apply a correction to the indicated values.

### 1.6.8 *Main rotor limitations*

The aircraft's Flight Manual laid down rotor speed limits as follows:

Power-on flight

Minimum rotor speed (transient). . . . . . . . . . . . . 220 rpm
Governed rotor speed . . . . . . . . . . . . . . . . . . . . . 265 ± 7 rpm

Autorotative flight

Maximum rotor speed ...................... 310 rpm
Minimum rotor speed:

– 220 rpm if I.A.S. below 200 km/h (108 knots)
– 240 rpm if I.A.S. above 200 km/h (108 knots)

1.6.9 *Take-off procedures*

BHL stated that the group A, clear area, take-off procedure laid down in the SA330J Flight Manual, Supplement 8, would have been the one employed. (See diagram at appendix 5). The relevant performance data applicable under the conditions pertaining have been extracted from the Flight Manual and are given below. For the Puma the critical decision point (CDP) on such a take-off is defined by a critical decision time.

Take-off safety speed (VTOSS): a minimum of 43 knots
Critical decision time (Tc): 6 seconds
Acceleration-stop distance (d): 730 feet
Take-off distance required (D1): 850 feet
Single engine rate of climb at VTOSS, wheels down: 650 ft/min
Best rate of climb speed (Vy): 70 knots

The Flight Manual, Supplement 8, described the action to be taken following an engine failure on take-off at or after the critical decision point as follows:

"(2) At critical decision point.

In case of failure of an engine at the critical decision point, it is possible either to land following the previously described procedure, or to proceed with take-off.

In this case, the procedure is as follows:

| | |
|---|---|
| THE PILOT: | Reduces pitch to 13° (with 13° pitch, rotor rpm will stabilize at about 255 rpm; it is recommended to maintain this value) |
| THE CO-PILOT: | Announces rotor rpm |
| THE PILOT: | Accelerates the aircraft speed up to VTOSS if necessary. Maintains speed until aircraft is 500 feet above the terrain, maintaining 255 rpm minimum rotor speed. At this height, accelerates, in level flight, up to Vy = 70 knots. |
| THE CO-PILOT: | Retracts the landing gear. |
| THE PILOT: | Continues to climb at Vy, setting the engine to intermediate contingency rating. |

(3) After the critical decision point.

Should an engine fail after the critical decision point continue take-off."

### 1.6.10 *Autorotative landing procedure*

The Flight Manual, Section 3, Emergency Procedures described the method of carrying out an autorotative landing as follows:

"Intentional autorotative landings should be avoided.

However, in case of necessity, comply with the following procedures:

A. ON LAND

- Collective pitch at low stop
- Extend the landing gear
- Lock the nose wheel
- Zero the horizon miniature if not already zeroed
- Final approach into wind at 150 km/h (80 knots)
- Take a 15° nose-up attitude – at about 40 m (130 ft) if weight is heavy
  – at about 20 m (70 ft) if weight is light
- On approaching the ground increase collective pitch and lower the nose of the aircraft to about 10° nose-up attitude in order to hold a forward speed on touch-down of about 80 km/h (45 knots) at a weight of 7,000 kg (15,400 lb)
- Maintain the stick in this position when the main wheels contact the ground
- After nose wheel touch-down, gently reduce collective pitch and apply the brakes (when IAS is lower than 60 km/h – 32 knots)"

NOTE: On touch-down, the rotor rpm can be sufficiently low to produce complete failure of the ac generating system power supplies and AP system, (battery bus-bar remains powered)."

## 1.7 Meteorological information

A weather observation taken at Aberdeen Airport at 0902 hrs on the day of the accident was as follows:

| | |
|---|---|
| *Wind:* | 070°/7 knots |
| *Visibility:* | 4,000 metres |
| *Weather:* | Drizzle |
| *Cloud:* | 1 okta at 400 feet<br>5 oktas at 500 feet<br>8 oktas at 900 feet |
| *Temperature:* | 11°C, dew point 10°C |
| *Pressure:* | QNH 1012 QFE 1004 |

## 1.8 Aids to navigation

Not relevant.

## 1.9 Communications

An examination of the recording of the Aberdeen Airport Tower frequency of 118.1 MHz revealed that all radiotelephony (RTF) messages passed between Aberdeen and G–BJWS were normal from the start of the flight until 0855 hrs when WS was given clearance to take-off and join the circuit once again. No further full communication was received from WS but at 0855:56 hrs a truncated 'Mayday' call was made, only 'MAYD' being audible.

The signal content of this transmission was examined by means of a spectrum analyser. Although the brevity of the call made identification of extraneous sounds impossible, one relevant fact was established. The characteristic 'transient' when a press-to-transmit button is released, was not present. This indicates that the message was terminated by a sudden disruption of the transmitter, thus fixing the time of the ground impact at the time of the transmission.

## 1.10 Aerodrome information

Aberdeen airport is 215 feet amsl. Runway 17 is 1,829 metres x 46 metres.

## 1.11 Flight recorders

G–BJWS was not equipped with a flight data recorder or a cockpit voice recorder, nor were these required to be fitted.

## 1.12 Wreckage and impact information

### *1.12.1 On-site examination*

The aircraft crashed into a level grass field approximately 550 metres from the end of runway 17. The tail rotor had struck the ground some 10 metres beyond the airfield boundary hedge. The fuselage impacted approximately 5 metres further on, leaving the nose undercarriage embedded in the earth. The main wreckage was in a fairly compact area centred some 30 metres from the first point of impact. The furthest wreckage item was the left undercarriage, which came to rest 20 metres beyond the main wreckage. The two cabin entry doors and the liferaft (which had inflated as it was ejected at impact) were lying close to the main wreckage, and were virtually the only wreckage items unaffected by fire. It was evident that the tail boom had become detached at the initial impact point, although it had come to rest in the main wreckage area.

The short distance of 22 metres between the first point of impact, made by the tail rotor, and the main wreckage suggested a low forward speed just prior to impact. The distance between the tail skid mark and the main undercarriage impact point was slightly less than the distance between these two parts of the airframe. Thus it was concluded that the mainwheels struck the ground fractionally before the tail rotor. The airframe geometry indicated that the impact attitude must have been less than 15° nose-up. The distance between the main undercarriage marks and the nosewheels was almost twice the wheelbase of the helicopter. This confirms the nose-up impact attitude, as the aircraft rolled for a short distance on its mainwheels before the nosewheels dug in. The precise point of impact for the nosewheels could not be determined as the ground had been disturbed by the fuselage underside. The tail and fuselage ground marks did not follow exactly parallel tracks, and were angled in a manner which indicated that the aircraft was yawing to the right at the time of impact. The score made by the left main undercarriage occurred 2 metres earlier than that made by the right leg, thereby indicating that the aircraft was rolled slightly to the left at the time of impact. The disposition of the wreckage suggested that the fuselage continued to yaw to the right after impact, and that the aircraft remained upright.

Only one main rotor strike could be discerned in the ground surrounding the wreckage. It was considered that as the aircraft remained upright after impact, the rotors only contacted the ground when the fuselage rolled over to the right as a result of the ground fire consuming much of the structure, with all the blades successively striking the ground at the same point.

The wreckage had been substantially consumed by the post impact fire. It was nevertheless possible to determine several cockpit settings as follows:

> No 1 engine's fuel flow control lever (FFCL) was in its forward detent, ie engine in governed range
>
> No 2 engine's FFCL was in a retarded position, some 0.8 inches forward of its aft detent; this was the position to which it is moved to simulate an engine failure.

The rotor brake lever, the hydraulic system change-over lever, and the two emergency fuel shut-off levers were all in the forward ie normal positions. The undercarriage selector was in the down position. Subsequent detailed examination led to the conclusion that all of these were the pre-impact positions. It was noted that the engine fire extinguisher buttons had not been operated, and that the switch that isolates the electrical supplies had also not been operated.

It was established that the free power turbine of the No 1 engine had suffered visible heat damage to the tips of the blades, also that the inside of the jet pipe was covered with metal spatter. The T4 temperature probes were also found to be coated with solidified molten metal to the extent that they could not be withdrawn through the holes in which they were located in the engine casing. A boroscope examination subsequently confirmed that the gas generator turbine had suffered severe damage, with approximately 60% and 50% respectively of the span of the first and second turbine stages having disappeared.

A superficial examination of the No 2 engine revealed no defect. The fuel flow lever on the fuel control unit was noted to be in broad agreement with the flight deck lever position, ie it was at a low rpm setting.

### 1.12.2 *Subsequent detailed examination*

The wreckage was recovered from the accident site in order to facilitate a more detailed examination. The engines were sent to Turbomeca at Tarnos, France, where they were subjected to strip examination under AIB supervision.

*1.12.2.1 Engines*

No 2 engine

External damage was comparatively minor, largely being confined to fire-blackening and some slight crushing of the oil tank. Considerable quantities of mud were found in the intake area. Mud shavings created by the axial compressor were discovered and mud had penetrated as far as the diffuser area, some also being found in the dilution holes of the combustion chamber. It was thus evident that the engine was rotating at the time of the accident, the mud probably entering the intake when the aircraft rolled onto its side. Examination of the gas generator turbines revealed nothing of note; in particular there were no signs of blade contact on the shrouds. There was evidence of the free power turbine blades rubbing on the casing. This was in the form of light scoring on the left hand arc, and fairly severe scoring on the right hand arc. It was considered that the latter occurred when the aircraft rolled over and the rotor system, and hence the free power turbine, came to an abrupt halt.

The remainder of the No 2 engine was found to be in generally good condition and no evidence was found to suggest that there had been any failure or malfunction.

No 1 engine

Disassembly confirmed that the outer 60% of the gas generator first stage turbine blades and 50% of the second stage blades had disappeared. In addition, the tips of all the free power turbine blades had been eroded away, resulting in increased clearance between the blade tips and the casing. This increased clearance explained the absence of rubbing marks on the casing, such as was observed on the No 2 engine. The leading edges of the first stage nozzles were in good condition but the trailing edges, together with the second stage and free power turbine nozzles, had suffered impact damage from turbine blade fragments.

The remainder of the engine components were in generally good condition, and it was clear that there had been no mechanical failure elsewhere within the engine that could have caused the damage in the gas generator turbine.

Turbine blade samples from the No 1 engine were examined at Turbomeca's laboratories and later by the Royal Aircraft Establishment, Pyestock. Consideration of the metallic phases present in the microsections led to an estimate of the temperature experienced in the gas generator turbine stages. Both analyses were in broad agreement indicating a first stage temperature of 1000 to 1080°C, and a second stage temperature of approximately 1250°C. A similar assessment of the free power turbine blades resulted in a figure of 1100°C. Turbomeca considered that the temperature in the first turbine stage could have been reached by the use of maximum contingency power with engine anti-icing in use.

At the end of their examination Turbomeca concluded that the failure of the No 1 engine had resulted from the effects of first stage gas generator turbine tip rub caused by excessive creep. However when the investigation was thought to be complete new evidence was introduced which raised a serious doubt. Evidence was given by Rolls-Royce Ltd to the effect that from their experience of the Turmo engine they did not consider the failure could have been the result of a gas generator turbine blade tip rub. It was therefore decided to conduct a further examination of the turbine casing. Accordingly, sections were cut from the casing in the area of the first stage turbine and subjected to an examination by the RAE, Pyestock. This revealed the presence of a thin oxide layer between the casing material and the solidified molten spatter that had originated from the turbine blades. Even in areas where the oxide layer was absent (ie the solidified spatter was adhering directly to the casing material), it was evident that it had merely been displaced by the effects of the high impact speed of the molten fragments. The oxide layer is normally present on the surface of the casing material unless abraded away during a tip rub. It was therefore concluded that a severe tip rub had not taken place.

Turbomeca, when asked to comment on these findings, introduced the subject of microscopic cracks that can occur within the blade material on turbines that have suffered creep – see Section 1.17.5. They undertook to conduct a series of tests whereby a number of specimens were cut from the No 1 engine gas generator turbine wheel and subjected to steady tensile stresses at varying temperatures, noting the time to failure. Some of the specimens were notched to simulate the notch effect caused by microcracks.

The results were not conclusive; however there were certain similarities between a typical rupture surface on one of the blades to those on notched specimens that had been subjected to creep tests at 760°, 850° and 1000°C. Turbomeca also examined another Turmo IVC that had been operated in Kenya, and which had suffered considerable erosion and oxidation on the leading edges of the gas generator turbine blades. A considerable amount of creep had also occurred, the turbine wheel diameter having increased by 0.95 mm, and the engine had been rejected from service at 932 operating hours. In the laboratory, one of the turbine blades was failed by impact loading it at room temperature. The resulting fracture face revealed a network of microcracks near the leading edge; the impact loading had exploited these intergranular cracks, and a brittle fracture had occurred over the rest of the blade. Once again, these were similar in character to those observed on some of the turbine blades from G–BJWS's No 1 engine.

Turbomeca's conclusion was that failure of the G–BJWS engine was in fact the result of a slow damage process, namely creep plus erosion and intergranular oxidation, which was followed by a brittle fracture due to operating the engine with a high turbine entry temperature ($T_3$).

*1.12.2.2 Fuel control units*

After confirming that the drives between engine and fuel control units were secure, the latter were removed and subjected to a rig test by Turbomeca under AIB supervision. Both units had been externally fire blackened, and the one from the No 1 engine was observed to have a fuel leak. However it was considered that this was the result of the effects of fire. It was not possible to determine the pre-crash positions of the over-power solenoids, as the relatively low impact speed would not be expected to result in any witness marks being made on these components. Both solenoids were found in the de-energised (stop disengaged) position, which is the position to which they are spring loaded in the absence of electrical power. The associated pilot-operated switches are of the spring-loaded push-button type, which do not have discrete positions.

Rig tests were performed to function the gas generator and free power turbine governors, and confirmed the correct adjustment of the take-off and maximum contingency stops. The acceleration control units were also tested; these were found to be sluggish in operation, the one from the No 1 engine particularly so. Subsequent strip examination revealed no evidence of a fault, and it is considered that the poor results were due either to the effects of fire, or the introduction of air and/or contaminant particles during the time the units were being prepared for mounting on the test rig. In the case of No 1 engine acceleration control unit, replacement of the fabric diaphragm cured the problem and the acceleration values on test were then within test schedule limits.

*1.12.2.3 Engine bleed air*

Cabin heating air is bled via a valve on each engine, both of which are controlled by a single electric actuator. The position of the valves confirmed that the cabin heating was switched off. Engine anti-icing air is controlled by means of an electrically operated valve on each engine. Examination of the wreckage revealed that both valves were open ie anti-icing was switched on.

*1.12.2.4 Airframe*

Much of the aircraft structure had been completely consumed in the intense post-crash fire. The only major portions remaining were the lower rear fuselage and the cabin roof, which was still attached to the engine bay structure and the rotor head. It was therefore impossible to obtain an accurate assessment of the structural distortion that had occurred at impact.

The lower rear fuselage structure contained the substantially intact fifth tank (albeit holed and burnt) and the main undercarriage attachments. Examination revealed that both main undercarriage retraction/extension jacks had suffered overload failures, thereby allowing the landing gear to fold rearwards with consequent ground contact of the fuselage underside.

The two crew seats appeared to have remained intact during the accident although the frameworks were severely heat distorted. Both cabin entry doors became detached from the aircraft at impact and were virtually undamaged. The crew doors were found together with some of the nose structure underneath the wreckage of the cockpit area. Lying thus, they had been partially protected from the post-crash fire, and it was evident that they had suffered some pre-fire distortion. This was probably as a result of the nose of the aircraft contacting the ground heavily after the nose undercarriage broke off during the impact. There was no evidence to suggest that the doors had been opened or jettisoned. The liferaft, which is stowed in the cabin, had deployed when the pack was ejected at impact along with the cabin door.

Considerable effort was expended in finding the remains of the snow shield. This is a glass-fibre screen attached to the cabin roof forward of the air intakes. Consideration was given to the possibility of the shield having become partially detached and causing intake blanking of No 1 engine. Due to the fire, very little remained. However the centre and right hand attachments were recovered and identified, and their integrity was confirmed. The left hand attachment was also recovered, but the effects of the fire were such that it was not possible to establish whether or not it was attached to the roof. The possibility of its detachment was considered to be remote; nevertheless Aerospatiale considered that the shield would remain secure with only two of the three attachments intact.

*1.12.2.5 Fuel tanks*

A diagram of the fuel system is at appendix 6.

The aircraft was equipped with the standard four under-floor tanks; these comprised the forward transverse, the rear transverse, and the left and right longitudinal tanks. Also fitted were the optional fifth tank under the floor in the rear fuselage, and two sponson tanks. Apart from the fifth tank, and a section of wreckage containing most of the right hand longitudinal tank, very little of the floor/tank structure survived the post-crash fire. It was thus not possible to establish the degree of damage that the fuel tanks had sustained during the impact. It is considered however, that the outlet from the fifth tank would have been one major source of fuel release. The sponson tanks had been substantially burnt during the post-impact fire, although it is considered probable they suffered little significant damage during the impact; this was because the main undercarriages failed in a rearwards as opposed to an upwards direction.

The fuel tank bladders were virtually all consumed in the fire. A portion of the fifth tank survived; it was found to have an average thickness of 0.94 mm and was bonded to a single ply of looseweave nylon fabric.

A few gallons of fuel and post-accident rainwater were found in the fifth tank when the wreckage was removed. The rest of the fuel had burned in the post-impact fire.

*1.12.2.6 Examination of fuel and oil samples*

A sample from the bulk fuel installation and the daily fuel sample from the aircraft were analysed by the Materials Quality Assurance Directorate (MQAD) at Harefield. The installation sample met the specification of Avtur, the aircraft sample failed to meet the specification requirement for water reaction. However, BHL stated that the sample room became flooded as a result of heavy rain and that water contamination may have occurred. In any case, the nature of departure from specification could not have had any bearing on the failure experienced by the No 1 engine.

A sample of oil was recovered from the reservoir of engine S/N 509 and analysed by MQAD. No significant anomaly was reported, the sample being consistent with ASTO 750.

## 1.13 Medical and pathological aspects

Autopsy examinations were carried out on both pilots. No evidence was found to suggest a causative or contributory medical factor in this accident.

The pilots had sustained no injuries which would indicate an inability to escape. The only injuries resulting from impact were observed on the commander in the form of bruising to the right side of the chest and the disruption of the 2nd lumbar intervertebral disc; both injuries could have been received following acute flexion or, in the case of the disc injury, heavy vertical impact. Both crew members had significant concentrations of carbon monoxide in their blood, indicating survival following impact. The cause of death in both cases was given as asphyxiation in the post-crash fire.

It was the opinion of a consultant pathologist from the RAF Institute of Pathology and Tropical Medicine that the spinal injuries suffered by the commander were by site and nature very similar to those experienced by pilots on ejection, particularly if not in tight contact with their seat pack. This indicated a peak acceleration of not less than 14 'g' at the bottom of the commander's seat.

## 1.14 Fire

### *1.14.1 Origin and severity*

The original source of the fire was within the gas generator section of the No 1 engine. Witnesses initially observed an orange flame emitting from its jet pipe, which continued for approximately 5 seconds. External fire was then reported in the jet pipe area of the fuselage. They reported that there was a loud explosion on impact, the helicopter being immediately engulfed in an intense fire. The cabin entry doors were the only parts of the airframe that escaped the post-impact fire and examination showed them to be free of soot. It was considered that the flames that featured in witness reports were the result of excess fuel being supplied to the No 1 engine as it wound down and that the structure of the helicopter was not on fire before impact.

The post-impact fire was extremely severe, with much of the light alloy airframe components being consumed; the fuselage structure collapsed allowing the still turning rotors to contact the ground. 2,700 lbs of fuel were aboard the aircraft at the time of the accident. The fire had its source in a massive release of fuel when the No 5 tank, and probably one or more of the others, ruptured on impact. The cause of ignition could not be identified but may have been the flames being emitted from the No 1 engine.

### *1.14.2 Emergency services*

ATC notified the on and off airfield emergency services, via the 'omni-crash' line, at 0856:40 hrs. The omni-crash line is an emergency telephone which simultaneously connects ATC with the Aerodrome Fire Service (AFS), the airport telephone exchange, the Grampian Police and the Grampian Fire Brigade. The crews and fire appliances responded immediately and the AFS proceeded to the crash site via the main runway and the slip road parallel to it, arriving approximately one minute later. Four appliances attended and used 115 gallons of 'AFF' foam.

The fire was quickly extinguished and a search revealed the bodies of the two crew members.

## 1.15 Survival aspects

Neither crew member received a serious injury in the impact, the cause of their deaths being asphyxiation in the post-crash fire.

## 1.16 Tests and research

### *1.16.1 Flight tests*

On 26 October 1982 flight tests were carried out by Aerospatiale at Marignane, France. The helicopter used was a SA330G (the military version of the SA330J) fitted with equipment which recorded time, collective pitch angle, main rotor speed, engine gas generator speeds, and indicated airspeed. The helicopter's weight varied from 15,700 lbs to 15,500 lbs during the tests. The No 1 engine overpower stop was in, thus limiting the engine to a theoretical 99.8% Ng as is normal practice during BHL single engine training flights. No 2 engine fuel flow control lever was withdrawn from the forward flight idle (governed) gate and set at an intermediate 'zero-torque' quadrant position. IAS was reduced to approximately 45 kts and collective pitch was applied to top out No 1 engine and to droop the main rotor to 255 rpm. This configuration simulated the parameters of a continued take-off under training conditions. As the aircraft climbed through 1,500 feet, No 1 engine FFCL was rapidly retarded, and between a half and one second later the No 2 engine FFCL was advanced, the handling pilot raised the collective to control rotor speed as engine power increased, and recovery was made to a single engine climb.

Following failure of the No 1 engine, the average rotor droop was to 220 rpm over 1.5 seconds; the average time before reaching the lowest altitude during recovery to a single engine climb was 11 seconds; and the height lost varied between 250 and 325 feet.

*1.16.2 Computer simulation of Puma power off landing performance*

As part of the investigation into this accident the Royal Aircraft Establishment, Farnborough, carried out computer simulations of the behaviour of the Puma helicopter following a complete failure of engine power under the conditions believed to have been experienced by G–BJWS. The simulation runs were commenced with the helicopter at a height of 225 feet at 45 knots, 14° collective pitch and a rotor speed of 255 rpm, but with a zero rate of climb. With a pilot response delay of 1 second before starting to reduce collective pitch rotor speed fell to a minimum of 228 rpm at 2.5 secs after engine failure, and then recovered to 275 after 7 seconds by which time the helicopter was near to zero feet.

Next, constant attitude type landings were simulated. In these the initial speed of 45 knots was maintained down to 70 feet when a flare to a touch-down attitude of +10° was adopted. The maximum rate of descent reached was 2,900 fpm at 5.6 seconds after the power failure. Descent rate was reduced for touch-down by increasing collective pitch from the fully down position of 7.5° at a rate of 12° per second, commencing at heights of 15, 30, 45 and 60 feet. This resulted in rates of descent on touch-down of 2,000, 1,600, 1,150 and 1,000 fpm respectively. The total time from the power failure to touch-down varied between 8 and 9 seconds.

Finally, to explore the possibility of achieving a lower rate of descent at touchdown the increase of collective pitch was initiated at 90 feet. To avoid an excessively low rotor speed being reached the lever was raised in two stages; initially pitch was increased at rates of 2 and 3 degrees per second at 90 feet with a final pull up being initiated at 30 feet. The rates of descent on touchdown were about 1,000 ft per minute. (Aerospatiale stated that this is beyond the design limit of the helicopter and its undercarriage absorbing capacity).

Because the computer programme did not include a model of dynamic stall the results obtained are on the pessimistic side. Had dynamic stall been included in the model, rates of descent of about 800 fpm on touchdown might have been achieved.

### *1.16.3 Engine tests*

Subsequent to the accident, Turbomeca have bench tested an engine, the gas generator turbine of which already displayed 0.75 mm of creep. During the test the engine was run for 2½ minutes at maximum contingency power with anti-icing bleed air selected on. A further test ran the engine for 2½ minutes at the fuel flow limit of 590 litres/hour, during which an Ng of 101.6% was achieved. No failure was experienced.

## 1.17 Other information

### *1.17.1 Training*

G–BJWS was the only SA330J on the United Kingdom register at the time of the accident. There was no flight simulator available. When converting to the aircraft, pilots were trained to meet the CAA requirement that they carry out 'an autorotative descent through at least 1,500 feet with recovery to forward flight over a pre-determined spot'. Recovery was made at a minimum height of 200 feet, and a similar exercise formed a part of each subsequent base check.

### *1.17.2 SA330J Type Certificate*

At the time of the CAA's certification of the SA330J Puma in 1977 the Puma type was already certificated by the French Direction Generale de l'Aviation Civile (DGAC) to United States Federal Aviation Regulations (FAR) part 29, Airworthiness Standards – Transport Category Rotorcraft. This certification had been carried out in 1970. The CAA procedure therefore followed the standard practice of accepting demonstration of compliance with FAR's as satisfying British Civil Airworthiness Requirements (BCAR's) where common requirements exist. In 1977 the FAR level of fuel system crashworthiness was to a higher standard than that of BCAR's, although the latter have been amended since to a similar standard. The SA330J fuel system was thus certified by the CAA as complying with a standard of FAR which was at least equivalent to the applicable BCAR.

During the certification process the CAA looked at the overall design of the fuel system. One of the aspects discussed in detail was the integrity of the sponson fuel tanks, in view of the fact that the undercarriage support brackets are fixed to the tank walls. This point was eventually resolved as satisfactory when taking into account strengths of the attachments at the critical forward landing speed, together with the fact that with the undercarriage extended the weak point was at the swivel actuator.

The present BCAR does not have any significant change from the standards applied when the SA330J was certificated by the CAA.

### *1.17.3 Crashworthy fuel systems*

Since the 1960's much progress has been made in the development of crashworthy design concepts for military helicopters. Throughout this period the United States Army has been a leader in the field. Although aspects such as structural strength, impact energy absorption, interior furnishings, and aircrew clothing have formed an important part of their work, the achievement in the particular area of crashworthy fuel systems is worthy of especial attention in the context of the accident to G–BJWS.

These are summarised in the following extracts from a paper published in May 1982 in lecture series No 123 of the Advisory Group for Aerospace Research and Development (AGARD)[1]:

> "*Crashworthy Fuel System (CWFS)*
>
> In our opinion, the most significant advance to reduce the fire hazard has been the development of the crashworthy fuel system in common use in US Army rotary-wing aircraft. Beginning in 1970, all new helicopters that were manufactured were equipped with a crashworthy fuel system. At the same time, an extensive retrofit program of older aircraft was begun and is now complete. The ideal crashworthy fuel system is one that completely contains its flammability both during and after the crash sequence. To accomplish this, all components of the system must resist rupture regardless of the degree of failure of surrounding structures. Success of such a system depends on proper selection of materials and design techniques in the areas of fuel tanks, fuel lines, and supporting components and sub-systems such as valves. The ideal system would also reduce or eliminate potential ignition sources. The hydraulic system would be similarly protected or non-flammable hydraulic fluid would be used. Fuel tank location, fuel tank shape, and fuel tank materials will be carefully engineered. Tank fittings and tank attachments will be designed to shear or break away from associated structures without causing secondary damage. Valves will be self sealing. In some instances attachment strength to prevent breakaway will have to be increased to as much as 80 percent of the failure level of supporting structures. Fuel lines may have to be flexible and armored and coiled or wrapped to allow extension and distortion without rupture. Fuel line attachment clips will be breakaway and not cause secondary cutting or tearing of fuel lines. The routeing of fuel and hydraulic lines requires design attention in the early stages of airframe layout. Passage holes through bulk heads that can distort and cut a line may need to be larger than normal. Supporting components play a vital role in that they should be capable of preventing spillage in accidents with crash forces equal to or better than tank strength. The fuel tank vents must prevent fuel leakage in the event of aircraft roll over. The same applies to fuel filler necks and quantity sensors found in the fuel tanks themselves."

The effectiveness of the overall US Army programme to reduce thermal-related fatalities is summarised below, the figures being derived from the AGARD paper.

*US Army helicopter crashes*

| *1967–1969* | No of accidents | Post-crash fires | Thermal related fatalities |
|---|---|---|---|
| Survivable – No CWFS | 1,000 | 133 | 159 |
| Non survivable – No CWFS | 68 | 57 | 96 |

| *1970–1976* | No of Accidents | Post-crash fires | Thermal related fatalities |
|---|---|---|---|
| Survivable – No CWFS | 1,160 | 43 | 159 |
| Survivable – With CWFS | 1,258 | 16 | 0 |
| Non survivable – No CWFS | 61 | 42 | 31 |
| Non survivable – With CWFS | 32 | 18 | 0 |

1 "Human Response to Fire" by Stanley C Knapp, MD, and Francis S Knox, PhD, of the US Army Aeromedical Research Laboratory, Fort Rucker, Alabama.

*1976 Onwards*

"Since 1976, in US Army helicopters equipped with crashworthy fuel systems involved in accidents classified as survivable, there have been no thermal related injuries or deaths. There have been post-crash fires, but because of the protective microenvironment provided to the aviators, they have been able to successfully escape without injury."

The experience gained from military programmes has resulted in design and equipment improvements for civil helicopters. Nevertheless helicopter manufacturers (eg Bell Helicopters and Aerospatiale) who do offer crashworthy equipment, such as high integrity fuel systems and impact absorbing seats, as optional alternatives to the standard equipment designed to meet FAR's and BCAR's experience great difficulty in persuading customers to select the crashworthy equipment. The most appropriate example in the context of this accident is the case of the Aerospatiale AS332L Super Puma. The manufacturer offers a crashworthy floor, crew seats, and fuel tanks, all built to a much higher standard than those required by FAR's and BCAR's, as a customer option. The crew seats are stated to limit acceleration to a maximum of 20g for a drop speed of up to 10 m/s, and the tanks are certificated to drop speeds of 14 m/s. However none of the three UK operators who have purchased the Super Puma have fitted any of the optional crashworthy equipment.

In addition fuel tank materials are available which meet significantly higher standards than do the tanks in other currently certificated civil helicopters. These standards are laid down in the USA MIL–T–27422B and the UK DTD 3966/7/8 specifications. Such materials are as suitable for fabricating replacement tanks for helicopters currently in service as for use in new machines.

*1.17.4 Other occurrences*

The investigation has revealed no instances of engine failure occurring in circumstances directly comparable to those surrounding G–BJWS. However, there have been two engine failures that have been attributed to wrongly adjusted fuel control units. In both cases, the maximum contingency rating of 100.9% Ng was exceeded, leading to failure of the gas generator turbine. Subsequent strip examination revealed that the engine components were in a similar condition to those of engine S/N 509. However, the failure that occurred in the latter could not have been due to overspeeding, as the fuel control unit was found to be correctly set and performed satisfactorily under test.

Another incident concerned an engine which had fallen off a stand during 1983 and was returned to the manufacturer for a shock loading inspection. During strip examination of the engine, which had achieved only 463 hours, evidence of overtemperature was found on the gas generator turbine blades. The leading edge corners of the first stage blades had burned away. Turbomeca could not ascribe a cause to this condition but stated that there was a possibility that it had resulted from a hot start.

In addition to the above, another significant incident emerged during the course of informal discussions with Turbomeca at a very late stage in the investigation. This concerned a Puma (not operated by BHL) both engines of which were rejected due to rubbing noises being heard on shut-down. Excessive creep was discovered on the gas generator turbines during strip examination, even though both engines had only achieved some 450 hours. Turbomeca have stated that they were aware that Flight Manual limitations had not been adhered to by the operator.

### 1.17.5 *Factors affecting turbine blade life*

Factors affecting turbine blade life include the effects of erosion through, for example, operation in a dusty environment; corrosion through an oxidation process which may be assisted by impurities such as sulphur in the fuel; and from the effects of creep. The latter may be described as a tendency for the blades to undergo plastic elongation due to the combined effects of high temperatures and centrifugal stress, and can be the result of very high temperatures over a short period of time or lesser temperatures over a longer time. Creep is cumulative and if allowed to continue unchecked, the clearance between the blade tips and the casing would reduce to the point where contact was made. Normal operation should ensure that adequate creep life remains at the end of the overhaul life.

In metallurgical terms, an idealised picture of the creep process is one of three separate stages. When an alloy is stressed, there is a relatively high initial strain rate due to plastic adjustments at points of stress along the grain boundaries and at flaws within the alloy structure. These initial adjustments are followed by a second, steady state, stage of creep – the strain rate being higher at elevated temperatures. Before finally rupturing, a higher rate, tertiary stage of creep occurs due to an effective reduction of area caused by intergranular cracking. Oxidation occurs within these cracks, which also create a 'notch effect', thereby further weakening the material. Normal engine operation should ensure that turbine blades never reach the tertiary creep stage.

Turbomeca have stated that the creep life of the Turmo IVC turbine blades has been assessed as 9 hours ± 20% for engine operation at a turbine entry temperature of 1030°C. They further stated that any more severe operating conditions, such as the use of maximum contingency power, or the use of take-off power with airbleed – ie with engine anti-icing or cabin heating on – would result in accelerated creep.

Turbomeca have also indicated that the intergranular cracking mentioned earlier, which they term 'microcracks', are frequently associated with high time engines and on engines that have been subjected to an over-temperature condition, but that no failure has previously been attributed solely to microcracks. Turbine tip rub on engine shut-down, resulting from the creep that can be associated with the microcracks, has always provided adequate advance warning of the condition of the turbine blades.

Turbomeca stated that BHL engines returned for overhaul tended to exhibit slightly more creep than those of other operators. Turbomeca considered that apart from take-offs made with engine anti-icing on, this might also be due to the amount of single engine training done by BHL. BHL engines had an average of approximately 0.24 mm creep (expressed as an increase on the diameter of the turbine wheel) at overhaul. This is well short of the figure at which failure might be expected to occur, which is approximately 0.9 mm. Turbomeca said that turbine rub could become apparent during engine shut-down where creep exceeds approximately 0.75 mm. One engine of the BHL sample displayed this amount of creep, and had been rejected at 1900 hours due to turbine rub. No reason had been established as to why this particular engine displayed so much creep in relation to the fleet mean.

### *1.17.6 Safety measures adopted during the investigation*

Following the accident BHL decided to carry a third pilot during single engine training on all twin-engined helicopters; this is a practice that Aerospatiale recommends to all operators. The third pilot's function is to advance the retarded fuel flow control lever in the event of an emergency. BHL also required that there be a sufficient clear area available on single engine training exercises to allow a take-off to be rejected from a height of 300 feet. Finally when a replacement Puma was brought onto the UK register BHL prohibited single engine training when the temperature was below 10°C and there was visible moisture.

In February 1984 a meeting was held with representatives from Turbomeca, the DGAC, BHL, the CAA, and AIB. Turbomeca was informed of the reservations of the CAA, BHL and the AIB on the use of anti-icing and the effect of single engine training on the creep life of the gas generator turbines.

Subsequent to this meeting, the CAA issued an Emergency Airworthiness Directive, No 008–02–84 dated 16 February 1984 (see Appendix 7). This clarified the use of bleed air and maximum contingency power, and also introduced a turbine creep check. Similar action was taken by DGAC. These measures were formalised by Turbomeca Service Letter No 979/84/214 (which dealt with the creep check) and by a revision (No 3B Code 03–84) to the Flight Manual. The latter laid down Ng limits with engine anti-icing on and off for two-engine and one-engine operation for each engine rating.

A significant change was the introduction of reduced Ng limits for two-engined operation when engine anti-icing was in use. These were 95.4% Ng for the maximum continuous rating and 97.9% Ng for the take-off rating. The revision also pointed out that the selection of anti-icing bleed air caused a T4 increase of approximately 30°C over the no-bleed value.

Following the issue of the EAD, BHL withdrew the restriction on single engine training mentioned above.

## 1.18 New investigation techniques

None.

# 2. Analysis

## 2.1 Flight data recorders

The investigation was considerably handicapped by the absence of direct evidence as to the crew's intentions, of their actions before and after the emergency commenced, of the performance of the engines, and of the helicopter's flight path and rotor speed during the descent and crash landing. Such evidence could only have been obtained from cockpit voice and flight data recorders. While cockpit voice recorders have been mandatory equipment for medium and large helicopters in the Transport Category from June 1984, the introduction of flight data recorders remains an urgent requirement for the satisfactory investigation of helicopter accidents.

## 2.2 The cause of the accident

The available evidence indicates that the crew were carrying out a Group A clear area take-off procedure during which the commander simulated a No 2 engine failure initiated immediately after the critical decision time. It is apparent that shortly afterwards, with the No 2 engine retarded, the No 1 engine emitted flames and lost power, forcing the commander to perform an emergency power-off landing.

The commander was faced with a formidable emergency which arose suddenly at a critical stage of flight. He had experienced a sudden and complete loss of power at about 225 feet when the helicopter was at a high weight, an airspeed of about 45 knots, and with rotor speed possibly already drooped to 255 rpm as a result of the practice continued take-off manoeuvre. The helicopter was thus in a condition and position from which a recovery could not be made by increasing power on the No 2 engine, from which normal autorotation could not be fully established, from which only a constant attitude type of touchdown could be attempted, and from which a severe impact was inevitable about eight seconds after the engine failure. Moreover the commander was flying a helicopter in which he had no experience of power-off landings and whose Flight Manual (in common with those of many others) described only the flare type of autorotative landing – from an airspeed of 80 knots. As the take-off had been made only some 630 metres short of the runway end, the landing had to be made in a grass area containing substantial obstructions comprising the airfield boundary fence, the ILS localiser aerial and hut, and approach lights. In the circumstances the commander did well to avoid hitting these and to limit the severity of the inevitable impact to a level at which neither he nor the co-pilot sustained any significant injury. This achievement owed much to his skill as a pilot and no doubt also to his experience as a helicopter instructor, especially that gained in the Royal Navy where he would have carried out many power-off landings of both the flare and the constant attitude types. It is especially tragic that following this skilled performance an intense post-crash fire cost the crew their lives.

The cause of the accident was, therefore, the failure of the No 1 engine when the crew were carrying out a practice single-engine continued take-off with the No 2 engine retarded, and when the helicopter was at a height and airspeed too low to permit either a recovery using the No 2 engine or an autorotative landing.

## 2.3 Operational aspects

Following the accident BHL required that when single engine training was to be done a third pilot be carried who must keep one hand close to the shutdown engine's control lever to enable it to be advanced rapidly if required, a practice which is recommended by the manufacturer. Although the flight tests carried out by the manufacturer following the accident show that WS was too low to permit the No 2 engine to be accelerated and a safe transition made to a climb, this would be an option when such failure occurs at a greater height. Even if the height is insufficient to avoid ground impact, advancing the control lever of the throttled engine would be beneficial as torque would be transmitted to the rotor system so making it easier for the pilot to limit the severity of the ground impact. An alternative to carrying a third pilot is for the training captain to keep his hand on the retarded engine control lever until a safe height is reached, but this is not always possible. A second change made by BHL was to require a sufficient area to be available to permit the take-off to be safely rejected from 300 feet in the event of a malfunction of the working engine. Such methods of reducing the risk involved in continued take-off exercises are clearly sensible, as is the practice of carrying out as much such training as is possible in a flight simulator.

Four other points concerning the operation of helicopters in these circumstances are worthy of consideration. The first concerns helicopters, such as the Puma operating from Aberdeen, which have a single engine climb performance which can be in practice significantly greater than the scheduled gross data. This is especially true when the helicopter is operated at training weights of 95% maximum total weight authorised (MTWA) and when the MTWA is not weight, altitude or temperature (WAT) limited. The question arises as to whether in such circumstances the single engine procedure laid down in the Flight Manual is as safe as can be devised in that, it can result in rotor speed being reduced to 255 rpm whilst the helicopter is climbing in the critical low speed/low height region following an engine failure on a normal two engine take-off, whether or not the increment in climb performance so obtained is required for obstacle clearance. If it is unnecessary to select a collective pitch angle such that rotor speed droops to about 255 rpm to obtain a satisfactory climb performance then the helicopter is needlessly exposed to extra hazard should the second engine fail, perhaps as a consequence of the failure of the first.

Secondly, it is apparent that whilst BHL Change Sheet No 1 to the Flight Manual stated that a reduced maximum power setting of 99.5% Ng should be used in training to reduce possible accelerated engine deterioration, the Flight Manual recommended technique for a continued take-off could result in the live engine attaining a speed of 99.8%.

Thirdly, the execution of a survivable power-off landing from low speed and low altitude is an extremely difficult task for a pilot, especially if other conditions are also adverse. The pilot must therefore be aware of the optimum technique applicable to his helicopter, especially such information as the height and rate at which the collective pitch should be increased to reduce the rate of descent on touchdown to a minimum. The importance of this is well illustrated by the results of the RAE computer simulations. While all helicopter flight manuals include a recommended procedure for power-off landings of the flare type this technique cannot be used from low speed and low altitude, where the constant attitude technique alone is possible. Such a technique should be included in the flight manuals of all helicopters.

Finally, if pilots are to be expected to carry out survivable touchdowns following any event which forces them to carry out a power-off landing they should be given training in the approved techniques. On light helicopters this is done by carrying out actual power-off landings. Whilst this is not practicable in larger machines, where a suitable flight simulator is available such training should be given. Tests in a Sikorsky

S–61N simulator have indicated that realistic practice can be obtained even under flight conditions as adverse as experienced in this accident.

## 2.4 Technical aspects

### *2.4.1 The engine failure*

The exhaustive examination of both engines, the power train components, and the fuel control units, did not reveal any engineering evidence regarding the failure of the gas generator first stage turbine in engine S/N 509. However, it was possible to exclude the possibility of air starvation by blockage of the intake, of foreign object ingestion, or of mechanical failure outside the gas generator as causal factors.

The gas generator components of engine S/N 509 were similar in appearance to those of two engines which had each failed as a result of a wrongly adjusted fuel control unit causing the Ng to rise to 100.9%. However, both fuel control units from G–BJWS were found to be satisfactory in operation and adjustment. A degree of 'stiction' was found in the acceleration control unit from engine S/N 509, although this has been attributed to either the effects of fire or the introduction of contaminant particles during preparation for test. In any event, had this been a pre-existing defect, engine acceleration would have been noticeably erratic. According to the sequence of events that has been established for the accident, the engine had sustained the aircraft from shortly after the critical decision point to a height in excess of 200 feet. The acceleration control unit has no effect on the engine during a steady state condition. It was concluded that there was no evidence that the engine failure was due to a malfunction of the fuel control system.

First stage gas generator turbine blade tip rub due to extreme creep is a known consequence of exceeding Ng/T4 limits, and in certain cases this had led to engine failure. However, the (original) Turbomeca theory that the failure of engine S/N 509 was due to this phenomenon had to be discarded when examination of sections cut from the turbine excluded the possibility of sustained tip rub.

Turbomeca's definitive view at the conclusion of the investigation was that the engine failure was the end result of a slow damage process to the first stage gas generator turbine blades involving creep combined with erosion and inter-granular oxidation, the culmination being a fracture of one or more blades when the engine was under severe stress during the continued take-off with anti-icing in use. In support of this last point Turbomeca stated that the temperature to which the first stage turbine blades had been subjected (confirmed by laboratory examination to be 1000 to 1080°C) could have been achieved by operating the engine in this way. Had this been the case, then the T4 indication would have been in the red sector, ie above 790°C. Whilst this is a plausible theory, it would be unreasonable to draw a conclusion on the basis of the temperature assessment, since the conditions immediately after blade failure are likely to have been complex, as evidenced by the high temperatures (1250°C) found to have been experienced by the second stage turbine. This was probably due to the increasingly rich fuel-air mixture (as the Ng reduced) causing aft migration of the flame front. There is therefore a possibility that this phenomenon also caused the temperature at the first stage turbine to rise above that at which failure occurred.

Turbomeca's definitive theory as to the cause of the failure in engine S/N 509 is unique in that they have not discovered any other failure resulting from an inter-granular rupture in some 3.5 million hours of Turmo operation. Engine S/N 509, at 2368 operating hours, had suffered its failure at a considerably higher time than the engines involved in other known incidents, and no tip rub had been detected during its service life. In the absence of engine monitoring equipment, it has not been possible to determine whether this engine had experienced an unnoticed over-temperature condition at any time in its life, and even if this could have been established it would then have been difficult to explain why such an event should lead eventually to an intergranular rupture rather than excessive creep. This in turn leads to a suggestion that it may have suffered an unusually severe period of erosion or corrosion, the operational circumstances of which cannot be determined. If such was the case the effects could subsequently have been exacerbated by high power, high temperature, operation such as was achievable during a take-off or a rejected take-off exercise with engine anti-icing in use before the issue of Revision 3B to the Flight Manual in March 1984.

At the end of the investigation, therefore, although the theory that the failure of the No 1 engine (S/N 509) was due to intergranular rupture of one or more first stage gas generator turbine blades appeared credible, the evidence was insufficient to permit the cause of the failure to be attributed to this.

However, if this was the failure cause, then the risk of a similar occurrence in the future should be reduced by the measures taken in the Emergency Airworthiness Directive, and subsequently by the Flight Manual Revision.

### 2.4.2 *Engine condition monitoring*

It is apparent that due to differences in interpretation of the sections of the Flight Manual dealing with the use of engine anti-icing some BHL Puma engines had on occasion been exposed to more severe operating conditions than those envisaged by Turbomeca when the Turmo IVC's overhaul life was established, although there is no evidence that any individual engine limitation had been exceeded.

This is a case of the type in which engine condition monitoring has a part to play in enabling assumed use to be compared with reality, so enabling early corrective action to be taken. Moreover the investigation itself would have been considerably aided if this type of data covering engine S/N 509's life since its last overhaul had been available. It is therefore recommended that the practicability of fitting engine monitoring equipment to the larger public transport helicopters should be examined.

### 2.4.3 *Survivability*

The evidence indicates that despite the high rate of descent at touchdown the crew would have been able to make their escape without suffering other than minor injury if the helicopter had not been instantly engulfed in a severe fire. This raises important questions of crash survivability, even though at first sight the circumstances of the accident might appear to have little relevance to passenger operation because they involved the unique situation of a genuine engine failure superimposed on a simulated one, and because the probability of a genuine twin-engine failure would normally be considered to be extremely remote. However, it is apparent that the condition of engine S/N 509 at the start of the last take-off was such that if G–BJWS had been returned to passenger carrying duties immediately prior to the accident flight it would have been exposed to a similar accident risk in the event of a loss of power or drive from the No 2 engine. Finally, failures other than that of total power loss can, and have, caused twin-engined helicopters to crash onto hard surfaces (ie the ground or offshore installations) at potentially survivable impact velocities. The survivability lessons which arise from this accident are therefore considered important.

Although the cause of ignition of the large quantity of kerosene released by impact damage to the fuel system could not be identified, the flame being emitted from the No 1 engine provided a ready source. The speed with which the fire developed, together with its severity, gave the crew no chance of escape. The outcome is typical of many helicopter post-impact fires, and arises partly because of the close proximity of fuel, ignition sources, and occupants. The only method of preventing such fires currently in use is the crashworthy fuel system, in which fuel is contained during an impact instead of being released.

Notwithstanding it cannot be deduced that a severe fire would have been avoided if the helicopter had been fitted with a typical current type of crashworthy fuel system, it can be said that the probability of such a fire would have been considerably reduced. The US Army's experience readily proves this. Even where post-impact fire does occur, it appears that crashworthy fuel systems result in a reduced rate of fuel leakage thereby slowing down the rate of flame propagation and providing extra time for the occupants to escape.

It is noteworthy that whilst significant improvements in the survivability of helicopter accidents taking place onto water have been achieved by industry to meet the requirements of operators and the CAA, in the case of accidents onto land, industry is capable of offering equipment of a standard that is considerably in excess of current airworthiness requirements. Heavy duty undercarriages, impact absorbing structure, and crashworthy seats, can ensure that severe impacts can be survived providing the fire risk is minimised. Although in this accident the death of the crew was due to fire and so attention tends to focus on the fire aspects of crashworthiness, these necessarily go hand in hand with consideration of impact forces and their attenuation.

Whilst it is a long term matter to up-grade the relevant requirements and thus affect the design of new helicopters, the gap between what is now achievable and what BCAR's and FAR's lay down is so large, and the potential gain so significant, that urgent action is indicated. Moreover, the demonstrated success of the US Army fleet modification programme establishes that valuable crashworthy modifications to existing helicoptor types can be incorporated and should be considered. Crashworthy fuel systems have cost and other penalties attached to them which must significantly account for the market resistance experienced by helicopter manufacturers when such systems are offered as options on civil aircraft. The retrofitting of existing aircraft types, such as the Puma or the Sikorsky S–61, with crashworthy fuel systems is sometimes said to be prohibitively expensive. Nevertheless, a significant improvement could probably be economically achieved with a straightforward replacement of the tanks with types constructed from material such as that meeting the USA MIL–T–27422B or the UK DTD 3966/7/8 specifications. However, it is unlikely that operators will risk placing themselves at a commercial disadvantage to their competitors by investing individually in crashworthy equipment. That being the case the numbers of UK helicopters so equipped are unlikely to be significant until crashworthy fuel systems are made mandatory. An example of this is the recent introduction of the Super Puma by three UK operators without either the crashworthy fuel system or the crashworthy crew seats being fitted.

Thus it is considered important that the CAA initiate an urgent review of certification requirements with a view to a significant raising of those dealing with crashworthiness, including impact as well as fire risks. It is also considered important that the CAA review public transport helicopter types on the UK register to determine what improvements can be made to make them less susceptible to post-crash fire.

# 3. Conclusions

*(a)* *Findings*

(i) The helicopter had been maintained in accordance with the approved maintenance schedule, its Certificate of Airworthiness was valid, and it was correctly loaded.

(ii) Both pilots held valid licences which were appropriate to the training flight being carried out.

(iii) Weather was not a factor in the accident.

(iv) Following a number of practice rejected take-offs the crew carried out a practice continued take-off with engine anti-icing selected on, the No 2 engine's fuel control lever being retarded at or shortly after the critical decision time.

(v) Shortly afterwards the No 1 engine suffered a failure of its gas generator turbine causing total power loss from that engine. The cause of this failure could not be established.

(vi) When the No 1 engine failed the helicopter was at a height and airspeed too low to permit recovery using the No 2 engine, from which autorotation could not be fully established, from which only a constant attitude type of power-off landing was possible, and from which a severe impact with the ground was inevitable.

(vii) Although he had no experience of power-off landings in the Puma and although the Flight Manual did not include a procedure for a constant attitude technique, the commander was successful in avoiding obstacles in his path and in limiting the severity of the impact to a level at which neither pilot sustained any significant injury. This was an outstanding achievement.

(viii) The impact was survivable but fire caused the death of both pilots.

(ix) The instant and intense post-impact fire occurred when fuel released by rupture of the fuel system was ignited, possibly by the existing fire in the jet pipe area of the No 1 engine.

(x) The fuel system of the SA330J Puma cannot be considered as crashworthy in the light of the technical progress made since its certification although it then complied with the applicable standard of FAR's and BCAR's, which remain substantially the same today.

(xi) ATC and the emergency services acted with promptness and in an efficient manner.

*(b)* *Cause*

The accident was caused by the failure of the No 1 engine when the crew were carrying out a practice single-engine continued take-off with the No 2 engine retarded, and when the helicopter was at a height and airspeed too low to permit either a recovery using the No 2 engine or an autorotative landing.

# 4. Safety Recommendations

It is recommended that:

4.1 The CAA review the crashworthiness of the fuel systems of public transport helicopters already on the UK register, and initiate action with other airworthiness authorities to formulate improved design criteria for helicopter fuel systems.

4.2 Consideration be given to requiring that the larger public transport helicopters be fitted with engine monitoring equipment.

4.3 A recommended technique for carrying out a power-off landing from low height and airspeed be included in the Flight Manual of every helicopter, and pilots given experience of it in flight simulators where this is practicable.

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