

**AIRCRAFT ACCIDENT REPORT 2/2004**

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**Department for Transport**

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**Report on the accident to  
Sikorsky S-61N, G-BBHM  
at Poole, Dorset  
on 15 July 2002**

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**Department for Transport  
Air Accidents Investigation Branch  
Berkshire Copse Road  
Aldershot  
Hampshire GU11 2HH**

March 2004

*The Right Honourable Alistair Darling  
Secretary of State for Transport*

Dear Secretary of State

I have the honour to submit the report by Mr P T Claiden, an Inspector of Air Accidents, on the circumstances of the accident to Sikorsky S-61N, G-BBHM, which occurred at Poole, Dorset on 15 July 2002.

Yours sincerely

**Ken Smart**  
Chief Inspector of Air Accidents

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## GLOSSARY OF ABBREVIATIONS USED IN THIS REPORT

AAD	Additional Airworthiness Directive	km	Kilometre(s)
		kt	Knot(s)
AAIB	Air Accidents Investigation Branch	MDP	Magnetic Drain Plug
AD	Airworthiness Directive	MDS	Main Drive Shaft
agl	Above ground level	MGB	Main Gear Box
AGB	Accessory Gear Box	MHz	Megahertz
amsl	Above mean sea level	MM	Maintenance Manual
ASB	Alert Service Bulletin	NTSB	National Transportation Safety Board (United States)
ATC	Air Traffic Control	N <sub>f</sub>	Engine free turbine speed
BCAR	British Civil Airworthiness Requirements	N <sub>g</sub>	Engine gas generator speed
CAA	Civil Aviation Authority	N <sub>r</sub>	Main rotor speed
CAP	Civil Aviation Publication	psi	Pounds per square inch
C of G	Centre of Gravity	psig	Gauge pressure (pounds per square inch)
CRS	Certificate of Release to Service	RPM	Revolutions per minute
CVFDR	Combined Voice and Flight Data Recorder	SAR	Search and Rescue
CVR	Cockpit Voice Recorder	TBO	Time Between Overhauls
DAPU	Data Acquisition and Processing Unit	TRE	Type Rating Examiner
ECD	Electrical Chip Detector	TSN	Time Since New
EGT (or T5)	Exhaust Gas Temperature	TSO	Time Since Overhaul
EMRSA	Engine Mounting Rear Support Assembly	TT	Total Time
FAA	Federal Aviation Administration	UK CAA	United Kingdom Civil Aviation Authority
FACTAR	Follow-up ACTION on Accident Report	US	United States
FDR	Flight Data Recorder	UTC	Co-ordinated Universal Time
FE	Finite Element	VHF	Very high frequency
FFT	Fast Fourier Transform	ZFW	Zero Fuel Weight
FM	Frequency modulation	°C, °F	Degrees Celsius, Fahrenheit
fpm	Feet per minute	°M, °T	Degrees Magnetic, Time
FRC	Flight Reference Card		
FSI	Flying Staff Instruction		
g	Normal acceleration		
GE	General Electric		
HARP	Helicopter Airworthiness Review Panel		
HUMS	Health and Usage Monitoring System		
Hz	Cycles per second (Hertz)		
HRC	A hardness value based on the Rockwell 'C' scale		
IFSD	In Flight Shut Down		
IRE	Instrument Rating Examiner		
IHUMS	Integrated Health and Usage Monitoring System		
kHz	Kilohertz		



**Sikorsky S-61N G-BBHM, Poole 16 July 2002**

## **Air Accidents Investigation Branch**

**Aircraft Accident Report No: 2/2004**

**(EW/C2002/7/3)**

Registered Owner and Operator	Bristow Helicopters Limited
Aircraft Type	Sikorsky S-61N
Nationality	British
Registration	G-BBHM
Place of Accident	Poole, Dorset UK
Date and Time	15 July 2002 at 1515 hrs (all times in this report are in UTC)

### **Synopsis**

The accident was notified to the Air Accidents Investigation Branch (AAIB) by Bristow Helicopters Limited at 1700 hrs on 15 July 2002 and the investigation began that evening. The following inspectors participated in the investigation:

Mr P T Claiden	Investigator in Charge
Mr R W Shimmons	Operations
Mr A P Simmons	Engineering
Mr J R L James	Flight Recorders

G-BBHM, which was based at Portland, was being operated in the Search and Rescue role. Following the first alert of the day, G-BBHM had been airborne for about 40 minutes over Poole Harbour when the two rear crew members became aware of an unusual noise. Almost immediately, the pilots saw the 'NO 2 ENG FIRE WARN' light illuminate accompanied by the audio alert. The pilots commenced their emergency procedures, including shutting down the No 2 engine and activating the fire extinguisher, and initially set heading for Bournemouth Airport. However, with the 'FIRE' light still illuminated and indications of hydraulic failures from both tactile and warning systems, the co-pilot alerted the commander to a suitable nearby landing area. The commander called for an immediate landing and made a successful approach and touchdown; during the approach, the pilots became aware that 'NO 1 ENG FIRE WARN' was also illuminated. After touchdown, the pilots shut down

No 1 engine and the crew quickly vacated the helicopter. G-BBHM was destroyed by fire shortly after they were clear. The time between the onset of the original fire warning and touchdown was 82 seconds.

The investigation identified the following causal factors:

- 1 The No 2 engine had suffered rapid deterioration of the No 5 (location) bearing of the free turbine, causing failure of the adjacent carbon oil seal and mechanical interference between the Main Drive Shaft Thomas coupling and the Engine Mounting Rear Support Assembly tube, which completely severed the support tube.
- 2 A severe fire, outside of the engine fire zone, was caused because the released engine oil was ignited either by this mechanical interference, or by contact with the hot engine exhaust duct.
- 3 The No 2 engine's No 5 bearing failed because of unusual and excessive cyclic loading conditions arising from shaft vibration. The bearing deterioration was exacerbated by a reduction in its oil supply during the same period, when the live oil jet fractured as a consequence of the vibration.
- 4 It is probable that the Main Drive Shaft vibration was caused by damage or distortion sustained during one or more previous No 2 engine starts involving a high torque rotor engagement.
- 5 There was no specific torque limitation published in the manufacturer's Flight Manual, used by Bristow Helicopters Limited, during rotor engagement after engine start.

Thirteen safety recommendations were made during the course of this investigation.

# **1 Factual information**

## **1.1 History of the flight**

### **1.1.1 Flight background**

The helicopter, which was based at Portland, was being operated in the Search and Rescue (SAR) role by a commercial company on behalf of HM Coastguard. The allocated crew comprised two pilots and two rear crew members. Although both pilots were qualified as captains, one was assigned as the commander and the other was assigned as the co-pilot. The two rear crew members were also both qualified as winch operators but one was assigned as the winchman, the other as winch operator.

Following their arrival on duty, the crew completed their pre-flight checks, which included the commander preparing G-BBHM for a scramble start. He also checked the aircraft documentation and found no reported unserviceabilities apart from a minor deferred defect relating to one of the windows. The crew reported on standby at 0800 hrs.

### **1.1.2 Accident flight**

At 1425 hrs, the crew were alerted to look for a possible person in the water in Poole Harbour. After an uneventful engines start, G-BBHM was airborne within five minutes.

The commander was the handling pilot in the right seat and he established the helicopter in the search pattern over Poole Harbour at approximately 200 feet amsl and at an airspeed varying between 20 kt and 80 kt. The co-pilot was in the left cockpit seat and controlling the radios. He was operating with Bournemouth Radar on frequency 119.47 MHz, but also monitoring the emergency frequency of 121.5 MHz and the coastguard channel. The winch operator was seated in the front left seat of the cabin and was also monitoring the coastguard channel as well as the crew intercom. The winchman was wearing a 'dispatcher harness' and was seated in the aperture of the open cargo door; he was also monitoring the crew intercom and the coastguard channel.

After about 40 minutes, the crew were requested to investigate reports of a vessel emitting a lot of smoke to the north of their position. From the helicopter location to the west of Brownsea Island, the commander headed 350° (M) at 80 kt and at 200 feet amsl.

Shortly afterwards, the two rear crew members noticed an unusual noise (like "escaping gas" and an "expiration of breath") and commented on this on the

crew intercom. The pilots heard these comments but, almost immediately, the commander saw 'NO 2 ENG FIRE WARN' and 'No 2 T HANDLE' lights illuminated; he called "Fire on engine No 2". The co-pilot also heard the audio alert and saw the fire lights. He looked at the engine instruments but could see no unusual indications and, on being instructed by the commander, put his hand on the 'No 2 SPEED SELECT' lever. The commander, after confirming that this was the correct lever, asked the co-pilot to retard the lever to 'flight idle'. By now, the commander had looked in the rear view mirror and saw greyish/white smoke coming out under pressure from the area of the No 2 engine exhaust. The rear crew also called that there was smoke in the cabin. The commander called "Confirm Fire, Stopcock No 2". By now the co-pilot had retarded the 'No 2 SPEED SELECT' lever to idle and, on hearing the confirmation fire call, retarded the 'No 2 SPEED SELECT' lever further to the 'Cut-off' position. After re-affirming with the commander that he wanted the fire extinguisher to be fired, he then pulled the 'No 2 T HANDLE' and activated the fire extinguisher. One of the rear crew members advised the coastguard of their problem and, concurrently, the commander had initiated a climbing turn to the right with the intention of heading for Bournemouth Airport.

Within the cabin, the winchman had stood up, closed the cargo door and released his 'dispatcher harness'. He strapped into a seat on the right of the cabin just in front of the airstair door and transmitted to the coastguard radio operator that they had an engine fire. As the situation developed, he updated the coastguard, initially to say that they would be landing at Bournemouth and finally that they would be landing immediately.

The co-pilot noted that 'NO 2 ENG FIRE WARN' was still on and called to the commander that there was a suitable field for landing out to the left. About then, the 'TRANS OIL PRESS' light illuminated on the caution panel accompanied by the 'MASTER CAUTION' light. The commander visually acquired the possible landing field (part of the Royal Marines Barracks complex) and made the decision to land. Shortly after, the 'PRI SERVO PRESS' light illuminated followed by the illumination of the 'AUX SERVO PRESS' light. The commander was also aware of an uncommanded lateral movement of about three inches on the cyclic control. He called "Immediate Landing" and established his approach to the selected sports field to the west of Poole. On final approach, the co-pilot extended the landing gear and heard the commander call "Fire in No 1". Within the cabin, the smoke was becoming more dense as the commander made a successful run-on landing at about 10 kt. After landing, the commander tried to apply the foot brakes but there was little apparent pressure and no resulting retardation. Nevertheless, the helicopter came to rest after a ground roll of about 30 metres. On the ground roll, the co-pilot selected the 'No 1 SPEED SELECT' lever to cut-off and applied the rotor

brake. He did not think that the rotor brake was having any effect and, as the helicopter came to rest, he switched off both batteries.

Within the cabin, as the helicopter came to rest, the crew heard the instruction to evacuate. The winch operator opened the cargo door and exited through it, and the winchman lowered the airstairs and vacated the helicopter. The co-pilot evacuated through the cargo door and the commander exited through his cockpit emergency door. Once clear of the helicopter, the crew moved away upwind. As they did so, there were a series of explosions and the helicopter appeared well alight. Some Royal Marine personnel arrived with hand held extinguishers but the crew kept them well clear of the burning helicopter. Shortly after, the Fire Service arrived and were advised by the crew that all of the occupants were clear of G-BBHM.

## **1.2 Injuries to persons**

Injuries	Crew	Passengers	Others
Fatal	Nil	N/A	Nil
Serious	Nil	N/A	Nil
Minor/none	Nil	N/A	Nil

The crew evacuated the aircraft without difficulty, and there were no injuries either to the occupants of the aircraft or to persons on the ground.

## **1.3 Damage to aircraft**

During flight, the helicopter had sustained damage to the rear mounting and free turbine shaft assembly of the No 2 engine and the associated transmission Main Drive Shaft (MDS). Fire damage then occurred in flight to systems in the Main Gear Box (MGB) bay, including those hydraulic systems which power the flying controls. Most of the helicopter was subsequently consumed in the ground fire.

## **1.4 Other damage**

The helicopter landed on a military sports field. The ground fire, together with the resulting fuel and oil contamination, caused considerable localised damage to the specially prepared surface of the field.

## 1.5 Personnel information

1.5.1	Commander:	Male, aged 41 years
	Licence:	UK Airline Transport Pilot's Licence (Helicopters)
	Medical certificate:	Class 1, issued on 30 May 2002
	Flying experience:	Total all types: 4,600 hours Total on type: 2,750 hours Total last 28 days: 24 hours Total last 24 hours: 1 hour
	Previous rest period:	Off duty: 9 July 2002 On duty: 0900 hrs on 15 July 2002

The commander was one of five pilots (three of whom were captain qualified) in the unit. He was also the designated Type Rating Examiner and Instrument Rating Examiner (TRE/IRE).

1.5.2	Co-pilot:	Male, aged 43 years
	Licence:	UK Airline Transport Pilot's Licence (Helicopters)
	Medical certificate:	Class 1, issued on 14 February 2002
	Flying experience:	Total all types: 5,592 hours Total on type: 4,342 hours Total last 28 days: 24 hours Total last 24 hours: 4 hours
	Previous rest period:	Off duty: 2100 hrs on 14 July 2002 On duty: 0900 hrs on 15 July 2002

The co-pilot was the chief pilot of the organisation and was one of three designated captains.

## 1.6 Aircraft information

### 1.6.1 Leading particulars

Manufacturer:	Sikorsky Aircraft
Type:	S-61N
Constructor's Number:	61713
Year of manufacture:	1973
Powerplants:	2 General Electric CT58-140-2 turboshaft engines
Total airframe hours:	29,853:55 at 14 July 2002
Total airframe cycles:	26,465 landings at 14 July 2002
Certificate of Airworthiness No:	003136/006
Category:	Transport Category (Passenger)
Validity:	2 November 2001 to 1 November 2004
Issuing Authority:	UK CAA
Certificate of Registration No:	G-BBHM/R2
Registered Owner:	Bristow Helicopters Limited
Issued:	10 May 1993
Issuing Authority:	UK CAA

The helicopter Zero Fuel Weight (ZFW) was 15,812 lb. At the time of the accident, there was a total of 2,000 lb of fuel on board. The helicopter weight was therefore 17,812 lb, which was less than the structural limitation of 20,500 lb. The helicopter Centre of Gravity (C of G) was also within the normal limits.

### 1.6.2 Engines

Number 1 engine	General Electric CT58-140-2
Serial number:	295271
Year of manufacture:	1979
Time Since New (TSN):	14,558:20 hours at 14 July 2002
Time Since Overhaul (TSO):	2,108:10 hours, 8,052 cycles at 14 July 2002

Number 2 engine	General Electric CT58-140-2
Serial number:	295289
Year of manufacture:	1980
TSN:	13,417:14 hours at 14 July 2002
TSO*:	1,562:58 hours, 8,002 cycles at 14 July 2002

\* Based on last statement of life used on life limited parts (light overhaul).

### 1.6.3 General description

The Sikorsky S-61N is a large twin-engined helicopter of conventional configuration and construction. Each of its two General Electric CT58 turboshaft engines consists of a gas generator section and a single stage free power turbine connected to the respective Main Gear Box (MGB) input pinion. Each MGB input has a freewheel unit to cater for engine failure cases. The power turbine of each engine is connected to the respective MGB input by a shaft, known variously as the High Speed Shaft, the Input Drive Shaft or the Main Drive Shaft (MDS). The latter term, given in the Maintenance Manual (MM), is used throughout this report. This shaft turns at approximately 19,000 RPM and is required to be accurately balanced, together with its couplings. The MDS runs inside the Engine Mounting Rear Support Assembly (EMRSA), which is a large diameter tube which supports the rear of the engine, and which is mounted through vibration absorbing mounts to the MGB. A diagram of this overall arrangement is shown at Appendix A, Page 2. There is approximately ¼ inch clearance between the outer diameter of the forward coupling (the Thomas coupling) and the inner diameter of the EMRSA tube. The EMRSA also forms part of the firewall between the engine bay and the MGB bay. Measurement of engine torque is achieved through a sensor within the MGB, at each engine input. The power turbine assembly is supported by the No 4 and 5 bearings (Appendix A, pages 3 and 4), which together with the rear carbon seal, are lubricated by oil jets contained within the No 4/5 bearing chamber.

### 1.6.4 Main Gear Box bay

The MGB consists of a magnesium alloy casing mounted within the bay behind the two engines. The bay is protected from each engine fire zone by firewalls. The MGB bay is not a fire zone, and therefore has no fire detection or suppression systems, except that the fire bottles for the engine bays are mounted in the MGB bay. Hydraulic pipes and fuel pipes in the MGB bay are not required to be fire proof. There are a number of sensors, including those for the fuel and oil pressure systems and wiring within the bay, which provide electrical signals to captions on the flight deck.

### 1.6.5 Flying controls

The flying controls are powered by two hydraulic systems, 'Primary' and 'Auxiliary'. Normally, both systems are active. Either system may be deactivated by energising its fail-safe electrical solenoid. The application of electrical power to the solenoids is controlled by a three-position switch on each collective control stick grip. The switch positions are marked 'PRI OFF', 'ON' and 'AUX OFF'. The system is designed so that it is not possible to select both systems off simultaneously. A falling pressure in either system activates the other system regardless of the switch selections made by the crew.

### 1.6.6 Fire protection

Fire detection and suppression is provided for each engine bay and detection is by means of a separate fire-sensing loop for each bay. The fire-wire runs on the centre and canted firewalls within the engine bays and on the inside of the bay doors. In case of a temperature rise to above 575°F at the fire-wire, a control unit detects a resistance change in the sensor loop and illuminates a warning light on the flight deck. In addition, there is a synthesised audio voice warning which draws attention to the fire and identifies the engine concerned. There is no fire detection system within the MGB bay.

Fire suppression is provided by the engine fire extinguisher system with the two fire extinguisher containers mounted aft of the MGB. Discharge tubes for each engine bay are connected to both containers to provide a main and reserve supply when used for only one engine bay fire. When either of the 'FIRE EMERGENCY SHUT-OFF' selector handles is pulled, fuel flow to that engine is shut off and the 'FIRE EXT' switch on the overhead panel is energised. The 'FIRE EXT' switch may then be selected to 'MAIN' or 'RESERVE' as necessary to discharge the extinguishant into the appropriate engine. Each container has a main and a reserve cartridge. The Maintenance Manual contains a warning that firing a cartridge in an already discharged container may result in damage to the container.

### 1.6.7 Maintenance information

A Maintenance Check 'B' was completed at 29,754:40 airframe hours and the Certificate of Release to Service (CRS) was dated 29 November 2001. An 'A' check was completed at 29,828 hours on 16 May 2002. The next 'A' check was due at 29,868 hours, and the next 'B' check was due at 29,934:40 airframe hours.

The last major work in the area of the main rotor transmission had been a MGB change carried out on 24 November 2001 at 29,754:40 hours TSN, about

100 hours before the accident. The MGB was found to have excessive metal in the scavenge and pressure filter. The MGB was removed and MGB serial number A14-987 was fitted. That gearbox had been overhauled and full load tested, and issued with Federal Aviation Administration (FAA) form 8130-3 Airworthiness Approval on 13 July 2001. It had accumulated 18,129:27 hours Total Time (TT) at the time of the accident, and 158 cycles and 99:15 hours Time Since Overhaul (TSO). An MDS balance check was carried out at the time the gearbox was installed, and the vibration levels at the No 2 engine were well inside the published limits at all speeds tested from 80% to 108% free turbine speed ( $N_f$ ). Some hours after this balance check, the torques of the four 'T' bolts (Appendix A, page 2), which connect the MDS to the MGB input flange, were rechecked and found to be satisfactory. Since the MGB change, only routine minor maintenance had been carried out in the affected area.

At about 113 hours before the accident, the No 2 engine had been removed for lack of power. The problem was found to be in the compressor and, although the power turbine stage was removed, it was refitted to the rectified engine without any other work being carried out. The engine was re-fitted to the aircraft, and an MDS balance check was completed satisfactorily at that time.

The Magnetic Drain Plugs (MDP) on the No 2 engine (and incidentally on the No 1 engine) were checked at 29,848:50 hours, approximately six hours before the accident. There was no contamination of the MDPs with metal particles at that inspection.

From the Technical Log, there were no significant defects recently recorded, and there had been nil defects in the last six sectors.

#### 1.6.8 Health and Usage Monitoring System (HUMS)

HUMS was introduced into the UK in response to recommendations in the CAA's Helicopter Airworthiness Review Panel (HARP) report, published in 1984 (Civil Aviation Publication (CAP) 491). Such systems consist of onboard equipment (sensors, data processor, data recorder, etc) and a ground station for the processing and archiving of data. An Integrated Health and Usage Monitoring System (IHUMS), a specific proprietary system, was fitted to G-BBHM and this was used to monitor the health of the transmission and rotor systems of the aircraft with a view to providing early warning of any degradation. As part of the installation, vibration transducers were fitted at two locations on each engine, including one position on each power turbine diffuser. After flight, the data, which had been acquired and stored on a removable memory card was downloaded on to a ground based computer system. That system analysed the data for abnormalities. In particular, it

looked for any parameters which had exceeded pre-set thresholds. Had a threshold been exceeded, an alert would have been generated and this would have required investigation by engineering personnel. HUMS generally are complex systems making use of elaborate algorithms and, historically, have been subject to false warnings. However, on several notable occasions, they have provided timely warning of an impending serious failure. Normally, it is fairly straight forward to download the basic information together with any alerts. While HUMS systems have been constantly evolved and improved, it was not intended that line engineers should enter into detailed interpretation of that data.

Flight crews operate the IHUMS system by inserting a memory card into a card reader in the ground station before flight. They initialise or 'pre-flight' the card from this station and take the initialised card to the aircraft. During data acquisition the system looks for particular flight parameters, essentially straight and level cruise flight. The process requires approximately 20 minutes of cruise flight, although this need not be continuous should a cruise segment be less than 20 minutes. In that circumstance, the system will wait until the next time cruise parameters are established, before continuing with data acquisition. This process will terminate, however, if the aircraft lands. The flight crew have no indication of when the data acquisition process is complete. Engine vibration files are the last to be acquired during the sequence. During the acquisition process, data files are written from the Data Acquisition and Processing Unit (DAPU) to the memory card. After landing and before electrical shutdown, the closing data files are written and the card can be removed from the aircraft. The flight crew then insert it into the ground station to download the data before handing it over to engineering staff. The only evidence of an incomplete acquisition process is the number and size of the files downloaded into the ground station. For SAR operations, such as those conducted by G-BBHM, it was not unknown for some data not to be acquired due to the limited time spent in straight and level flight within the acquisition parameters. For other helicopters operating in the North Sea environment, this is not a problem and data downloads are routinely performed at the end of each day. The card is then erased prior to its being inserted into the IHUMS on the aircraft at the start of the next day.

The system was originally certificated for use on a 'No Hazard, No Credit' basis. This meant that the system was not certified against any functional health monitoring requirements or objectives. No warnings are provided to the flight crew (with the exception of Flight Manual exceedences) from which a decision about the airworthiness could be made in-flight. IHUMS has now been granted credit in respect of CAA AAD 004-10-93 and AAD 001-05-99, as a means of health monitoring.

The principles of the vibration analysis that formed the basis of the system fitted to the aircraft are still subject to improvement. Several trials have been conducted with ‘seeded defects’ to determine and widen the fault detection envelope of the system. Further development is also being carried out to reduce the number of spurious warnings caused by mechanical or electrical unreliability, and by inadequacies in the detection algorithms.

The IHUMS fitted to the aircraft was designed to perform a number of automatic functions:

- Monitor shaft health
- Monitor gear health
- Monitor for bearing defects
- Monitor main rotor track and balance
- Monitor tail rotor balance
- Provide crew with flight manual exceedence warnings.

In addition, a facility was provided to enable the crew to request the taking of a snapshot of data by using a cockpit mounted control panel.

#### 1.6.8.1 Principles of Operation

The IHUMS system relied on the fact that every rotating part of the transmission system in the aircraft has an associated vibration signature due to imperceptibly small manufacturing and assembly anomalies. Once the aircraft became established into a particular phase of flight the system automatically sampled and logged the vibration signatures by sequencing through the output of various sensors and recording the measurements. Ten separate phases of flight were allocated, each with their own selection of vibration sources to measure.

A number of accelerometers were mounted on the aircraft in positions determined to best pick up particular vibration sources. The nature of the accelerometers was such that they picked up all vibrations within their locality and so signal processing was required to extract the vibration signature of a particular rotating part. This processing required knowledge of the speed of rotation of the source and so the sampling of the accelerometer signal was governed by a master timing signal derived from the number one engine turbine. Signal averaging was also used by taking a large number of samples. This helped to reduce the level of vibration components that were not synchronous with, and hence not relevant to, the vibration originating from the source being measured. Due to the transient nature of some of the flight phases, such as hovering, the number of signal averages was reduced to enable time for more sources to be measured. In addition, the number of sources to be measured and the large number of averages used in the cruise condition meant that the time

taken to complete a full cycle of measurements could have been up to 20 minutes. Health data is acquired only in the cruise; during hover, only rotor track and balance data is acquired.

The ground station was capable of analysing the data and would alert the operator if the vibration from any particular source exceeded a predetermined limit. All IHUMS threshold values are reviewed, at least, on an annual basis and these limits were initially set during the design of the IHUMS, after assessing each vibration source and its likely vibration level during normal operation. The ground station would then print out a report detailing areas of concern in order to aid aircraft maintenance. Further facilities enabled the operator to extract time histories of parameters, and the newer Windows NT version of the software is capable of generating multiple time histories simultaneously. However, the operator must initiate this function but with so many available parameters, there must be a threshold exceedence warning, or other reason, to alert the operator.

Vibration sources of importance in this accident were those associated with the MDS, and formed part of those engine vibration files most recently acquired during the last sampling sequence.

#### 1.6.9 Limitations

##### 1.6.9.1 Engine start procedure

There are no specific limitations within the Flight Manual relating to starting procedures. Limitations during start are subject to the overall figures in the Manual. For example, upon rotor engagement the torque limit is stated as 123%; this is based on flight conditions rather than during start. However, amongst the operator's crews, there was a widespread acceptance that, during start and rotor engagement, torque should normally be about 40 to 50% and should not rise above about 80%; this 'operator limitation' was not published. The control of torque during start was achieved by appropriate use of the engine power lever. After the accident to G-BBHM, the operator introduced a formal requirement to observe a 40-60% torque limit during engine start. This requirement was included in a Flying Staff Instruction (FSI); the FSI is detailed in paragraph 1.18.3.

##### 1.6.9.2 Operator procedures

Single engine starts, where the start of the second engine is delayed until after the rotor is engaged, have not historically been the norm for S-61 operations. The operator reportedly introduced them during the 1970s in order to minimise the time spent with two engines idling while conducting pre-flight activities. At

the time, the normal mode of operation of the helicopter was scheduled passenger operations. Within the UK, the company operating G-BBHM was the only one using single engine starts. It is unclear how many operators world wide employ single engine starts.

At the time of the accident, the normal company procedure was for the designated handling pilot to start one of the two engines with the rotor brake engaged. When  $N_g$  (gas generator speed) and T5 (exhaust gas temperature) were stabilised at ground idle, the handling pilot advances the power lever and, with a 1 to 2% rise in  $N_g$ , releases the rotor brake. During this procedure, the power lever would be advanced progressively until normal  $N_g$ ,  $N_f/N_r$  (free turbine/main rotor speed) and T5 were achieved. During this stage, the torque gauge and T5 gauge would be closely monitored. As a standard operating procedure, the order of engine start would be varied with No 1 engine being started first on odd numbered days of the month; No 2 engine would be started first on even numbered days.

Company start procedures were adhered to for the accident flight, with No 1 engine being started first, and no abnormalities were noted by either pilot. However, during the investigation, IHUMS records showed that three instances had occurred when the 'unwritten' 80% rotor engagement torque limitation was exceeded. These occurred on the day prior to the accident to different pilots from those operating on the accident flight. On these start sequences, torque values above the 'normal' unwritten company starting torque limitation were noted. The crew involved were not aware of the 'unwritten' 80% limit and the torque recorded did not exceed any published limitation. IHUMS recorded these torque readings but no action was subsequently taken since no warnings were generated and in neither case was the threshold for maintenance action exceeded.

#### 1.6.9.3 Single engine climb torque limit

A torque limit of 115% is specified for the single engine climb case at inter-contingency power. In some conditions of atmospheric temperature and pressure, this torque limit could inadvertently be exceeded while operating within the engine's  $N_g$  and T5 limits. There is a further margin between the torque limit and the torque at which damage would occur. Any torque above 123%, depending on the duration and degree of overtorque, may require maintenance action. Any torque exceeding 150% for more than 20 seconds requires maintenance action, as damage is predicted in this case.

While this torque limitation (115%) is intended to protect the transmission and gearbox input in the single engine climb case, excessively high torque in any phase of the operation could cause similar damage. There is no data available

from Sikorsky on any additional adverse effects arising from high torque at lower than normal  $N_f$  speeds.

## **1.7 Meteorological information**

The weather was good on the day of the accident. The conditions were CAVOK (no cloud below 5,000 feet amsl and visibility greater than 10 km) with an outside air temperature of 24°C, no reported turbulence and a surface wind of 270°/10 kt.

## **1.8 Aids to navigation**

Not applicable

## **1.9 Communications**

At the time of the accident, the crew were in contact with Bournemouth Radar on VHF and with HM Coastguard on FM. A radio recording of the Bournemouth Radar frequency was available but contained nothing relevant to identifying the cause of the accident.

## **1.10 Aerodrome and approved facilities**

Not applicable

## **1.11 Flight Recorders**

### **1.11.1 Crash-protected recordings**

The crash-protected recorder fitted to the aircraft was a Penny and Giles Combined Voice and Flight Data Recorder (CVFDR) type D51506 and formed part of the aircraft's IHUMS. The CVFDR was installed in the tail boom of the helicopter and had suffered only superficial fire damage during the accident. Following recovery from the accident site, the recorder was taken to the AAIB where the magnetic tape was removed and replayed successfully.

The CVFDR was designed to, and had maintained a record of, the most recent five hours of aircraft data and one hour of three channels of audio; the commander, co-pilot and area microphone. The voice recording method used on the aircraft was the 'hot microphone' system where the two cockpit crew's microphones were always live regardless of whether a radio push-to-talk button was depressed or not. Through use of the intercom system, speech from the two crew members in the rear of the helicopter was also recorded.

The cockpit area microphone was located on top of the main instrument panel, just to the right of centre. A separate pre-amplifier was provided to boost the audio signal to a level acceptable for recording on the CVFDR. The design of the area microphone recording channel enabled the identification of audio signals with frequencies between 40 Hz to in excess of 5 kHz as long as the signal amplitude was larger than the background noise level. The audio recording started three minutes before the landing prior to the accident flight and covered the entire accident flight itself. A period of engine running on the ground between these two sorties was also recorded.

The data recording system comprised many transducers, distributed throughout the helicopter, designed to convert physical parameters into electrical signals. These signals were then multiplexed in a predetermined order into a serial data stream prior to being recorded on the CVFDR. The data retained on the CVFDR commenced with information from operations conducted on the 13 July, two days prior to the accident. Data recording terminated after the accident flight at the same point as the voice recording. As part of the IHUMS, the CVFDR recorded flight manual exceedences concerning the engines and gearbox oil temperatures. None of the IHUMS data recorded on the CVFDR was of relevance to the investigation of this accident.

A small subset of the parameters recorded, together with their transducer details, was of particular relevance to this investigation and is tabled in Appendix B.

#### 1.11.2 Data recorded prior to the accident flight

The CVFDR had retained the last five hours of data, including the accident flight, of which four and a quarter hours of data had been recorded prior to the accident flight. This five hour period included four engine starts with the first three, carried out on the 14 July, having commenced with No 2 engine. The last, on the morning of the day of the accident, commenced with No 1 engine being started first but was for ground running only and the rotors were not engaged.

##### 1.11.2.1 High torque rotor engagements

The first start recorded on the 14 July had an associated maximum transient torque of 87% (at 74%  $N_r$ ) at rotor engagement, whereas the second and third had resulted in higher transient engine torques being recorded as the rotors increased in speed. During the second start, a peak No 2 engine torque of 115.2% at 87%  $N_r$  was recorded whilst the figures for the third start (No 2 engine) were 104.5% torque at 83%  $N_r$ , as shown in Appendix C. On the day of the accident, 15 July, the final start recorded, at the beginning of the accident flight, indicated a peak transient torque of 72% at 86%  $N_r$ .

### 1.11.2.2 Variation in $N_f$

Once the entire five hours of data had been plotted out, it was apparent that there was some unusual variation in the No 2 engine recorded  $N_f$  values. This variation appeared to be in the form of short term peak to peak changes (jitter), and did not manifest itself in the recordings of No 1 engine  $N_f$ , or  $N_g$  of either engine, or  $N_r$ . Furthermore, the magnitude of the jitter, having remained relatively constant up to the point of the first high torque start, appeared to increase for the remaining two hours of the recording.

A statistical analysis of the  $N_f$  fluctuations was carried out by taking the average deviations from five consecutive recorded samples of  $N_f$ . Engine starts and other rapid, crew action induced, changes were excluded and the results plotted for the entire five hour recording. A graph of the results is shown in Appendix D and clearly shows an increasing trend in No 2 engine  $N_f$  deviation towards the end of the accident flight.

### 1.11.2.3 Variation in torque

It was also observed that the recorded values of torque for No 2 engine varied more than those for engine No 1. There was no discernible increase in the amount of variation over the five-hour period of the recording but periods were evident where torque splits of 10% or more between the engines had been recorded. No consistency was observed in which of the two torque readings (No 1 and No 2 engine) was higher at any given moment but, in general terms, the peak to peak variability of No 2 engine torque was larger. No correlation was evident between this torque variation and the No 2 engine  $N_f$  fluctuations described previously.

## 1.11.3 Accident flight recordings

### 1.11.3.1 Flight prior to onset of the event (15 July 2002)

The CVFDR started to operate once electrical power had been applied and, almost immediately, engine No 1 was started and the rotors engaged 30 seconds later. No 2 engine was started, once the No 1 engine had stabilised at 88%  $N_g$  with T5 at 510°C. Both starts were uneventful, with a maximum transient torque of 72% being recorded on the No 1 engine.

The aircraft lifted within three minutes of electrical power being applied and departed to the north-east towards Poole Harbour. The helicopter accelerated to about 115 kt and climbed to 760 feet amsl. The search area was reached nine minutes after departure. Once in the search area the helicopter slowed and descended. For the first quarter of an hour, the search pattern was flown at

300 feet agl with airspeeds ranging between 30 kt and 75 kt. The remainder of the search was flown at 200 feet agl, or below, and at similar airspeeds.

### 1.11.3.2 The onset of the event

Approximately 44 minutes after the flight commenced the helicopter had just completed a 90° left turn and had descended to 140 feet. As airspeed increased to 87 kt, the torque produced by the No 2 engine began to fall below the values of those associated with the other engine. In addition, the recorded values of  $N_f$  for the No 2 engine began to show a much larger peak to peak variation.

Twenty seconds later at the start of a climb, with torques of 40% and 29% on engines No 1 and No 2 respectively, a momentary change in value was observed in some parameters. Engine No 2 torque dipped to 4% before increasing to 30% whilst that for engine No 1 increased to 51%. Immediately following this, over a period of 2 seconds (sampling interval of 0.5 seconds), the following values of  $N_r$  and  $N_f$  from both engines were recorded:

<b>Sample</b>	<b>Main rotor <math>N_r</math> (%)</b>	<b>Engine 1 <math>N_f</math> (%)</b>	<b>Engine 2 <math>N_f</math> (%)</b>
1	102.0	102.2	102.3
2	101.1	100.4	96.5
3	99.9	100.7	104.7
4	101.0	101.9	99.0
5	101.7	102.4	103.2

Just after these variations the crew started to discuss that they had felt something but their conversation was interrupted within two seconds by the activation of the aural warning “<chime>, <chime>, <chime> fire engine 2”.

Within ten seconds the crew had diagnosed and confirmed the No 2 engine fire warning and retarded the engine to ground idle; torque on the No 1 engine began to increase. They then shut down the No 2 engine by pulling its ‘T handle’. As the No 2 engine  $N_g$  and  $N_f$  shaft speeds began to reduce, the aural fire warning repeated but this time announced “<chime>, <chime>, <chime>, fire engine 2, fire engine 1”. The presence of fire warnings on both engines was confirmed by the change in state of the two engine fire parameters recorded in the data on the CVFDR but the crew made no comment about the activation of the fire warning on their remaining engine. At that time they were actioning the fire drill on No 2 engine. The fire warnings for both engines continued at ten-second intervals for the remainder of the flight. As No 2 engine  $N_g$  reduced through 63%, the ‘LOW OIL PRESSURE ENGINE 2’ warning was activated

which also caused a 'MASTER CAUTION' indication. The helicopter temporarily levelled at 240 feet agl.

The helicopter resumed the climb and started to turn right but, as it was passing 300 feet agl, the commander observed that the 'MGB LOW OIL PRESSURE' (triggered at less than 7.5 psi) caption had illuminated. This resulted in a further 'MASTER CAUTION' indication. The recorded data confirmed that the low oil pressure discrete parameter had changed to indicate a warning but the recorded values of actual oil pressure showed 44 psig, a value within the normal range of operation. As the low pressure indication became active, actual oil pressure increased to 48 psig as the emergency lubrication system was activated. Simultaneously with the 'LOW OIL PRESSURE' and 'MASTER CAUTION' indications, the No 1 engine fire discrete parameter reverted to a 'no fire detected' state. MGB oil temperature, which had remained relatively constant until that time at 94°C, began to rise.

The helicopter levelled at 360 feet agl as the crew observed that there was now smoke in the back of the cabin and they highlighted the possibility of landing on the Royal Marines parade ground. As they reviewed their current status of low transmission oil pressure and an engine fire, the commander observed that the primary hydraulic system now indicated low pressure and elected to carry out an immediate landing. The recorded data showed that, almost simultaneously with the low primary hydraulic pressure warning, indications of the emergency floatation bags fired, secondary hydraulic system low pressure and secondary hydraulic system deselect warnings were activated. The floatation bags fired discrete parameter reverted to a 'no warning present' state within three seconds of its activation. The helicopter began to descend and the winchman informed the Coastguard of their intention to make an immediate landing on the sports field.

At 280 feet agl, five seconds after the start of the descent, the No 1 engine fire warning discrete parameter re-activated. At the same time the MGB low oil pressure, followed by the primary and secondary hydraulic system warnings, reverted to 'no warning present' states. It was observed that, due to hydraulic system switching, disturbances in the position of the lateral cyclic control occurred at the same time as the hydraulic system warnings changed state. The handling pilot later confirmed that the cyclic had 'kicked' at those times. The landing gear was lowered at 210 feet agl, airspeed started to reduce and the helicopter turned left by 40° onto a heading of 310°M for the landing. Once on the ground, the No 1 engine was shut down immediately and the commander called for everyone to get out of the helicopter. The data showed that the rotor brake was applied but no braking effect was observed in the recording of main rotor speed.

The time between the initial engine fire warning and when the helicopter landed was 82 seconds and the CVFDR recording terminated 15 seconds later as power was removed. Main rotor speed had reduced to 62% whilst MGB oil temperature had increased to 129°C at the end of the recording.

Pertinent parameters recorded during the accident flight are shown in Appendix F, Figures 1 to 3.

#### 1.11.3.3 Low engine oil pressure parameters

Each engine was fitted with a low engine oil pressure switch, the states of which were recorded on the CVFDR. Engine oil pressure was not recorded. Each oil pump is driven from the engine gas generator and, as the associated engine was started or shutdown, the switch changed state. In an attempt to correlate the rate of pressure increase during start-up or pressure decay during shutdown, the engine  $N_g$  value at each change of state of the switch was noted. In total, five starts and six shutdowns for each engine on G-BBHM were recorded on the CVFDR. These were compared with results from an additional four pairs of starts and five pairs of shutdowns from another aircraft. The results, plotted in Appendix E, show that No 2 engine low oil pressure switch activated at a much higher  $N_g$  during the in-flight shutdown than during any other recorded shutdown, whereas the No 1 engine switch did not.

#### 1.11.3.4 Time correlation of recordings

The audio and data recordings from the CVFDR were time correlated, primarily with reference to the engine parameters and their corresponding spectral characteristics. The usual method of correlation would have been to relate the timings of radio transmissions made by the crew to those of the activation of the radio push-to-talk discrete parameters, but no suitable changes of state were recorded in the data.

#### 1.11.3.5 CVFDR noise analysis

The AAIB and the aircraft manufacturer conducted an analysis of the noises recorded on the area microphone channel of the CVFDR. The aircraft operator also arranged for a representative test flight to be conducted on another S-61N with an identical type of recording system in order that acoustic comparisons could be made without introducing differences associated with microphone location or recording system characteristics. The aircraft used for the test flight was G-BPWB.

The CVFDR audio record from G-BBHM, being of only one hour in duration, did not contain the period associated with the higher torque starts referenced in

paragraph 1.11.2.1, but did cover the period of the accident. What was evident from spectral analysis was a high amplitude peak varying between 316 Hz and 335 Hz throughout the recording. The presence of this peak during the previous flight was apparent only during the last few minutes of flight and was coincident with an increase in  $N_f/N_r$  and engine torque during the approach and landing (refer to Appendix G). The first two harmonics of this fundamental frequency were also evident during this short period.

This frequency range is associated with a 'once per revolution' (1/rev) of the MDS of each engine, (19,200 RPM or 320 Hz). During the engine starts before the accident flight, the peak became evident as soon as the  $N_f$  of the No 2 engine exceeded 90%. Subsequent starting of the No 1 engine did not increase the amplitude of the peak. The aircraft manufacturer also observed that this signal was the loudest of the peaks recorded whilst their previous experience has been that the peaks associated with the planetary mesh (at approximately 660 Hz) were expected to have been the highest. In comparison, the recording from G-BPWB did not show the high level of the main drive shaft 1/rev frequency but had the highest amplitude signal occurring at 660 Hz. This was consistent with the experience that the manufacturer had gained from acoustic tests on other aircraft. By plotting a spectrogram of frequencies versus time it was possible to derive a near continuous trace of  $N_f$  as opposed to the sampled values obtained from the data recording. It was apparent that this continuous trace did not exhibit the same jitter in the recorded  $N_f$  signal described previously.

Shortly after G-BBHM departed from Portland the main rotor speed reduced to 100% and the MDS 1/rev frequency peak became significantly less pronounced. That associated with the planetary mesh was still evident. As the aircraft reached the search area,  $N_r$  increased marginally, the MDS frequency peak grew in amplitude and the harmonics at the MDS 2/rev and 3/rev frequency reappeared, but were intermittent in nature. Also from that point additional, higher frequency peaks became apparent and varied in frequency in line with that of the MDSs. The only frequencies of interest were at the 100%  $N_f$  equivalent of 981 Hz, 1,798 Hz and 2,166 Hz.

The manufacturer stated that the 2,166 Hz frequency peak was very close to the No 5 bearing inner race ball passing frequency. It was observed that this peak remained evident until the helicopter landed, significantly after the time that the No 2 engine had been shut down, thereby indicating that it had originated in the No 1 engine.

Approximately 20 minutes before the No 2 engine shutdown, the harmonics of the MDS frequency became more pronounced and the 1,798 Hz signal reduced in amplitude below the background noise level. Approximately seven minutes

before engine shutdown the  $N_f$  harmonics grew stronger with pronounced sidebands associated with the noise of the main rotor blade passing frequency.

All of the  $N_f$  related harmonics were evident after the unusual noise event noted by the crew and only decayed, rapidly, in both amplitude and frequency as the No 2 engine was shut down. The origins of the 981 Hz and 1,798 Hz peaks could not be determined from the standard set of frequency signatures provided by the manufacturer.

#### 1.11.3.6 Previous investigations

In 1980, the AAIB conducted similar audio analysis during an investigation into a S-61N accident involving a MDS. That helicopter was G-BEID and was fitted only with a Cockpit Voice Recorder (CVR), which was of an earlier design than that on G-BBHM. The differences in audio response of that recording system made it difficult to draw direct comparisons with the evidence from G-BBHM but some similarities were noted. In particular, both recordings showed an increase in the amplitude of the fundamental frequency of the MDS and the development of the associated second harmonic nearer to the time of failure. Additionally, a higher frequency peak at 1,970 Hz was observed which, although it did not correspond directly with a No 5 inner race ball passing frequency, was attributed as being from that source, after allowing for probable bearing degradation.

#### 1.11.3.7 Analysis of HUMS data

Although the data card for G-BBHM had been lost in the fire, all the available IHUMS data already downloaded was reviewed by the equipment specialists following the accident. Interrogation of this data held by the operator showed that the engine vibration data had not been acquired on the penultimate flight due to the short flight time. In particular, the (Aft Broadband) vibration parameter from the No 2 engine was of interest. The threshold for this parameter (peak to peak) was set at approximately 28 g. Although there was no threshold exceedence or clearly defined trend in the data, there was an increasing variability and a rise from the datum (approximately 14 g) to over 20 g during the last 30 hours. Detailed analysis of the data, using Fast Fourier Transform (FFT) techniques did not show clear evidence of an association with any degradation or the subsequent engine failure. The ground station graphical output for this parameter is shown in Appendix H.

### 1.12 Examination of wreckage

Preliminary examination of the wreckage at the landing site rapidly established that the Thomas coupling at the forward end of the MDS on the No 2 engine had

severed the EMRSA tube, by means of a heavy rotational rubbing contact mechanism around the inner circumference of the tube. This suggested that the Thomas coupling had run eccentrically. Radial movement was observed at the rear of the power turbine, supporting this view. This in turn implied damage to the carbon oil seal at the back of the power turbine assembly, and the release of engine oil into the region where the heavy rotational contact had occurred.

On site, it was observed that one of the fire extinguisher bottles mounted in the MGB bay was found to have been penetrated by the firing of one of its own cartridges.

#### 1.12.1 Engine strip examination

The engines were delivered to the operator's maintenance facility at Redhill for strip examination, under the supervision of the AAIB. The power turbine assembly of the No 2 engine was completely stripped, and some other parts of each engine were examined, notably the bearings.

The engine aluminium and magnesium alloy components had been destroyed in the ground fire. Therefore no oil, and only the Accessory Gear Box (AGB) magnetic drain plug (MDP), was recovered for the No 2 engine. The other three MDPs including the 4/5 bearing scavenge MDP had been destroyed by ground fire. The AGB MDP was heavily contaminated with blackened magnetic (ferrous) deposits. The oil filter element was dismantled and the 40-micron filter discs were found to be heavily blocked with fine debris, black in colour with some bright specks. The level of contamination and blockage suggested that the filter would have gone into bypass mode.

The No 2 engine's power turbine module was removed from the engine and mounted on a work stand. The severed flange of the mounting tube was removed. The turbine would not turn due to either debris or contact with the casing, or both. The MDS was removed at the Thomas coupling and the shaft was found to be cracked (see paragraph 1.12.4). The Thomas coupling itself showed no evidence of failure prior to contact with the ERSMA tube. However, a crack was observed in several laminations of the coupling at one of the bolt positions. The turbine rotor was released and the carbon face seal was found to have broken up, with all of the carbon element missing. Damage was observed to the No 5 bearing and both associated oil jet assemblies (see Appendix A, Figure 4, for a detailed description), where the cantilevered tubes which supply oil to the carbon seal, were broken off. Corresponding damage was found to the right angle drive gears into which one of the tubes had fallen. The No 5 bearing was found to contain eleven balls (the correct number) but initially the bearing cage was not evident. The balls measured between 0.52 to 0.45 inches diameter (nominally 0.5625 inches diameter). The inner race halves were heavily

worn and smeared and the shaft exhibited surface oxidation, blue in colour. The No 5 bearing was assembled the correct way round in the housing but the screws securing this housing were found to be loose. This was thought to have resulted from the in-flight vibration, but may also have been due to the effects of the ground fire. The turbine had run eccentrically in the No 5 bearing, causing it to 'orbit', and sustain damage and heavy wear to the turbine blade tips and retainers at the blade roots. The No 4 bearing at the front of the turbine, and the remainder of the assembly, was undamaged.

The No 2 engine's front frame section accessory drive was checked by removing the engine's starter drive, as this area has been known to provide early indications of engine distress arising from a number of possible causes. No evidence of pre-existing deterioration in this area was found.

#### 1.12.2 No 5 bearing

It was confirmed from the bearing part number, P/N 6051T67P02, that the bearing was the latest standard, with M50 specification steel balls and races with a 4340 specification steel cage. Modification actions had been taken after the G-AZRF and G-BEID events (see Appendix I) to bring the No 5 bearing to this standard. It had been installed at the last overhaul and had completed 1,563 hours TSN. This part has a 6,000 hour life.

The engine manufacturer prepared a metallurgical report which showed that the bearing had been traced back to production batches which experienced no unusual manufacturing issues and which were not the subject of any in-service reports. The balls showed no evidence of spalling (a fatigue mechanism in which small chunks or flakes of the bearing are released). Of the eight balls measured (three had been used for metallographic analysis) all had been uniformly reduced in diameter by about 11%, a mean reduction of 0.0565 inches. As a result, radial free play in the order of 0.10 inches may have existed at some time before engine shutdown. The bulk microstructure of the balls was overtempered martensite of hardness value 57 or 58 on the Rockwell 'C' scale (HRC). The near surface microstructure exhibited significant plastic flow and microstructural alteration, with HRC values of 66 to 68. As manufactured, the values for this batch were between 61 and 61.7 HRC.

The outer race showed no evidence of spalling. Smeared material consistent with that of the balls and inner race was found between the 4 o'clock and 8 o'clock positions. The microstructure was consistent with properly processed M50 steel and the hardness of 62/63 HRC was within drawing limits of 60 to 64 HRC.

The inner race exhibited severe, asymmetric wear and some cage material had been smeared on to the inner race. The bulk microstructure of both inner race halves exhibited an overtempered martensitic structure with a hardness value of 53 to 55 HRC. The near surface microstructure showed significant plastic flow, oxidation and microstructural alteration. Near surface hardness was measured as 60 to 64.5 HRC. The relatively undeformed forward half of the bearing cage was found in a cavity forward of the bearing housing. Due to the distortion of the inner race and the dimensions of the cage, it was evident that the cage had broken up and passed forward over the inner race, but before the damage to the inner race became severe. There was evidence that the inside diameter of the cage and the shoulders of the inner race halves had made contact with each other prior to the cage fracture.

The manufacturer's report concluded that oil starvation of the bearing was not consistent with these observations. Specifically, it concluded that the lack of heat damage to the forward half of the cage, and the lack of microstructural alteration of the outer race material, were not consistent with a general lack of lubrication but were consistent with dynamic radial imbalance loading of the bearing.

The United States National Transportation Safety Board (NTSB) was asked to examine the No 5 bearing, together with several other parts. Their report indicated that the outer race exhibited some localised subsurface microstructural alteration at sections taken from the eight and eleven o'clock positions (looking aft). The section made by the engine manufacturer was taken at the 4.30 o'clock position and did not exhibit subsurface microstructural alteration. The NTSB identified localised corrosion beneath the smeared material on the outer race.

### 1.12.3 Oil jet assemblies

Two oil jet assemblies, as shown in Appendix A, page 4, are installed in the No 4 and 5 bearing chamber on each engine. One is installed at the 12 o'clock position, the other at the 6 o'clock position, but only the lower of the two (the 'live' assembly) is supplied with oil. The 'dead' assembly is fitted to the engine to allow the power turbine module to be installed in either the No 1 or No 2 engine, whilst always retaining an 'active' oil jet at the 6 o'clock position. When operating, oil under pressure exits from relatively small holes at the end of the oil tubes, and jets directly on to the inner race areas of No 4 and No 5 bearings, and a further nozzle provides an oil supply to the free turbine carbon oil seal.

The oil jet assemblies associated with the No 4 and No 5 bearings from the No 2 engine were examined by the engine manufacturer. Their laboratory report concluded that:

*‘Both the live and dead oil jets had failed in fatigue;*

*The dead oil jet, located at the 12 o’clock position on the engine, had fatigue initiations in the 1 to 3 o’clock area;*

*The live oil jet, located at the 6 o’clock position on the engine, had fatigue initiation sites in the 11:30 to 12:30 area;*

*The fracture surfaces of both oil jets were completely oxidised. No microscopic features could be clearly discerned;*

*Bearing material, including M50 steel and silver, were found on the dead oil jet tube. No foreign debris was found on the live oil jet nozzle;*

*Both nozzles were confirmed to be 300 series stainless steel.’*

The NTSB report indicated that both oil jets had fractured because of multiple initiations at or near the upper side of the tube. In both cases the multiple origins and relatively large over-stress regions were considered typical for fatigue under relatively high stress amplitudes at initiation and at final fracture.

#### 1.12.4 Main Drive Shaft assemblies

The AAIB commissioned a number of material examinations and analyses. In addition, analyses were conducted in the United States by the NTSB and the airframe, engine and bearing manufacturers.

The MDS for the No 1 engine was damaged where the ‘T’ bolts attached its rear flange to the MGB input pinion coupling. One ‘T’ bolt was still in place and secured with its normal fixing hardware. The bolt had sustained some permanent bending distortion. Part of the input coupling lug was under the head of the bolt, the lug having broken. The corresponding lug on the Main Drive Shaft showed severe bending in the same area. All the remaining ‘T’ bolts had apparently broken and all the remaining lugs were distorted to a greater or lesser extent. Only part of one other ‘T’ bolt was recovered, the head was found less most of its shank, which had failed in overload.

A circumferential crack in the No 2 MDS was found, in an orientation opposite the asymmetric rub on the turbine and associated with deformation of the shaft due to bending. Eccentric wear on the inner race of the No 5 bearing was consistent with, and oriented to, orbital motion of the shaft. The NTSB report identified areas of fatigue in the crack and multiple origins were found at the outer surface of the shaft, with the fatigue affected area extending through about 50% of the shaft wall thickness. At the aft end of the shaft, each of the

four attachment lugs was deformed and the NTSB report concluded that these deformations were consistent with all the 'T' bolts being in place while the deformation occurred. Sikorsky formed a similar conclusion in their examination of the shaft. The AAIB examination noted that three of the holes were stretched, and the fourth was burred in a manner which indicated that all the bolts had been in place when the shaft bending occurred.

At aircraft hours 29,853:20, the TT at time of the accident, the No 2 MDS had completed 5,310:15 hours TSO and 16,793:20 hrs TSN. The Time Between Overhauls (TBO) is 6,000 hours. The shaft, part number S6135-20860.001, serial number 66W, was repaired in 1992 at 13,467.17 hours TSN, but the reason for the repair was not recorded on the component log card. It was first fitted to G-BBHM on 14 June 1999.

#### 1.12.5 'T' bolts

In March 1999, the airframe manufacturer had issued an Alert Service Bulletin 61B30-14 advising that some 'T' bolts could have suffered hydrogen embrittlement during the cadmium plating process, and should be removed from service. Identification of the affected parts was to be by the appearance of the cadmium plate finish. The operator, however, was unable to distinguish between satisfactory 'T' bolts and those which could have been affected. Therefore, a Type Engineering Directive 49A was issued by the operator to change all the 'T' bolts on its S-61 fleet. G-BBHM was not so checked or reworked due to an administrative error. It is possible, therefore, that affected 'T' bolts were still fitted to the aircraft at the time of the accident. Examination of the parts by NTSB, AAIB and Sikorsky indicated, from the damage to the bolt holes, that all the four 'T' bolts associated with the No 2 installation had been in place at the time of the ground fire. Historically, loss of a 'T' bolt has resulted in shaft imbalance sufficient to destroy the associated MGB input pinion journals (see events involving aircraft registered G-BCLD and military registration 61786 in Appendix I).

#### 1.12.6 Thomas coupling

Each engine's power turbine shaft is connected by a flexible Thomas coupling to its respective MDS. The Thomas coupling consists of two flanges, one on each shaft, bolted together through eleven very thin steel laminations to permit some flexibility. The coupling is therefore able to accommodate small misalignments between the two shafts.

The Thomas coupling between the No 2 engine turbine shaft and its MDS had a heavy rub on one corner, consistent with rubbing contact with the EMRSA tube.

In addition, cracks were found in five of the eleven laminations at one of the bolt positions. A certain amount of cracking in these laminations is acceptable.

#### 1.12.7 Engine Mounting Rear Support Assembly (EMRSA)

The EMRSA consists in the main of a support tube with a large flange at its forward end, which is bolted to the engine, so that the tube surrounds the MDS and its couplings. The EMRSA on the No 2 engine had been severed around its entire circumference adjacent to the Thomas coupling. The normal clearance between the outside diameter of the Thomas coupling and the inner diameter of the tube is about ¼ inch. The severed ends of the tube, like the Thomas coupling itself, had markings consistent with a heavy rub. The tube was also distorted from contact with the shaft as the engines and gearbox had collapsed during the ground fire. The assembly had an overhaul life of 6,000 hours, and had completed 3,326 hours TSO and 27,195 hours TSN.

#### 1.12.8 Other bearings

The No 2 bearing from engine No 2 and the No 5 bearing from engine No 1 were examined by an independent metallurgist. Neither bearing showed any signs of unusual wear. In addition, these bearings and the No 4 bearing from engine No 2 were examined by the NTSB metallurgist and by the bearing manufacturer. No significant anomalies were found in any of these bearings. Some slight corrosion was evident, but this was attributed to the effects of the ground fire and the large amounts of extinguishant foam which was used. No pre-accident damage or deterioration of these bearings was indicated.

#### 1.12.9 Main Gear Box

The AAIB undertook a reconstruction of the MGB moving parts. Only the steel components, such as gearwheels, shafts and bearings, had survived the ground fire; the casing, which was made from magnesium alloy, was destroyed. These remaining parts were assembled in a wooden space frame in their original geometrical relationships. From this reconstruction, it could be seen that there was no evidence of any pre-accident mechanical problems with the MGB. None of the gear teeth showed any evidence of abnormal wear, and the shafts and bearings appeared to be free of any major distress prior to the fire. The epicyclic gearbox was itself complete, being contained in a steel casing which supports the outer ring gear. It was also apparent from the reconstruction that no damage to the input pinions, consistent with any in-flight loss of a 'T' bolt, had occurred.

The rotor brake and its friction discs (pucks) were recovered and these showed no signs of distress. In particular, the rotor brake disc was free of warps or

distortion, and carried no abnormal witness marks. The pucks also were free of abnormal wear or witness marks. Abnormal friction between the rotor brake disc and the pucks is known to result in heating and distortion of the brake disc, and damage generally to the rotor brake system.

### **1.13 Medical and pathological information**

Not applicable

### **1.14 Fire**

The local Poole Fire Service had been alerted and reached the scene within a few minutes of the helicopter coming to rest. They contained the already well developed fire but, by then, some 75 to 80% of G-BBHM had been consumed.

### **1.15 Survival aspects**

Once the helicopter had come to rest and No 1 engine was secured, the commander ordered an evacuation. This was expeditious and, once clear, the crew stayed away from the burning helicopter. The crew also kept other 'helpers' well clear and awaited the arrival of the Fire Service.

### **1.16 Tests and research**

#### **1.16.1 Metallurgical tests**

Metallurgical reports commissioned by the AAIB included the examination of the No 2 MDS and Thomas coupling, the bearings of both engines, the oil jets from both engines, the oil pump, MDP and filter of the No 2 engine and the condition of the MGB input pinions and journals. No evidence was found of any pre-ground fire deterioration of the MGB input pinions. The oil jet fractures were attributed to a fatigue process and, although the cracks appeared to propagate from one side of the tubes only, a resonance mechanism was considered to have been the only plausible mechanism by which this could have occurred.

The independent metallurgist commissioned by the AAIB, considered that the No 5 bearing had been damaged as a result of a lack of lubrication, possibly due to the failure of the live oil jet, if that had preceded the bearing failure.

In general, however, there was broad agreement between the reports from the other parties, except over the mode of bearing failure. The engine manufacturer submitted data (Appendix J) supporting the view that the bearing had failed as a

result of unusual, excessive cyclic loading, and this view was broadly supported by the NTSB metallurgist.

The strongest evidence opposing this view, and supporting the scenario of a loss of, or inadequate, lubrication, was to be found in the records of the accident to G-AZRF on 16 September 1976. In brief, following maintenance activity within the No 2 engine power turbine, a test flight was conducted during which the live oil jet failed. There was a subsequent No 5 bearing failure, severance of the EMRSA tube and an in-flight fire which, on that occasion, was extinguished. The oil jet had been damaged during assembly, and the damage to the No 5 bearing was very similar to that which occurred in the accident to G-BBHM. In a letter dated 28 April 1977, the engine manufacturer stated:

*‘.....the failure of the subject power turbine was caused by misdirected or reduced oil flow from the oil nozzle which resulted in oil starvation of the No 4 and 5 bearings.’*

Both the FAA and CAA accepted this conclusion.

The NTSB metallurgist concluded that G-BBHM's oil jet tubes had fatigued as a result of a resonance mechanism associated with high stress reversals. Similarity was found with the other accidents where the oil jets had fractured, in that the origins were generally at either top or bottom, or both, of each jet. In four accidents, a total of eight jets had been fractured or cracked. In each case loss of location of the MDS had occurred and three of the four cases involved No 5 bearing distress. No recorded cases of jet cracking or fracture, where the MDS remained located normally, have been found.

#### 1.16.2 Electron microscopy

Visual examination of the No 2 engine oil filter elements and the AGB MDP appeared to show a mixture of fine debris and flake-like particles. Three flakes were removed from the MDP, and another three from one of the filter elements, and examined in a Scanning Electron Microscope. The flakes removed from the filter element were mainly iron but with considerable silver, in the form of a surface layer, and some copper. The flakes removed from the MDP were largely iron with a small amount of silver, predominantly on the edge of one flake, and some other smaller elemental traces. Two of the flakes had a chemical composition consistent with the M50 steel used in the races and ball bearings, and one was consistent with the 4340 steel used for the cage. The cage is plated with silver on a copper base.

The remaining debris, together with the MDP and filter elements, was sent to the engine manufacturer for separate analysis. The report on that work made broadly similar findings, also identifying a flake of low alloy steel, probably not from the bearing. By comparison with typical chips seen as a consequence of bearing spalling, the report concluded that the debris was a result of wear, not spalling. It also concluded that a magnetic chip detector would be able to detect such wear debris successfully, although the amount of warning time it might provide before bearing failure could not be established.

#### 1.16.3 Illumination of flight deck captions

Tests were conducted on the wiring for certain flight deck captions, using a similar helicopter. It was confirmed that the transmission oil pressure caption illuminated when its wiring at the MGB sensor was grounded. Likewise, the Primary and Auxiliary hydraulic system low pressure warnings illuminated when the corresponding wiring was grounded. Any degradation of the insulation of these wires, due to the fire, which allowed them to short to earth, would have illuminated the associated caption.

#### 1.16.4 Oil jet flow testing

The engine manufacturer conducted tests which showed that a broken oil jet tube would allow an excessive flow rate through the oil jet assembly. This had the effect of lowering the oil pressure within the assembly, reducing the pressure at the nozzle supplying the No 5 bearing and also reducing the system oil pressure as indicated on the flight deck.

The engine manufacturer carried out a numerical analysis of the reduction of oil flow to the No 5 bearing with the fractured oil jet assembly. This analysis was backed with further rig tests and a reduction of oil flow to 28% of normal to the No 5 bearing was determined to occur. Using this reduced flow as an input for a mathematical model of the bearing operating conditions, a bearing component temperature rise of 25-30°F was predicted. While this is not a significant temperature rise, the analysis did not take account of any other effects on the oil supply such as aeration or impaired accuracy of the direction of the oil jet.

#### 1.16.5 Oil jet resonance testing

The fatigue fractures of the oil tube assemblies led to much test work to establish the resonant frequencies and vibration modes of these parts. In the first instance, this work was carried out by the engine manufacturer using analytical techniques. The work suggested that a number of resonant peaks occurred, but these were all well above the natural frequency of the MDS of 316 Hz at 100%  $N_f$ .

This work was backed up by resonance testing carried out for the AAIB by AWE, a UK government agency based at Aldermaston. A representative oil tube was mounted on a solid base and tested by exciting the assembly at different points whilst recording the output from up to six miniature accelerometers. It was then re-tested using a single accelerometer. From this work the various mode shapes were established. However, the frequencies of the modes were likely to have been affected by the combined mass of the accelerometers. Therefore, the oil tube assembly was also tested while mounted in a bearing chamber, which was suspended freely. For this, a single accelerometer was mounted on the chamber and the chamber was excited close to the oil tube. This test was repeated without the oil tube assembly installed, for comparative purposes.

The AWE tests gave a good indication of the shapes of the various modes, with indications that relatively high stresses might occur at the brazed joint between the tubes and the oil jet housing where the fractures had actually occurred. The frequencies of these modes were, however, difficult to ascertain. When the oil jet assembly was mounted in the bearing chamber, its behaviour was much modified and, in particular, a strong 430 Hz mode reduced in frequency to around 400 Hz. This raised questions as to how much further the frequency might be reduced if the rear support tube were bolted to the chamber, as is the case when the engine is installed. Testing on the complete airframe was considered but was deemed unlikely to provide worthwhile additional results.

The engine manufacturer also carried out some resonance testing, but without using the bearing chamber. The results tended to confirm the earlier theoretical analysis, but it was difficult to show precise correlation between the GE and AWE tests. During the AWE tests it was found that mounting conditions had a larger than expected effect on the results.

The engine manufacturer also carried out a large amount of additional work to try to determine the reason for the fracture of the oil jets. Ultimately this work was inconclusive but a significant possibility concerned the consequences of higher than normal vibration at three times the shaft fundamental frequency. This frequency, designated 'PT 3/rev' corresponded to about 948 Hz at 100%  $N_f$ . The HUMS data and CVFDR spectral analysis both showed 'signatures' at about this frequency, as measured at the exhaust casing, with the CVFDR showing an initial amplitude increase at about 20 minutes before the engine was shut down, with a further increase seven minutes before shutdown. By revisiting the data for the G-BEID 1988 accident, it was seen that in that case too the PT 3/rev signal had become apparent and increased in amplitude some two minutes before the engine failure. No HUMS data was available for the last one hour 42 minutes on G-BBHM, and the CVFDR spectral analysis did not show a strong PT 3/rev signal until the last few minutes of flight.

Tests on the oil jets using strain gauges showed an unusual response near to this PT 3/rev frequency, which also corresponded to one of the response modes identified in the earlier AWE work. The strain gauged oil jet was mounted to a vibration table and a series of zero to 3,000 Hz frequency sweeps were performed, each sweep at a different 'g' level. These tests showed a high stress mode occurred at between 1,000 and 1,080 Hz. For this mode alone, the mode frequency was seen to significantly reduce as the input 'g', and therefore stress level, was increased. Extrapolation of the data showed the oil tube would reach its average material stress limit of 44,000 lb/in<sup>2</sup> at a frequency of approximately 950 Hz. It is pointed out that the degree of extrapolation is such that the results are tentative, and the vibration levels implied are very high. The results are shown at Appendix K.

#### 1.16.6 IHUMS

An investigation of the operator's IHUMS system and its status and *modus operandi* was carried out.

It was established that after the short penultimate flight, when the crew removed the memory card from the aircraft, the closing files had not been written. When engineering personnel inserted the card into the ground station, a message 'Insufficient data to facilitate download' was displayed. No further action was taken, as this was thought by engineers to be a problem related to short flights. They were unaware, at that time, that the system could be forced to download by returning the card to the aircraft and setting the system up as for a 'Rotors Running Turn Around' (RRTA). This procedure is now employed to secure such data whenever this problem occurs.

If a download had been forced, data from the penultimate flight would have been available for investigation. However, the dataset may have been incomplete. It is necessary for the file names and file sizes to be manually checked to determine if the data is complete and this additional task is now conducted on a per flight basis by the operator.

The IHUMS system software in use by the operator is routinely upgraded, and has been upgraded to version 4.0 since the accident. This version is able to provide improved automatic graphing of engine and gearbox parameters. Because of the short flight times in SAR operations, it was impossible to comply with the requirements for a daily review of the IHUMS data, since a full data set was not always acquired. The operator was granted a dispensation in this respect, in 1999, but for SAR operations only. In addition, CAA AAD 001-05-99 allowed for up to 25 flying hours between data downloads, which effectively superseded the dispensation. Since the accident, the operator has introduced a requirement for 20 minutes of acquisition time on

every flight to allow full compliance with the requirement and routinely interrogates and graphs this data. HUMS has evolved into a valuable tool, providing useful data to both base and line level engineering staff.

## **1.17 Organisational and management information**

Not applicable

## **1.18 Additional information**

### **1.18.1 Previous events**

From archives of earlier AAIB investigations, a summary of about 40 possibly relevant events has been compiled and is shown at Appendix I. Information on most of these events is too limited to form firm conclusions about their relevance, or lack of relevance, to this accident. The majority of these events were to versions of the helicopter in US military service. Most of this information had been received by letter during the earlier investigations and, as actual reports were not usually made available to the AAIB at the time, details are brief. During the course of this investigation, a request was made to the US Marine Corps Naval Safety Center for information on possibly related events. Their database begins in about 1980. No instances were found relating to damage to the EMRSA tube during that period.

Documented cases of in-flight failures of the No 2 engine associated with severe damage to the No 5 bearing and damage to the ERMSA have been largely limited to the UK fleet. Such events occurred to G-AZRF in September 1976, G-BEID in July 1988 and to G-BBHM. One similar case occurred in 1985 to a USAF HH-3E, registration 69-8504; in that case, there was also an uncontained failure of the power turbine.

In-Flight Shut-Downs (IFSDs) of No 1 engine coupled with severe drive train damage have occurred to UK registered S-61s in January 1986 (G-LINK), May 1989 (G-BFFJ) and October 1990 (G-BCLD). One case occurred to a foreign operator in the Far East in February 1989. In these cases damage occurred to the drive train, notably in the area of the gearbox input coupling. In two cases, the EMRSA was damaged, but in no case was it severed and in no case was the No 5 bearing damage more than 'minor'. In two cases at least, records suggest that a loose or missing 'T' bolt may have been the cause.

In the 1990 incident (G-BCLD) both carbon seal oil jets were damaged by fatigue cracking, even though there was little or no damage to the No 5 bearing. In that incident, the oil jet fatigue damage was a consequence of vibration in the drive train due to MGB input pinion damage and not No 5 bearing damage.

In Canada, a number of events involving total power loss to the rotor during logging operations have been experienced, but no similar total power loss has occurred to a UK registered S-61. The UK S-61 fleet has accumulated a total time of about two million engine hours and a mean time between events similar to the accidents to G-BBHM, G-BEID and G-AZRF, of only about 660,000 engine hours. The commercial fleet in the rest of the world, some 100+ aircraft, has accumulated some four and a half million engine hours for no known similar events. S-61s are still operated commercially in the US, Canada, Europe and the Far East. In addition, there is a substantial military fleet of S-61 variants and broadly similar installations are to be found in Boeing-Vertol and Karman built helicopters using T58 engines, with over 22 million engine hours accumulated to date. Available accident data, particularly for the military fleet, is not comprehensive, and reporting deficiencies may play a part in the apparent difference between UK experience and the rest of the world. For example, looking at the civil S-61 fleet, the UK rate would appear to be some 15 times worse than that for the rest of the world

Appendix L is a chart illustrating the fleet-wide experience of similar events, which was prepared by Sikorsky for a presentation to the FAA.

#### 1.18.2 AAIB Safety Recommendations arising from previous events

Many AAIB Safety Recommendations were made in the reports on the above events. Amongst the most significant were the Safety Recommendations arising from the reports on G-BEID (27 Safety Recommendations) and G-BCLD (eight Safety Recommendations). In addition, some relevant Safety Recommendations regarding HUMS were made in a report on an incident to a Super Puma, G-PUMH.

Particular Safety Recommendations arising from the AAIB report 3/90 on G-BEID, and of relevance to this accident, were as follows.

That the CAA should:

*‘Require, for UK registered public transport S-61N helicopters, that measures be taken to ensure that excessive deterioration of the No 5 bearing of the engine shall not result in failure of the engine mounting rear support assembly.’ (Recommendation 4.11)*

The CAA accepted this Recommendation, and in their Follow-up ACTION on Accident Report (FACTAR) F3/90 responded:

*‘Further to discussions between the Authority, and the S-61 aircraft and engine manufacturers, a programme to investigate and develop a means of monitoring and providing an early warning of No 5 bearing deterioration has been initiated. When this programme is concluded to the satisfaction of the Authority, action will be taken to mandate the installation of a No 5 bearing monitoring system on UK registered public transport S-61N helicopters.’*

CAA Status - OPEN

The UK CAA subsequently issued Additional Airworthiness Directive (AAD) 004-10-93. This required implementation of a means of monitoring, on a flight-by-flight basis, the mechanical condition of the system. Vibration monitoring was considered acceptable. Although HUMS was not mandated at the time, all UK operators were using such systems and this was considered an acceptable means of compliance.

*‘Require, for all UK public transport helicopters, the early provision of a facility to continuously monitor the vibration of all high-speed rotating equipment whose integrity is critical to flight safety (made 21 November 1989).’ (Recommendation 4.14)*

The CAA accepted this Recommendation, and in FACTAR F3/90 responded:

*‘Changes to Airworthiness requirements are being prepared for consideration by the Joint Aviation Authorities Engine Study Group requiring the provision of vibration monitoring equipment on turbo-prop and turbo-shaft engines to monitor rotor unbalance. Requirements for the provision of such equipment on Turbo-jet engines are already in place. For existing helicopter types, the Authority is considering requiring the provision of vibration monitoring equipment where this is warranted by service experience. Therefore the Authority accepts this Recommendation for the new helicopter types and is reviewing the situation for helicopters already in production and service.’*

CAA Status - OPEN

CAA AAD 001-05-99 introduced a UK mandatory requirement for Health Monitoring on non-Design Assessed helicopters (generally older types) equipped to carry more than nine passengers. Until the release of this AAD, use

of health monitoring systems, although invariably installed in UK S-61s, had not been mandated.

*Require, for all public transport helicopters, the provision of cockpit indications of engine oil systems reservoir contents and chip-detector warnings. (Recommendation 4.17)*

The CAA responded:

*It is an Airworthiness requirement that a failure analysis of the engine be conducted to identify any potential Major or Hazardous effects of engine failure and to establish that the probability of occurrence of such failures is remote.*

*The engine manufacturer is required to design the system to achieve the required level of integrity. It would therefore be inappropriate and unnecessarily restrictive for the Authority to specify that provision for cockpit indications of engine oil system reservoir contents and chip detector warnings should be made, unless shown to be necessary by failure analysis or service experience. Therefore the Authority only partly accepts this Recommendation.*

CAA Status - CLOSED

As a result of the accident to G-BBHM, it is considered that there remains an unacceptable risk of a hazardous failure mode, involving the loss of location of the power turbine shaft and severance of the EMRSA tube, which can lead to a serious in-flight fire. The intent of this recommendation, as it applies to chip detectors, will be served by the introduction of the Electrical Chip Detectors (ECDs) and associated cockpit warnings, as described in paragraph 1.18.3.

*Require, for UK public transport S-61N helicopters, a review of the standard of engine condition monitoring, and the improvements necessary to achieve an adequate level. (Recommendation 4.18)*

The CAA responded:

*The Authority accepts this Recommendation. Following a review of the S-61 engine condition monitoring requirements and discussion with the aircraft and engine manufacturers, work has been initiated to provide a means of monitoring No 5 bearing deterioration as outlined in the Authority's response to Recommendation 4.11, above.*

*In addition, the Authority identified areas in the engine Maintenance Manual where improvements could be made. The engine manufacturer was informed and the manual has been amended accordingly.*

*Based on the available service experience data, (the CT58 engine has accrued over 2 million commercial operating hours on the S-61) the Authority is satisfied that these changes are sufficient to ensure a satisfactory level of engine condition monitoring.*

*CAA Status - CLOSED*

*Require a review of S-61N engine bay firewall integrity, to ensure that significant gaps in the fireseal arrangement at the points where the engine mounting rear support assembly tubes pass through the canted firewalls are eliminated when the aft cowls are closed. (Recommendation 4.23)*

The CAA responded:

*The Authority accepts this Recommendation.*

*The manufacturer has conducted a thorough review of the design of the canted firewalls and fire seals, and concludes that the design is basically sound. The manufacturer also concludes that the integrity of the assembly is satisfactory if it is installed and maintained in accordance with published procedures. However, the manufacturer has undertaken to publish warnings in the maintenance manual of the need to be vigilant for gaps in the assembly.*

*CAA Status - CLOSED*

*Require measures to improve S-61N engine bay firewall integrity by blanking the inspection hole in each engine mounting rear support assembly tube. (Recommendation 4.24)*

The CAA responded:

*The Authority accepts this Recommendation. A modification to blank off the inspection hole when it is not in use is being developed.*

*CAA Status - OPEN*

The above two recommendations arose from the AAIB concerns about the integrity of the firewalls. The modification to blank the inspection hole was considered unnecessary by the manufacturer and was not subsequently developed. Renewed AAIB concerns about the integrity of the firewall following bearing failure and severance of the EMRSA tube prompted the AAIB to make Safety Recommendation 2002-53 early in this investigation, which is referred to in paragraphs 1.18.3 and 2.2.2, and is given in full in Section Four of this report.

*Review the fire protection provision needs of helicopter main gearbox bays, including the fitment of thermal isolation means, fire detection and extinguishing systems, and flammable fluid shut-off systems. (Made 21 November 1989). (Recommendation 4.25)*

The CAA responded:

*The Authority accepts this Recommendation which is similar to Recommendation 4.2 made following the accident to BV234 G-BWFC in February 1983 (AAIB Aircraft Accident Report 7/84 and CAA FACTAR F4/85 dated 2 April 1985 refer).*

*At that time, attention was focussed on the problems of rotor brakes as the main potential cause for initiating fires in the transmission area, notwithstanding that the source of the BV234 fire was within an engine transmission gearbox. Accordingly, proposals have been made in Draft BCAR Paper 29E-22 for new requirements to address the precautions to be taken in respect of fire hazard when installing rotor brakes. The Authority has in hand a review of the fire precaution standards applicable to zones containing main rotor transmission systems.*

*CAA Status – OPEN*

The Authority's review indicated that thermal isolation systems were undesirable, that a requirement for fire detection and extinguishing systems in a non fire zone could not be justified, and that flammable fluid cut-off systems presented difficulties in helicopters where systems such as lubrication and hydraulic power were essential to continued safe flight.

Particular Safety Recommendations relevant to this accident arising from the G-BCLD report (EW/A343) in AAIB Bulletin 12/91, were that the CAA should:

*Consider the need for measures aimed at providing significantly greater margin between the natural frequency of the S-61N engine*

*mounting rear support assembly tube and the normal rotational frequency of the main gearbox input drive train. (Recommendation 2)*

The CAA responded:

*The Authority accepts this Recommendation. The Authority has requested that the FAA considers the need for embodying modifications made to the US military variant of the S61 which, it is understood, were aimed at providing a greater margin between the natural frequency of the engine mounting rear support assembly tube and the normal rotational frequency of the main gearbox input drive train. The FAA response is awaited'.*

An outcome of this was that the CAA issued AAD 005-10-93, which required the replacement of the engine mount rear support assembly isolator (elastomeric) spacers with new spacers within 2,000 hours and repeat replacement every 2,000 hours to prevent deterioration. The elastomeric mounts had been introduced to modify the response of the EMRSA, and by ensuring their regular replacement, the possibility of unintended reduction of the natural frequency was reduced.

*'Require, for UK registered public transport and aerial work helicopters, the early provision of a facility to continuously monitor the vibration of high-speed rotating equipment whose integrity is, or may foreseeably be, critical to flight safety. (Made 27 November 1990. A similar recommendation was made on 18 June 1991 in relation to the accident to AS355-2 Twin Squirrel G-WMPA near Birmingham on 30 December 1990 (AAIB Bulletin 12/91); on 21 November 1989 in relation to the accident to S-61N G-BEID in the North Sea on 13 July 1988 (AAIB Report 3/90); and on 25 November 1987 in relation to the accident to Bell 222 G-META at Lippitts Hill on 6 May 1987 (AAIB Report 3/88)'. (Recommendation 8)*

The CAA responded:

*'The Authority accepts this Recommendation. As stated in the response to Recommendation 4 the Authority is committed to harmonisation with its JAA partners and that a common perception of the role for technologies such as HUMS applied to all public transport helicopters does not exist. However in consideration of the large majority of UK Group A operations being over hostile terrain or city centre areas it is expected that the JAA partners will support the need for additional safeguards for this type of operation.*

*To this end, as the result of response to a widely circulated discussion paper on 'The Airworthiness of Group A Helicopters' the Authority for JAA consideration proposals on the retro application of revised safety objectives and consequent design assessment requirements, in particular suggesting that those helicopters operating over hostile terrain and city centres need to be addressed.*

*It is anticipated that the health monitoring package stemming from the above requirements would include the monitoring of vibration of high speed rotating components'.*

In addition, the CAA introduced AAD 004-10-93, which required the use of a health monitoring system to monitor, on a flight by flight basis the vibratory health of main rotor gearbox high speed input drive shafts and associated parts, including engine power turbine No 5 bearings.

In 1999, the CAA issued AAD 004-05-99, which made the installation and use of HUMS mandatory for UK registered helicopters in the Transport Category (Passengers). This applied to helicopters with more than nine passengers, unless subject to a 'Design Assessment' as part of the certification process (BCAR 29, JAR 29).

Following the accident to G-BBHM, AAD 002-12-2002 was introduced which required the introduction of electrical chip detectors to replace the magnetic drain plugs at the power turbine accessory drive, together with daily continuity checks of the electrical chip detectors.

Particular Safety Recommendations relevant to HUMS arising from the G-PUMH report 2/98 were:

*The CAA should review, with associated helicopter operators and manufacturers, the function and trigger thresholds of the ground based IHUMS software with the aim of introducing procedures which will be able, routinely and without substantial operator intervention, to highlight adverse trends. (Recommendation 98-08)*

#### CAA Response

*The Authority accepts this Recommendation. The Authority will review with operators and manufacturers the outputs of the ground based HUMS software, and will consider the introduction of procedures that will facilitate the identification of adverse trends. It is intended to complete this review by 31 December 1999.*

*CAA Status- Open*

As one consequence of the subsequent review, CAA issued AAD 001-05-99, which mandated HUMS for non-Design Assessed helicopters equipped to carry more than nine passengers. This AAD required the implementation of 'acceptable procedures' covering all aspects of data collection and analysis. The latest iteration of IHUMS software is more user-friendly, and makes graphing and use of data considerably easier.

*'The CAA should consider means by which ready access could be provided to fleet wide trend data which would identify abnormal trends on a particular aircraft against an operator's whole fleet. (Recommendation 98-09)*

#### CAA Response

*The Authority accepts this Recommendation. The Authority will review means by which ready access could be provided to fleetwide trend data which would identify abnormal trends on a particular aircraft against an operator's whole fleet. It is intended that this review will be completed by 31 December 1999.*

*CAA Status – Open'*

The same CAA review, with operators and manufacturers, did not identify clear advantages of such a capability. AAD 001-05-99, which came out of the review, incorporated requirements which were considered to meet the intent of the Safety Recommendation. To some extent, the comparison with fleet wide data is achieved by the use of thresholds that are themselves based upon whole fleet data. There are some difficulties with this approach, for example, because parameter stability is probably more important than its actual level, even if it is close to the prescribed threshold. There are also existing means of comparing data across the fleet at specialist, if not line, level.

*'In order to utilise the data available from IHUMS systems on board Public Transport helicopters to maximum effect to avoid serious accidents, the CAA should develop the concept of providing flight deck display of IHUMS exceedance information, including vibration, to flight crew as previously proposed in CAA HARP (CAP491) of June 1984.'* (Recommendation 98-11)

#### CAA Response

*'The Authority accepts this Recommendation. At the Authority's request, the Helicopter Health Monitoring Advisory Group has formed a Working Group to assist with developing the concept of a flight deck*

*display of Integrated Health and Usage Monitoring Systems (IHUMS) exceedance information, including vibration. It is intended that this Group will present its findings by 31 December 1998.'*

*CAA Status – Open*

The current CAA view on this is that every flight deck warning must have a suitable corresponding crew action, and that it is very difficult to proceduralise such actions unless the integrity of the warning is sufficiently high. For those events where the consequences of the necessary crew action can itself be hazardous, eg, ditching, then the integrity of the warning must be correspondingly high. At the present state of the art, the CAA advises that it is not possible to be sufficiently confident of the HUMS systems warnings to permit flight deck displays.

### 1.18.3 Safety actions

In August 2002, the AAIB issued Special Bulletin S2/2002 which detailed the circumstances of the accident and the initial findings. This is reproduced at Appendix M.

On 21 November 2002, the AAIB issued Safety Recommendations 2002-51, 2002-52 and 2002-53 by means of letters to the CAA and FAA. These Safety Recommendations were aimed at interim and long-term protection of the fleet. The Safety Recommendations are attached at Appendix N. In a response dated 12 February 2003, the CAA advised that it accepted Safety Recommendations 2002-51 and 2002-52, but did not accept 2002-53. The CAA response is attached at Appendix O. In a Memorandum dated 9 April 2003 the FAA responded that they concurred with the first recommendation but did not concur with the other two. Their full response is at Appendix O.

The operator, in consultation with the CAA and the airframe and engine manufacturers, initiated a programme to install Electrical Chip Detectors on a trial basis. CAA AAD 002-12-2002 made this a UK requirement. These ECDs were to be checked for electrical continuity (debris) after each flight, and included no cockpit warning. These trials were subsequently extended across the operator's S-61 fleet with a view to eventually providing cockpit warnings. The operator also evaluated other systems for monitoring mechanical deterioration of the system.

On 30 April 2003, the aircraft manufacturer issued a letter to all operators advising of a forthcoming Alert Service Bulletin (ASB) which would introduce a plug type chip detector, with a flight deck warning, to monitor the engine oil for metallic chips generated by a degraded bearing. This would be wired to the

master caution panel and a separate caution advisory within the cockpit. The ASB was expected to be mandated by FAA Airworthiness Directive (AD) and compliance was expected to be required by 31 December 2003. On 9 June 2003, All Operators Letter CCS-61-AOL-03-005 informed operators of the release of ASB61B30-15 and that all parts were available. The compliance date was 31 December 2003.

Immediately following the accident the operator initiated a one-time check for oil leaks at the EMRSA and, at the time of writing, this check is part of the operator's daily inspection.

On 17 July 2002, the operator published an Alert Message AM/S61/02/004. This required an immediate fleet wide inspection of the engine and MGB drive shaft area for engine oil leaks, and a check of the IHUMS trends for specific parameters related to the MDSs.

On 24 July 2002 the operator published an Alert Message AM/S61/02/005 which required daily inspections of the MDSs and MGB input area for damage, security and oil leaks or loss of grease. It also required manual checks of IHUMS trends for a number of engine bearing related parameters, and introduced a requirement to confirm the file sizes of certain engine and gearbox parameters, to ensure that full data download had taken place. It further required that the flight crew operate the aircraft so as to ensure that at least one full set of IHUMS engine and gearbox data is automatically collected per day. This data takes about 20 minutes to acquire and, during this period, the aircraft must be flown at more than 90 kt with a bank angle of less than 7 degrees, a rate of descent or climb of less than 1,000 fpm and a total torque of between 96 and 160%.

Following the discovery of the high torque starts during rotor engagement, the operator introduced a start torque limit. This was formalised in a Flying Staff Instruction, in part B 2.3 of the Operations Manual. It stated...

*'S61 Torque during rotor engagement. Pilots are reminded that single engine torque MUST NOT exceed 60% during rotor engagement. 40% should be considered the optimum for rotor engagement. 40-60% may be used for engagement during gusting conditions'.*

#### 1.18.4 Comparison with Sea King/Gnome experience

The Sikorsky S-61N and its General Electric CT58 engines are very similar to the licence-built Westland Sea King with its Rolls-Royce Gnome H1400 engines. Detail differences in the designs, which might be relevant to this accident, are as follows:

On the GE CT58, the No 5 bearing has evolved to have a split inner race and a steel cage. This requires changes to the inner race geometry and cage inner diameter, and theoretically requires the oil jet alignment to be different. The life of the No 5 bearing on the Rolls-Royce Gnome is limited to 2,000 to 2,500 hours to coincide with the engine overhaul life. The CT58 bearing has a 6,000 hour life.

The Westland Sea King has a steel mass damper weight mounted on the forward end of the EMRSA. The Sikorsky design originally used a similar mass damper, but later introduced a revised mounting at the MGB, including an elastomeric gimbal ring (the Sea King gimbal ring is steel with elastomeric bushings), and dispensed with the mass damper. The wall thicknesses of the EMRSA tubes are similar for both designs. However, limits on the tube wall thickness and design variants mean that the Sikorsky tube could be marginally thicker. These detail changes mean that the vibration ‘signatures’ of the two designs could be significantly different. In addition, the MGB input pinions use roller bearings on the Sea King, rather than the plain journal bearings of the S-61N MGB. The oil jet assemblies appear to be very similar, if not the same, on both engine types.

The Westland Sea King fleet has accrued over 1.6 million flying hours (3.2 million engine hours) over some 30 years, all with Gnome H1400 engines. During that period, there have been two No 5 bearing failures arising from MDS problems; in one case rag was left around the shaft and in the other, incorrect and unbalanced bolts were fitted to the shaft. In addition, the Gnome installation on the Wessex has had a number of No 5 bearing failures attributed to problems with the couplings, but where the bearing was not the prime cause.

The Sea King Flight Reference Card (FRC) drill for rotor starting states:

*‘No 2 speed select lever ... Set 98%  $N_r$  using 40 to 60% torque during engagement.’*

#### 1.18.5 Shaft misalignment

The couplings in the Main Drive Shaft allow for any probable misalignment of the shafts. Such misalignment can arise from (a) the normal limits on the airframe build and installation of the engines and MGB and, (b) movement of the engines and gearbox in operation. The MGB will typically rotate slightly against the torque of the rotor head in operation, and may take up a permanent ‘set’ in the airframe due to this factor. The engines themselves may move due to the torque they generate. Therefore some level of misalignment will normally exist, and it is likely that this will increase at high power and torque, reducing as the torque reduces, but is unlikely to become zero. While the

couplings are able to accommodate typical alignment variations, some resultant forces and moments, normally insignificant, will exist.

#### 1.18.6 Dynamic behaviour of the Main Drive Shaft

The AAIB commissioned a dynamic analysis of the power train between the free turbine and the MGB input pinion. The analysis was based on a Finite Element (FE) model and made a number of assumptions and approximations based on measurements of actual parts. Without comprehensive supporting design data, the results can be regarded as an indication only of the shaft behaviour.

The analysis indicated that the system might be susceptible to damage if high torque is experienced below the normal operating speed of 19,200 RPM. The FE model (Appendix P, page 1) showed that significant elastic distortion occurred in the Thomas coupling and the associated flanges at high torque values. A typical case showing these deflections (greatly exaggerated for clarity) is given at Appendix P, page 2.

The natural frequency of a perfectly balanced shaft, at which it will 'whirl', varies with its rotational speed. A Campbell diagram, which shows the relationship between the balanced whirl mode natural frequency and the shaft rotational frequency, which provides the dominant excitation force, is given at Appendix P page 3. This shows that the natural frequency of the shaft and its rotational frequency coincide at speeds below the normal operating speed.

In itself, this will not cause the shaft to 'whirl' but, should there also be an imbalance, significant whirl is likely. The application of high torque will generate elastic deformation in the coupling. A very small out of balance component of 0.04 ounce-inches, equivalent to about 0.001 inches of eccentricity, can in some cases in the analysis generate damaging responses in the system, involving high loads for the No 4 and No 5 bearings. The analysis indicated that the whirl mode introduced large deflections in the area of the Thomas coupling. A typical set of response curves is shown in Appendix P, page 4. If the bearing stiffness value is reduced, the whirl modes occur at lower shaft speeds.

It was considered that these conditions could cause serious consequent residual imbalance in the shaft assembly.

During the later stages of the investigation, however, the manufacturer advised that in 1964 they had conducted a two year programme of evaluations for the US Government, including engine operations in the range of 90 – 110%  $N_f$ , during which no MDS whirl mode was identified. They suggested as an

alternative mechanism that if high radial loads were applied to the No 5 bearing during a 'high torque' rotor engagement, as suggested by the AAIB analysis, could lead to a failure of the bearing cage. The effect of such a cage failure would be to allow the balls to skid, become asymmetrically disposed around the bearing, and consequently generate unusual bearing loads, shaft displacement and vibration.

In addition, the stiffness of the elastomeric elements (isolators) of the gimbal ring is an important factor determining the natural frequency of the EMRSA assembly itself. Deterioration or contamination of the isolators could result in a large change of natural frequency, possibly causing it to move towards the fundamental frequency of the shaft. The elements had been installed new at the MGB change less than 100 hours previously, and it is unlikely that they had deteriorated significantly in that time.

## 2 Analysis

The accident resulted in G-BBHM being destroyed by fire shortly after the crew made a successful forced landing. Nevertheless, with the successful landing on land, the investigation was able to use evidence from the wreckage and the crew to identify the most likely source of the problem. The fact that the crew were able to successfully land G-BBHM within 82 seconds of the first indication of a serious emergency, reflects extremely well on their training and professional ability. While the commander was in overall command of the flight, he was well supported by his very experienced co-pilot (chief pilot) and by the two rear crew members.

The investigation determined that the No 5 bearing, which located the free turbine in the No 2 engine, had severely deteriorated during the flight. This allowed the shaft and its Thomas coupling to orbit, cutting through the EMRSA tube and releasing engine oil which then ignited. This was a similar failure mode to that seen in several previous accidents, the most recent being in 1988, and the actions taken after each of those events had been inadequate to prevent a recurrence.

### 2.1 Operations analysis

#### 2.1.1 Conduct of the flight

The flight was normal up to the point at which the rear crew became aware of an unusual noise. Almost immediately, the No 2 engine fire warning illuminated. The CVFDR confirms the recollection of the crew in that the subsequent actions were in accordance with their training, with each crew member making a valuable contribution to the successful landing.

Almost immediately after the unusual noise noted by the two rear crew members, the 'NO 2 ENG FIRE WARN' light illuminated accompanied by the audio alert. Using the rear view mirror (standard installation following recommendation made after the accident to G-BEID in July 1988), the commander was able to see smoke coming out from the area of the No 2 exhaust. While the co-pilot was carrying out the 'Fire Drills', the commander initiated a climb and turn towards Bournemouth Airport, some 7 nm away. Analysis of the CVFDR indicates that the 'NO 1 FIRE WARN' light and audio came on shortly after the activation of the 'NO 2 FIRE WARN' alert. Of note is that, although the audio warning for the 'NO 1 FIRE WARN' activated regularly during the rest of the flight, the light was not on continuously. The crew were not aware of any warnings of fire relating to the No 1 engine until the co-pilot saw the warning light as the helicopter was on final approach to the selected landing site. As discussed in the engineering analysis, this illumination

of the 'NO 1 FIRE WARN' light was probably caused by fire damage to electrical wiring. At this stage, all crew members were taking appropriate action in response to the No 2 engine bay fire warning and it is understandable that none were immediately aware of this other apparent engine bay fire. If they had been aware, the most sensible reaction might have been to ignore the warning and concentrate on the impending landing. It may be relevant that both warning lights are close together on the same panel, which is located on the right upper side of the main instrument panel. Although the light was not always illuminated, its location could cause confusion as to which engine is on fire. Modern design requirements recognise the importance of positioning warnings appropriate to the system failure. Although the S-61 is an old design, it would be appropriate for consideration to be given to improving the presentation of the FIRE warning indications so as to minimise the possibility of confusion in the event of an engine bay fire.

Accordingly, it is recommended that:

The aircraft manufacturer, Sikorsky, should re-locate the No 1 and No 2 engine bay fire warning lights on the main instrument panel of the S-61N helicopter, with the intention of ensuring as far as possible that unambiguous information is presented to both flight crew members in the event of an engine bay fire. (Safety Recommendation 2003-83)

The priority for the co-pilot was to secure the engine, activate the appropriate fire extinguisher and monitor the fire warning light for the No 2 engine. The CVFDR confirms that this action was completed with the commander verbally confirming each necessary step. In the cabin, the open door was closed, the two crew members strapped into suitable seats and the winchman informed the coastguard of the emergency. Thereafter, the winchman transmitted the commander's intentions during the developing situation and both rear crew members prepared for the forthcoming landing. The commander's initial intention was to head for Bournemouth Airport but, with indications of further serious system failures, he made a positive decision to land as soon as possible. While these warnings were probably false and due to fire damage to wiring, this decision to land as soon as possible was the most sensible option. The extent of the developing fire indicated that controlled flight could only have continued for a very short time. The decision of the commander to land as soon as possible was dependent on the selection of a suitable landing site. His decision was simplified because the co-pilot had identified a site during the developing emergency and alerted the commander. Thereafter, the commander controlled the helicopter to a successful landing. This was accomplished with continuing indications of a serious fire but with each crew member operating in a calm and highly effective manner.

After landing and the helicopter coming to rest, the crew continued to operate effectively as a crew with each aware of his responsibilities. Evacuation was completed promptly and the crew remained well clear of, and kept helpers away from, the burning helicopter.

## **2.2 Engineering analysis**

### **2.2.1 Introduction**

From Section one of this report it was established that that the following important abnormalities had occurred before landing:

- (i) The EMRSA tube of the No 2 engine had been severed adjacent to the Thomas coupling;
- (ii) The No 5 bearing of the No 2 engine had worn excessively and suffered cage failure, allowing the power turbine shaft to lose location and to 'orbit';
- (iii) The 'live' and 'dead' oil jet assemblies of the No 2 engine had both experienced fatigue fractures of the rear carbon seal oil tube;
- (iv) There had been recorded indications of anomalies in  $N_f$  of the No 2 engine over the preceding two hours of flight, but this was not available to the flight crew;
- (v) There were several flight deck warnings and cautions during the last 82 seconds of flight;
- (vi) There had been at least two No 2 engine starts associated with higher than normal torques at rotor engagement on the previous day.

Also, there was damage to a fire bottle found after the ground fire.

### **2.2.2 Analysis of anomalies**

- (i) EMRSA tube severance

The tube had been severed adjacent to the Thomas coupling, and considerable local distortion at the cut was evident. In addition, the Thomas coupling showed evidence of rubbing contact around the inner wall of the tube. In order to make contact it was necessary for the normal running clearance between the coupling and the tube, approximately ¼ inch, to be lost. This required that either the No 4 and/or No 5 bearing ceased to locate the power turbine correctly and, since the

No 4 bearing was in reasonable condition, it is evident that the severance of the EMRSA tube was a consequence of the degradation of the No 5 bearing.

In addition to contact between the Thomas coupling and the EMRSA tube, degradation of the No 5 bearing would have resulted in severe damage to the carbon oil seal at the rear of the turbine assembly. This in turn would have released engine oil into the EMRSA tube in the area where rubbing contact was to take place. The possibility that deterioration of the carbon oil seal could have been the prime cause of the failure was dismissed because, in that case, the daily inspections would have detected an oil leak well before the bearing failed. In addition, a leak large enough to affect the bearing would probably have resulted in more general distress to the turbine itself. Also supporting this view was the lack of over-temperature damage on much of the bearing cage, indicating that it had broken up early in the failure sequence.

It is considered that the high speed rotating contact of the Thomas coupling with the EMRSA tube, and the elevated temperature of the exhaust duct, could both have provided localised ignition sources for the released oil to be ignited. A consequence of this conclusion is that the fire occurred in an area which, technically, is not within the engine fire zone but is within the MGB bay, which is not a fire zone. This allowed the fire to propagate essentially unchecked and to adversely affect other important systems.

The possibility that elevated temperatures within the No 5 bearing could have provided an ignition source has not been entirely dismissed. It seems less likely, however, given that prior to cage failure, the bearing temperature was not excessively high, and that a large heat sink effect existed at the bearing due to its assembly within the engine itself. These conclusions regarding the basic failure mechanism were, for the most part, reached early in the investigation. In view of the intense nature of this fire, and its rapid development to catastrophic proportions, the AAIB issued the following three Safety Recommendations, published in a Safety Recommendations letter:

‘The US Federal Aviation Administration, in conjunction with UK CAA and the airframe and engine manufacturers, implement a means of providing a suitable warning to aircrew and/or engineering staff, of any impending loss of integrity of the drive shaft system of the S-61N helicopter which could lead to failure of the engine rear support mounting tube.’ (Made 21 November 2002) (Safety Recommendation 2002-51)

‘The US Federal Aviation Administration, in conjunction with UK CAA and the airframe manufacturer, ensure that the integrity of the engine fire zones on the S-61N helicopter is not breached by a failure of the engine rear support mounting tube.’ (Made 21 November 2002) (Safety Recommendation 2002-52)

‘The US Federal Aviation Administration, in conjunction with UK CAA and the airframe manufacturer, devise a means of protecting essential systems in the main rotor gearbox bay of the S-61N helicopter from the effects of fire.’ (Made 21 November 2002) (Safety Recommendation 2002-53)

The Safety Recommendations document is reproduced at Appendix N and CAA and FAA responses given at Appendix O.

(On 9 June 2003, in response to these recommendations, the airframe manufacturer issued an Alert Service Bulletin (ASB 61B30-15) which required the installation of the engine chip detector system, with associated indication in the cockpit, as previously described in paragraph 1.18.3.)

(ii) No 5 bearing failure

Six hours before the accident, when the MDPs were last checked, there was no debris seen which could have indicated an impending bearing failure. All the deterioration took place in the subsequent period.

Common reasons for bearing failure include material or manufacturing defects, improper installation, foreign object ingress, inadequate lubrication and unusual or excessive loading. Material or manufacturing defects and improper installation would have resulted in bearing failure, or evidence of impending failure well before the 1,563 hours of operation which this bearing had accumulated. The metallurgical examinations also discounted manufacturing or material defects, and the strip examination of the engine did not reveal any installation errors.

Foreign object ingress can result in bearing spalling, a fatigue mechanism in which chips are released from the balls or races. Although analyses were made of the debris released from the bearing and recovered by the MDP or trapped by the filter, no spalling chips were identified, rather the debris was a result of heavy wear. While foreign object ingress could not be entirely discounted, there was no evidence to support this possibility.

Loss of lubrication was considered by one of the metallurgists to be the initial cause of the bearing failure. This view was supported by the generally smooth

appearance of all except one of the balls, and the general condition of the bearing was similar to the No 5 bearing in the case of G-AZRF (see Appendix I). This was attributed at the time to a loss of lubrication. However, this view was not supported by the generally good condition of the forward half of the bearing cage, which was in one piece with most of the silver plate intact and untarnished. This indicated that the cage had fractured before high overall bearing temperatures had occurred. The cage showed evidence of radial motion of the balls, indicating that high levels of shaft vibration had existed immediately before cage failure, and it was likely that this vibration caused the fatigue fracture of the oil jets and a general deterioration of the bearing. The history of the S-61 shows that problems in this area have arisen mainly from MDS imbalance, and the engine manufacturer provided an analysis of the bearing condition which strongly suggested that the initial cause of the failure was shaft vibration. (Appendix J). Such a possibility goes some way towards reconciling the views of the various specialists, and would also explain the short period (less than 6 hours) between the last MDP check and the bearing failure.

(iii) Oil jet failure

Since there is no support or evidence of pre-fracture mechanical contact with the oil jet tubes aft of the plane of fracture, vibration or resonance is the only plausible cause of the fatigue in these tubes. Any such fatigue mechanism would be expected to result in fatigue origins on opposite sides of the tube. However, if the vibration were severe and the fracture fast enough, the fatigue damage might not be evidenced by cracking on both sides. This view is supported by the three other cases of fracture of the oil jets, all of which were accompanied by shaft vibrations, while no cases are known of similar oil jet fractures in isolation. It proved difficult to validate this theory by testing, as neither the resonance testing nor the stress measurements were fully conclusive. This testing merely provided an indication of possibilities.

Corroborating evidence for the active oil jet failing during the accident flight (after the start of main drive shaft jitter increase) may be taken from the analysis of low engine oil pressure switch activations against gas generator speed. The activation during the No 2 engine in-flight shutdown occurred at a significantly higher  $N_g$  (and hence oil pump speed) than other shutdowns. Although based on a statistically limited sample, it is considered possible that, with an increase in the oil flow rate due to a fractured jet, the oil pressure may have decayed more rapidly than if the constriction of the jet nozzle had been in place. There were also marked similarities between the four cases (including this one) of oil jet failures, ie, similar locations of the fatigue origins at generally the top and/or bottom of the tubes, no distinction between the 'live' and 'dead' tubes, and both tubes suffering fatigue failures in each case. It seems probable, therefore, that the oil jet fractures were consequent upon the shaft vibration, and were not the

primary cause of the No 5 bearing deterioration. However, as noted above, the fracture of the oil jet tube supplying oil to the carbon seal resulted in a reduction of the oil supply, both volume and pressure, to the No 5 bearing, and misdirection of that supply. This would particularly have affected the inner race, at which the oil jet is directed. In such a case, the reduction of oil supply of some 72% to the bearing, together with the likely misdirection, could have resulted in a rapid acceleration of the bearing deterioration, and this would have precluded the opportunity for engineering personnel to detect any damage before final failure. Therefore, it is recommended that

The FAA, CAA, and engine manufacturer should introduce a modification to the oil jet assembly that, in the event of fracture of the tube which supplies oil to the carbon seal, would prevent a large reduction in supply pressure to the nozzle which supplies oil to the No 5 bearing. (Safety Recommendation 2003-84)

(iv) FDR anomalies

Short term variability (jitter) in the No 2 engine recorded  $N_f$  value

The only evidence recorded on the CVFDR that gave any indication of deterioration in the health of the power train, was the jitter present on the No 2 engine  $N_f$  values. A mechanical assessment of the main drive shaft takeoff to the transducer was made to establish whether radial or axial free movement could give rise to the jitter. It was considered unlikely that there had been much axial free movement due to the drag of the free turbine in the gas path forcing the shaft aft against the bearing faces. Conversely, with a mean reduction in No 5 bearing ball diameter of 0.0565 inches and similar wear on the inner race, a radial free play in the order of 0.10 inches may have existed at some time before engine shutdown. This represented a possible radial motion at the worm gear contact faces of approximately 0.023 inches. At 100%  $N_f$ , and in consideration of the worm and gear wheel geometry, this corresponds to a change in angular velocity of the 'radial' shaft to which the transducer is ultimately coupled of approximately 0.36%.

Typical mean deviations calculated for the  $N_f$  jitter and shown in Appendix D, are in the order of 0.5% towards the end of the flight. It should be noted that this is an overall figure for the measurement uncertainty of the recording system, and will include transducer errors in addition to the variation induced by the radial play at the worm gear contact faces. If the latent deviation (before the noticeable upwards trend) is subtracted from the total value, it can be seen that the residual is entirely consistent with the effect that radial play would have had on the measurements. Based on this relatively coarse measurement system it is possible to observe a noticeable change in main drive shaft behaviour at least

one hour before the failure. While this may prove to be too inconsistent a parameter for this purpose, it may be possible with appropriate signal conditioning, to use it as an input to a health monitoring system. This could then provide early warning and a trend of deterioration of the free turbine shaft location. It could detect such loss of integrity arising from a number of different possible causes, including impending failure of either shaft bearing. Therefore, the following Safety Recommendation is made:

The CAA, together with the FAA, airframe and engine manufacturers, should consider the possible value of measuring short term variability in the recorded  $N_f$  speed on S-61 helicopter engines, in order to provide early warning of loss of integrity of the drive shaft system, which could lead to failure of the engine mounting rear support assembly tube and subsequent fire. (Safety Recommendation 2003-85)

#### *Torque Variations*

The geometry of the gearing of the engine torque measurement system was such that it would not have been affected by the same MDS radial play that was likely to have induced errors in the measurement of No 2 engine  $N_f$ . It has not been possible to propose any alternative theory for the variations recorded.

#### *No 1 engine fire warning*

Once the audio and data recordings had been time correlated, the continued aural announcements of the No 1 engine fire warning, as opposed to the temporary interruption of the discrete warning for the same parameter in the data, could be assessed. The discrete was sampled and recorded at an interval of one second. This meant that the presence of the actual fire warning may have been interrupted for a period of between 15 and 17 seconds. The way that the aural warning system interpreted and announced warnings differed depending upon the number of concurrent warnings and their pre-determined priority.

In this particular instance, as the two engine fire warnings were of the same priority, their respective announcements were sequenced so as not to interrupt each other. This sequencing, together with the associated delays introduced to time-separate concurrent messages, allowed repeated announcements of both fire warnings whilst in reality one of the warnings had deactivated temporarily. Had this temporary deactivation of the No 1 engine fire warning lasted longer than 17 seconds, this would have been reflected in the aural announcements made and recorded on the CVFDR.

(v) Flight deck warnings and cautions

During the emergency, the crew received warnings and cautions for the transmission oil pressure, the primary servo pressure and the auxiliary servo pressure. The activation of the 'flotation fired' discrete parameter was also present in the recorded data. The tests detailed in paragraph 1.16.3 showed that these captions could have illuminated either because of loss of pressure in the respective systems, resulting in a genuine warning or if fire damage to electrical wiring had allowed earthing to take place. The control 'twitches' experienced during landing could have been due either to switching between the two hydraulic systems for flight controls, or due to fire damage to the systems themselves. The remoteness of the wiring of the emergency flotation bags system from the MGB and engines area made it unlikely that the temporary activation of the flotation fired discrete was due to fire damage. However, it has not been possible to propose a reasonable, alternative theory.

As the MGB oil is circulated by external non-metallic flexible pipes, and is piped from the MGB to input No 1, then to input No 2 and then to the pressure transducer, the loss of oil pressure indication could have been due to either loss of oil pressure within the gearbox, fire damage to non-metallic oil pipes or grounding of the pressure transducer wiring.

(vi) High torque rotor engagements

The engine manufacturer's analysis of the data for the start in which the torque reached 115.2%, indicated that it was a non-typical rotor engagement in which  $N_g$  had possibly hit 'topping', a mechanical fuel flow limit prior to the  $N_f$  speed governor cutting back the fuel flow. The review of the engine parameters did not reveal any obvious anomalies when comparing these with each other, suggesting that the engine and transmission had behaved as expected, given the values of the engine parameters of fuel flow,  $N_g$  and  $N_f$  speeds, T5 and the conditions of the day.

2.2.2.1 Fire bottle damage

While carrying out the fire drill on No 2 engine, the crew had fired the main cartridge in the No 2 bottle, to release the extinguishant into the No 2 engine. Consequently each bottle would have had at least one live cartridge remaining after landing, and the subsequent ground fire would have ignited these cartridges. The Maintenance Manual warning concerning firing a cartridge in an already discharged container indicates the most likely explanation of the damage found to one of the containers, rather than any in-flight failure. Such damage to the fire bottle in-flight would have resulted in the extinguishant being released into the MGB bay rather than into the engine bay.

### 2.2.3 Scenarios considered

During the investigation there were two primary scenarios postulated. The damage to the No 2 engine drive train indicated that the initial deterioration of components had occurred within the turbine stage of the No 2 engine. In one scenario the failure of the No 5 bearing was postulated as being the earlier event, whilst in the alternative scenario, complete bearing failure would have occurred after the fatigue fracture of the live oil jet. The evidence supporting each of these possibilities was complicated by the fact that in either scenario, the second event introduced its own 'signature' into the body of evidence. If the bearing was the earlier failure, then the subsequent reduction of lubrication generated its own typical wear characteristics but, if the jet had failed first, the subsequent bearing deterioration itself introduced shaft vibration capable of failing the bearing. Therefore, the difficulty for the metallurgists was to look beyond the final condition of the heavily worn parts in order to determine the initial failure mode, for which the evidence was obscured.

The evidence from the FDR oil pressure switch activation showed that the oil jet was not fractured at engine start, but was fractured by the time the crew shut down the engine during the emergency. When the jet fractured it seems most probable, from the tests and dialogue with the engine manufacturer, that lubrication of the No 5 bearing would be impaired sufficiently to rapidly accelerate the bearing deterioration. This view is supported by the circumstances of the accident to G-AZRF, where the oil tube supplying the No 5 bearing was considered to have fractured shortly after lift off, and the engine subsequently failed during the flight. Thus it was considered that the majority of the evidence supported deterioration of the No 5 bearing as the earlier cause.

Several possible reasons for bearing failure have been considered earlier and dismissed, including assembly errors and material defects. It is possible that the ingress of foreign matter initiated bearing failure but no evidence for this, such as spalling 'flakes', was found in the material analyses, although there may have been some early spalling for which the evidence has been lost. The generally spherical condition of ten of the eleven balls argues against spalling, as does the nature of the majority of the recovered debris, which was associated with heavy rub or wear.

The most likely remaining reason for bearing failure was adverse loading, of which cyclic or vibration loading was a likely case. Again, lack of spalling evidence could be considered to argue against this. The forward half of the cage, however, showed clear evidence that ball radial motion occurred before an elevated temperature was reached, suggesting that vibration of the bearing preceded the significant reduction of lubrication.

The measured and recorded data suggests that, up to about two hours before the accident, vibration levels were normal. However, during that two hour period the FDR data, and later the CVR audio spectrum, showed evidence of a developing problem. Although HUMS data for the last one hour 42 minutes was not downloaded, up to the point that the data ended, some increasing variability in the data was evident. It is considered, in the light of the overall body of evidence, that the failure of the bearing came about because of an increase in shaft vibration during that two-hour period. It was also considered that the jet failure occurred as a result of a resonance mechanism generated by severe vibration of either the bearing or the MDS. This view, however, raises some questions which have not been fully answered, despite the considerable effort expended to understand the dynamic behaviour of the oil jet assembly.

The cause of the onset of shaft vibration in the last two hours of operation of the No 2 engine, was not positively identified. The only possibly relevant factor which the investigation revealed was evidence of unusually high rotor engagement torques during two of the engine starts the day before the accident, both of which involved the No 2 engine. These two events occurred within the time-scale of the recorded deterioration of the system, and well after the last MDP check when no bearing debris was found. The airframe manufacturer has suggested that the torques achieved would not result in deterioration or imbalance of the MDS. However, there were several factors, including alignment errors, gyroscopic and dynamic effects, which would not normally cause shaft damage/balance problems in flight, when the system is running at rated speed, but might precipitate such problems at the higher torque values when running at lower speeds. The dynamic analysis commissioned by the AAIB indicated that this is a possible explanation of the subsequent bearing failure. The high No. 5 bearing loads predicted by the dynamic analysis could, alternatively, have lead to cage failure which, in turn, would have given rise to vibration. In either scenario, there is a prudent reason to limit rotor engagement torque after engine start.

Accordingly, the following Safety Recommendations are made.

The FAA and CAA should require Flight Manuals for all variants of the S-61 and similar types to include an appropriate engine torque limitation during rotor engagement. (Safety Recommendation 2003-86)

(Sikorsky have stated that for consistency and thoroughness, they will add a recommended limit to their civil Flight Manuals.)

The FAA and CAA, together with the airframe and engine manufacturers, should investigate the dynamic behaviour of the S-61 MDS and associated high speed rotating components in support of the introduction of an appropriate torque limitation during rotor engagement. (Safety Recommendation 2003-87)

#### 2.2.4 Fire

It has been shown above that the fire propagated inside the EMRSA tube at the Thomas coupling, and was being fuelled by engine oil released by the damaged carbon seal at the rear of the bearing housing. This area normally operates at somewhat elevated temperatures, being surrounded by the exhaust duct itself. Together with the localised heating from the friction between the coupling and the tube, and the breach of the ERMSA tube, mechanisms were provided for vaporising as well as igniting the oil. Once the tube was severed, it was possible for air from the engine compartment to enter the tube and provide a front to rear ventilation path, helping to sustain combustion. The space inside the tube was essentially outside the engine fire zone. The combustion within the tube would have been directed towards its open aft end, where it interfaces with the magnesium alloy gearbox. In addition, fuel and oil pipes in the immediate area, which are constructed from flexible non-metallic materials, when impinged upon by the fire, would have released additional combustible fluids into the gearbox bay. This would have continued to fuel the fire after the shut down of the No 2 engine. The rapid development and intensity of this fire was indicated by the fire warnings for both engines; since the fire detection wires are exclusively within the engine fire zones and there was no fire directly in these zones, it follows that the warnings were triggered by heat transfer into the engine fire zones from the fire within the MGB bay.

In view of the intense nature and speed of propagation of this fire, the AAIB made the three Safety Recommendations 2002-51,52 and 53, described earlier, in the course of this investigation (Appendix N and O).

#### 2.2.5 HUMS

There was no requirement for ground engineering staff to force a download of incomplete data, and it was expected that, in cases where the data was incomplete, that the following flight would collect sufficient additional data to permit a download. Alternatively, the card would be re-initialised and the data collected again in its entirety. The engineers were, at that time unaware of the possibility of simulating at turnaround with the rotors running in order to complete the data acquisition.

Checking the completeness of the data set is an additional manual task which was unknown to engineering staff at the time, although it is now conducted on a per flight basis. In future HUMS software, this function should occur automatically from the ground engineering 'general acquire' operation on the ground station. Therefore, the AAIB makes the following Safety Recommendation:

The CAA, in conjunction with the HUMS systems designers, should require the incorporation into future software versions the capability of providing, automatically, appropriate information about the recorded parameters and the integrity and completeness of the data. (Safety Recommendation 2003-88)

The operator now routinely interrogates graphs of engine and gearbox parameters. In the past, this work was only carried out by system specialists. For an initial interpretation of the data this is satisfactory but, when a potential problem is identified, there is no immediate guidance to the engineer. Maintenance activity needs to be based on properly interpreted data, and if this is to be, even initially, conducted by local engineering staff, then appropriate training is essential. It is therefore recommended that:

The CAA should require, for operations where HUMS is expected to contribute to the safe operation of the aircraft, improved training for the engineering staff to facilitate useful and meaningful 'first level' interrogation and investigation of the data. (Safety Recommendation 2003-89)

In addition, even where there are no threshold exceedences or trends, changing signal characteristics may indicate a pending deterioration. It is therefore recommended that:

The CAA, together with HUMS system designers, should incorporate in future HUMS software versions, algorithms which can identify changing signal parameters, other than levels, such as frequency changes and the development of harmonics. (Safety Recommendation 2003-90)

Following the serious incident to AS332L Super Puma G-PUMH, Safety Recommendation 98-09 called for 'means by which ready access could be provided to fleet wide trend data which would identify abnormal trends on a particular aircraft against an operator's whole fleet'.

There are still problems with comparing data across a fleet. Although the latest Windows NT software has a more capable user interface which can compare

data, the ground stations are often not connected, so data must be transferred by modem, or physically, between remote stations. This requires substantial operator activity and there is, in any case, no requirement for even modem data links between ground stations. The importance of comparing data raised in the report on the accident to G-PUMH remains. Therefore, the following safety recommendation is made:

It is recommended that the CAA, together with HUMS systems designers, should incorporate in future HUMS requirements, a requirement for a suitable infrastructure to facilitate the comparison of stored HUMS data between aircraft. (Safety Recommendation 2003-91)

The instruction to crews to operate for a sufficient time to acquire a full set of IHUMS data, is an additional burden to SAR crews, made more difficult by the lack of any system indication of when the data is complete. However, for most operations where a longer time is spent in the cruise, this is not an issue. Nevertheless, the AAIB makes the following Recommendation:

The CAA, in conjunction with HUMS system designers, should consider in future design, the incorporation of modified DAPUs which provide an indication of the completion of the data acquisition cycle. (Safety Recommendation 2003-92)

### **2.3 Previous Safety Recommendations**

The Safety Recommendation 4.11 made in 1990 following the similar accident to G-BEID, called for *'measures be taken to ensure that excessive deterioration of the No 5 bearing of the engine shall not result in failure of the engine mounting rear support assembly.'* This has been achieved, in part, by health monitoring of the No 5 bearing. The implementation of ECDs with cockpit warnings will improve the standard of health monitoring, but the severe consequences of bearing failure, should it occur, remain unaltered. For this reason, the AAIB made Safety Recommendation 2002-52 which called for action to ensure that the integrity of the engine fire zones would not be breached by failure of the EMRSA tube. However this recommendation has been rejected by the FAA and the manufacturer. The AAIB does not propose to make a further recommendation on this subject, but considers that the response to the already published Safety Recommendation 2002-52 is not sufficient to either prevent loss of a helicopter from the same cause, should a further case of bearing failure occur, or address fully the intent of Safety Recommendation 4.11.

Safety Recommendation 4.25, also made in 1990 and contained in the report on the accident to G-BEID, called for a review by the CAA of the fire protection

provision needs of helicopter main gearbox bays. This was accepted and a review conducted. However, the review concluded that thermal isolation systems were undesirable, that a requirement for fire detection and extinguishing systems in a 'non fire zone' could not be justified, and that flammable fluid cut-off systems presented difficulties in helicopters where systems such as lubrication and hydraulic power were essential for safe continued flight. Safety Recommendation 2002-53, which was made during the course of this investigation, was based on the same concerns addressed by Safety Recommendation 4.25. Safety Recommendation 2002-53 was rejected by the FAA and the manufacturer but, in the event of a further No 5 bearing failure, it is considered that there remains a strong probability that a potentially catastrophic and uncontrolled fire could occur.

### **3 Conclusions**

#### **(a) Findings**

- 1 The crew was properly licenced and medically fit to operate the flight.
- 2 The crew operated highly effectively during the rapidly developing emergency situation.
- 3 The performance of the crew reflected extremely well on their training and on their individual professional competence and ability.
- 4 The crew were not aware of the No 1 engine fire warning until the final approach to land.
- 5 The present positioning of the No 1 engine fire warning light is not optimum.
- 6 The No 1 engine fire warning did not remain on continuously.
- 7 Most of the failure warnings displayed to the pilots were caused by fire damage to electrical wiring.
- 8 The aircraft was properly certificated, maintained, and prepared for the flight. There were no relevant reported unserviceabilities.
- 9 The helicopter weight of 17,812 lb, was within the structural limitation of 20,500 lb, and its Centre of Gravity was also within normal limits.
- 10 The last major maintenance activity on G-BBHM had been a MGB change carried out about 100 flying hours before the accident. Since this change, only routine minor maintenance had been carried out in the area of the No 2 engine and drive train.
- 11 A Main Drive Shaft balance check was carried out at the time of the MGB change and this was well inside the published limits.

- 12 The engine MDPs were checked approximately six hours before the accident. There was no contamination of the MDPs with metal particles at that inspection.
- 13 The Technical Log recorded that there were no significant defects recently recorded, and there had been nil defects in the last six sectors.
- 14 During flight, the helicopter sustained damage to the ERMSA tube, free turbine shaft assembly of the No 2 engine and associated MDS. An intense in-flight fire then occurred which affected flight control and other systems in the MGB bay.
- 15 The No 5 bearing of the No 2 engine had worn excessively and suffered cage failure in flight, allowing the power turbine shaft to lose location and to 'orbit'.
- 16 The Thomas coupling at the forward end of the MDS of the No 2 engine severed the EMRSA tube, by means of a heavy rotational rubbing contact mechanism.
- 17 The 'live' and 'dead' oil jet assemblies of the No 2 engine had both experienced fatigue fractures of the rear oil tube in flight, as a consequence of the No 5 bearing deterioration and associated shaft vibration.
- 18 It was concluded that the initial cause of the failure was shaft vibration.
- 19 Deterioration of the No 5 bearing preceded the fracture of the oil jets.
- 20 There had been recorded indications of anomalies in the No 2 engine  $N_f$  over the preceding two hours of flight, and some anomalies in the last recorded HUMS data at the start of the same period.
- 21 The recorded engine No 2  $N_f$  jitter was a consequence of free play (wear) in the No 5 bearing.
- 22 It is considered, in the light of the overall body of evidence, that the failure of the bearing came about because of an increase in shaft vibration during the two-hour period preceding the accident.

- 23 It was concluded that the initial cause of the No 5 bearing failure was not consistent with a general lack of lubrication, but was consistent with dynamic radial imbalance loading of the bearing.
- 24 There was no evidence of any pre-accident mechanical problem with the MGB.
- 25 The cause of the onset of shaft vibration in the last two hours of operation was not positively identified. The only possibly relevant factor revealed by the investigation was evidence of high engine torque at rotor engagement, during three of the engine starts the day before the accident, two of which involved the No 2 engine.
- 26 A high rotor engagement torque value, at less than the engine normal  $N_f$  operating speed and possibly combined with a whirl mode of the MDS, could cause residual shaft imbalance or lead to No. 5 bearing cage damage.
- 27 It is considered that the high speed rotating contact of the Thomas coupling with the EMRSA tube, or the elevated temperature of the exhaust duct itself, provided a localised ignition source, allowing the released engine oil to be ignited.
- 28 The fire initiated in an area which was not within the engine No 2 fire zone, but within the MGB bay, which is not a designated fire zone.
- 29 The fire rapidly propagated unchecked and adversely affected flight critical systems.
- 30 The damage to one of the fire bottles occurred during the ground fire.
- 31 The history of the S-61 type shows that problems in the area of the drive train have arisen mainly from MDS imbalance
- 32 Data relating to the civil S-61 fleets world wide indicates that the UK rate for similar failures is apparently some 15 times higher than that for the rest of the world. Reporting deficiencies may account for some of this difference.

- 33 The Sikorsky S-61N and its General Electric CT58 engines are very similar to the licence-built Westland Sea King with its Rolls-Royce Gnome H1400 engines. The Sea King/Gnome fleet has not experienced any similar events.
- 34 The operator alternated starts on No 1 and No 2 engines as a matter of standard operating practice.
- 35 Single engine starts/rotor engagements have not historically been the norm for civilian S-61 operations.
- 36 Amongst the operator's flight crews, there was a widespread understanding that, during start, torque should not be allowed to rise above about 80%. The operator has now introduced a formal requirement to observe a 40-60% torque limit during the start.
- 37 There was no limitation contained within the Sikorsky S-61N Flight Manual used by BHL on torque during start, but other similar helicopter types did have a published limitation.
- 38 The flight crew involved in the recorded high torque starts were not aware of any special requirement to limit the torque during start.
- 39 Interrogation of the IHUMS data held by the operator showed that data from the previous flight had not downloaded.
- 40 The measured and recorded data suggests that, up to about two hours before the accident, vibration levels were normal.
- 41 During the two hour period before the accident, the recorded data showed evidence of a developing anomaly.
- 42 The need to check the completeness of the HUMS data set is an additional manual task which was unknown to engineering staff at the time of the accident, although this is now conducted on a per flight basis.
- 43 There is no requirement for data links between ground stations to facilitate the comparison of parameters across a fleet. Substantial operator intervention can be required.

- 44 During data acquisition, the flight crew currently have no indication of when the data acquisition process is complete.
- 45 The instruction to crews to operate for sufficient time to acquire a full set of IHUMS data is an additional operational burden, made more difficult by the lack of any system indication of when the data acquisition is complete. The DAPU could be modified provide an indication of the completion of the acquisition cycle.

**(b) Causal factors**

The investigation identified the following causal factors:

- 1 The No 2 engine had suffered rapid deterioration of the No 5 (location) bearing of the free turbine, causing failure of the adjacent carbon oil seal and mechanical interference between the Main Drive Shaft Thomas coupling and the Engine Mounting Rear Support Assembly tube, which completely severed the support tube.
- 2 A severe fire, outside of the engine fire zone, was caused because the released engine oil was ignited either by this mechanical interference, or by contact with the hot engine exhaust duct.
- 3 The No 2 engine's No 5 bearing failed because of unusual and excessive cyclic loading conditions arising from shaft vibration. The bearing deterioration was exacerbated by a reduction in its oil supply during the same period, when the live oil jet fractured as a consequence of the vibration.
- 4 It is probable that the Main Drive Shaft vibration was caused by damage or distortion sustained during one or more previous No 2 engine starts involving a high torque rotor engagement.
- 5 There was no specific torque limitation published in the manufacturer's Flight Manual, used by Bristow Helicopters Limited, during rotor engagement after engine start.

## 4 Safety Recommendations

The following Safety Recommendations were made in the course of this investigation:

- 4.1 **Safety Recommendation 2002-51** (made 21 November 2002): The US Federal Aviation Administration, in conjunction with UK CAA and the airframe and engine manufacturers, implement a means of providing a suitable warning to aircrew and/or engineering staff, of any impending loss of integrity of the drive shaft system of the S-61N helicopter which could lead to failure of the engine rear support mounting tube.
- 4.2 **Safety Recommendation 2002-52** (made 21 November 2002): The US Federal Aviation Administration, in conjunction with UK CAA and the airframe manufacturer, ensure that the integrity of the engine fire zones on the S-61N helicopter is not breached by a failure of the engine rear support mounting tube.
- 4.3 **Safety Recommendation 2002-53** (made 21 November 2002): The US Federal Aviation Administration, in conjunction with UK CAA and the airframe manufacturer, devise a means of protecting essential systems in the main rotor gearbox bay of the S-61N helicopter from the effects of fire.

The following additional Safety Recommendations are made in this report:

- 4.4 **Safety Recommendation 2003-83**: The aircraft manufacturer, Sikorsky, should relocate the No 1 and No 2 engine bay fire warning lights on the main instrument panel of the S-61N helicopter, with the intention of ensuring as far as possible that unambiguous information is presented to both flight crew members in the event of an engine bay fire.
- 4.5 **Safety Recommendation 2003-84**: The FAA, CAA, and engine manufacturer should introduce a modification to the oil jet assembly that, in the event of fracture of the tube which supplies oil to the carbon seal, would prevent a large reduction in supply pressure to the nozzle which supplies oil to the No 5 bearing.
- 4.6 **Safety Recommendation 2003-85**: The CAA, together with the FAA, airframe and engine manufacturers, should consider the possible value of measuring short term variability in the recorded  $N_F$  speed on S-61 helicopter engines, in order to provide early warning of loss of integrity of the drive shaft system, which could lead to failure of the engine mounting rear support assembly tube and subsequent fire.

- 4.7 **Safety Recommendation 2003-86:** The FAA and CAA should require Flight Manuals for all variants of the S-61 and similar types to include an appropriate torque limitation during rotor engagement.
- 4.8 **Safety Recommendation 2003-87:** The FAA and CAA, together with the airframe and engine manufacturers, should investigate the dynamic behaviour of the S-61 MDS and associated high speed rotating components in support of the introduction of an appropriate torque limitation during rotor engagement.
- 4.9 **Safety Recommendation 2003-88:** The CAA, in conjunction with the HUMS systems designers, should require the incorporation into future software versions the capability of providing, automatically, appropriate information about the recorded parameters and the integrity and completeness of the data.
- 4.10 **Safety Recommendation 2003-89:** The CAA should require, for operations where HUMS is expected to contribute to the safe operation of the aircraft, improved training for the engineering staff to facilitate useful and meaningful ‘first level’ interrogation and investigation of the data.
- 4.11 **Safety Recommendation 2003-90:** The CAA, together with HUMS system designers, should incorporate in future HUMS software versions, algorithms which can identify changing signal parameters, other than levels, such as frequency changes and the development of harmonics.
- 4.12 **Safety Recommendation 2003-91:** It is recommended that the CAA, together with HUMS systems designers, should incorporate in future HUMS requirements, a requirement for a suitable infrastructure to facilitate the comparison of stored HUMS data between aircraft.
- 4.13 **Safety Recommendation 2003-92:** The CAA, in conjunction with HUMS system designers, should consider in future design, the incorporation of modified DAPUs which provide an indication of the completion of the data acquisition cycle.

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Inspector of Air Accidents  
Air Accidents Investigation Branch  
Department for Transport  
March 2004

Unless otherwise indicated, recommendations in this report are addressed to the regulatory authorities of the State having responsibility for the matters with which the recommendation is concerned. It is for those authorities to decide what action is taken. In the United Kingdom the responsible authority is the Civil Aviation Authority, CAA House, 45-49 Kingsway, London WC2B 6TE