AIRCRAFT INCIDENT REPORT 2/98

Air Accidents Investigation Branch

Department of the Environment, Transport and the Regions

Report on the incident to Aerospatiale AS332L Super Puma, G-PUMH over North Sea on 27 September 1995

This investigation was carried out in accordance with
The Civil Aviation (Investigation of Air Accidents and Incidents) Regulations 1996

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Department of the Environment, Transport and the Regions
Air Accidents Investigation Branch
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26 February 1998

The Right Honourable John Prescott MP
Secretary of State
for the Environment, Transport and the Regions

Sir,

I have the honour to submit the report by Dr E J Trimble, an Inspector of Air Accidents, on the circumstances of the incident to Aerospatiale AS332L Super Puma, G-PUMH over North Sea on 27 September 1995.

I have the honour to be
Sir
Your obedient servant

K P R Smart
Chief Inspector of Air Accidents

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GLOSSARY OF ABBREVIATIONS USED IN THIS REPORT

AAIB	Air Accidents Investigation Branch
agl	above ground level
amsl	above mean sea level
ATC	Air Traffic Control
BCAR	British Civil Airworthiness Requirement(s)
CAA	Civil Aviation Authority
CVR	Cockpit Voice Recorder
CVFDR	Combined Voice and Flight Data Recorder
°C	degrees Celsius
DGAC	Direcion Generale de l'Aviation Civile
dN.m	decaNewton metres
DRA	Defence Research Agency
FDR	Flight Data Recorder
FMECA	failure mode, effect and criticality analysis
FRC	flight reference card
hrs	Time (24 hour clock)
HSE	Health and Safety Executive
Hz	hertz
IHUMS	Integrated Health and Usage Monitoring System
JAR	Joint Aviation Requirement(s)
kg	kilogram(s)
kt	knot(s)
MHz	Megahertz (frequency)
mm	millimetre(s)
MPa	MegaPascals
NDT	non-destructive test
nm	Nautical mile(s)
RPM	revolutions per minute
UTC	Universal Time Co-ordinated

Air Accidents Investigation Branch

Aircraft Incident Report No: 2/98 (EW/C95/9/4)

Registered Owner and Operator:	Bond Helicopters Ltd
Aircraft Type:	Aerospatiale AS332L Super Puma (Tiger) helicopter
Nationality:	British
Registration:	G-PUMH
Place of Incident:	Over North Sea Latitude: 57° 52' North Longitude: 000° 48' West
Date and Time:	27 September 1995 at 0730 hrs All times in this report are UTC

Synopsis

The incident was notified to the Air Accidents Investigation Branch (AAIB) by the operator at 0930 hrs on 27 September 1995. The investigation was conducted by Dr E J Trimble (Investigator-in-Charge), Mr P N Giles (Operations), Mr A P Simmons (Engineering) and Mr J R James (Flight Recorders).

The helicopter departed Aberdeen at 0702 hrs for a flight to the Tiffany Platform, 136 nm to the north-east, with the first officer as the handling pilot. At 0729 hrs, while cruising at 3,000 feet amsl and 120 kt, there was a sudden onset of severe airframe vibration. The commander diverted to Longside Airfield and, after transmitting a MAYDAY call at 0752 hrs, the helicopter landed at 0821 hrs. The passengers were evacuated without injury and shut-down was achieved by use of the General Cut Off Handle at 0821:32 hrs.

Subsequent examination of the helicopter revealed that a tail rotor blade flapping hinge retainer had fractured on one side.

The investigation identified the following causal factors:

1 Maintenance inspections conducted over a period prior to the incident flight did not detect a developing surface crack in the Blue tail rotor blade flapping hinge retainer, despite additional work on the associated tail rotor drive shaft assembly to rectify a tail rotor vibration problem, which was detectable as a trend recording within the Health and Usage Monitoring System some 50 flying hours previously and was the subject of an associated alert 5 hours before the incident.

2 The undetected fatigue crack extended during the flight, fracturing one side of the flapping hinge retainer and causing excessive and potentially critical tail rotor vibration.

3 The fatigue crack had been initiated by fretting and corrosion of the flapping hinge retainer bore induced by abnormal cyclic loading of the retainer, which was attributed to the effects of a defective flap needle-roller bearing during some previous period of the tail rotor drive shaft's life.

4 The inspection provisions within the aircraft Maintenance Manual and associated Maintenance Requirements did not specify periodic visual inspections of such retainers, since they had been designed and certificated on a 'safe-life' basis.

Six safety recommendations have been made as a result of this investigation.

1 Factual information

1.1 History of the flight

The helicopter, callsign 32E, departed Aberdeen at 0702 hrs for a flight to the Tiffany Platform, 136 nm to the north-east, with the first officer as the handling pilot. At 0729 hrs, while cruising at 3,000 feet amsl, 120 kt and heading 063°, there was a sudden onset of severe airframe vibration. The commander immediately instructed the first officer to reduce main rotor pitch and so he lowered the collective lever to reduce the main rotor pitch from 15.6° to about 12.2°, in addition to applying left yaw pedal to maintain the heading. The commander reported "major vibration" to Aberdeen Radar on the frequency in use, 134.1 MHz, and requested an immediate return to Aberdeen. The controller cleared 32E to turn left, or right, and to maintain 3,000 feet. The commander acknowledged this and advised that they would route towards Longside which is near Peterhead, 21 nm to the north-east of Aberdeen. The controller then advised 32E that the Buchan Platform was in their 12 o'clock position at 22 nm. However in view of the weather and the relatively difficult approach to the platform, the commander decided to continue with his decision to return to the mainland. He then briefed the passengers and reminded them to 'zip-up' their survival suits, put on their hoods and tighten their seat belts.

Meanwhile, the first officer had further reduced collective pitch to 11.1°, allowed the speed to reduce to about 68 kt and had started to turn left onto a heading of 237°. With the reduced pitch, the aircraft started to descend at 450 fpm; the commander informed ATC that they were descending to 2,000 feet. During the turn, the first officer raised the collective to 12.8° and increased the speed to about 90 kt. He then started a turn onto 326° to avoid weather ahead of the helicopter.

Both crew thought that the vibration was associated with the main rotor and the commander decided to carry out the FREQUENCY ADAPTOR FAILURE drill. His first action was to tell the first officer to reduce collective pitch to below 12°. Recorded data showed that, at this point, the helicopter was still turning towards a heading of 326° and the collective pitch was reduced to 12.1°. When the turn was complete, the pitch was further reduced to 11° for 5 seconds, during which time the tone of the associated noise recorded on the area microphone became less harsh. The crew perceived a slight reduction in the level of vibration and decided to continue the flight to Longside. The handling pilot then adjusted the pitch to 12.4° while the commander read aloud the following extract from the drill:

> "Reduce collective pitch slowly to the setting which reduces the vibration to a level acceptable for continued flight - approximately 12 to 14 degrees pitch - if vibration level does not decrease to an acceptable level - land/ditch immediately - comply with ditching - with floatation/gear - warn passengers and switch on seatbelt/no smoking signs - transmit PAN call."

The ATC controller had arranged for another helicopter in the area to act as an escort for 32E and, at 0733 hrs, confirmed that it was in visual contact. The controller noticed that 32E was "tracking northbound" and asked the commander to confirm his intentions. He confirmed that they had deviated north to avoid weather and then gave the first officer a heading of 241° for Longside. At this point the helicopter was slowly descending through 1,800 feet amsl. The handling pilot made several small pitch adjustments in an attempt to find a setting which reduced the level of the vibration, however this appeared to make no significant difference, although the commander thought it was not as bad as when it had initially occurred. He then gave the passengers an update on the situation and reminded them to keep their survival suits zipped-up with their hoods on. The crew then reviewed the situation and commented that the vibration appeared to be constant. They noted that the engine indications were normal, with no fluctuation in the torque readings.

At 0736 hrs, the ATC controller asked the commander if he wished to declare an emergency and a state of Urgency (PAN) was then declared. The helicopter was still descending slowly through 1,200 feet amsl and the airspeed was between 60 and 80 kt. The commander then armed the floats and reviewed the landing/ditching advice given in the drill. This recommended that, on land, the passengers should be disembarked from a 'low hover' with the wheels held lightly on the surface.

The commander tried unsuccessfully to make direct contact with his company at Aberdeen and Longside to inform them of the situation. The information was eventually relayed by a third aircraft. During this exchange, the 'Check Height' warning was heard, indicating that the helicopter was below 1,000 feet amsl. The aircraft was then flown at about 1,000 feet amsl for the remainder of the flight to Longside. No further collective pitch changes were made and the airspeed remained between 65 and 85 kt until the approach.

Although the crew remained convinced that the vibration was emanating from the main rotor, the symptoms were confusing and they were not able to positively identify the source. The commander considered the vibration period to be nearer 4R (ie four times main rotor speed) than 1R. The main rotor tracking appeared to be normal. The possibility of the source being an engine mounting was discounted because the engine torque and other engine parameters were normal and steady. The general airframe vibration was severe, but there was no obvious lateral component. No significant vibration was felt through the collective lever and that through the yaw pedals seemed to be in sympathy with the airframe. The most significant control vibration was felt through the cyclic control; this was greater fore and aft than laterally. The first officer considered that, despite the vibration, he had full control movement and normal response.

At 0752 hrs, the commander decided that the level of vibration had increased and accordingly upgraded the state of emergency to distress (MAYDAY). The escorting helicopter, which was by this stage alongside 32E, was not winch-equipped. However at 0753 hrs, in response to the MAYDAY, a Royal Air Force Sea King helicopter, Rescue 137, was sent to the area from Blair Atholl where it had been operating.

At about 0759 hrs, the commander established contact with Longside and was passed the weather; the surface wind was 270°/10 to 15 kt. He briefed the controller on his intention to disembark the passengers from the hover and it was suggested that this be accomplished on the threshold of Runway 28. The commander then told the passengers that they had 15 nm to go to Longside.

At 0816 hrs, 32E crossed the coast and the commander disarmed the floats and lowered the landing gear. He then briefed the passengers on the planned evacuation procedure; his intention was to establish in a low hover with the wheels touching the ground and the passengers were to leave through the left door only. He asked them to leave 'one at a time' in order to make it easier for the handling pilot to maintain lateral balance; unfortunately the public address (PA) system had become unserviceable and the briefing was not heard. The commander realised this and asked Longside to instruct the emergency services to open the left door and initiate the evacuation; this was acknowledged by 'Fire One'. The helicopter was established in a low hover at 0820:44 hrs and the passengers were evacuated without injury. The helicopter subsequently landed and shut down was achieved by use of the General Cut Off Handle at 0821:32 hrs.

1.2 Injuries to persons

Injuries	Crew	Passengers	Others
Fatal	-	-	-
Serious	-	-	-
Minor/none	2	15	-

1.3 Damage to aircraft

The aircraft damage was confined to a tail rotor shaft flapping hinge retainer which had fractured and had caused consequent vibration damage to the Integrated Health and Usage Monitoring System (IHUMS) instrumentation and to the PA amplifier, which had broken free from its mounting rack and had suffered internal failures.

1.4 Other damage

There was no other damage.

1.5 Personnel information

1.5.1	Commander:	Male, aged 38 years	
	Licence:	Airline Transport Pilot's Licence (H)	
	Medical certificate:	Class 1, valid to 31 May 1996	
		Limitations: None	
	Instrument rating:	Valid to 12 March 1996	
	Base check:	VMC - Valid to 7 January 1996	
		IMC - Valid to 7 January 1996	
	Line check:	Valid to 15 March 1996	
	Crew Resource Management:	Refresher 3 March 1995	
	Flying experience:	Total all types:	3,900 hours
		Total on type:	3,800 hours
		Total last 28 days:	50 hours
		Total last 24 hours:	7 hours
	Previous rest period:	Off duty:	1245 hrs on 26 September 1995
		On duty:	0700 hrs on 27 September 1995

1.5.2	First officer:	Male, aged 45 years	
	Licence:	Airline Transport Pilot's Licence (H)	
	Medical certificate:	Class 1, valid to 30 April 1996	
		Limitations: Lenses for distant vision	
	Instrument rating:	Valid to 10 July 1996	
	Base check:	VMC - Valid to 10 December 1995	
		IMC - Valid to 2 November 1995	
	Line check:	Valid to 24 August 1996	
	Crew Resource Management:	Refresher 3 February 1995	
	Flying experience:	Total all types:	7,365 hours
		Total on type:	1,798 hours
		Total last 28 days:	55 hours
		Total last 24 hours:	4 hours
	Previous rest period:	Off duty:	1330 hrs on 26 September 1995
		On duty:	0700 hrs on 27 September 1995

1.6 Aircraft information

1.6.1 General information

Manufacturer:	Aerospatiale (now Eurocopter)
Type:	Aerospatiale AS332L Super Puma
Aircraft serial no:	2101
Year of manufacture:	1983

Certificate of Registration:	Issued on 3 August 1983 in the ownership of Bristow Helicopters Ltd
Certificate of Airworthiness:	Issued on 23 May 1995 in the Transport Category (Passenger) and valid until 22 May 1996.
Engines:	2 Turbomeca Makila 1A turboshaft engines
Total airframe hours:	14,105.25 hours
Total flight cycles:	16,662

Aircraft weights, balance and fuel:

Maximum take-off/landing weight:	8,600 kg
Take-off weight (TOW):	8,597 kg
Estimated weight at time of incident:	8,300 kg
Landing weight (LW):	7,800 kg

1.6.2 Maintenance records

The aircraft records showed that the following significant maintenance work had been carried out in the period leading up to the incident:

TASK	AIRFRAME HOURS	DATE
Rebalance of tail rotor	14,050	13/9/95
50 hour check	14,053	14/9/95
50 hour check	14,105	22/9/95

3,000 hour 'on condition' check of tail rotor head requirement issued	14,105	22/9/95
IHUMS trend investigation - Additional Work Sheet raised for removal of tail rotor blades, pitch change spider, fairings.	14,105	22/9/95
3000 hour 'on condition' check of tail rotor head completed	14,105	26/9/95
Rebalance of tail rotor	14,105	26/9/95

The 500 hour and 1,000 hour checks were out of phase with the 3,000 hour check. Each of these checks referred to the Maintenance Manual (MET) at 64.20.00.603 para C which required 'external visual examination for corrosion, impact, scoring, cracks or dents'. The last 500 hour check had been carried out at 13,818 hours, and the last 1,000 hour check at 13,390 hours.

The helicopter was operated in such a way that flight cycles (landings) did not equate to engine startup-shutdown cycles. The Technical Log showed that typically a flight cycle averaged slightly less than one hour, however there would be a number of such flights before the aircraft was shut down and this often included refuelling with rotors running. The average flight time between shutdowns in the period leading up to the incident was a little less than three hours.

1.6.3 IHUMS overview

The aircraft was fitted with an Integrated Health and Usage Monitoring System (IHUMS) as an aid to maintenance. The system consisted of onboard equipment (sensors, data processor, data recorder, etc) and a ground station for processing and storage of data. IHUMS was used to monitor the health of the transmission and rotor systems of the helicopter with a view to providing early warning of any degradation and also to provide a record of life usage of various components.

The system was installed on the operator's initiative and was certified on a 'No Hazard, No Credit' basis. This meant that the onboard IHUMS equipment did not present a hazard to the operation of the aircraft and the data that it gathered could only be used as an aid, without providing any other advantage or 'credit', for maintenance. With the exception of Flight Manual exceedances, the IHUMS provided no output to the crew to indicate malfunction of any of the systems which it monitored.

The data generated was used as a non-mandatory maintenance aid at the operator's discretion. Associated alerts required interpretation on a 'case by case' basis and any related maintenance action was determined from the data available to the operator, in conjunction with the IHUMS supplier.

Since the system was not installed by the airframe manufacturer, it did not utilise the airframe manufacturer's full knowledge of all modes of component failure and their associated means of detection.

Current design certification requirements for airframe manufacturers now include a full Design Assessment, in the form of a failure mode, effect and criticality analysis (FMECA). Each failure mode is now assessed for criticality and a suitable detection or prevention procedure stipulated to deal with its occurrence. This procedure may involve combinations of inspection, replacement of lifed components after a designated number of operating hours, or continuous monitoring for degradation by suitable vibration analysis systems, such as IHUMS.

The principles of the vibration analysis which formed the basis of the IHUMS system are still being improved. 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 vibration limits for triggering exceedance warnings are being reduced to provide earlier detection of faults, although this can also increase the incidence of spurious warnings.

1.6.4 History of tail rotor shaft

The information in this section is summarised in tabular form at Appendix A.

The tail rotor shaft (Appendix D, D-1) part number 330A.33.3165 E1, serial number M1941, was manufactured in October 1984. These shafts currently have an approved service life (scrap life) of 16,500 flying hours.

The shaft was fitted to gearbox serial number M290 in the same month as it was manufactured. That gearbox entered service and was subsequently returned to the manufacturer for overhaul near the end of its 3,000 hour overhaul life. The gearbox and shaft had, at that time, accumulated 2,965 hours total time since new (TTSN).

The shaft then passed through the normal overhaul and inspection process current at that time. It was accepted as serviceable and was fitted to gearbox serial number M296 with 13,535 hours remaining. That gearbox was subsequently removed for overhaul in October 1990 by the manufacturer when it had accumulated 3,031 hours, at which time the tail rotor shaft had accumulated 5,996 hours TTSN. The shaft was again overhauled and inspected before being fitted to gearbox serial number M187 between July 1992 (the date of the relevant Works Order) and October 1992 when the gearbox was released. Gearbox M187 was eventually fitted to G-PUMH at 11,151.15 airframe hours. At that time the shaft had 10,504 hours remaining. Appendix B shows the shaft and gearbox M187, after removal following this incident.

During the approximately 50 hours flying time before the incident, some vibration in the vertical plane of the tail rotor had been recorded by the IHUMS system, however this parameter did not exceed the alert threshold until the 22 September 1995, 5 days before the incident, when an exceedance alert occurred. (Note that the airframe hours at that time were 14,105 and that only 25 minutes was logged between the 22 and the 26 September 1995). At that time the personnel operating the ground-based IHUMS system had intervened and the trend had been plotted and observed to be increasing (see Appendix C). The IHUMS indications were attributed by the engineer involved to slight 'free play' in the tail rotor gearbox shaft bearings. The vibration was temporarily resolved by rebalancing, but on the day before the incident the manufacturer advised that the gearbox should be withdrawn from service. However this message was received the following day, after the incident had occurred. Final separation of the fracture face and the onset of severe vibration occurred about 27 minutes into the incident flight. At the beginning of that flight the aircraft had accumulated a total of 14,105.25 airframe hours, with gearbox M187 having accumulated 2,955 hours and the shaft a total of 8,951 hours TTSN.

The tail rotor gearbox assembly was covered by a 'Power-by-the-Hour' pool arrangement. Any technical problems or rectifications arising were recorded in the relevant documents, and for items such as flap bearings this meant that the defects were recorded in the Technical Log of whichever aircraft had this gearbox installed at the time and the rectifications detailed on Additional Work Sheets, which were kept with the respective aircraft records. Consequently the full flap bearing history could only be traced by reviewing the records of all the aircraft to which this gearbox had been fitted. Fortunately in this case all of the relevant aircraft were identified, and although it was by no means certain that all the relevant flap bearing defects had been recorded, or even found, the records did contain one item of interest. On 5 March 1989, while this shaft was fitted to gearbox serial M296 on aircraft G-TIGJ, metal particles were found in the grease of the 'Blue' tail rotor blade flap bearing. The associated bearings and bushings were renewed. Although no other information was recorded, it was slightly unusual for such log entries to refer to 'metal particles'.

1.6.5 Tail rotor shaft overhaul procedures

When the tail rotor gearbox serial number M296 (ie the second gearbox to which this shaft was fitted) reached its overhaul life, the tail rotor shaft was removed from the gearbox for inspection. This inspection required the removal of the flap needle roller bearings and bushes (see Appendix D, D-2). The removal of the bushes had become an overhaul requirement in March 1991; prior to that date the shafts had been examined without removing the bushes, and therefore the inside faces of the bores (where this crack had originated, see section 1.16.1) were not inspected prior to that date. It was known that the bores could be subject to fretting damage arising from the tail rotor blade lead/lag loads and it was mainly for this reason that the requirement to remove the bushes was introduced. After disassembly, the shaft was cleaned and visually inspected. Dimensional checks were carried out on the five bores and on that occasion the component records showed that the dimensions and condition of the bores were such that no re-machining was required. The shaft was then cleaned in a trichlorethylene-based solvent and subjected to a magnetic particle inspection. The length of the shaft and each bore were separately checked. According to the manufacturer's records, this shaft was initially rejected by Quality Control for an unknown reason. The relevant documentation was not found and so the reason for that rejection was not determined. This shaft, together with another from the same overhaul batch which was also rejected, was subjected to a second overhaul. The documentation showed that no re-machining of the bore was carried out. This shaft then met the inspection requirements and was accepted as serviceable.

1.6.6 Frequency adaptor and failure checklist

The frequency adaptors are lead/lag dampers fitted to the main rotor blades. These dampers are primarily elastomeric and failure of the elastomeric element can give rise to high vibration levels. Following the only recorded case of a frequency adaptor failure in flight, in July 1987, an appropriate drill was introduced into the Flight Reference Cards (FRC). There are no other vibration related drills in the FRC. The frequency adapter design was subsequently modified to reduce the probability of failure. The symptoms of a frequency adapter failure were described in the drill as 'Severe lateral vibration (approximately once every main rotor revolution/4 times every second)'.

1.7 Meteorological information

The synoptic situation at 0300 hrs showed a depression centred near the Shetland Islands. It was moving slowly south-east at about 8 kt, maintaining a showery westerly airstream across all routes north-east of Aberdeen.

Surface wind	260°/25 kt, gusting 30 to 40 kt
Visibility	20 km
Cloud	Scattered/Broken, base between 2,000 and 2,500 feet
Weather	Scattered showers
Temperature	+10°C
Mean pressure	995 mb

The 0600 hrs weather at the Buchan platform was:

Surface wind	230°/28 kt
Visibility	20 km
Cloud	Overcast, base 1,000 to 2,000 feet
Temperature	11°C
Pressure	995.8 mb

1.8 Aids to navigation

Not applicable.

1.9 Communications

At the time of the incident the helicopter was in contact with the Aberdeen Radar Heli 2 Offshore Controller on frequency 134.1 MHz; this frequency was recorded. All non-essential traffic was transferred to another frequency. 32E remained on this frequency until 0817 hrs when it transferred to Longside. There were no communications problems.

1.10 Aerodrome information

Not applicable.

1.11 Flight recorders

1.11.1 Flight recorder installation

The recorder fitted to the helicopter was a Penny and Giles Combined Voice and Flight Data Recorder (CVFDR) type D51506 and formed part of the aircraft's IHUMS. The CVFDR maintained a record of the last 5 hours of aircraft data, together with the last hour of 3 channels of audio from the commander, first officer and cockpit area microphones. Following the incident, the recorder was returned to the AAIB where it was replayed successfully using normal replay techniques.

The CVFDR voice recording started 6 minutes before the onset of the incident and contained the remainder of the flight, in addition to approximately 3 minutes of recording during which an assessment of aircraft damage was made by maintenance personnel. The voice recording system used on the aircraft was the 'hot microphone' system where the crew's microphones are always live and each crew member is recorded on a separate CVR voice channel. However, the intercom facility was 'permanently selected' and so the voice recordings were not channel-separated but were distinguishable due to the differing voice characteristics of the commander and first officer.

The data recorded on the CVFDR started during the last flight on the 22 September 1995 and terminated after the incident flight at the same point as the voice recording. As part of the IHUMS, the CVFDR recorded main rotor track and balance timing data. Flight Manual exceedances concerning the engines and gearbox oil temperatures were also recorded. None of the IHUMS data recorded on the CVFDR was of relevance to the investigation of this incident.

Pertinent aircraft parameters and voice extracts recorded by the CVFDR at the time of the onset of the severe vibration are shown in Appendices E and F.

1.11.2 Incident flight

The CVFDR showed that, at the time of the onset of the vibration (0729 hours), the aircraft was at 3,000 feet amsl, at 120 kt and on a magnetic heading of 063°.

Within 2 seconds of the onset of the vibration (which was audible on the area microphone recording) the collective lever was lowered from 15.6° to 12.2°. Left yaw pedal was also applied. Both altitude and airspeed began to reduce and the collective pitch was lowered further to 11.1°. The collective pitch was maintained below 13° for the remainder of the flight.

1.12 Post-incident inspection of the helicopter

1.12.1 On-site

The helicopter was examined on the airfield at Longside shortly after the incident. The tail rotor gearbox had been removed and dispatched to Bristow Helicopters, Aberdeen, for inspection. Examination of the airframe showed that the damage was confined to the tail rotor gearbox assembly, the public address amplifier and its mountings, and some instrumentation in the tail boom, specifically the IHUMS tail rotor optical azimuth transducer wiring and an accelerometer installed at frame 9000. The tail rotor gearbox was examined later at Aberdeen and it was observed that the flapping hinge retainer at the Blue blade position (each of the five blades is colour-coded for identification) had fractured on one side in the plane of the greasing point and had opened up, under centrifugal loads, by approximately 6 to 7 mm. The surfaces of the fracture showed some corrosion and 'concoidal' markings typical of fatigue progression (see Appendix G). It was also noted that there was a small amount of radial play in the tail rotor shaft, which was abnormal. The flap hinge bolts were torque-checked and were found to be set to not less than the specified torque range of 7.5 to 9.0 decaNewton metres (dN.m). The bolts associated with the Yellow and White blades were found to be torqued to 11.5 and 10.6 dN.m respectively, however the bolt for the Blue blade was set at 9.0 dN.m. Low torques values can give rise to increased fretting damage, but high torque values would have little adverse effect since the compressive loads are transmitted directly through the inner race and the use of polyurethane washers in the assembly prevents any adverse stacking tolerances from generating significant additional loads.

The bolts were then removed. The flapping hinge needle roller bearings and associated races at all five positions were inspected and were found to be in good condition. The bushes at the Blue blade position were removed and these were found to be distorted and damaged. It was concluded that this distortion and some of the damage, at least, had occurred after the crack had opened-up in flight. Damage to the flap stop fitting in the blade spindle assembly was noted. This was due to repeated hard engagement with the integral flap stop, and is shown in Appendix H. The retainer was then separated by cutting it approximately opposite the fracture. The small portion was returned to the Defence Research Agency (DRA) Farnborough for metallurgical examination; the shaft with the other half of the fracture was returned to the manufacturer, still assembled to the gearbox, for metallurgical examination and dimensional checks. No damage was noted to the blades or spindles, apart from the flap stops on the Blue blade spindle, as described above.

1.12.2 Manufacturer's examination

The tail rotor gearbox and shaft were inspected in detail by the manufacturer with the AAIB in attendance. Prior to disassembly, the radial play in the tail rotor shaft was measured and found to be 0.07 to 0.08 mm. As it seemed probable that the radial play was due to shaft 'end-float', the gearbox was fitted to a test rig which measured the end-float while axially loading the shaft. The end-float was found to be 0.29 mm; this was considered to be due to free motion in the taper roller bearings. In order to confirm this, the gearbox was stripped. No defects, other than shaft free play, were noted and it was concluded that the radial free play had been a result of the service operation of the gearbox and was not related to the incident.

Each tail rotor blade spindle assembly was then subjected to a 'pull-off' check in which the torque required to change the pitch angle was continuously measured. In addition, the starting torque was measured. Excessive torque values could give rise to abnormal cyclic stresses in the retainers, however no excessive torques were recorded and the condition of the spindle bearings was found to be satisfactory.

Following this work, the shaft was checked dimensionally, particularly the flap hinge bores and the bushes. It was confirmed that the four undamaged bores were within the range 41.631 to 41.657 mm diameter, which conformed to the nominal specification for an un-reworked shaft. Since the retainer had fractured the bore could no longer be dimensionally checked, however the bushes were checked and found to be consistent with nominal dimensions of 41.675 to 41.690 mm diameter and the same nominal size as the bushes in the other bores. It was thus concluded that the degree of interference was most probably within limits and it was therefore likely that the interference fit in the Blue blade retainer bore had been within the intended range. A loose fit would have resulted in excessive fretting damage. The bore surface was therefore examined for fretting and corrosion damage. Both were found, however the fretting was not as great as would have been expected had the bushes been insufficiently tight in the bore and the corrosion, which was 7 to 10 microns in depth, was not as deep as the manufacturer considered necessary for this to be a cause. Some small amount of fretting damage existed in one other bore. The shaft was then checked for cracks using the same magnetic particle technique normally used on overhaul, but no additional cracks were found.

Subsequently the fracture surface was subjected to metallurgical examination by the manufacturer. Residual internal stresses were measured and found to be 48 hectobar, a low value. Later, the half of the fracture examined by the DRA was also sent to the manufacturer for examination. The metallurgical examinations carried out at the DRA Farnborough and by the manufacturer are described in section 1.16.1.

1.13 Medical and pathological information

Not applicable.

1.14 Fire

Not applicable.

1.15 Survival aspects

Not applicable.

1.16 Tests and research

1.16.1 Metallurgy - DRA and Eurocopter reports

The fracture surfaces were examined at Aberdeen by a metallurgist from the DRA Materials and Structures Department. Later, one half of the fracture was examined in more detail at DRA, Farnborough. The DRA metallurgical report stated that the shaft was manufactured from 30 NCD 16 steel with a strength in the range 1,220 to 1,370 MegaPascals (MPa). Hardness tests showed that the strength was consistent with the material specification. The crack had clear evidence of growth by fatigue over most of its length from an origin close to one end, on the inside face of the bore (Appendix G). There was corrosion pitting which was about 7 to 10 microns in depth in the region of the origin, some 7 mm from one end. The total crack length was 67 mm, having propagated in a slow, stable manner in both directions from the origin. The crack had propagated a maximum distance of some 60 mm from the origin. The remaining material had failed in a ductile rupture mode. The report concluded that the fatigue markings visible might be related to the number of flight cycles since the crack began to propagate. Additionally, the metallurgist was able to estimate the crack length at the time the IHUMS data began to show a divergent trend, from the striations, as about 54 mm.

The manufacturer examined the other half of the fracture and later the half of the fracture which had been held at Farnborough. The associated metallurgical report generally concurred with the DRA report. It added that there was evidence of slight rotation of both bushes within the bore. The report indicated that the failure had initiated due to fretting. Further discussions with the manufacturer's senior structural specialist indicated that this fretting damage was such as would occur if the rotation of the bushes had been caused by unrelieved flapping moments. Damage to the protective treatment would then occur, initiating the corrosion.

The DRA report could not confirm or refute, from the metallurgy alone, the manufacturer's assessment regarding the fretting as this required a detailed knowledge of the operating characteristics available only to the Design Authority.

Both reports indicated that over two hundred load cycles had been required for the crack to propagate to failure. Since the major loading was due to centrifugal forces, it was concluded that one fatigue cycle had occurred for each startup-shutdown cycle.

1.16.2 Non-destructive test (NDT) technique trials

Shortly after the incident, the operator and manufacturer devised a non-destructive test (NDT) eddy current technique. This technique was used on the operator's fleet. No crack indications were found. The nature of the technique gave rise to concerns that eddy current methods might produce erroneous results due to the use of steel bushes against the bore surfaces where a crack might originate. Some further experimentation with ultrasonic pulse echo techniques was carried out, but this was not particularly successful.

Another operator developed both eddy current and ultrasonic pulse echo techniques. These were demonstrated to the AAIB using a test specimen with an artificially generated crack. The ultrasonic technique was particularly effective in the trial, but in use it gave false crack indications when the flap bearings were in place. That operator carried out a 'one off' eddy current inspection across its Super Puma fleet and found no cracks.

The tail rotor shaft is protected with an epoxy primer and paint treatment as manufactured. In service, this protective treatment may be repaired from time to time. It is possible, therefore, that subsequent layers of paint may not adhere fully to the original finish and that this could result in subsequent cracks at the surface of the metal remaining invisible under such paint layers. Oil and grease in the area could also obscure crack indications. Magnetic particle NDT techniques would be effective in such circumstances.

The continuing airworthiness of the fleet is currently controlled by daily visual inspections.

1.17 Organisational and management information

Because the company had found it necessary to commit some of the Super Puma commanders to an overseas contract, a number of senior first officers were promoted to commander for a limited period to cover their absence; this period was from February to the end of September 1995. These pilots were chosen for their command potential and were given the normal conversion training. The commander of the incident flight was one of these pilots and reverted to the rank of senior co-pilot on 1 October 1995. This was in accordance with this agreement and did not reflect any dissatisfaction on the part of the company with the way in which he had handled this emergency as commander.

The operator's Maintenance Organisation was approved under Joint Aviation Requirements (JAR) 145. The engineers involved reported through a typical hierarchical structure headed by the Technical Director. They were generally aware of the day-to-day condition of the fleet and the IHUMS trend information was made available to assist trouble-shooting of the vibration problem.

1.18 Additional information

1.18.1 Integrated Health and Usage Monitoring System

1.18.1.1 IHUMS capabilities

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

Monitor shaft health
Monitor gear health
Monitor for bearing defects
Maintain a record of life usage of up to 60 components
Monitor main rotor track and balance
Monitor tail rotor balance
Provide crew with flight manual exceedance warnings
(not including vibration)

In addition, a facility enabled the crew to record a 'snapshot' of data by using a cockpit mounted control panel, and the IHUMS ground station provided a means of permanently storing the data downloaded from aircraft.

The airframe flying hours calculated by IHUMS for component usage monitoring is based on the length of time that the weight-on-wheels signal is not active. The time is then rounded up to the nearest minute. The airframe hours detailed in the aircraft's logbook are compiled from the flight times noted by the crew and are slightly longer. For this reason, there is a disparity between the IHUMS and logbook hours of approximately 0.5%. At the time of the incident, IHUMS hours lagged those of the logbook by some 72 hours. A facility existed within the IHUMS to resynchronise with the logbook hours, but this was not used by the operator.

The ground based element of the IHUMS system was not optimised for the detection or plotting of trend data, which for a given aircraft could be obtained by operator intervention. Across the fleet no such facility was available, so in order to detect a parameter on an individual aircraft which might be significantly different from the fleet as a whole, related data were manually extracted for each aircraft and then plotted by hand.

1.18.1.2 Health monitoring, principles of operation

The system relies on the fact that every rotating part of the transmission system in the aircraft has an associated vibration signature. Once the aircraft is established in a particular phase of flight, the system automatically samples and logs the vibration signatures by sequencing through the various sources and taking measurements. Ten separate phases of flight are allocated, each with a particular selection of vibration sources to measure. During long, stable flight conditions a timer is set to count down from one hour and once elapsed, a further sampling sequence is initiated. The system also has the capability of assessing main rotor track and balance using an optical sensor, and tail rotor imbalance in two orthogonal planes using two airframe mounted accelerometers and an optical azimuth detector.

A number of accelerometers are mounted on the aircraft in positions determined to best sense particular vibration sources. These are shown at Appendix I. The characteristics of these accelerometers are such that they pick up all vibrations within their locality and therefore associated signal processing is required to extract the vibration signature from a particular source. This processing requires information on the speed of rotation of the source and so the sampling of the accelerometer signal is governed by a master timing signal, derived from the number one engine turbine. Signal averaging is also used by taking a large number of samples. This helps to reduce those vibration components that are 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 is 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 mean that the time taken to complete a full cycle of measurements can be up to 12 minutes.

Once processed, the resulting vibration signatures are stored in the IHUMS memory. At the end of a flight when the rotors have slowed to less than 100 RPM and the engines have run down to less than 10,000 RPM, the IHUMS memory is transferred to non-volatile memory on a removable card. Due to the large number of individual vibration sources on the aircraft, considerable vibration data is accumulated during flights and so the card is removed at the end of each flight and the data downloaded into a ground station for further analysis. Such cards are then erased, prior to being inserted into the IHUMS on aircraft at the start of the next flight. These cards were not specific to a particular aircraft.

The ground station is capable of automatically analysing the data and will alert the operative if the vibration from any particular source exceeds a predetermined limit. This limit was set during the design of the IHUMS after assessing each aircraft vibration source and its likely vibration level during normal operation. The ground station can then print out a report log detailing areas of concern to aid aircraft maintenance. Further facilities enable the ground-based IHUMS system user to extract time histories of vibration signature levels for establishing trends in the data by manual analysis.

1.18.1.3 Tail rotor health monitoring

The particular vibration source which was of importance in this incident was the tail rotor shaft, with the direction of the vibration in the plane of the tail rotor disc. The positions for all three accelerometers are shown in Appendix I. Two separate accelerometers capable of registering this vibration were positioned near the tail rotor; ie the transmission accelerometer (number 11) mounted on the tail rotor gearbox, and the vertical axis airframe accelerometer (number 7) attached to the structure of the tail boom. The lateral axis airframe accelerometer (number 8) was attached to the tail boom structure in a position that was within 150 mm of that of the number 7 accelerometer. During maintenance, just prior to the incident flight, it was discovered that the two airframe accelerometers (numbers 7 and 8) had been crosswired and that the IHUMS Tail Rotor Lateral Vibration was measuring in the vertical plane, and vice versa.

1.18.1.4 Limitations of health data obtained prior to incident flight

Although the IHUMS equipment generated a clear trend of progressive abnormality within its data over some 50 flight hours and exceedance of a predetermined threshold prior to the incident, three data management issues arose that compromised its effectiveness in preventing this tail rotor failure:

1. The IHUMS system did not provide for data to be either manually or automatically examined for developing trends as a routine.
2. The specified alert threshold was, with hindsight, set too high.
3. Specific maintenance actions relating the alert to the potential failure modes and the most relevant inspection methods did not exist.

As a result, the associated alert was not timely in relation to the damage progression and the engineer involved received no direct advice as to the location, or nature, of the defect.

Section 1.18.3 'Maintenance activity' includes further information on the Health and Usage Data obtained before the subject flight.

1.18.1.5 Health data from G-PUMH during incident flight

As a result of the general cut off handle being pulled at the end of the incident flight, electrical power was cut from the IHUMS processing unit before it recognised the shutdown condition (ie gas generator RPM Ng on both engines less than 10,000 RPM and rotor speed less than 100 RPM). This prevented the accumulated flight data from being transferred from system memory to the non-volatile memory in the removable card. However, by using another card encoded with the relevant incident flight details, the IHUMS data was successfully downloaded and analysed by the ground station.

The resulting report log showed no abnormalities with respect to the tail rotor lateral and vertical vibrations, or to the vibration level associated with the tail rotor gearbox output shaft. The data for both of these sources was gathered before the final fracture in the tail rotor flapping hinge retainer and hence were of an acceptable level.

Further data was not gathered during the flight as the IHUMS one hour timer used to schedule the start of a new data acquisition cycle had not elapsed prior to aircraft shutdown.

1.18.2 Design and certification of the tail rotor shaft

The certification of the AS332L was validated by the CAA on 13 April 1982 and the certification basis in force at that time did not include the current Design Assessment requirements. The latest British Civil Airworthiness Requirements (BCAR) and Joint Aviation Requirements (JAR) call for a Design Assessment of the rotor and transmission system (which would have included the tail rotor gearbox output shaft) and includes a detailed failure analysis to identify the parts with potential for hazardous or catastrophic failure. The 'compensating provisions' necessary to minimise their likelihood of failure are then assigned and must be substantiated by tests, or calculation. For those helicopter types that have already met these requirements, health monitoring is exploited as a 'compensating provision' and mandatory maintenance actions, which directly relate to the failure modes being monitored, are defined for each alert.

The tail rotor shaft part no 330 A 33.3165.00 used on the Super Puma was significantly different to the corresponding shaft on the earlier Puma. It was visually apparent that the flapping hinge retainers were larger. The likely consequence of complete loss of one tail rotor blade was that the tail rotor gearbox assembly would have detached from the airframe, causing a large forward shift in the centre of gravity and associated pitch-down moment. This situation was assessed as 'catastrophic' for airworthiness purposes and therefore the design was required to provide a level of risk of failure better than 1 in 1×10^9 per hour, categorised for airworthiness purposes as 'extremely improbable'. In order to achieve this level of reliability, a number of potential events leading to failure were considered: these were external damage, fatigue, corrosion and fretting; jamming of the flapping hinge was also considered. These possibilities were then controlled by use of appropriate safety factors in accordance with normal design practice, and the results were then justified by tests. Finally the manufacturing processes for the assembly called for appropriate protective treatments, to prevent corrosion, and regular inspections to assure integrity of the component throughout its service life.

The major loading on the flapping hinge retainers arises from the centrifugal force generated by the tail rotor blades. This force is some 6 to 8 tonnes per blade. It is rotor speed-dependent and so one cycle occurs only once per startup/shutdown cycle. The main cyclic loads occur due to the drag moment generated by each blade; this cycle occurs once per revolution. The component is so designed that the stress levels generated by these forces fall well below those required to initiate fatigue cracking within the service life of the shaft. The bending moments induced by flapping and drag moments, and the mean and safe fatigue curves are shown in Appendix J. The cyclic drag moment generates small deflections of the retainers in the drag plane, and this can cause fretting between the retainer bores and the bushes within them. The fretting can lead to crack formation and so the degree of fretting is controlled by the interference fit of the bushes and justified in tests. The stress levels in the retainer are controlled to provide a large safety margin. Taking these effects into account, the unfactored life of the retainer would be 129,000 hours. In other areas the shaft is limited to a life of 16,500 hours. The large safety factors which exist in the retainer are required to achieve the necessary level of structural integrity. Therefore significant levels of corrosion or fretting damage must occur before the stresses become high enough to cause crack initiation and propagation. The manufacturer advised that the corrosion found in the retainer was not deep enough to have reduced the margins sufficiently to have caused cracking to initiate, and that much larger amounts of fretting damage than those found would have been required to initiate cracking.

One issue which was part of the design study was the examination of cyclic loads imposed in the flapping plane by a seized, or partially seized, flapping hinge. These effects are also shown in Appendix J, J-1. Normally, no significant stresses are induced in the retainer because of the flap bearing, a double needle roller bearing, which attaches each blade spindle to the hub. The manufacturer advised that stiffness in this bearing theoretically introduces significant loads into the retainer in the direction of flap and that these loads could completely erode the fatigue margin. This failure mode was assessed as potentially catastrophic and the prevention mode was defined as "Periodic Inspection", ie daily inspections, 50 hour inspections and periodic disassembly.

1.18.3 Maintenance activity

The last significant opportunity to detect the crack occurred some 287 hours before the incident when the previous 500 hour check had been carried out. This check specifically required an inspection for cracks in the tail rotor shaft. From the metallurgical reports it appears that the crack was much shorter at that time, ie probably approximately one inch in total length. This was the last time, prior to the incident, that a mandatory requirement to inspect for cracks arose since at the time that the 3,000 hour check was carried out (ie on 26 September, the day before the incident flight) the next 500 hour check was due in a further 213 hours.

On 13 September 1995 at 14,050.35 airframe hours (13,975 IHUMS hours) the two airframe accelerometers picked up a difference between the tail rotor lateral 1T (once per revolution) and vertical 1T vibration at Minimum Pitch On Ground (MPOG) as shown in Appendix K. At this stage it was not known that the two airframe accelerometers had been crosswired and so the larger of the two vibration levels at MPOG was in the lateral plane, not the vertical plane as shown. This IHUMS warning resulted in maintenance action to rebalance the tail rotor. The work was carried out using Chadwick 8500 equipment and was satisfactory. Checking of the rotor balance must be carried out in the open and preferably not in bright light since a stroboscopic light source is used. The Chadwick 8500 unit then produces a printout which shows how much weight must be added to each blade. The weights were added at locations some distance from the area in which the (unknown) crack existed, and it is unlikely that during this work the engineer would have been able to position himself so as to be able to see the retainer face on which the crack was present. At this time the crack would have been approximately 54 mm long.

On 21 September 1995 at 14,089.45 airframe hours (14,021 IHUMS hours) the IHUMS log report showed five data integrity warnings associated with the tail rotor vertical vibration sensor. These warnings are given when the sampled data fails to meet checks designed to detect faulty instrumentation. Associated maintenance action resulted in the detection and rectification of the crosswired accelerometers at the end of the following day.

On Friday 22 September, five days before the incident, a work requirement was raised for a 3,000 hour 'on condition' check of the tail rotor gearbox, as detailed in the Maintenance Manual. At this time the aircraft had accumulated 14,105 airframe hours (14,033 IHUMS hours). This work requirement was passed by the shift supervisor to one of the licenced aircraft engineers. At that time the IHUMS log report showed a suspected shaft imbalance defect associated with the tail rotor gearbox output shaft. This warning was triggered by the level of vibration associated with the fundamental shaft order (ie one per revolution) of the tail rotor gearbox output shaft exceeding the predefined level of 0.3g, which had occurred some 5 hours earlier. The graph of vibration level versus IHUMS airframe hours was selected manually by the ground station operator, printed out and passed to the maintenance section for investigation of possible causes. This printout, shown in Appendix C, showed a rising trend in vibration level from 50 flying hours before the incident, whereas the warning triggering point occurred only 5 hours before the start of the incident flight. The engineer was asked to inspect the tail rotor gearbox generally to see if any associated cause could be found. In accordance with the normal procedure for the 3,000 hour check, he cleaned the hub using rag, or paper towels, and removed the flapping hinge pins and the inner races. He inspected the inner races and roller bearings and found them to be satisfactory. He then noted that there was vertical 'play' in the tail rotor shaft assembly, which he suspected to be due to wear in the pitch change 'spider'. After removal of the spider, he established that the play was within the gearbox itself. As he considered this to be abnormal, he advised the shift supervisor that the gearbox was unserviceable and discontinued any further work. He raised Additional Work Sheet (AWS) line entries which covered the removal of the tail rotor blades, pitch change spider and various fairings. He did not specifically look at the inboard face of the tail rotor shaft for damage, nor was he required to do so by the 3,000 hour check requirements At this time the crack was probably about 59 mm long. This work was carried out during the day shift.

Over the weekend period, the manufacturer was informed of the gearbox shaft play. The shift supervisor determined that the gearbox could remain in service if the vibration could be reduced by rebalancing. Another engineer was tasked with re-assembling the tail rotor. That engineer correctly completed the earlier AWS line entries and certified the work, also conducting one of the two duplicate inspections. The reassembly and first duplicate inspection were conducted during the night shift. A third engineer carried out the second duplicate inspection. On 26 September, the tail rotor was re-balanced. 48 grams were added to the Black blade, and 30 grams to the White blade; the total weights on each of the blades and the distribution of the weights were within Maintenance Manual limits. Following this rebalancing, the vibration had reduced to an acceptable level and the shift supervisor was satisfied that the gearbox could continue in service. The associated documentation reflected that all the requirements of the Maintenance Manual had been complied with. No further investigation of the IHUMS vibration indication was made. Since the 3,000 hour check had thus been completed and a line entry made and signed off, the aircraft was released to service.

After a short test flight, a further download resulted in an IHUMS log report which detailed two data integrity warnings. These warnings arose from omitting to redatum the IHUMS system after rebalancing. The IHUMS log report showed that 26 grams had been added to the White blade and 42 grams to the Black blade. These weights were calculated from the IHUMS vibration analysis and were very close to the actual weights added. No further relevant IHUMS data on tail rotor vibration were printed or passed to the maintenance section before the incident flight, although the related information was extracted during the investigation and a composite graph compiled from the airframe accelerometer data. This graph is shown as Appendix K and compares the vibration levels observed 55 flying hours before the start of the incident flight (13,975 IHUMS hours) to those observed at 14,017, 14,021 and 14,029 IHUMS airframe hours on 21 and 22 September before the accelerometer crosswiring was rectified and the tail rotor balanced. The graph shows an increasing level of vibration in both the lateral and vertical planes of the tail rotor as time progressed.

After the helicopter had departed Aberdeen on the morning of the 27th, a reply was received from the manufacturer advising that the gearbox be withdrawn from service.

The 3,000 hour check had been carried out in the hanger with normal staging and lighting, however to see the crack the engineer would have had to place himself between the tail rotor and the pylon at which point he would have had limited visual access to the crack if it was on the far side of the rotor, and none if it was on the near side.

The lighting in the hangar where the work had been carried out consisted of sets of high mounted sodium discharge lamps and fittings, typical of the style and intensity of lighting used in many hangars. The Health and Safety Executive (HSE) publishes 'Lighting at Work' (HS/G38) which suggests that lighting intensities of 50 to 100 lux, typical of many hangars, is sufficient for 'work requiring limited perception of detail' and that a minimum of 200 lux and an average of 500 lux is required for 'work requiring the perception of fine detail'. The HSE recommended levels of illumination are for the avoidance of visual fatigue. Additionally, the proximity of the tail rotor assembly when working on the staging used precluded a close inspection of the cracked area. Appendix L is a photograph taken from the position, between the tail rotor and the pylon from which an engineer would have to inspect the rotor. The staging used for access to the tail rotor area was described as being uncomfortable when working close to the tail rotor in order to see the area in question. This involved standing near the edge of the staging between the rotor and the tail boom with the body twisted at an angle. A small gap existed between the staging and the tail boom and due to the need to position the staging close to the tail and past the plane of the tail rotor blades, no handrail was fitted on that side. Also, when standing at the edge of the staging, albeit with the tail boom itself providing support and security, the staging was in its least steady condition, contributing to a feeling of unease. From this position the cracked area could not be seen without a mirror if it was on the nearest blade, and would be partly obscured and several feet away if inspecting the other blades.

1.18.4 Fleet history

By December 1994 the Puma and Super Puma fleet comprised a total of 1,103 aircraft, which had accumulated a total of 3,784,100 flying hours. The Super

Puma AS332 fleet of 398 aircraft had accumulated 1,177,300 hours. This incident arose from the only recorded case of fracture of the tail rotor hub flapping hinge retainer.

1.18.5 Public address system amplifier

The public address system amplifier was mounted in the tail boom of the aircraft in an anti-vibration mounting rack. During the incident, the system had failed to operate. On inspection it was found that the anti-vibration mountings had failed and the rack itself had suffered significant damage due to the amplifier moving, predominantly in a vertical plane. The amplifier was stripped and repaired by an approved organisation, which reported that the unit had ceased to operate as a result of several defects, possibly related. The most significant of these was the release inside the unit of a large capacitor. This showed evidence of repeated impact damage from striking other components within the unit, and also evidence of electrical arcing. An associated fuse had failed and the report indicated that this had been due to contact between the capacitor positive terminal and the amplifier case. In addition, an integrated circuit within the power supply section had failed. It was also noted that a variable resistor, which provided overall 'gain' adjustment and was accessible from outside the case by means of an appropriate screwdriver, had been damaged. The associated adjustment screw had been broken-off at the top of the case in a manner suggestive of unskilled use, or through use of an inappropriate tool. It was further noted that, as received, the security seal was broken and there was no evidence of the tiewraps which should have been used to secure the capacitor to its mounting plate. Subsequent enquiries found that, following the incident and before the seriousness of the event had been fully realised, the amplifier had been opened-up for inspection. A check of all the public address amplifiers in the operator's possession was conducted to inspect the security of the capacitor to its mounting plate. No similar cases were found.

1.18.6 Post-incident airworthiness promulgations

On 30 September 1995 the manufacturer issued to all operators Information Telex No 10043/0272/95 calling for a visual inspection of the flapping hinge retainers for cracks before next flight, in addition to subsequent daily visual inspections.

On 29 January 1996 the manufacturer issued Mandatory Service Bulletin No 05.00.41. This was approved by the French Direcion Generale de l'Aviation Civile (DGAC) and required visual or other locally approved inspections daily after the last flight, and detailed operator-required actions in the event of a crack being found. It also advised of corresponding revisions to the Maintenance Manual (MET) and Maintenance Requirements (PRE).

On 27 March 1996, the DGAC issued an Airworthiness Directive number 96 074-057(B), effective date 6 April 1996, which mandated the requirements of Service Bulletin No 05.00.41. This was transmitted to UK operators by the CAA in a letter dated 4 April 1996.

1.18.7 Detection of damaged flap bearings using IHUMS

Shortly after this incident, the IHUMS system successfully identified a worn flap bearing in the tail rotor of another Bond AS322L, G-PUMB, when trend information showed a steadily increasing vibration in the vertical plane. When the flap bearings on that helicopter were stripped, one was found to be 'brinelled'. The similarity of the IHUMS trace to that of G-PUMH just before the incident flight was marked. The trace from MB is shown at Appendix M.

Subsequently another operator successfully identified three brinelled flap bearings on three of its Super Puma helicopters using similar information obtained from its IHUMS system.

1.18.8 Vibration monitoring and previous safety recommendations

The AAIB has previously made a number of safety recommendations on the subject of vibration monitoring.

Following an accident to a Chinook, G-BWFC, on 6 November 1986 the AAIB recommended that **'The CAA should report on the progress that has been made towards the early incorporation of a specification for suitable condition monitoring systems into airworthiness requirements for helicopters and indicate the timescale and scope of likely developments.'** (AAIB AAR 2/88 para 4.2). The report stated : *'The CVR tape showed an abnormal frequency signature throughout its duration..... The abnormality was only audible to the crew for the final 60 seconds, by which time they were unable to take any action to prevent the accident.'* (para 3(a)(viii)).

After an accident to a Bell 222, G-META, on 6 May 1987 the AAIB recommended that the CAA **'Require, for all aircraft types, the early provision of a facility to continuously monitor the vibration of all high speed rotating equipment whose integrity is critical to flight safety.'** (AAIB AAR 3/88 para 4.13). The report stated: *'It was considered likely that the developing failure.... would probably have generated an abnormal vibration signal, possibly over an extended period of operation.'*

A similar safety recommendation was made following an accident to a Sikorsky S61N, G-BEID, on 13 July 1988. The recommendation stated that the CAA should **'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.'** (AAIB AAR 3/90 para 4.14). In the associated findings, the report stated: *'Spectral analysis of the CVR recording revealed abnormalities in the vibration signature from 15 minutes before the bang.'*

Subsequently, following an accident on 10 November 1988 to another S61N, G-BDES, the AAIB recommended that **'The Civil Aviation Authority require, for all public transport helicopters, the provision of a facility to continuously monitor the vibration/audio 'signature' of all high speed rotating equipment whose integrity is critical to flight safety.'** (AAIB AAR 1/90 para 4.7). In the associated findings, the report stated: *'A vibration/audio monitoring system optimised to detect changes in the characteristic 'noise' emitting from the transmission might have been able to provide the crew with a warning early enough to have avoided a ditching or at least to have permitted a more controlled one.'* The recorded data indicated that the noise was detectable from about 8 minutes after lift-off, for about 20 minutes.

A further accident on 11 May 1989 to a Sikorsky S61N, G-BFFJ, caused the AAIB to state: **'It is recommended that UK public transport S61N helicopters be fitted with a means of continuously monitoring the health of the main gearbox input pinion plain bearings'** (AAIB Bulletin 7/90). The report stated that a signal of abnormally high amplitude was present throughout the period of engine running. The total period, from start to shutdown, was only 4 or 5 minutes. The aircraft had earlier flown 50 minutes.

Following an accident on 9 October 1990 to another S61N, G-BCLD, the AAIB recommended that **'The CAA 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.'** (AAIB Bulletin 12/91). The report stated: *'The evidence suggests that a system capable of monitoring gearbox vibration....would....probably be able to provide adequate pre-warning of excessive deterioration.'* The report indicated that this information should be available to the flight crew.

On 30 December 1990 an AS355-F2 Twin Squirrel, G-WMPA, crashed following failure of a high speed input to the main gearbox. The associated AAIB Bulletin recommended that the CAA consider requiring **'The early provision**

of a facility to monitor continuously the vibration of high speed rotating equipment whose integrity is, or may foreseeably be, critical to flight safety.' (AAIB Bulletin 12/91). This Bulletin stated that the vibration levels would have been abnormally high for some time.

Because of a wider concern about helicopter safety standards in the late 1970s and early 1980s, the CAA formed the Helicopter Airworthiness Review Panel, which reported in June 1984 (CAP 491). The HARP report contained several pertinent statements:

Paragraph 8.5 (page 23) *'What the panel would wish to propose is a philosophy based on the argument that where full redundancy is not possible in the design of helicopters warning of likely failure (at some reasonable period ahead in time, maybe only an hour or two) could provide the equivalent overall safety level.'* and *'We would hope to see airborne equipment with computerised analysis to simplify the display of information to the crew.'*

Also, *'Telemetry....may have potential on helicopters to remove some responsibility for monitoring critical elements from the pilot to a ground specialist.'*

The HARP report recommended that a working party should be set up with experts from the CAA Airworthiness Division, the Ministry of Defence and specialists from universities and industry, to draw up proposals for the parameters to be measured and to consider requirements for new or improved condition monitoring devices. The Report of the Working Group on Helicopter Health Monitoring, CAA paper 85012, was published in August 1995. The report recognised that 'major benefits' could be obtained not only on future helicopters, but by retrospective application to those currently in service. Five sub groups were set up to consider different areas of helicopter design. One of the terms of reference given to each sub group was *'To determine whether awareness is necessary on the flight deck., or during maintenance activity (or both), and recommend suitable presentation.'* It is not clear from the report how this matter was addressed, however the remainder of the report gives the impression that flight deck indications were not perceived as the primary means of addressing this issue. Nonetheless, the report stated: *'In-flight indications should be restricted to 'need-to-know' information relevant to the current flight. Between flights maintenance information should be displayed to ground crews. Reliance upon off-aircraft analysis of health monitoring data should be minimised. For techniques requiring off-aircraft analysis, the consistency in analysis, advanced warning capability and sampling intervals should be such that airworthiness is not compromised.'* (Report of the Industry sub-group, para 7.10). In the main body of the report, in the conclusions, para 4.1 (vii) stated: *'Flight deck indications from the Health Monitoring systems need to be integrated with the warning philosophy for the particular aircraft and should normally be limited to parameters where there is a 'need-to-know'.*

Other CAA responses have included discussion papers and consultation with the FAA, JAA and industry on retrospective requirements; the establishing of a Helicopter Health Monitoring Advisory Group; research and development programmes and operational trials; and the issue of BCAR Section G Blue Papers G778 and G780 during 1985, which included requirements for rotor and transmission design assessments. Early warning, i.e. health monitoring, credits were clarified in January 1989 in G811. Also relevant has been the introduction of Design Assessment requirements in JAR 29 and specific mention of health monitoring as a 'compensating provision' in the associated Advisory Circular Joint (ACJ) material. The most significant outcome of this activity has been the introduction of HUMS. Much of this activity has been directed at new design, with the involvement of the airframe manufacturers. For existing types, development of HUMS and particularly the definition of maintenance actions related to specific indications, has been inhibited by the difficulties of obtaining and integrating design data from the airframe manufacturers. Currently, for the AS332L there are no mandatory systems or specifications, no requirement for retrospective action and no in-flight indications of vibration available to crews. A structured mechanism does exist within the certification process in the form of the Design Assessment which could maximise the effectiveness of HUMS through proper integration with helicopter maintenance programmes. The CAA have declared their intent for retrospective application of Design Assessment to current helicopter types.

2 Analysis

2.1 Flight crew actions

Although the commander, whose appointment was only temporary, was nearing the time when he was due to revert to first officer status this did not result in any problem on the flight deck and the crew co-operated well during the emergency. The commander permitted the first officer to continue as handling pilot and the latter responded by flying the aircraft to a high standard of accuracy.

2.1.1 Crew analysis of the vibration

If the crack in the tail rotor flapping hinge retainer had led to detachment of the Blue tail rotor blade in flight, the resultant massive rotational out-of-balance forces would have caused rapid detachment of the tail rotor and its gearbox. With the advantage of hindsight, the safe option would have been to have ditched the helicopter as soon as possible after the onset of the vibration, however separation of a tail rotor blade is considered to be an extremely improbable event. Had the commander suspected this possibility, there is no doubt that he would have immediately ditched the aircraft.

His initial diagnosis of a frequency adapter failure as the source of the vibration may have been stimulated by the fact that it was the only drill in the FRC related to vibration. There had been only one recorded previous case of such a failure, in July 1987, and the drill had been written as a result of this; the frequency adapter design was subsequently modified to reduce the probability of such a failure. Although reducing collective pitch to 12°, the first action in the drill, did appear to reduce the vibration slightly, there was little or no lateral component present. It may be that the drill tended to mislead the crew rather than assist them in their attempted diagnosis. This raised the question of whether it was necessary to retain a drill in the FRC which was specific to failure of a frequency adapter, if such an occurrence is now unlikely.

Because the cyclic, collective and yaw pedal controls act through hydraulic servo units any feedback from them, which might have helped the crew in their attempt to identify the source of the vibration, would have been reduced; the vibration felt at the controls seemed to be in sympathy with that felt generally through the airframe. With the controls appearing to function normally, the only source of evidence was the nature and frequency of the vibration and its relation to main rotor (265 RPM) or tail rotor (1,279 RPM) rotational speeds. The main rotor blades rotate 4.4 times per second and therefore a 1R vibration would have a frequency of 4.4 Hz; a 4R vibration would have a frequency of 17.6 Hz. A vibration associated with the tail rotor (1T) would have a frequency of 21.3 Hz. The closeness of these frequencies is such that a pilot would not normally be able to differentiate between a 4R vibration and a 1T vibration and, in the absence of any other evidence, the problem could be associated with either rotor system. In addition, various frequencies will beat together to produce a resultant vibration which could have a frequency close to one of the basic frequencies such as, for example, the 4.4 Hz of the 1R case. It is thus extremely difficult for crew to attempt to analyse the source of a vibration from its frequency alone.

Although, as the flight progressed, the crew became less convinced about a frequency adapter failure, in the absence of any other evidence they continued to regard the vibration source as associated with the main rotor; the fact that there was no 'buzz' through the yaw pedals appeared to have been the main factor which supported their belief.

The Operations Manual gives no guidance to help pilots in determining the source of an unusual vibration and it is generally felt that it would not be practical to do so because of the large number of variables involved.

2.1.2 Commander's decision to return to the mainland

The commander's decision to prepare for immediate ditching but to continue the flight, while the aircraft appeared to respond normally to control inputs, was reasonable. His decision to track towards the nearest point on the mainland rather than to the Buchan platform was sound. The weather and the possibility of a major failure while landing on the platform were the main factors he took into consideration. The first officer was in agreement with both these decisions.

2.1.3 Declaration of the emergency

A subsequent action in the checklist, following the frequency adapter drill carried out by the crew, required the non-handling pilot to transmit a PAN call. The commander did not do this after completion of the drill since he was endeavouring to diagnose the source of the vibration and preparing for a return to land. However a PAN call was prompted by the ATC controller at 0736 hrs, although clearly a state of at least Urgency existed and was tacitly assumed by both the commander and the controller from the start of the incident. It is, however, common for pilots to understate an emergency. In this incident, the commander upgraded the state to Distress at 0752 hrs, 22 minutes after the onset of the vibration. The MAYDAY call stimulated the immediate diversion of a rescue helicopter to the scene, however as it had over 100 nm to fly, 32E had landed before it arrived in the area.

The commander was aware of the possibility of having to ditch at any time and it is considered that it would have been prudent for the commander to have made a MAYDAY call at the onset of the vibration, thus alerting the emergency services some 20 minutes earlier.

2.1.4 Evacuation procedure

The commander's planned evacuation procedure was to establish the helicopter in a low hover with the landing gear wheels touching the ground and to instruct the passengers to leave through the left door only. He thought that it would be less confusing for the passengers if they used the same door as for normal entry and exit. However, this then necessitated the instruction to leave 'one-at-a-time' to assist lateral balance. While it may have been more efficient to evacuate the cabin through both doors, which would have alleviated the balance problem and speeded egress, a risk inherent in this would have been that a passenger using the right door could have, in the heat of the moment, turned right and moved dangerously close to the tail rotor.

The approach, low hover and landing were all skilfully flown and the prime objective was achieved in that the aircraft sustained no further damage and both the passengers and crew escaped without injury.

2.2 Area microphone noise analysis

2.2.1 Analysis method

The audio recording section of the flight recorder fitted to the aircraft was a magnetic tape based system which had a defined audio bandwidth of 150 Hz to 5,000 Hz. Within this range the frequency response of the system was essentially 'flat'; ie the replayed level was the same as the recorded input level. For audio frequencies outside this range, the signal did not record as well and so replayed at a lower level than the original source. This effect was progressively more marked with frequencies well outside the defined band with. In addition, very low replay levels are indistinguishable from the inherent system 'noise'. The audio frequencies of interest in this helicopter incident originated from the main and tail rotors, and ranged from around 4 Hz to 100 Hz, all of which fell below the stipulated bandwidth of the flight recorder. However, by taking a number of samples of the low level signals and averaging them, the general background noise level could be artificially lowered, thus enhancing the discrete frequency peaks of interest.

Samples of the audio signal recorded from the cockpit mounted area microphone were assessed for their frequency content at discrete points during the incident flight. In each case, four successive samples of the signal were taken by a dynamic signal analyser. The analyser calculated a root mean square (RMS) average of the four samples to enhance the dominant frequency components. The resulting frequency spectrum was then displayed and plotted using the same vertical and horizontal scales in all cases for comparison purposes. No correction for the out-of-band signal reduction was made and so it should be noted that the signal peaks towards the left of the associated plots (see below) appear lower than was actually the case. Allowing for this 'frequency weighting', the height of each peak can be taken as a guide to the amount of each frequency component within the overall recorded signal.

2.2.2 Audio analysis results

The first averaged sample was taken at the point just prior to the onset of the severe vibration. The frequency spectrum plot, shown in Appendix F, Figure 1, shows the fundamental frequency of the main rotor blades passing the aircraft structure at 17.375 Hz (4R), together with the first four harmonics at 34.75 Hz, 52.125 Hz, 69.5 Hz and 86.875 Hz. No other discrete frequencies were discernible.

In contrast, the second sample, shown in Appendix F, Figure 2, was taken after the co-pilot had lowered the collective lever from 15.6° to 12.2° as a result of the onset of the severe vibration. The main rotor 4R fundamental and associated harmonics are reduced due to the reduction in the angle of attack of the main rotor blades. However, a further set of discrete frequencies were then dominant, ie those of the tail rotor. At 21 Hz, the frequency component resulting from each revolution of the tail rotor (1T) is visible. This signal had harmonics at 42 Hz, 63 Hz and 84 Hz. All of the frequency peaks, other than those above, are separated by 3.625 Hz and are the 'beat frequencies' of the 4R and 1T signals, and their harmonics.

Following the further reduction of the collective to 11.1°, the plot in Appendix F, Figure 3, shows a significant reduction in the level of the 1T fundamental at 21 Hz. This was also due to the reduction in angle of attack of the tail rotor blades resulting from the first officer applying less right yaw pedal. If a vibration level is kept constant and only the frequency altered then the induced movement halves for every doubling in frequency. It is for this reason that, although the level of the vibration harmonics remained relatively constant after the collective was lowered from 12.2 to 11.1°, the crew perceived only the reduction in the 1T fundamental.

The combination of believing that the reduction in collective alone had reduced the severity of the vibration, together with the near impossible task of distinguishing a 17.375 Hz (4R) vibration from a 21 Hz (1T) vibration led the crew to believe that their aircraft had suffered a frequency adaptor failure.

During the return to Longside the first officer maintained the collective lever position at around 12.5°. Some 30 minutes after the severe vibration had started, both crew felt that the vibration had increased slightly and so declared a MAYDAY. Analysis of the audio sample taken just after the MAYDAY call does not show an increase in the level of the 1T fundamental, although the 4R fundamental is reduced thus emphasising the vibration component at 21 Hz originating from the tail rotor.

2.3 Fracture mode of the tail rotor shaft

2.3.1 Metallurgical findings

The metallurgical reports showed that the tail rotor shaft was of the correct material specification, 30 NCD 16 steel, with (according to one of the reports) a strength slightly below specification at 1,174 MPa. This was not considered to be significant. The reports identified several pertinent features on the fracture surfaces. Growth in pure fatigue had occurred over some 200 or more cycles determined by counting the associated striations on the fracture faces. Near the origin it was not possible to discern the striations, and it was apparent that these 200 or so cycles which were discernible began more or less coincident with the time that the crack had reached the surface. It was not possible to identify how long the crack had existed before then, although it is considered likely to have been a considerable time. Until the crack had penetrated the surface for a significant distance, it was not detectable during daily visual inspections. The metallurgical reports also identified a significant degree of corrosion and/or fretting damage on the bore of the flapping hinge retainer.

2.3.2 Fretting damage

The manufacturer advised that the degree of fretting damage was considerably less than that generated during certification testing, and could therefore be excluded as the primary reason for failure. During certification tests, the fretting had been produced by cyclic blade loads in the drag plane and testing had established the levels of fretting damage required to materially reduce the fatigue life. Fretting which arises from relative movement between the bushes and the bore would not normally occur due to blade flapping unless the flap bearing was stiff. Since the possibility of a flap bearing failure had been considered during design and recognised as potentially catastrophic, this risk was controlled by the requirements of the daily inspections.

2.3.3 Corrosion damage

The manufacturer's experience of in-service fleet operation included cases where considerable corrosion had occurred and it was concluded that the corrosion pitting in this case, which was about 7 to 10 microns in depth, was relatively minor and could not alone, or in combination with the fretting damage, account for the initiation of fatigue cracking.

2.3.4 Probable failure mode

The evidence of rotation of the bushes in the bore indicated that the flapping moments had not been eliminated by the flap bearing. The only mechanism identified by which this could occur was if the flap bearing had become stiff at some time. In that case, the unresolved flapping moments would have induced stresses in the flapping hinge retainer. The manufacturer had demonstrated by analysis that the shaft was not able to withstand such stresses, which were capable of inducing eventual failure of the retainer (see Appendix J, J-1). Furthermore, associated rotation of the bushes would have damaged the primer and mastinox sealed surfaces at the retainer faces, allowing the ingress of moisture and the onset of corrosion.

2.3.5 Evidence of flap bearing defects

The records contained evidence of a few flap bearing problems, however not all such problems may have been reported since they may not have been obvious. Any flap bearing defects or rectifications were recorded in the aircraft maintenance documents, making it necessary to review the records of all the relevant aircraft to obtain the full component history. The implication for operators who make use of Power-by-the-Hour schemes is that they will be unaware of earlier problems which could have an effect on the serviceability of components. In this case, it was suspected that an earlier stiff flap bearing had overstressed the flapping hinge retainer, rendering the component unsuitable for continued operation. In this context, the recorded finding of metal particles in the grease of the Blue blade flap bearing in March 1989 (section 1.6.4) was notable. This evidence may have indicated that an unusual bearing failure mode occurred at that time, giving rise to the initiation of the fatigue damage. Such damage would have consisted of micro-cracking, undetectable by normal in-service test methods.

2.3.6 Daily inspections

The daily inspection included moving the tail rotor blades by hand to check the flap bearings for freedom and damage. However such checks do not simulate the effects of centrifugal force (some 6 to 8 tonnes). A defective bearing might therefore become significantly stiff in operation, but remain free when moved during this daily manual inspection.

The experience with ultrasonic and eddy current NDT techniques following this incident, and the predictable crack growth rate, both supported the manufacturer's view that daily visual inspections were adequate. In view, however, of the nature of daily inspections and the size and location of this type of crack, it might be considered advisable to supplement them with periodic dye penetrant checks, for example.

2.3.7 Failure sequence

The evidence of the rotation of the bush and the manufacturer's analysis indicated that at some time in the life of the shaft the Blue blade flap bearing became stiff and initiated fatigue cracking, assisted by fretting and corrosion, within the bore. No means of detecting this damage was available until the crack had reached the surface and progressed sufficiently far to be detectable by visual inspection. When the crack had extended to the external surface and reached perhaps 50 mm in length, it would have started to open up significantly under centrifugal loads, causing the increased vibration which was apparent in the IHUMS traces. It is likely that with the aircraft shut down and the centrifugal loads removed, the crack would have 'closed-up' again. During this period, two 500 hour inspections and some balancing operations were carried out, but the crack was not detected for a variety of reasons, as discussed later. At the last rebalancing, the effect of the undetected crack was eliminated by increasing the weights on the opposite blades. During the subject flight, the crack progressed to such a length that the remaining material fractured in ductile overload. This caused the crack to open up by some 6 mm due to plastic deformation, giving rise to severe vibration in the plane of the tail rotor. This was the first event perceived by the crew.

2.4 Certification implications

As a result of this incident, the service life (safe life) certification basis for this part of the shaft design could no longer be justified and the flapping hinge retainers are now controlled by a 'damage tolerance' philosophy ie on the basis that any retainer which develops a crack will continue to function until such time as the daily inspection of the tail rotor detects the crack. The manufacturer maintains that a margin exists between the crack becoming visible and final fracture which is greater than the normal industry practice for a damage-tolerant critical part.

2.5 Maintenance issues

The aircraft documentation showed that the maintenance requirements of the AMS were fully met. Although this investigation identified factors, such as the staging and lighting, which were not ideal, there was no evidence that these factors affected the failure of inspections to find the crack on the external surface of the flapping hinge retainer.

In practice it is not probable that a hairline crack in an area which is not easy to inspect will be detected, unless there is specific related information available from service history or manufacturer's publications. Although the task was carried out in a hangar under lighting conditions adequate for general purposes, it was less than that recommended for the 'perception of fine detail' by HSE. If a requirement to examine a particular location for cracks had been applicable, the engineer would have been expected to more closely inspect that area, with supplementary lighting if so required. Additionally, the proximity of the tail rotor assembly when working on staging made a close inspection of the cracked area more difficult.

The IHUMS trace which was made available to maintenance engineering was the prime indicator of the impending failure. However, since IHUMS data is used on a 'no hazard no credit' certification basis, it was purely 'advisory'. The engineer involved followed the problem through until he reached the decision to reject the gearbox, at which point he had completed the task in a thorough and responsible manner. The subsequent re-assembly and acceptance of the tail rotor and gearbox assembly was in accordance with published requirements and available information, and so no further work to determine the source of the vibration was conducted, since it was believed that the source was the 'free play' in the shaft. It was unfortunate that the rebalancing was satisfactory, as this gave a false assurance that the problem had been identified and corrected. There remained, however, the question of why the tail rotor should have needed to be rebalanced when there had not been any obvious damage or other change.

The rebalancing appears to have worked because the required weights were applied to blades which balanced the displaced position of the Blue blade, and therefore compensated for that blade operating at a minutely increased radius and offset position as the crack opened under centrifugal load. The IHUMS trace showed a slight 'dip' before the final steady rise and this perhaps indicated that as the crack grew the overall balance initially improved, before starting to deteriorate.

The crack was not found during the work carried out in the period leading up to the incident because at no time did a specific inspection requirement coincide with the crack being of an easily visible length and width. What constitutes a detectable length is a matter of opinion, but the detection of quite long cracks has not always been reliable in the past. The balancing of the tail rotor did not present a realistic opportunity to inspect the retainer surface, although the crack would have been clearly visible at that time. The likelihood of finding the crack during the 3,000 hour check was reduced by the rejection of the gearbox and the criteria specified for its subsequent re-acceptance.

The two 50 hour checks carried out before the incident did not require an inspection of the flapping hinge retainers for cracks (although they did require inspection of the spindles for cracks). These checks were, however, an opportunity for a closer examination of the retainers under reasonably good conditions, and they occur much more frequently than the more extensive checks. The following safety recommendation is therefore made:

> The manufacturer of the AS332L Super Puma should revise the Maintenance Manual and Master Servicing Recommendations to require periodic detailed inspection of the tail rotor shaft flapping hinge retainers for crack indications during the 50 hour inspections described in MET 64.20.00.603 paragraph A3. [Recommendation 98-6]

2.6 Public Address (PA) amplifier

The failure of the PA amplifier was a result of the high level of vibration in the tail boom of the helicopter following the opening-up of the crack. In an emergency situation the availability of the PA system is very important, especially in a case such as this where there was no cabin crew and the flight deck crew were fully occupied with the emergency. The industry is investigating some possible solutions to this PA amplifier vulnerability, such as improving the integrity of the system and the provision of a duplex, or dual redundant, system.

In the light of these observations, the following safety recommendation is made:

> The CAA, in conjunction with North Sea operators, should review the integrity of helicopter Public Address systems and determine the most satisfactory way of significantly improving the reliability of such systems in conditions of severe vibration. [Recommendation 98-7]

2.7 Health and Usage Monitoring System

The operator was using the IHUMS system in the intended manner in which the primary function of the ground based software was to automatically highlight threshold exceedances. Although such a warning was generated five days before the incident, the importance of this was probably obscured by the very high number of false warnings which had previously been experienced. By manually initiating additional software functions, trend data was generated which showed that an unusual and divergent trend existed many hours before the incident. However, with so many parameters being recorded on each aircraft, and the possibility of generating many different types of output trace, it was impracticable for the ground based users of the system to routinely examine trend data.

The concept by which the IHUMS software identifies primarily discrete threshold exceedances does not fully address the reasoning behind the introduction of HUMS, which stemmed from a number of cases where a steadily deteriorating, but detectable, situation had preceded accidents. It follows that, in order to derive maximum benefit from the system, analysis of trend data is at least of equal importance to deriving discrete data. In addition, the discrete data trigger levels were, with hindsight, set too high. The following safety recommendation is therefore made:

> 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 98-8]

For fleet data, the individual datums across the fleet were being manually recorded and plotted on paper, and although this information was seen by the operator as valuable in identifying unusual parameters on a given aircraft, it was difficult and time-consuming to obtain. In view of the demonstrated usefulness of fleet-wide data to the operator, it is further recommended that:

> The CAA should consider 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. [Recommendation 98-9]

A major limitation of existing systems has been the lack of involvement of the airframe manufacturer in providing technical information for the Design Assessment. This has prevented the development of optimised systems for existing types and the development of formal maintenance instructions related to specific indications. Therefore the operator is able only to apply engineering judgement in the light of available data. These limitations are removed for later designs where the manufacturer has been closely involved in the Design Assessment. In this incident the current standard of HUMS was successful in providing an early warning of the incipient failure, but there were no specific and required associated maintenance actions. Other critical failure modes may exist where current IHUMS could provide useful indications of impending failure. The following safety recommendation is therefore made:

> In order to maximise the effectiveness of HUMS through proper integration with the maintenance programmes of existing helicopter types, the CAA should require all group A helicopter types on the UK register to be subject to evaluation against the latest BCAR/JAR rotor and transmission Design Assessment requirements. [Recommendation 98-10]

2.8 In-flight vibration monitoring

From the accidents referred to in section 1.18.8 'Vibration monitoring and previous safety recommendations' it can be seen that the AAIB has frequently called for means of continuously monitoring vibration levels. Although IHUMS does monitor vibration parameters continuously, the information is not presented to the crew. In this instance the period of deterioration was around 50 hours, but in many cases it could be less than the duration of a flight. The crew's analysis of the situation was severely impaired because information which was being automatically recorded throughout the incident was not available to them.

The HARP report accepted the concept of IHUMS derived presentation on the flight deck and considered the amount of reduced data which might be displayed. This concept was carried forward in the Working Group Report, CAA paper 85012. However, subsequently the concept of flight deck presentation has not been developed as part of IHUMS implementation. While it may be (and presumably has been) argued that ground based analysis alone is adequate, such a view can only be sustained on the basis that all potentially critical in-flight failures can be diagnosed successfully in advance by ground based IHUMS analysis. Indeed, this is the very concern voiced by the Industry sub-group when they stated: *'For techniques requiring off-aircraft analysis, the consistency in analysis, advanced warning capability and sampling intervals should be such that airworthiness is not compromised.'* This also assumes that effective maintenance action will always be taken, which can never be safely presumed and was demonstrably not the case in this instance. Had this incident resulted in a critical loss of control due to tail rotor gearbox detachment, with attendant loss of life, the lack of associated IHUMS vibration exceedance information on the flight deck would have been a causal factor.

The following safety recommendation is therefore made:

> 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 (CAP 491) of June 1984.
> [Recommendation 98-11]

3 Conclusions

(a) Findings

(1) The crew were properly licenced, medically fit and rested to conduct the flight.

(2) The crew misidentified the source of the vibration, however this was not due to any shortcoming on their part; the helicopter IHUMS system had no facility to alert the crew to the source of the severe airframe vibration.

(3) The commander's decision to prepare for immediate ditching but to continue the flight while the aircraft appeared to respond normally to control inputs, was reasonable in the light of the information available to the crew, but could have resulted in a critical loss of control had the tail rotor gearbox detached.

(4) The commander's decision to track towards the nearest point on the mainland rather than to the Buchan platform was reasonable on the basis of the level of information then available to the crew.

(5) It would have been prudent for the commander to have declared a state of Distress at an early stage in the event.

(6) The approach, low hover and landing were skilfully flown and the aircraft was successfully evacuated with no injury to crew or passengers.

(7) The tail rotor shaft flapping hinge retainer of 'Blue' blade had fractured in fatigue as a result of abnormal cyclic loads in the flapping plane; this was the first such failure on the AS332L helicopter type.

(8) The tail rotor shaft was within its approved overhaul and total lives and had been properly overhauled on its previous return to the manufacturer's overhaul facility 5 years previously. At that time, some 2,955 hours before the incident, it was dimensionally within limits and no detectable cracks existed.

(9) The crack had taken over 200 flight cycles, representing well over 200 hours, to progress to failure.

(10) The helicopter Integrated Health and Usage Monitoring System (IHUMS) had recorded trend data which began to identify a developing vibration problem some 50 hours previously, culminating in an associated exceedance alert some 5 hours before the incident.

(11) Maintenance activity to identify the cause of the tail rotor vibration problem failed to identify the cracked retainer. This work was in excess of the mandatory maintenance requirements of the Approved Maintenance Schedule.

(12) The accessibility and associated hangar lighting levels, although typical for this type of work, did not assist the detection of the crack in the absence of specific information about its location.

(13) The ground based element of the IHUMS system required operator intervention for the detection or plotting of trend data on an individual aircraft, and no such facility was available to facilitate comparison of a specific aircraft with the fleet as a whole.

(14) As a result of this serious incident, the certification basis of the flapping hinge retainer has been changed from a 'safe life' philosophy to one of 'damage tolerance', reliant on daily visual inspections of such flap bearing lugs.

(15) The public address amplifier failed because of excessive tail boom vibration in the vertical plane generated by the tail rotor during the return to the mainland.

(b) Causes

The investigation identified the following causal factors:

(1) Maintenance inspections conducted over a period prior to the incident flight did not detect a developing surface crack in the Blue tail rotor blade flapping hinge retainer, despite additional work on the associated tail rotor drive shaft assembly to rectify a tail rotor vibration problem, which was detectable as a trend recording within the Health and Usage Monitoring System some 50 flying hours previously and was the subject of an associated alert 5 hours before the incident.

(2) The undetected fatigue crack extended during the flight, fracturing one side of the retainer and causing excessive and potentially critical tail rotor vibration.

(3) The fatigue crack had been initiated by fretting and corrosion of the retainer bore induced by abnormal cyclic loading of the retainer, which was attributed to the effects of a defective flap needle-roller bearing during some previous period of the tail rotor drive shaft's life.

(4) The inspection provisions within the aircraft Maintenance Manual and associated Maintenance Requirements did not specify periodic visual inspections of such retainers, since they had been designed and certificated on a 'safe-life' basis.

4 Safety recommendations

The following safety recommendations are made:

4.1 The manufacturer of the AS332L Super Puma should revise the Maintenance Manual and Master Servicing Recommendations to require periodic detailed inspection of the tail rotor shaft flapping hinge retainers for crack indications during the 50 hour inspections described in MET 64.20.00.603 paragraph A3.
[Recommendation 98-6]

4.2 The CAA, in conjunction with North Sea operators, should review the integrity of helicopter Public Address systems and determine the most satisfactory way of significantly improving the reliability of such systems in conditions of severe vibration.
[Recommendation 98-7]

4.3 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 98-8]

4.4 The CAA should consider 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.
[Recommendation 98-9]

4.5 In order to maximise the effectiveness of HUMS through proper integration with the maintenance programmes of existing helicopter types, the CAA should require all group A helicopter types on the UK register to be subject to evaluation against the latest BCAR/JAR rotor and transmission Design Assessment requirements.
[Recommendation 98-10]

4.6 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 (CAP 491) of June 1984.
[Recommendation 98-11]

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