AIRCRAFT ACCIDENT REPORT 6/94

Air Accidents Investigation Branch

Department of Transport

Report on the accident to Piper PA-31-325 C/R Navajo, G-BMGH 4 nm south east of King's Lynn, Norfolk on 7 June 1993

This investigation was carried out in accordance with
The Civil Aviation (Investigation of Air Accidents) Regulations 1989

London: HMSO

First published 1994

ISBN 0 11 551652 2

LIST OF RECENT AIRCRAFT ACCIDENT REPORTS ISSUED BY AIR ACCIDENTS INVESTIGATION BRANCH

1/93	Piper PA-28-161 Cadet, G-BPJT, at Oxford Airport, Kidlington, on 12 July 1992	April 1993
2/93	AS 332L Super Puma, G-TIGH, near the Cormorant 'A' platform, East Shetland Basin, on 14 March 1992	May 1993
3/93	Lockheed 1011 Tristar, 9Y-TGJ, near 'KIRN' VOR, Germany, on 9 March 1992	June 1993
4/93	British Aerospace 146-300, G-UKHP, at Aberdeen Airport, Dyce, Scotland, on 31 March 1992	August 1993
5/93	British Aircraft Corporation/SNIAS Concorde 102, G-BOAB, over the North Atlantic, on 21 March 1992	November 1993
1/94	Aerospatiale AS355F1 Twin Squirrel, G-OHMS Near Llanbedr Airfield, Gwynedd, on 8 December 1992	January 1994
2/94	RAF Tornado GR1, ZG 754 and Bell 206B JetRanger III, G-BHYW at Farleton Knott near Kendal, Cumbria on 23 June 1993	June 1994
3/94	Boeing 737-2Y5A, 9H-ABA at London Gatwick Airport on 20 October 1993	June 1994
4/94	Boeing 747-243, N33021 at London Gatwick Airport on 7 February 1993	August 1994
5/94	Cessna 550 Citation II, G-JETB at Southampton (Eastleigh) Airport on 26 May 1993	July 1994

These Reports are available from HMSO Bookshops and Accredited Agents

Department of Transport
Air Accidents Investigation Branch
Defence Research Agency
Farnborough
Hampshire GU14 6TD

12 October 1994

The Right Honourable Brian Mawhinney
Secretary of State for Transport

Sir,

I have the honour to submit the report by Mr E J Trimble, an Inspector of Air Accidents, on the circumstances of the accident to Piper PA-31-325 C/R Navajo, G-BMGH, which occurred 4 nm south east of King's Lynn, Norfolk, on 7 June 1993.

I have the honour to be
Sir
Your obedient servant

K P R Smart
Chief Inspector of Air Accidents

Contents **Page**

Contents (continued) Page

Contents (continued) **Page**

5 Appendices

6 Figures

Contents (continued)

6 Figures

GLOSSARY OF ABBREVIATIONS USED IN THIS REPORT

AAIB	Air Accidents Investigation Branch
AD	Airworthiness Directive
amsl	above mean sea level
AN	Airworthiness Notice
°C	degrees Celsius
CAA	Civil Aviation Authority
CAVOK	ceiling and visibility OK
cg	centre of gravity
DRA	Defence Research Agency (formerly RAE)
FAA	Federal Aviation Administration
FAR	Federal Aviation Regulation
hrs	hours
Hv	hardness-Vickers
IACS	International Annealed Copper Standard
KHz	kilohertz
kt	knots
lb	pounds
mb	millibars
MHz	megahertz
mm	millimetre(s)
MPa	megapascals
NDB	non-directional radio beacon
psi	pounds per square inch
RAE	Royal Aircraft Establishment
rpm	revolutions per minute
rps	revolutions per second
SB	Service Bulletin
SEM	scanning electron microscope
TBO	time between overhauls
UK	United Kingdom
UTC	Coordinated Universal Time
VHF	very high frequency

Photograph showing G-BMGH on site

Air Accidents Investigation Branch

Aircraft Accident Report No: 6/94 (EW/C93/6/3)

Registered owner:	Jet West Limited
Operator:	Prospair Air Charter Limited
Aircraft Type and Model:	Piper PA-31-325 C/R Navajo
Nationality:	British
Registration:	G-BMGH
Place of accident:	4 nm south east of King's Lynn, Norfolk Latitude 52° 42' N Longitude 000° 28' E
Date and Time:	7 June 1993 at 1801 hrs All times in this report are UTC

Synopsis

The accident was notified to the Air Accidents Investigation Branch (AAIB) at 1855 hrs on 7 June 1993 and an investigation began the same day. The investigation was conducted by Mr E J Trimble (Investigator in Charge), Mr P D Gilmartin (Operations) and Mr R Parkinson (Engineering).

The aircraft was operating on a scheduled passenger flight from Birmingham to Norwich, with a pilot and seven passengers on board when, as the flight passed south of King's Lynn, there was a loud 'bang' and the aircraft immediately rolled to the right and entered a tight spiral dive, or spin. The loud bang was caused by a blade, that had detached from the right propeller, penetrating the aircraft's nose baggage bay and exiting through the upper left fuselage structure. This blade then struck and removed the front of the left propeller assembly. The right engine tore away from the wing, precipitating the loss of control, and the left engine stopped. The commander managed to regain control of the aircraft and successfully carried out a forced landing in a field of cereal crop. All eight occupants evacuated the aircraft with no serious injury.

The investigation identified the following causal factors:

(i) Separation of the right engine, as a result of massive out-of-balance forces following fatigue failure of the right propeller hub and associated release of one blade, caused an immediate and critical loss of control which was only recovered and a successful forced landing accomplished by the exceptionally skilful handling of this commander.

(ii) The grease nipple holes in such Hartzell HC-()3Y()-() type propeller hubs had not been masked prior to the shot peening process at manufacture and had therefore suffered deformation of the associated threads, which weakened their fatigue resistance.

(iii) No detailed stress calculations from direct strain gauge testing had been undertaken, or had been required, on this propeller hub type at the time of its design and certification.

(iv) Fatigue cracking that emanated from deformed grease nipple threads and which broke through to the outside surface of the propeller hub may not have been visible at the last maintenance inspection prior to the in-flight failure of the right propeller hub.

(v) Despite the occurrence of fatigue cracking from grease nipple holes on such propeller hubs in service which had caused the manufacturer to redesign this type of hub in 1983, some 10 years prior to this accident, and to issue three related Service Bulletins in the period between October 1989 and September 1992 with the introduction of an optional eddy current inspection, in addition to visual inspection, the FAA had only issued one Airworthiness Directive (No. 89-22-05) requiring compliance with the initial SB 165. This had merely required periodic visual inspections and the FAA had not issued revised ADs to include eddy current inspections (as per SB 165A of 27 August 1992) or to emphasise the manufacturer's strong recommendation (SB 165B of 11 September 1992) for the replacement of such hubs with the improved post - 1983 type of hub.

(vi) During the last visual inspection to FAA Airworthiness Directive (AD) No. 89-22-05, no cracking was observed on the propeller hub; the grease nipples had been removed from the hub to facilitate inspection. Such removal was not a requirement of this AD (which did not, however, warn against such removal) and may have tended to 'close up' any crack(s) present, reducing the chances of such visual detection.

(vii) The original hub design was certificated in the knowledge that the vibration stresses on the left-hand rotating propeller of this type were generally higher than those on the right-hand rotating propeller, but were deemed acceptable.

(viii) Operators and pilots of affected aircraft had not been made aware that the sudden initiation of unexplained vibration or grease leakage could indicate a potentially dangerous defect on such propeller hub assemblies although related Service Bulletins had warned aircraft engineers of such symptoms subsequent to 27 August 1992.

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

1 Factual Information

1.1 History of the flight

The aircraft was being operated by Prospair Air Charter Limited, an aerial survey and public transport operator based at Birmingham. Their usual fleet comprised two Piper Navajos, one a -310 series, and the other a -350 Chieftain. The company operated a twice daily round trip scheduled passenger service between Norwich and Birmingham, using one aircraft.

Prior to the day of the accident, one aircraft had been engaged on survey work, and the other was due for a maintenance check. An additional aircraft, G-BMGH, a PA-31-325, was therefore leased from Navajo Support Services Limited based at Biggin Hill, who had in turn leased it from the owners.

The aircraft had been positioned from Exeter to Biggin Hill on 28 May by Navajo Support Services. It had then flown from Biggin Hill for two days of flying work in the period to 4 June. On 6 June, it was positioned to Norwich to take over the Prospair operation.

The commander commenced duty at 0555 hrs on the day of the accident. It was the first time that he had flown G-BMGH. He carried out an uneventful morning return service to Birmingham, and went off on a split duty rest period from 0910 until 1530 hrs. He then returned to duty and carried out an uneventful flight to Birmingham. The turnaround took some 30 minutes, during which seven passengers were boarded for the return flight to Norwich. One passenger occupied the front right seat, alongside the commander.

The aircraft took off at 1731 hrs, and climbed to flight level 50 for the transit. The commander noticed that the right propeller rpm indication showed small fluctuations throughout this flight, as he had also observed on his previous three flights on this aircraft, but there was no yawing associated with these fluctuations. The right propeller pitch lever was also difficult to move when attempting to adjust for propeller synchronisation. The left propeller lever was therefore used to synchronise the propellers. This had also been the method which he had employed on the previous three flights.

During the en route climb, the commander had also noticed some vibration which manifested itself through the control column and through the airframe as a high frequency vibration. He had not previously experienced such vibration on either this aircraft or other aircraft of the same type and was not aware that it was possibly indicative of a potentially dangerous defect on one of the propeller systems. The vibration continued to the top of the climb, and for about the first

10 minutes into the cruise, before it ceased. The commander noted that the vibration did not alter when he adjusted the engines and propellers to their cruise settings at the top of the climb, and did not appear to be associated with any engine indications. The vibration did not recur during the remainder of the flight.

The commander contacted Marham Military Aerodrome Traffic Zone Radar for a Radar Information Service at 1757 hrs. At 1800 hrs, he informed Marham that he was commencing a slow descent towards 3,000 feet, in order to be at that level on reaching the Norwich non-directional radio beacon (NDB). Cruise power had been set up to this point, which the commander recalled as being 31 inches of manifold pressure with slightly over 2,200 propeller rpm, giving an indicated airspeed of around 160 kt. As he slowly retarded the throttles towards 25 inches of manifold pressure, in order to commence the descent, there was a loud 'bang'. The aircraft rolled to the right and entered a steep spiral dive, or spin. The commander managed to regain control of the aircraft after two rotations, initially by use of full left rudder, eased the aircraft out of the steep dive, and reduced the airspeed towards 100 kt, a speed that he considered might be a reasonable glide speed. However, on reaching around 120 kt, the aircraft again began to roll to the right, despite the application of full opposite aileron and rudder. The dive angle was therefore increased to give a minimum satisfactory gliding speed of around 130 kt, which gave a steep glide angle but enabled adequate control for manoeuvring. During the descent the commander saw that the right engine had separated, that there was damage to the nose of the aircraft, and that the blades of the left propeller had stopped and were bent backwards at their roots. He issued a hurried 'MAYDAY' call at 1801:34 hrs, but did not initially indicate the nature of the emergency. The Marham Zone Radar controller immediately passed a heading to steer in order to reach the airfield, which was some 6 nm south south east of the aircraft's position. A second 'MAYDAY' call was transmitted, advising that there had been a double engine failure. The noise and vibration levels were high, however, and prevented the commander from hearing the reply transmissions from the Marham Zone Radar controller.

With the aircraft under control, the commander entered a gentle turn to the left in order to identify a suitable place in which to make a forced landing. The approach to the most suitable field was obstructed by a line of high tension power lines, but in the limited time available and in the absence of a more suitable landing site the turn was reversed towards the field, which was cultivated with a standing green crop. The commander managed to manoeuvre the aircraft so as to avoid the obstructions on the approach to the field. The landing gear was not extended since there was no time to operate the hand pump, and both hands were required to fly the aircraft despite the application of full left rudder and aileron trim. A successful forced landing was carried out into the chosen field, the commander having managed to level the aircraft off just above the surface and allowed the speed to decay until the aircraft sank gently into the crop. The fuselage touched down just as the right wing began to drop with the loss of airspeed. During the subsequent ground slide, the aircraft slewed through 90° to the left and continued to slide sideways to the right until it came to a halt. The aircraft remained upright, intact, and there was no fire.

The commander had attempted to reassure the passengers once control of the aircraft had been regained in the dive, and had shouted "BRACE, BRACE" just prior to the touchdown. He noted that the passengers had adopted the brace position that had been covered during his pre-flight passenger safety briefing. All the occupants quickly vacated the aircraft through both the rear left cabin door and the right over wing hatch. There were no injuries, with the exception of one passenger who subsequently complained of whiplash neck pains. Emergency procedures briefing cards were available for each passenger on board.

The commander returned to the aircraft shortly afterwards and transmitted to the Marham Zone controller that he had landed in a field and that the occupants were uninjured. A Royal Air Force Tornado aircraft which was recovering to land at Marham was requested to search the area in order to ascertain the exact location of the landing site. The crew located the aircraft in the field some 5 minutes later.

Cromer Radar was the closest radar installation to the accident location, but it was unserviceable at the time. A radar replay was obtained from the Debden Radar station, which received the mode 3/A transponder code from the aircraft, but not the mode C height encoding. It was not possible therefore to determine the vertical flight profile of the aircraft after the occurrence, but the radar derived ground track of the aircraft is shown plotted in Appendix 1.

1.2 Injuries to persons

Injuries	Crew	Passengers	Others
Fatal	-	-	-
Serious	-	-	-
Minor/None	1	7	-

1.3 Damage to aircraft

The aircraft was severely damaged in the air and subsequently during the emergency landing.

The airborne damage consisted of separation of the right-hand engine, complete with its cowlings; a horizontal gash (4 inches x 20 inches) through the nose baggage compartment; separation of the left-hand propeller spinner and blade pitch change mechanism; disruption and bending of the left-hand propeller blades; failure of the left-hand engine and bending upwards of the elevator aerodynamic balance horns.

1.4 Other damage

There was minor damage to the crop in the area of the emergency landing and subsequent damage to a hay cutting machine when the separated propeller blade was discovered some two weeks later.

1.5 Personnel information

Commander:	Male, aged 45 years
Licence:	Airline Transport Pilot's Licence
Aircraft ratings:	Piper PA-31, Piper PA-23
Medical certificate:	Class 1, valid until 31 July 1993
Instrument rating:	Renewed on 7 May 1993
Last base check:	7 May 1993
Last line check:	19 May 1993
Flying experience:	Total all types: 4,200 hours
	Total on type: 120 hours
Duty time:	13 hours (including 5 hours 20 min split duty)
Co-pilot:	Not required and not carried
Cabin Attendant:	Not required and not carried

1.6 Aircraft information

1.6.1 Aircraft details

Manufacturer:	Piper Aircraft Corporation, USA
Type:	PA-31-325 C/R Navajo
Airframe serial number:	31-7512045
Date of construction:	1975
Maximum all-up weight:	6,500 lb
Total airframe hours:	3,723:50 hours
Utilisation, hours/cycles:	See Appendix 2 and Section 1.17.1
Engines:	No.1 (left)-Lycoming TIO-540-F2BD piston engine
	No.2 (right)-Lycoming LTIO-540-F2BD piston engine
Engine hours:	No.1 (left)-1,974:20 hours; on 20% extension
	No.2 (right)-1,974:20 hours; on 20% extension
Time between overhauls (TBO):	1,800 hours
Propellers:	
No.1 (left):	
Manufacturer:	Hartzell Propeller Inc.
Type:	HC-E3YR-2ATF
Date of manufacture:	Not known
Serial number:	DJ4311

Hours:	
Since manufacture:	3,660:49 hours
Since overhaul:	870:00 hours
Since Airworthiness Notice 75 inspection:	62:10 hours
Since last FAA AD 89-22-05 inspection:	10:35 hours
No.2 (right):	
Manufacturer:	Hartzell Propeller Inc.
Type:	HC-E3YR-2ALTF
Date of manufacture:	1977
Serial number:	DJ4256
Blade serial numbers:	D59183, D59198, D59261
Hours:	
Since manufacture:	5,195:40 hours
Since overhaul:	508:55 hours
Since Airworthiness Notice 75 inspection:	508:55 hours
Since last FAA AD 89-22-05 inspection:	10:35 hours
Certificate of Airworthiness:	Transport Category (Passenger) Issued on 20 November 1992 Valid until 9 August 1993

Certificate of Registration:	Jet West Limited, Exeter Airport, Devon, UK Issued 7 June 1990
Certificate of Maintenance Review:	Issued on 28 May 1993 at 3713:15 hours and valid until 3,813:15 hours or 9 August 1993

Airframe:

Maximum weight authorised for takeoff:	2,948 kg
Actual take-off weight:	2,859 kg
Estimated weight at time of accident:	2,814 kg
Estimated fuel remaining at time of accident:	168 kg
Centre of gravity (cg) at time of accident:	Within normal loading envelope

1.6.2 General description of the propeller

The Hartzell HC-()3Y()-() three-bladed propeller is known as a 'Compact' type and was a development of the manufacturer's two-bladed Compact propeller which was their first type to use a forged aluminium alloy hub. This type of propeller represented a new concept in basic propeller design. It combined low weight with simplicity in design and rugged construction. The hub was made as compact as possible, utilising shot peened aluminium alloy forgings of 2014-T6 composition for most of the parts. The hub shell was made in two halves (Figure 1), bolted together along the plane of rotation, and carried the pitch change mechanism and blade roots internally. Lubrication of the blade pitch change bearings within the hub was accomplished by injecting lubricant through grease nipples mounted in the hub. These grease nipple ('zerk') holes, two per blade root, were drilled and threaded as tapered holes through the thickness of the hub prior to shot peening treatment of the hub surfaces (Figure 2). At the time that this hub was designed and manufactured there was no procedure to mask, or 'plug', the grease nipple holes prior to the shot peening process. The current manufacturing practice is to plug all holes prior to shot peening. The actuating cylinder, which provides power for changing the propeller pitch, is located at the front of the hub. The propeller utilises nitrogen, or air pressure, spring and blade counterweight forces to move the blades to the high pitch and feather positions; the opposing governor-regulated oil pressure moves the blades to the low pitch positions.

The hubs were produced with a number of extensions, dependant upon the engine and airframe manufacturers' requirements. These extensions varied the distance between the hub and the engine propeller flange. This particular propeller had the 'E' type extension, which added 5 inches to the basic hub extension length.

The original hub design was made for a right-hand rotating propeller, ie clockwise rotation (viewed from the rear), but in response to an aircraft manufacturer's requirement, a left-hand rotating model was also produced. The hub forging of the left-hand rotating model was identical to that of the right-hand rotating model.

When the HC-()3Y()-() propeller hub was designed, comprehensive stress and load tests were conducted on the associated propeller blades to meet the certification requirements in force at that time. These tests included static load and fatigue tests, and aircraft vibration tests. During these tests, stress levels were monitored using strain gauges mounted on the propeller blades. However, the loads on the hub were assessed by extrapolating the readings taken from the blades. The stresses on the hub forging were thus not directly strain-gauge monitored. The manufacturer has used strain gauges to monitor the hub stresses during their certification testing on all new hub designs since 1984. There was no certification requirement to conduct detailed analytical stress calculations of the geometry of the hub and none were carried out during the period of its design and production.

The three-bladed propeller was given its Type Certificate in 1965 and was certified to Federal Aviation Regulation Part 35, dated February 1965. The extended hub model of the propeller was given its Type Certificate in April 1968. The left-hand rotating model of the propeller did not require a specific Type Certificate, but was covered by a note on the original right-hand rotating model certificate.

1.6.3 Right propeller maintenance history

The log books for the right propeller recorded that this propeller had arrived in the UK in October 1977, having accumulated a total operating time of 40 hours and 30 minutes since manufacture. Examination of these log books did not show any evidence of the propeller having suffered an overspeed condition, or of any damage having occurred to the blades during the life of the propeller. The serial numbers of the blades fitted to the propeller at the time of the accident were the same as the serial numbers fitted at manufacture. Since manufacture, the propeller had been overhauled on three occasions, all by the same approved overhaul agency in the UK. There were no indications within the documentation associated with these overhauls that any shot peening or rework of the grease nipple hole threads had been carried out. The last overhaul took place in July 1990 during which an examination for evidence of cracks in accordance with the manufacturer's overhaul instructions was carried out by visual and non-destructive testing using dye penetrant techniques. During this overhaul the propeller hub was not painted by the approved overhaul organisation as this procedure was not specified in the manufacturer's overhaul requirements or Service Letter 144F that was applicable at that time. In August 1991, the propeller was fitted to this aircraft to replace a propeller that required a scheduled maintenance check. After the propeller was fitted, the aircraft was given a Certificate of Airworthiness flight test, during which all the documented propeller speeds were within their specified limits. The aircraft was given another Certificate of Airworthiness flight test in September 1992 and again all the documented propeller speeds were within their specified limits. During the last week of May 1993, the aircraft underwent a 50 hour maintenance check (Check 1) carried out by an approved maintenance organisation within the UK. During this check, FAA Airworthiness Directive (AD) No. 89-22-05 (see Section 1.17.6) was satisfied on both propeller hubs and no associated cracks were detected.

1.6.4 General description of the engine

The Lycoming 540 series engine is a six cylinder, direct drive, wet sump, horizontally opposed, air cooled aircraft engine fitted with crankshaft counterweights which was originally designed to drive a propeller in the right-hand direction. The TIO-540 series engine is a derivative of the O-540 with fuel injection, a turbocharger, and fifth and sixth order counterbalance weights. The LTIO-540 series engine is a TIO-540 engine that drives the propeller in the left-hand direction. There are no major differences between the TIO and LTIO series engines, except for the direction of rotation. The engine manufacturer did not carry out any torsional vibration testing of the left-hand rotation 540 series engine during certification of that engine type. They used the results of the torsional vibration tests from the left-hand rotating 360 series engine to support their case for not conducting these tests on the left-hand rotating 540 series engine.

1.6.5 Right engine maintenance history

The right engine had been imported into the UK with the aircraft in 1986 and had accumulated 1,425.35 hours since manufacture. At the time of this accident it had accumulated 1,974:20 hours, 174:20 hours in excess of the manufacturer's recommended time between overhaul (TBO). CAA Airworthiness Notice (AN) No. 35 allows a 20% extension to the TBO hours. This engine met the associated inspection criteria specified in CAA AN No. 35. No log books were available for the engine prior to its importation into the UK. Examination of the UK log book did not reveal that any significant unscheduled work had arisen, except for an oil pump failure which resulted in a loss of engine oil pressure in flight. The pilot had shut the engine down as soon as he was aware of the loss of oil pressure. The engineering organisation involved had replaced the oil pump and carried out a thorough examination of the engine for signs of damage, but none was found.

1.7 Meteorological information

An aftercast for the Gayton area was produced by the Meteorological Office after the accident. This indicated that the surface wind was south westerly at less than 5 kt. The weather conditions were CAVOK, with a surface temperature of +23°C, dewpoint +12°C. The mean sea level pressure was 1,017 mb.

The upper forecast winds were:

250°/11 kt, temperature +12°C at 3,000 feet amsl,
and
250°/16 kt, temperature +10°C at 5,000 feet amsl.

1.8 Aids to navigation

Not applicable.

1.9 Communications

The aircraft was receiving a Radar Information Service from RAF Marham Zone Radar on VHF frequency 124.15 MHz at the time of the accident. Tape recordings and transcripts of the communications were made available for this investigation.

1.10 Aerodrome information

Not applicable.

1.11 Flight recorders

No flight recorders were fitted, and none were required to be fitted.

1.12 Wreckage and site information

1.12.1 Site parameters and wreckage trail

1.12.1.1 Main accident site

The aircraft's initial contact with the crop was 900 feet into the field and had been made by the rear fuselage ground tie-down point, which had produced a witness mark within the crop for a distance of 780 feet, on a heading of 044 degrees magnetic. Approximately 1,700 feet into the field, the left wingtip had cut the crop, initially cutting the top one inch. As the aircraft had continued into the field, the crop had been cut closer to the ground until after a distance of 1,100 feet, (ie 2,800 feet into the field) this wingtip had contacted the ground. The initial ground contact by the left wingtip was of a light grazing nature, but within 150 feet this wingtip had produced a furrow and the fuselage had pivoted to the left. During this initial ground impact sequence, ground marks indicated that the left-hand propeller had not been rotating. The aircraft continued to slide to its right for a further 310 feet on a heading of 035 degrees magnetic, before coming to rest some 3,260 feet into the field. The following parts of the aircraft were missing from the main site: the right-hand propeller and engine, the left-hand propeller spinner and associated propeller pitch change piston and cylinder. In addition, there was a large tear across the nose baggage bay.

1.12.1.2 Wreckage trail

The right-hand engine, complete with the propeller hub, two of its three propeller blades and the associated engine cowlings were located within 24 hours of the accident approximately 3.75 miles to the south west of the main wreckage site (Appendix 1). Around the same time, items of personal property from the nose baggage bay and pieces of the left-hand propeller spinner material were found in an area not far from the right-hand engine and consistent with the track of the aircraft as recorded by radar. Approximately two weeks after the accident, the missing propeller blade from the right-hand propeller, in addition to the left-hand propeller pitch change piston and cylinder, were found by the local farmer whilst cutting the hay crop in a field close to where the right-hand engine had been found.

1.12.2 Examination of wreckage

1.12.2.1 Aircraft

The examination of the aircraft and associated damage indicated that the pilot had carried out the approach and initial touchdown at a relatively high speed with the wings level, nose pitched up and the landing gear and flaps retracted. Examination of the right engine bay showed that the engine had separated from the airframe mainly at the engine frame firewall area and by tearing out three of the four engine frame mounting points at their airframe attachments. All of the engine control and service systems had separated in the area of the firewall. Evidence from the engine frame attachment failures indicated that the engine had departed the airframe in a downwards and outboard direction. The left engine was still attached to the airframe, although both of the top engine crankcase attachment lugs had fractured. Examination of these fractures indicated that they had occurred as a result of the ground impact. There was no evidence of any disruption to this engine's operating systems. The left engine oil contents were checked and found to be sufficient for normal operation. Two of the three propeller blades were found attached to the left propeller hub, the third blade having failed in overload at about one third of that blade's span from its root. The separated blade section was found within the wreckage trail, close to where the aircraft came to rest. All three left propeller blades had severe rearward bending and spanwise scoring, indicating that they had been in a fine pitch position during the initial part of the ground slide. During the last part of the ground slide the propeller blades had been forced into the feathered position. However, there was evidence of the left propeller blades having struck some hard objects whilst they were rotating under power and at a pitch angle which was consistent with normal flight.

Both of the elevator aerodynamic balance horns had buckled in an upwards direction about a line that was almost 45 degrees to the chord line of the elevator. The right balance horn, which contained a balance weight, had buckled through almost 90 degrees. The left horn, which did not contain a balance weight, had only buckled through about 4 degrees. Neither of these balance horns had any witness marks to indicate that the bending had occurred as a result of an impact in the air, or on the ground. There was good evidence which indicated that this bending had occurred in flight.

Examination of the cockpit revealed that neither the fuel, engine fuel mixtures or magnetos had been selected 'OFF' and that the emergency landing gear lowering system handle was stowed, with the cover removed. The aircraft's electrical master switch was selected 'OFF'. The flying control trims were found set with the rudder and ailerons trimmed fully left and the elevator trimmed fully nose-up.

1.12.2.2 Right propeller

Initial examination of the fractured propeller hub was carried out in-situ on the right engine (Figure 3). It became evident during this examination that the fatigue cracks which had caused the failure had initiated at the inner end of a grease nipple hole. The failed propeller hub, together with the blade that had separated, were submitted for metallurgical examination to the Materials and Structures Department of the Defence Research Agency (DRA, formerly The Royal Aircraft Establishment), Farnborough. In addition, and in an effort to resolve certain aspects of the failure progression, these parts were also submitted to HT Consultants, Aldershot, UK and, on behalf of the propeller manufacturers (Hartzell) to A-LAB, Dayton, USA. The results of these examinations are summarised below.

1.12.2.2.1 DRA examination

A general examination showed that the crack had grown to a total circumferential length of approximately 85 mm at the outer surface, and 93 mm at the inner surface, before the final failure had occurred. Its origins were within a grease nipple hole, near the inner end of the thread as shown in Figures 4, 5 and 6. In that region, which was beyond the penetration depth of the nipple, the thread form had been damaged, the crest having been 'splayed-out' into the gap either side (Figures 5 and 6).

Damage was also evident on the inner part of the threads within many of the other grease nipple holes (two to each blade position) but, as illustrated in Figure 7, this damage was distinctly dissimilar, having clearly been caused by the shot peening treatment which had been applied to the inner surface of the hub. No cracking was detected at any of these holes. The grease nipple holes in the hub from the left-hand engine were also found to be free from fatigue cracks.

The thread at which the fatigue cracking had originated only exhibited the shot peening effect close to the inner surface of the hub, the major damage having been caused by some other, unidentified mechanism.

The plane of the fatigue fracture was predominantly at approximately 90° to the surface of the hub but, close to the outer surface, there was a sudden change to an angle of approximately 45°. Figures 8 and 9 show the appearance of this effect on the inner half of the fracture, and Figure 10 shows the associated raised rim on the outer half. This rim extended along most of the fatigue crack length, becoming particularly pronounced on the shorter side adjacent to the thickened clamping flange, arrowed in Figure 10, and being least distinct close to the hole on the side away from the thickened portion (Figures 8 and 9).

The 45° feature was caused by a change in the direction of the crack as it propagated to the surface and it seems to have been associated with the presence of the heavily worked peened layer, which appeared to have delayed surface breakthrough of the crack.

A detailed examination of the fatigue crack surface was conducted in a scanning electron microscope (SEM). Figure 11 shows the region at the outer surface, close to the grease nipple hole, where the crack had broken through with minimal influence from the peened layer over a length of 5 to 7 mm from the hole. Beyond this length, the 45° lip became more pronounced, as shown in Figure 12.

The crack surface close to the origin, up to a distance of approximately 3 mm, exhibited areas of so called 'mud cracking' which is characteristic of corrosion deposit that has dried out. There also appeared to be evidence of intergranular corrosion upon the crack surface. However, much of the remaining surface was finely pitted in a manner consistent with the ingress of dust, or fine debris, into the growing crack. This had the effect of obscuring most of the fine growth detail. Where fatigue growth features were visible, they showed considerable variability in appearance and clarity.

Particular attention was paid to the transition between the flat fracture and the 45° lip in order to establish whether there had been a sudden change in growth rate, consistent with a delayed breakthrough to the outer surface. However due to the surface damage effects, no useful growth details were visible in this region. Nevertheless, the general appearance of the fracture features at this transition did not suggest that a rapid acceleration in growth had occurred.

A section cut through the fracture, to include the region in which corrosion product had been observed, was polished and optically examined for evidence of corrosion attack upon the surface. No such evidence was found and there were no other features of significance. Qualitative X-ray analysis carried out on the section showed that the composition of the aluminium alloy was consistent with 2014 as specified, and hardness tests on the polished surface gave a result of 161 Hv(10), which was in accordance with the T6 heat treated condition.

This examination thus concluded that:

The failure of the propeller hub was caused by the growth of fatigue cracks from the inner end of a grease nipple hole. The cracks had originated on both sides of the hole in a region where the thread had been damaged by two mechanisms; one at the inner rim of the hole which had resulted from the shot peening of the inner surface of the hub and another, of indeterminate cause, which had resulted in distortion of the thread form over the unused length below the nipple.

As previously observed, a great deal of the fine detail of the fatigue growth marks on the crack surface had been obscured by corrosion deposit and by damage caused by ingestion of fine grit particles into the crack as it was extending. These effects, and the random nature of the remaining growth evidence, prevented any meaningful assessment of growth rate, or the establishment of a relationship between growth and operating loads. However, an attempt to calculate the number of individual cycles using the visible single striations, along a 17.5 mm long curved path between the origin and the surface, gave a figure of approximately 28,000 cycles. This result was, at best, an approximation and was only considered a rough guide to the number of loading actions which had caused the finest observable striations over this 17.5 mm length of crack. Whilst the source of these loads was, however, not known such a number was clearly far in excess of the start-up, shut-down or ground-air-ground load excursions, and was possibly more in accord with vibrational high frequency loading.

Because of the lack of data linking propeller loading with fatigue crack growth it was not possible to establish the length of time that the crack had taken to become visible at the outer surface and, consequently, its likely size when the hub was last inspected 10.5 hours before the accident. Although the actual growth rate of this internally generated crack could not be quantified, it was clear that it would have achieved a much greater length before it could be detected by the specified inspection methods than would a crack growing from the outboard end of a grease nipple hole, particularly if, as appeared possible in this case, breakthrough to the outer surface had been slowed in the latter stages by the sub-surface influence of the peening treatment. It did appear, however, that the crack may have become visible close to the hole over a length of approximately 5 mm soon after the initial quadrant reached the outer end of the hole.

This possible delay in detectability also raised the concern that by the time the crack had become visible it was likely to have been growing at a rate which could have resulted in failure in much less than the inspection period and, quite possibly, before it could have been discovered. Indeed, if during the later stages of crack development the load cycles producing the observed growth occurred at some frequency related to blade revolutions there would have been very little chance of detecting such a crack that had initiated internally.

1.12.2.2.2 HT Consultants examination

The fractured hub, which had been cleaned and sectioned by the DRA, was submitted together with the hub from the left-hand engine. The DRA had reported that the hardness of the fractured forging was 161 Hv(10). The forging conductivity was measured as being 38 IACS. This hardness/conductivity relationship is satisfactory for 2014-T6.

The cleaned half of one side of the fracture shown in Figure 13 was optically examined at magnifications of up to x40. It was seen that, as reported by the DRA, the fracture face had been generally damaged by fine pitting, but the marks typical of fatigue progression were still clearly visible. The damage present was considered typical of that produced by low cycle fatigue (high stress/low endurance). By using oblique illumination, it was possible to photograph and count stages in progression (bands) which were each made up of a number of much finer striations. From the similarity and nature of these bands, it was considered by HT Consultants that each band had resulted from an event in the service life of the propeller and that these events were, in all probability, flights.

The results of the examination described above are shown in Figures 14 to 19 inclusive. The detailed counting of bands was carried out on the fracture surface to one side of the grease nipple hole only. Figure 20 shows that the fracture surface on the other side of the grease nipple hole was similar.

From the number of events (or striations) per 0.1 inch measured at various positions on the fracture surface, it was estimated that approximately 150 events had been responsible for the progression of the fatigue crack from the position adjacent to crack initiation, where bands were first visible at x40 magnification, to final fracture of the remaining material due to overload. It was also estimated that the cracking would not have been visible on the outer surface of the hub until some 10 events prior to final separation.

After cleaning where necessary, the inner ends of all the other grease nipple holes in both left and right hubs were macroscopically examined. Three radial cracks were found, all associated with the heavily cold worked (shot peened) region at, and adjacent to, the inner end of the grease nipple hole diametrically opposite the hole from which the fatigue fracture had initiated and progressed (Figures 21 and 22). During this examination it was noted that at least one grease nipple had been fitted into the hub in a 'cross-threaded' condition.

The remaining grease nipple holes in the fractured hub cap were cut out and encased in polymethylmethacrylate. After suitable preparation, the material at, and adjacent to, their inner ends was examined for cracks. No cracks were observed at magnifications of up to x150.

In summary, HT Consultants considered that progression of the subject cracking, from approximately 0.25 inches in from the initiation position to final separation, was the result of a low cycle (high stress/low endurance) tension fatigue mechanism and that this damage had occurred during approximately 150 flights. It was also considered that the progressing fatigue crack would not have been visible on the external surface of the hub until about 10 flights before blade detachment occurred.

A section of the hub material parallel to, and just under, the fatigue separation was prepared and examined. The macro grain size was seen to be very large although

Hartzell maintained that this was typical of forged 2014-T6 material. Large grain size is associated with poor resistance to fatigue.

1.12.2.2.3 A-LAB examination

It was reported by HT Consultants that the crack would have not been visible until '10 events prior to final separation'. This conclusion was based on light fractography and was not accepted by A-LAB. The latter stated that the crack had penetrated through to the outside of the hub when it was approximately 1.4 inches long on the inside and had resulted in a crack of approximately 1.1 inches on the outside surface. A-LAB also stated that the cracking had propagated another 0.8 inches on the inside prior to the final fast failure in overload.

A-LAB maintained that the crack on the outside had extended approximately 2.1 inches from the grease nipple hole, in fatigue, prior to final fast failure. The coarse fatigue striations that HT Consultants had reported as 'events', were considered to have been due to a number of different conditions. Scanning electron fractography of the fatigue showed the coarse striations to have a consistent spacing, with very fine striations which were not visible under light microscopy. The number of cycles to failure estimated by A-LAB was 28,000 (minimum).

If the coarse striations had been caused by an exceptionally high load, A-LAB contended that a large 'jump' distance should have been observed on the fracture surface, as compared to the fine striations. Due to the highly varied loading conditions on the hub, including takeoffs, landings, feathering, turbulence and normal stresses, it was considered doubtful by A-LAB that the coarse striations could be used to estimate when the final failure occurred with respect to crack growth.

Following this statement a number of questions were raised with A-LAB. A second examination was carried out by them which led to the Report 93-090805, PO Number 71251 of 22 November 1993, which is reproduced at Appendix 2.

In summary, this report agreed that the origin of the fatigue had been located at the last (ie bottom) thread and 'initiated in a shot peened surface'. It agreed that the fatigue crack had extended under the shot peened outer surface of the hub for some 1.25 inches before it broke through to the outer surface, producing an external crack which was also some 1.25 inches in length from the grease nipple hole. This report stated that the crack had extended approximately another 1.0 inch externally and within the hub material section before the final 'instantaneous overload' failure of the hub. However, the report also illustrated scanning electron microscope evidence of small zones of fatigue within the fracture progression front that formed the final 1.0 inch of propagation and concluded 'the fracture surface does not reveal significantly enough evidence to determine the number of flights prior to instantaneous or catastrophic failure.'

Because a difference of opinion continued to exist between HT Consultants and A-LAB, a meeting was arranged between A-LAB, HT Consultants, Hartzell and AAIB. Prior to this meeting, A-LAB again examined the components in the SEM. During this examination many areas of the fracture face were examined and photographed. As before, many areas of the fracture face beyond the point where the crack became external were found to contain fatigue striations. No conclusions were drawn during this factual investigation of the fracture face.

1.12.2.3 Right engine

The right engine was stripped and examined in detail at an approved overhaul facility. It was noted that the engine crankcase had failed circumferentially in the area between the No.1 and 2 cylinders and that the crankshaft was bent in this area. No other defects were found within the engine and the degree of general wear was consistent with an engine that had completed 900 to 1,200 hours since manufacture/overhaul. Detailed inspection of the crankshaft counterweights and their mountings found that all associated items were of the correct type and fitted correctly in accordance with the manufacturers instructions. Some wear, which was consistent with the time in service of the engine, was found on the rollers of one of the counterweights. This wear was considered insignificant by the engine manufacturer and would not have affected the functioning of the counterweight system on this engine. The crankshaft 'run-out' was measured and was found to be excessively out of tolerance due to bending in the area of the crankcase failure. The crankshaft No.1 bearing shells were examined for indications of uneven wear which may have indicated excessive runout in service, none was found.

1.12.2.4 Propeller rpm gauge

The propeller rpm gauge that was mounted in the cockpit instrument panel was tested on approved and calibrated test equipment. The right-hand propeller rpm half of the gauge was found to function correctly with the rpm readings well within the acceptable tolerances throughout the rpm range.

1.12.2.5 Right propeller governor

The propeller governor from the right engine was taken to an approved test facility in an attempt to determine the rpm at which the unit was set to govern. Examination of the unit revealed that its mounting casting had been damaged to the extent that it could not be fitted to the test equipment. An attempt was made to replace the casting, but without success.

1.13 Medical and pathological information

Not applicable.

1.14 Fire

There was no fire.

1.15 Survival information

The passengers adopted the brace position in accordance with the pre-flight briefing given by the commander. The touchdown impact was relatively gentle, but the aircraft slewed sideways as the left wingtip dug into the ground. One passenger sustained minor whiplash injuries to her neck during the landing deceleration. A rapid evacuation was successfully achieved by use of both the main cabin door on the rear left side of the aircraft, and the over wing escape hatch on the right side. No problems were experienced with the operation of these exits. Passenger Safety Briefing Cards had been available for each passenger.

1.16 Tests and research

1.16.1 Visual crack detection

A short time after this accident had occurred the propeller manufacturer, Hartzell, received at their manufacturing facility a similar model propeller hub that had been retired from service because of a crack found coming from one of the grease nipple holes. The crack had been found due to the presence of grease on the external surface of the hub, near the grease nipple hole. After cleaning, the hub was visually examined, without the use of a magnifier, in the region around the grease nipple hole but no crack was observed. However, when examined using eddy current techniques, an external crack of 0.8 inches in length was detected. Several members of the manufacturer's engineering and product support staff examined the area using a x10 magnifier and none were able to say that they could see the crack.

Based on this finding that it was possible not to be able to detect an external crack by visual inspection only, Hartzell's prepared and released SB No. 165C on 9 July 1993. This SB replaced the 25 hour visual inspection with a 25 hour visual and eddy current inspection.

Subsequent to the release of SB No. 165C, Hartzell received another similar propeller hub that had been rejected during overhaul because an internal crack had been detected during a fluorescent dye penetrant inspection. When visually examined using a x10 magnifier, no external crack could be detected, but when examined using eddy current techniques a crack which was 1.5 inches long externally was found.

1.16.2 Laboratory fatigue tests

The propeller manufacturer conducted fatigue tests on two HC-()3Y()-() propeller hubs that had been returned from service with cracks which had initiated from the grease nipple holes in the front half of the hub forging. The tests were performed in order to understand the propagation rate of the cracks, in addition to the load levels required to propagate them. The detailed results from both of these tests were contained in Hartzell Propeller Inc. Engineering Report 1446.

Test 1 was performed in November 1990. The hub had an external 1.75 inch crack emanating from one of the grease nipple holes. It was intended to run a test which would simulate the application of high frequency inflight loads to determine the number of load cycles (and therefore operating time) remaining until final failure of the blade arm.

The hub was thus installed on a C-50 'vibratory shaker' and subjected to a centrifugal load of approximately 49,000 lb using an internal load cell. This load represented the sum of the typical steady loads applied to the propeller. The assembly was then vibrated such that a vibratory stress of ± 1,800 psi (measured in the blade retention pocket) was applied to the hub. This load was considered higher than actual inflight vibratory loads. Crack growth was recorded as a function of the number of applied load cycles and the hub tested to final failure.

Based on this test, it was estimated that this hub with an existing 1.75 inch crack would have lasted an additional 35.9 hours of operation in service.

Test 2 was performed in August 1993. After disassembly of the second hub, the extent of the crack was measured using an eddy current device and found to be 1.6 inches externally and 1.7 inches internally. It was intended to run a test to assess the effect of the steady, or ground-air-ground, load cycle on crack propagation and to assess whether it was as significant as the inflight vibratory loads.

This hub unit was assembled with an internal load cell installed. The hydraulic load cell was pressurised to simulate a centrifugal blade load of 42,500 lb on each hub arm. The load cell was repeatedly pressurised up to 100 cycles. After every tenth cycle the crack on the external surface was examined with a eddy current device. After the 100th load cycle the hub was disassembled and the internal crack indication examined. At the end of the test the eddy current inspection revealed zero crack growth as a result of this repeated load application. Based on this test, it was concluded that the ground-air-ground load cycle was not a predominant cause of crack propagation of an existing 1.6 inch external crack.

1.16.3 Stress analysis of the hub

Due to failures occurring in service, an analytical study of the geometry of the hub had been carried out by Hartzell in 1990, the detailed results of which were contained in Hartzell Propeller Inc. Engineering Report No. 879. The purpose of the study was to analyse the stresses in the region of the grease nipple holes within the hub and of two proposed modifications to the grease nipple holes. The predicted stresses in the direction across the threads of the holes were indicated to be low compared to the stresses in the plane of the threads. This study also indicated that failures were more likely to originate from the inner surface of the hub and propagate around the circumference of the blade socket. These results compared well with the majority of service incidents of cracks originating from the grease nipple hole. It was also found that when the grease nipple hole was reamed to remove the thread and the inside end of the hole was chamfered, the stresses at the inner end of the hole were reduced by 39%.

An independent stress analysis was also commissioned by the AAIB. This used the BERSAFE finite element system (see Appendix 3). The hub designs modelled in these analyses were the Hartzell HC-()3Y()-() propeller hub that was manufactured prior to 1983 and the model manufactured after 1983.

The loading used was the centrifugal load generated by the propeller blades rotating at 2,600 rpm (43.33 rps). The resultant end face bending loads were subject to three dimensional analyses. This gave negative, or compressive, 'Z' stresses on the outer surface of the hub and at the outer edges of both hole positions. Analysis indicated that the inside edges of these holes were the highest stressed points. For the 45° hole, crack propagation commenced in this area. The compressive stresses in the outer hub surface prevented the crack propagating straight to the outer surface. Because of the shot peening on the inside hub surface, compressive surface layer stresses would also affect associated cracks. Such stresses would tend to stop crack propagation locally, ie the crack would be forced away from the inside surface. This explained why the observed crack propagation stayed inside and grew through the hub material cross section away from both surfaces for some time before the final failure.

The stresses due to other values of rpm are proportional to the square of the rotational speed, and so could grow quite rapidly if a severe overload occurred.

Fatigue crack growth rates are proportional to a higher power of stress, typically of the 4th or 5th order, ie small increases in this maximum stress can induce rapid increases in growth rate.

Other modes of loading, eg the deflection of the propeller plane at rotation during takeoff, would affect other components of stress other than the 'Z' component, and so would only exert minor influence on the hole stress concentration factor and the crack growing mechanism.

The results of the finite element analyses showed that, for the three dimensional analysis of the pre-1983 model hub with the 45° hole, the maximum 'Z' component of stress was on the inside hub surface at the hole and equalled 558 MPa under the centrifugal loading at 2,600 rpm. Without the hole, the corresponding 'Z' stress was 252 MPa, hence the hole gave rise to a stress concentration factor of 2.21:1.

The corresponding finite element analysis for the hole situated in the hub flange of the post-1983 model calculated that the highest stress was again on the inside hub surface at the hole and equalled 432 MPa, which was only 77% of the 45° degree hole highest stress. Without the hole, the 'Z' stress at the -45° position (no hole) was 238 MPa, giving a reduced stress concentration factor of 1.82:1.

1.17 Additional Information

1.17.1 Utilisation history of G-BMGH

Enquiries were made with the aircraft's operators regarding the type of flying that had been undertaken. From August 1991, when the right propeller was fitted to G-BMGH, up until May 1993 the aircraft had been used as a 'company' aircraft carrying company personnel within Europe. Approximately 15% of the flying hours had been used for 'Type' and 'Currency' training of company pilots. From May 1993 until the accident the aircraft had, in the main, been used for ad hoc charter flights. No test or experimental flying, except those test flights required for the issuance of a Certificate of Airworthiness, had been conducted since August 1991. Appendix 4 shows the hours/sector numbers flown since 30 August 1991 and the associated maintenance check dates.

1.17.2 Information from pilots of previous flights in G-BMGH

A number of pilots, including the commander of the accident flight, who flew the aircraft between the last maintenance check (ie 50 hour check in May 1993) and the accident flight were asked for their comments on the aircraft's performance. All of them had noted that the propeller rpm gauge indications 'hunted' slightly during the engine run-up checks, but that this was not unusual for the aircraft type. None of them had noted any tendency for the propeller speeds to 'overswing' the red line on the rpm gauge. All of them had noticed that the right propeller lever was stiff and two of them commented that they had to synchronise the propellers inflight using the left propeller lever, because of the stiffness in the right lever. None of the pilots had noticed any unusual vibrations whilst flying the aircraft. However one passenger, who had been a pilot, had noticed a vibration through the cabin floor during two of the flights; one flight was 2 flying hours after the last inspection and the other 2 hours 40 minutes. This vibration had become noticeable on the first flight during the climb and appeared to stop halfway through the flight. On the second flight, the passenger had been aware of the vibration throughout the cruise. On both flights this passenger had removed her shoes. Neither the pilot or the other passengers, who were also pilots, felt this vibration when commented upon by the passenger. One of the pilots who flew this aircraft for 5 hours 35 minutes prior to the day of the accident did notice a slight grease leakage from the right-hand propeller during the pre-flight inspections. This grease leakage could be seen running along the propeller blades. The pilot reported the leakage to the operator's senior pilot and they both considered that the leakage had been caused by the fact that the weather was 'hot' and that the aircraft had recently undergone a maintenance check. The pilot had, on previous occasions, observed grease leakage around propellers on other aircraft just after maintenance checks had been completed.

None of these individuals had been aware that, when the condition initiated suddenly, unexplained vibration or the leakage of grease from a propeller hub could possibly indicate a potentially dangerous defect of that propeller system.

1.17.3 Last 50 hour maintenance check/FAA AD No. 89-22-05

The aircraft engineer and his licensed engineering supervisor who had conducted the last 50 hour check were interviewed at length regarding the external visual inspection of the right propeller hub in the area of the grease nipple holes. As a result of these interviews, it was considered that this inspection had been carried out competently by an experienced aircraft engineer. During this inspection, the engineer had removed the grease nipples from the hub to facilitate a thorough inspection of the threaded hole, from where he understood cracks could originate. However, such removal of the grease nipples was not required by the FAA AD No. 89-22-05 (Appendix 6) but neither was there any statement that they should not be removed. The aircraft engineer stated that he had not been interrupted during his examination of the right propeller and that it had been accomplished during the late morning work period. None of the grease nipple hole areas on the hub had any indications of grease leakage and the visual examination using a x10 hand-held magnifier had not revealed any evidence of cracks. The engineer did not remove any paint from the hub as required by the AD since the hub had not been painted during its last overhaul. Prior to replacing the propeller spinner, the aircraft engineer reported to his licence engineering supervisor, informing him that the check was complete and ready for his inspection. Sometime later the licensed engineering supervisor instructed the aircraft engineer to refit the propeller spinner. The licensed engineering supervisor could not specifically remember if he had, or had not, inspected the right propeller hub prior to the spinner being refitted.

FAA AD No. 89-22-05 had previously been carried out on the 25 March 1993, seventeen flying hours prior to the inspection in May 1993. No evidence of grease leakage had been seen and no crack found in the area of the grease nipples.

1.17.4 Previous Hartzell HC-()3Y()-() propeller hub failures

Various accident data bases were interrogated, including that of the manufacturer, for information on previous Hartzell HC-()3Y()-() propeller hub failures. Appendix 7a lists all known instances of fractures within the hub forging, some of which were found on inspection/overhaul and some of which resulted in propeller blade separation. Of the 24 instances listed in Appendix 7a, 14 hubs had developed cracking from the grease nipple holes, of which 6 resulted in in-flight hub failure. Five hubs had released propeller blades. Appendix 7b describes those instances where the hub casting suffered a major failure from the area of the grease nipple hole. If single-engined agricultural aircraft are excluded, it is apparent that in all the instances where a major failure had occurred, the hub was from the counter rotating model propeller with the 'E' type extension fitted, and to the Lycoming LTIO-540 series engine, ie left-hand rotating.

A number of metallurgical reports were examined which detailed examinations of fractures which had initiated at the inner end of the grease fitting holes. Although the fracture initiation was very similar to the fracture that is the subject of this report (Figure 23), their progression was very different (Figure 24). After initiation, the fracture progression fronts generally broke through to the outer hub surfaces when they were at about 45 degrees to that surface. After surface breakthrough, their progress continued over a large number of 'cycles' before final failure, which allowed time for their detection by inspection.

1.17.5 Design changes to Hartzell HC-()3Y()-() propeller hubs

During 1983, a hub design change was implemented by the manufacturer which relocated the grease fitting holes to an area where the hub material was thicker and thus exposed to lower stress levels (Figure 25). This design change was implemented as a result of the manufacturer's experience of cracks occurring in-service which originated from the grease fitting hole. In addition to relocating the hole, its design was altered by drilling and threading only to a depth sufficient to accommodate the grease nipple. The remainder of the hole that was drilled through to the inside of the hub was not threaded and was of a reduced diameter, sufficient to allow passage of the lubricant (Figure 26). All hubs manufactured after 1983 were of this modified design.

1.17.6 Service history of Hartzell HC-()3Y()-() propeller hubs

Prior to this accident, cracks originating from the grease nipple (zerk) hole were a known occurrence and as a result the manufacturer and the Federal Aviation Administration (FAA) had, respectively, issued three Service Bulletins (SBs) and an Airworthiness Directive (AD).

On the **3rd October 1989 Hartzell issued SB No. 165** (Appendix 5) which stated the requirement for an initial 25 hour visual inspection followed by a repeat 50 hour visual inspection.

FAA AD No. 89-22-05 of 20 October 1989 (Appendix 6) required compliance with Hartzell SB No.165 of 3 October 1989.

Hartzell SB No. 165A of 27 August 1992:

This SB made the following alterations to SB No.165 of 3 October 1989:

> **'DISCUSSION:**
>
> This Bulletin has been revised (replaces Service Bulletin 165) in order to show a more restrictive compliance requirement and optional eddy current inspection as a result of a recent failure. In order to eliminate the burden of repetitive inspections and the possibility of an inadequate inspection, a terminating action (hub replacement) is addressed in the compliance portion of this bulletin.
>
> There have been incidents of hub cracks in Hartzell three blade "compact" aluminium hub propellers. Cracks typically originate in the threads of a grease fitting hole on the side of the hub. The cracks are external and are observable with careful visual examination. As the cracks propagate around the blade arm of the hub, their progression accelerates and results in failure of one hub half which can then, potentially, progress to blade separation.

WARNING: UNEXPLAINED VIBRATION OR GREASE LEAKAGE INCIDENTS, WHERE THE CONDITION INITIATED SUDDENLY, DEMAND IMMEDIATE INSPECTION FOR POSSIBLE CRACKED HUB.

COMPLIANCE:

REQUIRED ACTION:

Inspection is required within the next 25 hours of operation from August 27, 1992 the effective date of revision A to this Bulletin or within 50 hours of operation from the last inspection, whichever occurs first as follows:

1. Perform visual inspection and thereafter at intervals not to exceed 25 hours of operation <u>or</u>;

2. Perform a combination of visual and eddy current inspection and thereafter at intervals not to exceed 50 hours of operation <u>and</u>;

3. In addition, if any abnormal or unexplained changes occur in propeller vibration or grease leakage, inspection must be performed prior to further flight.

NOTE: During 1983, a hub design change relocated the grease fitting holes near the hub parting line. The earlier design hubs are listed as affected serial numbers. However, there may be a few hubs listed that are of the later type. Any hub found to be of the current configuration does <u>not</u> require compliance with this Bulletin.

RECOMMENDED ACTION:

Retirement of affected hubs is recommended during propeller overhaul. Replacement with later style hub (post 1983) is terminating action for this Bulletin.

> NOTE: To encourage operators to replace hubs, special reduced pricing will be provided for replacement hubs through 1994.'

Hartzell SB No. 165B of 11 September 1992:

This SB was effectively the same as SB No. 165A of 27 August 1992 except for the following:

> 'RECOMMENDED ACTION:
>
> Retirement of affected hubs is strongly recommended during propeller overhaul. Replacement with later style hub (post 1983) is terminating action for this Bulletin.
>
> Note: To encourage operators to replace hubs, special reduced pricing has been established for replacement hubs and/or propeller assemblies.'

After the accident to G-BMGH occurred the manufacturer issued two SBs and the FAA and the United Kingdom (UK) Civil Aviation Authority (CAA) each issued an AD. The inspections required by these ADs and SBs were:

UK CAA Emergency AD No. 004-06-93 of 15 June 1993 required external visual inspection of the grease nipple holes within 5 hours of receipt of the AD, and then at intervals not exceeding 5 hours time-in-service from the last inspection.

Hartzell SB No. 165C of 9 July 1993:

> **'DISCUSSION:**
>
> This Bulletin has been revised (replaces Service Bulletin 165, 165A & 165B) in order to show a more restrictive compliance requirement and required eddy current inspection as a result of a recent hub failure. In order to eliminate the burden of repetitive inspections and the possibility of an inadequate inspection, a terminating action (hub replacement) is addressed in the compliance portion of this bulletin.

This revision imposes a rather severe repetitive inspection requirement. The more restrictive requirements are placed on aircraft models which have a history of cracked or failed hubs. Other models, such as the PA-31 (310 hp), have had no failures but are addressed in this Bulletin because of their similarity to applications which have a history. These models have a more liberal inspection requirement.

In Hartzell three blade "compact" aluminium hub propellers, cracks typically originate in the threads of a grease fitting hole on the side of the hub. The cracks are external and are observable with careful visual examination. As the cracks propagate around the blade arm of the hub, their progression accelerates and results in failure of one hub half which can then, potentially, progress to blade separation.

WARNING: UNEXPLAINED VIBRATION OR GREASE LEAKAGE INCIDENTS, WHERE THE CONDITION INITIATED SUDDENLY, DEMAND IMMEDIATE INSPECTION FOR POSSIBLE CRACKED HUB.

COMPLIANCE:

NOTE: During 1983, a hub design change relocated the grease fitting holes near the hub parting line. The earlier design hubs are listed as affected serial numbers. However, there may be a few hubs listed that are of the later type. Any hub found to be of the current configuration does <u>not</u> require compliance with this Bulletin.

REQUIRED ACTION for Piper PA-31-325, PA-31-350, T-1020; Aerostar PA-60-700P and Agricultural aircraft:

Inspection is required within the next 25 hours of operation from July 9, 1993 (the effective date of revision C to this Bulletin) as follows:

1. Perform a combination of visual and eddy current inspection. Repeat inspection at intervals not to exceed 25 hours of operation <u>and</u>;
2. In addition, if any abnormal or unexplained changes occur in propeller vibration or grease leakage, inspection must be performed prior to further flight.

TERMINATING ACTION:

Replacement with later style hub (post 1983) is terminating action for this Bulletin. Retirement of affected hubs on Piper PA-31-325, PA-31-350, T-1020; Aerostar PA-60-700P; and agricultural aircraft is required during propeller overhaul or by January 1, 1995, whichever occurs first. Manufacturing capabilities are limited. If later style replacement hubs are not available at the time of overhaul, to avoid aircraft grounding, it is acceptable to temporarily (for up to six months) continue operation with old style hubs.

NOTE: To encourage operators to replace hubs, special reduced pricing has been established for replacement hubs and/or propeller assemblies. Old style hubs removed from service are to be retired rather than used on other applications not affected by this Bulletin.'

Hartzell SB No. 165D of 6 August 1993 introduced some minor changes to the eddy current testing procedures in SB No.165C and altered the applicability.

FAA AD No. 93-16-14 of 18 August 1993 required compliance with Hartzell SB No.165D.

Following a meeting between AAIB, Hartzell, FAA, HT Consultants and A-LAB the propeller manufacturer issued SB No. 165E of 21 January 1994 which introduced significant changes to the inspection (see Appendix 8 and Section 1.17.13) and stated for the first time that the cracks typically originated in the threads of grease fitting holes 'on the inside of the hub'.

1.17.7 Metallurgical examination of other Hartzell HC-()3Y()-() propeller hubs

Following this accident, 61 Hartzell type HC-()3Y()-() propeller hubs that had been replaced at authorised propeller overhaul organisations by the later style hub (post-1983) were obtained by the AAIB and examined for evidence of cracking in the area of the grease nipple holes. Both ends of all the threaded grease nipple holes were visually examined at a magnification of x5, and 31 were eddy current tested for cracks using a Foerster Defectometer. No cracks were found in the areas of the grease nipple holes in any of the hubs examined.

1.17.8 Eddy current non-destructive testing

The principle of eddy current testing is that when an electrical coil is mounted in a suitable probe and subjected to an alternating current of known characteristics, if the probe is applied to a metal surface eddy currents are generated in the metal. These currents, also known as 'Foucault' currents, alter the inductive characteristics of the electrical coil in the probe and these changes may be indicated by means of a measuring bridge network, the indications being shown on a scale, or meter. When a fault in the metal is beneath the probe the eddy currents generated are different to those generated when there is no fault in the metal. The effective depth of eddy currents generated at the frequencies normally used for the detection of surface breaking cracks is relatively small. Eddy currents generated at 40 KHz penetrate to an effective depth of approximately 0.04 inches in aluminium based materials. The higher the frequency of the eddy currents used the smaller the depth of penetration. The depth of penetration also depends on the conductivity and permeability of the metal. The higher the conductivity the smaller is the depth of penetration and the greater the permeability the greater the depth of penetration. The generation of eddy currents is also dependant on the closeness of the probe to the metal surface and a rough metal surface will lift the probe, reducing the effective depth of the eddy currents generated.

Very low frequency eddy current equipment has been developed to locate sub-surface cracks, but there are very few of these units available. Special probes and techniques would have to be developed and approved before such equipment could be used generally within the aviation industry.

1.17.9 Propeller Type Certification requirements

The Hartzell HC-()3Y()-() propeller was Type Certified to Federal Aviation Regulation (FAR) Part 35, dated 1965. The following are relevant extracts from this regulation which has been amended through to 1980:

> 'Design features
>
> The propeller may not have design features that experience has shown to be hazardous or unreliable. The suitability of each questionable design detail or part must be established by tests.
>
> Materials
> The suitability and durability of materials used in the propeller must;
> (a) Be established on the basis of experience or tests;
> and

(b) Conform to approved specifications (such as industry or military specifications, or Technical Standard Orders) that ensure their having the strength and other properties assumed in the designed data.

Durability

Each part of the propeller must be designed and constructed to minimise the development of any unsafe condition of the propeller between overhaul periods.

Blade retention test

The hub and blade retention arrangement of propellers with detachable blades must be subjected to a centrifugal load of twice the maximum centrifugal force to which the propeller would be subjected during operations within the limitations established for the propeller. This may be done by either a whirl test or a static pull test.

Fatigue limit tests

A fatigue evaluation must be made and the fatigue limits determined for each metallic hub and blade, and each primary load carrying metal component of non-metallic blades. The fatigue evaluation must include consideration of all reasonably foreseeable vibration load patterns. The fatigue limits must account for the permissible service deterioration (such as nicks, grooves, galling, bearing wear, and variations in material properties).

Endurance test

Variable-pitch propellers. Compliance with this paragraph must be shown for a propeller of the greatest diameter for which certification is requested. Each variable-pitch propeller (a propeller the pitch setting of which can be changed by the flight crew or by automatic means while the propeller is rotating) must be subjected to one of the following tests:

a) A 100-hour test on a representative engine with the same or higher power and rotational speed and the same or more severe vibration characteristics as the engine with which the propeller is to be used. Each test must be made at the maximum continuous rotational speed and power rating of the propeller. If a take-off rating greater than the maximum continuous rating is to be established, an additional 10-hour block test must be made at the maximum power and rotational speed for the take-off rating.

b) Operation of the propeller throughout the engine endurance tests prescribed in Part 33 of this sub chapter.

Functional test

(a) Each variable-pitch propeller must be subjected to the applicable functional tests of this section. The same propeller used in the endurance test must be used in the functional tests and must be driven by an engine on a test stand or on an aircraft.

(b) Manually controllable propellers. Five hundred complete cycles of control must be made throughout the pitch and rotational speed ranges.

(c) Automatically controllable propellers. One thousand five hundred complete cycles of control must be made throughout the pitch and rotational speed ranges.

(d) Feathering propellers. Fifty cycles of feathering operation must be made.'

1.17.10 Propeller flight loads

Flight test data obtained from a leading propeller manufacturer indicated that in a normal flight the highest stresses placed on a propeller hub fitted to a gas turbine engine were during the rotation phase of a maximum power takeoff with the aircraft at maximum take-off weight and with zero headwind.

1.17.11 Shot peening

Shot peening is a cold working process in which the surface of a part is bombarded with small spherical media, called 'shot'. Each piece of shot striking the material acts as a tiny peening hammer, imparting to the surface a small indentation, or dimple. In order for the dimple to be created, the surface of the material must be yielded in tension. Below the surface dimple a hemisphere of cold-worked material is generated which is highly stressed in compression. Overlapping dimples develop an even layer of metal in residual compressive stress. Cracks will not usually initiate or propagate in a compressively stressed zone. Since nearly all fatigue and stress corrosion failures originate at the surface of a part, compressive stresses induced by shot peening provide a considerable increase in part life. The maximum compressive residual stress produced at, or under, the surface of a part by shot peening is at least as great as half the yield strength of the material being peened. Many materials will also increase in surface hardness due to the cold working effect of shot peening. The standard practice when shot peening a surface that contains a drilled/machined hole, threaded or not, is to plug the hole prior to the shot peening. The inside of the hole, which may include thread forms, can be shot peened separately from the main surface. This is achieved by the use of a probe that ejects the shot at right angles to the probe and ensures uniform peening of the bore of the hole.

Benefits obtained by shot peening are the result of the effect of the compressive stress and the cold working induced. Compressive stresses are beneficial in increasing resistance to fatigue failures, corrosion fatigue, stress corrosion cracking, hydrogen assisted cracking, fretting, 'galling' and erosion caused by cavitation. Benefits obtained due to cold working include work hardening, intergranular corrosion resistance, surface texturing, closing of porosity and testing the bond of coatings. Both compressive stresses and cold working effects are used in the application of shot peening in forming metal parts.

1.17.12 Vibration tests

In 1972, the propeller manufacturer conducted vibration testing to determine the vibratory stress levels of the Hartzell model HC-E3YR-2F/FC8468-6R three-bladed propeller fitted to a Lycoming TIO-540-J2BD engine (left-hand mounted engine); and the Hartzell model HC-E3YR-2LF/FJC8468-6R three-bladed propeller fitted to a Lycoming LTIO-540-J2BD engine (right-hand mounted engine) both of which were in a twin engined Piper PA-31-350 aircraft, with counter rotating propellers (Hartzell Propeller Inc. Engineering Report No. 310). The engine crankshaft assemblies included two dynamic counterweights, one tuned to the 5.1 order and the other tuned to the 6.3 order. Strain gauges were mounted along the propeller blade surfaces, from the tips to the blade roots. The results of these tests showed that the vibratory stress levels were within the allowable certification parameters. However it was noted that the vibratory stress levels on the right-hand mounted propeller were generally higher than those of the left-hand mounted propeller. The results from the tests conducted at maximum power in level flight (Appendix 9) showed that the right-hand propeller tip stresses crossed the allowable limit at 2,600 rpm and rose to a peak that was well above the allowable limit at 2,675 rpm. However since this peak was beyond the 2,575 rpm engine rating, it was considered satisfactory by the certification authority (FAA). It was also noted that the propeller shank stresses did not rise proportionally with the tip stresses.

1.17.13 Hatrzell's review of SB No. 165D inspection requirement

Following a meeting between AAIB, Hartzell, FAA, A-LAB and HT Consultants which included the appraisal of the data collected from a variety of tests and assessments, Hartzell conducted the following review of the inspection requirements called for in SB No. 165D:

'Although the metallurgical consultants from A-LAB and HT Consultants could not agree on a specific number of hours, both agreed that a short period of time existed between the point at which the crack propagated through to the external surface and final fast overload. Therefore, it was Hartzell's position that a shortened inspection interval was required. It was also Hartzell's position that an existing crack might not be visible with a 10 power glass and that Eddy Current was the preferable inspection method.

A metallurgical examination of three cracked hubs was carried out by A-LAB which showed that the cracks penetrated through to the outside of the hubs when they were approximately 1.3 inches, 1.4 inches and 1.5 inches long internally which resulted in cracks on the outer surface approximately 0.8 inches, 1.1 inches and 1.1 inches long respectively. This A-LAB investigation also indicated the crack in the failure incident hub propagated approximately an additional inch before reaching final fast overload. Their examination of the other two hubs indicated these cracks had propagated approximately an additional 0.65 inches and 0.8 inches after the cracks had become external before being discovered (note: these hubs had not propagated to failure in service).

The following inspection interval was computed based on a conservative combination of these crack lengths. Assuming a crack is 1.1 inches long when it becomes external and using a propagation length of 0.65 inches an inspection interval can be determined from the results of the high cycle crack propagation test (described in Section 1.16.2). A crack propagation analysis of the results of this test showed approximately 41 hours of operation to grow a crack from 1.1 inches long to 1.75 inches (1.1 inches + 0.65 inches) long.

Dividing this value by 3 (to allow three inspections during this time) provides for an inspection interval of approximately 14 hours. This test did not account for the high once per flight ground-air-ground load cycle, and even though the second test indicated this to be a minor affect, the inspection interval was decreased to 10 hours in SB No. 165E of 21 January 1994 ' (Appendix 8).

1.18 New investigation techniques

None.

2 Analysis

2.1 Aircraft handling

After the in-flight detachment of a right propeller blade, consequent separation of the right engine and secondary damage to the airframe and left propeller, the aircraft effectively became a glider with no available power and in a condition outside its normal loading envelope in terms of the lateral and longitudinal centres of gravity. The detailed aerodynamics of such a configuration are unknown. The commander therefore found himself in a unique and extremely critical situation, for which he had no training or guidance.

The commander resorted to basic flying skills and employed standard techniques to attempt recovery of the aircraft from the spiral dive, or spin, which then occurred. Fortunately the aircraft recovered from its right spin, or spiral, after some two rotations with the commander using full left rudder. He had then eased the seriously damaged aircraft out of the resultant steep dive and carried out what was effectively a low speed handling check. This involved a reduction in airspeed towards what he estimated might be a satisfactory gliding speed of 100 kt. However on approaching 120 kt, despite having full left aileron and rudder control applied, the aircraft again began to roll to the right. The commander therefore elected to increase the airspeed to around 130 kt, giving a steeper glide angle and higher rate of descent, but assessed that there was a reasonable degree of controllability for the subsequent forced landing.

The use of flaps and landing gear, both of which were hydraulically driven on this aircraft, would have required relatively prolonged use of the hand pump, however this was not a practical proposition in the short time available. It was also very fortuitous that the weather conditions at the time were good, that it was daylight, and that the catastrophic failure of the right propeller had occurred at cruise altitude over suitable terrain for a forced landing, away from congested urban areas.

There was very little time available to the commander in which to locate a suitable landing site and plan an approach, but the chosen flight path and subsequent landing were successful in preventing a more serious accident which could have resulted in the total loss of the aircraft and serious, or fatal, injury to the occupants. The commander is therefore to be greatly commended for his very skilful recovery of the aircraft, his concern for the passengers, and for carrying out the subsequent forced landing with such a high level of proficiency.

2.2 Failure sequence

At about the mid-point of the climb after taking off from Birmingham the commander had noticed an unusual vibration from the aircraft which manifested itself through the control column and the airframe as a high frequency vibration. This vibration continued to the top of the climb and for about 10 minutes into the cruise. The subsequent events strongly suggested that this vibration had been caused by a pre-existing crack rapidly progressing within the right propeller hub. The commander was not aware of the previous failure history of such propeller hubs and was therefore unaware of the significance of such vibration. In-flight vibration and grease leakage from the right propeller had been noticed prior to the accident flight, but the individuals involved had not been aware that these symptoms could indicate a potentially dangerous defect within one of the propeller systems. In view of such findings, it is recommended that:

> Airworthiness authorities and manufacturers should ensure that when Airworthiness Directives and Service Bulletins are issued which contain important safety information which is also relevant to pilots, additional measures are taken to ensure that such pilots become aware of the relevant information.

(In this context it is noted that the latest Hartzell SB No. 165E of 21 January 1994 contained a statement that 'This issue must be made known to flight crew members as well as maintenance personnel').

The flight continued otherwise uneventfully for a further 20 minutes before the right propeller hub failed and allowed one of its three blades to detach. As a result of the massive out-of-balance forces thus generated the right engine tore away from the airframe, in a downwards and outboard direction. The detached blade had penetrated the aircraft's nose baggage bay and exited through the upper left fuselage structure before striking and separating the spinner and propeller pitch change cylinder/piston assembly from the left engine. These items, and the blade from the right propeller, had then gone through the left propeller disc, causing bending of the left propeller blades and stopping the left engine.

2.3 The hub failure

The right propeller hub failure was caused by the fatigue growth of a crack which had initiated at the inner end of the thread of a grease nipple hole in the forward half of the hub casting. The grease nipple was used to lubricate the blade pitch bearing mounted within the blade arm of the hub. The thread form at the site of the crack initiation had been damaged by shot peening of the inner surface of the hub and distortion of the thread form over the unused length below the grease nipple.

All three metallurgical examinations that were conducted upon this failed hub were in agreement as to the origins of the crack and that the progression from initiation to final separation was the result of a low cycle (high stress/low endurance) tension fatigue mechanism. All three examinations noted fine fatigue striations, the number of which were in excess of 28,000, but only HT Consultants and A-LAB referred to progression 'banding', each band containing a number of the finer striations. There was disagreement as to the progression of the crack after it had broken through to the outer surface of the hub. HT Consultants assessed that the progression of the fatigue crack from the position adjacent to its initiation to the final overload fracture had occurred over approximately 150 events, with the ultimate failure having been very rapid, within the order of 10 events after the crack had become visible on the outside of the hub. A-LAB and DRA felt that there was not enough evidence to determine the number of events, from the crack breaking through to the outer surface to the final instantaneous catastrophic failure, but A-LAB did assess the time period as being 'short'. A-LAB also indicated that it was doubtful that the coarse progression banding seen on the fracture surfaces could be linked directly to the number of flights and therefore could not be used to estimate when the final failure occurred with respect to the crack growth. From examination of the aircraft log book and documentation it was considered by the AAIB that as the aircraft had been operating for its last 137 hours, ie 159 flights, in a 'normal' commercial operating mode, with a small amount of flight training, and that the majority of these flights were of an average 50 minute duration, that these coarse striations were most probably produced on each flight and may have been associated with aircraft rotation during takeoff.

FAA AD No. 89-22-05, a visual examination of the hub using a x10 magnifier, had been carried out twice, once at 27:40 hours and again at 10:35 hours, prior to the failure and on neither occasion was a crack detected, or grease leakage seen. During the inspection 10:35 hours prior to the failure, the grease nipples were removed to allow inspection of the hub's surface up to the edge of the threaded hole. The grease nipple holes were tapered and the action of removing the grease nipples may have tended to cause any crack at the outer hub's surface to 'close-up' thereby making it more difficult to detect visually. It was also possible that if the crack had not broken through to the outside surface of the hub at the time that the grease nipples were removed for this inspection that the action of replacing them into the tapered holes may have caused the crack to break through. The removal of the grease nipples was not required in the ADs procedure, but neither did this AD include a warning not to remove them. It was noted that there was no mention in the AD that the cracks typically originated in the threads on the inside of the hub. It was also noted that in the latest SB No. 165E of 21 January 1994 there was a warning that the grease nipples should not be removed when conducting the inspection, and that the cracks typically originated on the inside of the hub. Soon after this accident, the propeller manufacturer received a hub that had been returned from service and which had a crack extending from one of the grease nipple holes. This external crack could not be seen visually, even when a x10 magnifier was used. Based on this finding, the manufacturer prepared and released SB No. 165C of 9 July 1993 which replaced the visual inspection with an eddy current and visual inspection. It was felt that if the crack in the failed hub had broken through to the outside, to the length determined by A-LAB, prior to the last visual inspection then there would most likely have been some grease leakage visible. The evidence from a pilot and a passenger who flew the aircraft within the 10 hours prior to the accident indicated that the crack may have broken through to the outer surface of the hub quite soon after the last inspection, if in fact it had not broken through prior to that inspection.

2.4 Fatigue mechanism

Two distinct types of fatigue striation were observed on the failure surfaces. There were progression bands, ie coarse fatigue striations, between which were very fine fatigue striations. The fine fatigue striations were very large in number, well in excess of 28,000, and were typical of the type produced by low cycle fatigue. The number of these fine striations was far in excess of the number of flight cycles that the propeller had completed since manufacture, which indicated clearly that during each flight cycle many such striations had been produced. The propeller manufacturer was able to propagate an existing crack in a hub by subjecting the hub to static, ie non-rotational, high cycle fatigue. This tended to indicate that vibrational high frequency loading could be instrumental in the failure mechanism. It was seen from the previous failures of the hub type that the majority were on the left-hand rotating hub. It was also seen from the results of the vibratory stress tests conducted by the manufacturer in 1972 that the stress levels detected on the left-hand rotating propeller/engine combination were significantly higher than those of the right-hand rotating combination. It was also noted that under certain conditions the vibratory stress levels at the blade tip exceeded acceptable limits, close to the allowable certification parameters. However, it was apparent that the stress levels at the propeller shank did not rise proportionally with the tip stresses. The right engine fitted to G-BMGH was a 'high time' engine which had accumulated 174:20 hours beyond the manufacturer's recommended overhaul period. The internal condition of this engine when strip examined was assessed as that of a 'mid-time' engine. However some wear, which was consistent with the time in service of the engine, was found on one set of crankshaft counterweight rollers. The engine manufacturer considered this wear to have been insignificant, such that it would not have affected the functioning of the counterweight system. However, there is the possibility that, under certain wear conditions in 'high time' engines, unacceptable vibratory stresses may occur.

The coarse fatigue striations were fairly evenly spaced along the crack progression path from initiation to the point where the final fracture had occurred due to overload. The number of these coarse fatigue striations, or events, were assessed by HT Consultants as being about 150. Since the propeller was fitted to this engine, the aircraft had flown 159 flights up to, and including, the accident flight. It was considered that the similarity between these two numbers could not be ignored. From flight test data, albeit related to a propeller mounted on a gas turbine engine, it was noted that the highest loads on a propeller in 'normal' flight occur during the rotation of the aircraft at takeoff. Such loads would be expected as a result of gyroscopic forces and hence would be applicable to all propeller driven aircraft, although it is recognised that vibrational stresses could be predominant on reciprocating engine installations. All previous hub failures that had occurred on the Piper PA-31 series aircraft involved the 'extended' model hub. It was thus considered possible that the loads applied to the hub at rotation during takeoff were increased by virtue of the fact that the hub extension placed the hub further from the point of rotation, ie the centre of lift of the aircraft's wing.

2.5 Inspection requirements

Prior to this accident, the manufacturer had initially required a visual inspection of the hub in the area of the grease nipple holes within 25 hours from the date of their SB 165 and then every 50 hours after that first inspection. The FAA issued an AD which required compliance with the manufacturer's SB. When the manufacturer updated the inspection requirement to a visual inspection every 25 hours, and/or an eddy current inspection every 50 hours, the FAA did not update their AD to reflect this new inspection requirement. It is therefore recommended that:

> When airworthiness authorities issue Airworthiness Directives (ADs) which require compliance with a manufacturer's Service Bulletin (SB), such ADs should be updated when the manufacturer updates the SB, or clearly state that the AD relates to the latest issue of the associated SB at any point in time.

After this accident, the CAA very quickly issued an Emergency AD requiring a visual inspection of such propeller hubs every 5 hours. The speed at which this Emergency AD was issued and the setting of the 5 hour inspection requirement is to be commended. The manufacturer and the FAA then, respectively, issued updated SBs and an AD requiring visual and eddy current inspections every 25 hours.

From the metallurgical examinations of this failure there appeared differing assessments of the action required to detect and prevent the loss of a propeller blade from the pre-1983 type hub. Initially, from the results of the work carried out by the manufacturer and A-LAB, a 25 hour visual and eddy current inspection were thought to be adequate to detect a crack before final failure occurred. Following a meeting between AAIB, Hartzell, HT Consultants, A-LAB and the FAA the propeller manufacturer re-appraised the inspection requirement and issued SB No. 165E on 21 January 1994 which reduced the inspection period to 10 hours on specific propeller/engine combinations. Ultimately, the only terminating action that can be taken to ensure that this particular mode of failure does not occur in the future is to replace the pre 1983 hubs that are in service with the post-1983 hub type.

In September 1992 the manufacturer had issued SB No.165B, the last part of which strongly recommended that pre-1983 hubs be replaced with the later style hubs at overhaul. However this evident concern by the manufacturer was not acknowledged and reinforced by associated FAA or CAA mandatory action. It is therefore recommended that:

> The CAA and FAA should seriously consider issuing Airworthiness Directives to make manufacturers' strong recommendations to replace components a mandatory requirement where it is apparent that failure to replace such components could result in a potentially major hazard to the safety of affected aircraft.

2.6 Design and manufacture

The three-bladed compact propeller hub was designed in the 1960s and was a development of the two-bladed propeller aluminium alloy hub which was the first commercial aluminium alloy hub produced by that manufacturer. At that time no detailed stress analysis calculations of propeller hub designs were required, or were carried out. The findings of this investigation have demonstrated that had a stress analysis been conducted upon the pre-1983 hub design at certification, the stress concentration factor associated with the original grease nipple hole locations would have been apparent (Section 1.16.3).

During the type certification testing, both static and dynamic, of this hub design stress levels within the hub forging were assessed by extrapolating the stress readings taken from strain gauges mounted on the propeller blades. Although this manufacturer has been strain gauge monitoring hub forgings since 1984 there is no requirement to do so within the associated Certification Requirements. In view of such findings, it is recommended that:

> Airworthiness authorities should require that detailed stress analyses and direct strain gauge monitoring are carried out on all propeller hubs as part of the associated Certification Requirements.

The grease nipple hole was drilled and threaded from the outer to the inner hub surface, although the grease nipple only utilised the outer half of the thread, thereby negating the presence of a threaded hole at the inner hub surface. There was no requirement to protect the threads of the grease nipple holes from being damaged during the shot peening of the hub surfaces. The modified hub, post-1983, repositioned the grease nipple holes to a lower stressed, thicker, part of the casting and were only drilled and threaded to a depth sufficient to house the grease nipples. The remainder of the holes, through to the inner surface of the hub, were of much smaller diameters sufficient to allow passage of the lubricant to the blade pitch change bearing.

The majority of all propeller hubs that are in use on reciprocating engines were designed and certified during the same era as the Hartzell HC-()3Y()-() propeller hub and therefore could have similar design weaknesses.

2.7 Previous failures

A number of previous failures of this hub type had occurred of which some had resulted in the separation of a blade from the hub. It would appear that the majority of these failures, not including agricultural aircraft, had occurred to the left-hand rotating propeller fitted to the Lycoming LTIO 540 series engine, as fitted to the Piper PA-31-325 and-350 aircraft.

3 Conclusions

(a) Findings

(i) The commander was properly licensed and medically fit to conduct the flight.

(ii) The aircraft had been maintained in accordance with an approved maintenance schedule, and the Certificates of Airworthiness and Maintenance Review were valid.

(iii) A propeller blade detached from the right propeller hub in flight and the resultant inbalance induced detachment of this engine.

(iv) The detached propeller blade penetrated and passed through the nose baggage bay, struck and damaged the left propeller assembly and stopped the left engine.

(v) Despite a consequent and potentially critical loss of control and associated handling problems, the commander skilfully regained sufficient control to carry out a successful forced landing.

(vi) The right propeller blade detachment occurred due to a fatigue failure of the propeller hub casting.

(vii) The fatigue failure was of the type which had been known to occur since the early 1980's and initiated at the inner end of a grease nipple hole in the forward half of the hub casting.

(viii) The site of the crack initiation was at a thread form which had been damaged by two mechanisms; one by the shot peening of the inner hub surface whilst the hole had not been protected and the other, of indeterminate cause, which had resulted in distortion of the thread form.

(ix) The presence of the fatigue cracking from one of the grease nipple holes in the right propeller hub was not detected during a mandatory visual inspection of this hub 10.35 hours before this accident due to the inadequacy of the visual inspection requirement in FAA AD No. 89-22-05. This AD had not been revised to reflect the introduction of eddy current inspection by the propeller manufacturer, or to reflect the latter's strong recommendation to replace such hubs with an improved design.

(b) Causes

The investigation identified the following causal factors:

(i) Separation of the right engine, as a result of massive out-of-balance forces following fatigue failure of the right propeller hub and associated release of one blade, caused an immediate and critical loss of control which was only recovered and a successful forced landing accomplished by the exceptionally skilful handling of this commander.

(ii) The grease nipple holes in such Hartzell HC-()3Y()-() type propeller hubs had not been masked prior to the shot peening process at manufacture and had therefore suffered deformation of the associated threads, which weakened their fatigue resistance.

(iii) No detailed stress calculations from direct strain gauge testing had been undertaken, or had been required, on this propeller hub type at the time of its design and certification.

(iv) Fatigue cracking that emanated from deformed grease nipple threads and which broke through to the outside surface of the propeller hub may not have been visible at the last maintenance inspection prior to the in-flight failure of the right propeller hub.

(v) Despite the occurrence of fatigue cracking from grease nipple holes on such propeller hubs in service which had caused the manufacturer to redesign this type of hub in 1983, some 10 years prior to this accident, and to issue three related Service Bulletins in the period between October 1989 and September 1992 with the introduction of an optional eddy current inspection, in addition to visual inspection, the FAA had only issued one Airworthiness Directive (No. 89-22-05) requiring compliance with the initial SB 165. This had merely required periodic visual inspections and the FAA had not issued revised ADs to include eddy current inspections (as per SB 165A of 27 August 1992) or to emphasise the manufacturer's strong recommendation (SB 165B of 11 September 1992) for the replacement of such hubs with the improved post-1983 type of hub.

(vi) During the last visual inspection to FAA Airworthiness Directive (AD) No. 89-22-05, no cracking was observed on the propeller hub; the grease nipples had been removed from the hub to facilitate inspection. Such removal was not a requirement of this AD (which did not, however, warn against such removal) and may have tended to 'close up' any crack(s) present, reducing the chances of such visual detection.

(vii) The original hub design was certificated in the knowledge that the vibration stresses on the left-hand rotating propeller of this type were generally higher than those on the right-hand rotating propeller, but were deemed acceptable.

(viii) Operators and pilots of affected aircraft had not been made aware that the sudden initiation of unexplained vibration or grease leakage could indicate a potentially dangerous defect on such propeller hub assemblies although related Service Bulletins had warned aircraft engineers of such symptoms subsequent to 27 August 1992.

4 Safety Recommendations

The following safety recommendations were made during the course of this investigation:

4.1 Airworthiness authorities and manufacturers should ensure that when Airworthiness Directives and Service Bulletins are issued which contain important safety information which is also relevant to pilots, additional measures are taken to ensure that such pilots become aware of the relevant information.
(Safety Recommendation No. 94-28, made September 1994).

4.2 When airworthiness authorities issue Airworthiness Directives (ADs) that require compliance with a manufacturer's Service Bulletin (SB), such ADs should be updated when the manufacturer updates the SB, or clearly state that the AD relates to the latest issue of the associated SB at any point in time.
(Safety Recommendation No. 94-29, made September 1994).

4.3 The CAA and FAA should seriously consider issuing Airworthiness Directives to make manufacturers' strong recommendations to replace components a mandatory requirement where it is apparent that failure to replace such components could result in a potentially major hazard to the safety of affected aircraft.
(Safety Recommendation No. 94-30, made September 1994).

4.4 Airworthiness authorities should require that detailed stress analyses and direct strain gauge monitoring are carried out on all propeller hubs as part of the associated Certification Requirements.
(Safety Recommendation No. 94-31, made September 1994).

E J Trimble
Inspector of Air Accidents
Air Accidents Investigation Branch
Department of Transport

September 1994