Department of Trade

ACCIDENTS INVESTIGATION BRANCH

Brantly Helicopter 305 G-ATLO
Report on the accident near Astley Village, Stourport-on-Severn on 1 October 1976

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5/78	Brantly Helicopter 305 G-ATLO at Astley Village, Stourport-on-Severn October 1976	*(forthcoming)*

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Department of Trade
Accidents Investigation Branch
Kingsgate House
66/74 Victoria Street
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22 June 1978

The Rt Honourable Edmund Dell MP
Secretary of State for Trade

Sir

I have the honour to submit the report by Mr R G Feltham an Inspector of Accidents on the circumstances of the accident to Brantly helicopter G-ATLO which occurred near Astley Village, Stourport-on-Severn on 1 October 1976.

I have the honour to be
Sir
Your obedient Servant

W H Tench
Chief Inspector of Accidents

Accidents Investigation Branch
Aircraft Accident Report No. 5/78
(EWC 575)

Operator:		Freeman's of Bewdley (Aviation) Ltd
Aircraft:	*Type:*	Brantly Helicopter
	Model:	305
	Nationality:	British
	Registration:	G-ATLO
Place of Accident:		Nr Astley Village, Stourport-on-Severn Hereford and Worcester
		52° 18′ 30″ N 02° 18′ 50″ W
Date of Accident:		1 October 1976. 1536 hrs
		All times in this report are GMT

Synopsis

The accident was notified to the Department of Trade by West Mercia Police on 1 October 1976. An Inspector's Investigation was carried out and the following groups were established: Operations and Engineering.

The aircraft was on a local test flight following work on the main rotor to reduce the vibration level. After an apparently normal take-off and climb to a height of approximately 800 feet a violent vibration commenced. Although a measure of control was initially retained, the pilot could not prevent the aircraft crashing into a small copse where it caught fire and was destroyed. The three persons on board escaped with slight injuries.

The report concludes that the accident occurred following a loss of control experienced when a main rotor gearbox attachment bolt failed in flight. The gearbox attachment bolt failed due to fatigue.

1. Factual Information

1.1 History of the Flight

The aircraft was on a local flight from the operator's private landing strip near Astley when the accident occurred. Earlier the same day vibration sensors and stroboscopic examination equipment had been installed on the aircraft and a number of adjustments had been carried out on the main rotor with the object of reducing the vibration level. The technical observations were carried out by a vibration specialist and the adjustments were carried out by a licensed aircraft engineer. According to those involved this work had been successful and the aircraft is reported to have flown quite well with very little vibration during the short flights that were carried out after the adjustments.

At the conclusion of the last of these short flights, the work on the aircraft was terminated but, before the test equipment was removed, the pilot elected to carry out a further flight with a heavier load on the aircraft and at speeds approaching Vne*. The technical observer agreed to continue his readings during this further flight. He occupied one of the rear seats of the helicopter, with the portable part of his equipment supported on his knees, the other rear seat being occupied by a heavy passenger to give the required greater weight. The pilot, the only other person on board, occupied the right hand front seat.

The take-off and climb of the helicopter appeared to be normal. When the aircraft reached a height of approximately 800 feet, the pilot levelled and began to accelerate the aircraft, calling the indicated speeds as he did so. According to the observer, as the aircraft reached an indicated speed of 110 mph, he noticed that an oscillation of the rotor tips had commenced. He notified the pilot of this and suggested that the speed be 'eased off'. At about the same time the pilot announced that the speed had increased to 115 mph and, simultaneous with this announcement, a violent vibration commenced which was so extreme that further reference to the test equipment or to any of the aircraft's instruments was not possible. However, the observer formed the impression that the vibration was aerodynamically induced and subsequently estimated that it had frequency of about 5 cycles per second and an amplitude of about 3 to 4 inches in the vertical plane.

The pilot's recollection of events following the onset of the vibration is unclear. According to the passengers, however, some degree of control of the aircraft was retained, the aircraft's speed was reduced and various manouvres were executed in an effort to reduce the vibration. These efforts were unsuccessful and the aircraft became established in a descent of varying steepness which continued until the aircraft struck the ground.

The first impact was with trees which tore away the main rotor assembly together with its gear box from the aircraft. The fuselage then continued for a short distance through the trees before striking the ground slightly nose down a few yards further on. Although all of the occupants were wearing seat belts, the pilot was knocked unconscious in the impact, the observer was slightly injured and the other passenger was thrown clear into some light vegetation. Immediately after impact the aircraft caught fire but the less injured of the two passengers was able to extricate the pilot before the fire developed. The other passenger made his own way away from the wreckage. The aircraft was severely broken up during the impact sequence and much of its structure was destroyed in the ensuing ground fire.

*Vne Velocity Never Exceed airspeed limitation as defined in the Flight Manual.

1.2 **Injuries to persons**

Injuries	*Crew*	*Passengers*	*Others*
Fatal	–	–	–
Slight	1	2	–
None	–	–	

1.3 **Damage to aircraft**

Destroyed.

1.4 **Other damage**

Several trees in the vicinity damaged by impact and fire.

1.5 **Pilot information**

Pilot:	Age 29 years
Licence:	Commercial Pilot's Licence (Helicopters & Gyroplanes)
Certificate of test:	29 May 1976
Last medical examination:	31 October 1975
Restrictions:	None
Total flying hours: (Helicopters)	869 hours 50 minutes
In command (Helicopters):	820 hours 25 minutes
Total command hours on type:	35 hours 25 minutes

The pilot also held a Private Pilot's Licence with an IMC rating for fixed wing aircraft. He had completed the following amount of fixed wing flying at the time of the accident:

Total flying hours (fixed wing)	412 hours 25 minutes
In command (fixed wing)	364 hours 40 minutes

1.6 **Aircraft information**

1.6.1. *General*

Type:	Brantly 305
Constructor's No.	1028
Manufacturer:	Brantly Helicopter Corporation Frederick, Oklahoma USA

Year of Manufacture:	1966
Certificate of Airworthiness: (C of A)	UK Civil Aviation Authority General Purpose Category. No. 1922. Valid until 20 May 1978
Certificate of Registration:	Registered in the name of Freeman's of Bewdley (Aviation) Ltd on 15 April 1976
Certificate of Release:	Dated 19 August 1976 at 346.35 hours Recorded as Valid for 45 days or until 370 hours.
Total recorded hours flown:	358 hours 20 minutes
Total recorded engine hours:	358 hours 20 minutes
Hours since last C of A:	56 hours 40 minutes

1.6.2 *Aircraft loading*

The calculated total weight of the aircraft, at 2,692 lbs, was below the maximum permissible total weight of 2,900 lbs. The centre of gravity was calculated to be at 103.48 inches aft of the datum point ie within the permissible limits of 101.0 to 107.2 inches aft of the datum.

1.6.3 *Aircraft history*

The aircraft was imported into the United Kingdom and first granted a British Certificate of Airworthiness in 1966. In 1967 the Brantly 305 temporarily lost its Type Certificate because of a number of accidents caused by main rotor torsion strap failures and G-ATLO was stored for nine months until mid-1968. The Type Certificate was restored in January 1968 when an Airworthiness Directive (AD) was issued requiring immediate replacement of all torsion straps. However, the manufacturing company ceased trading later that year and G-ATLO was again withdrawn from service when its Certificate of Airworthiness expired towards the end of 1969. At that time it had accumulated a total of 299 operating hours. It was held in storage until early 1976 when the aircraft was acquired by its final owner. Following maintenance action agreed with the United Kingdom Civil Aviation Authority (CAA) the aircraft was granted a British Certificate of Airworthiness in May 1976. Previous Certificates of Airworthiness had been issued for this aircraft in the Public Transport Category and also in the Private Category. The Certificate of Airworthiness granted in May 1976 was in the General Purpose Category and carried a condition that the aircraft must be maintained in accordance with the Approved Maintenance Schedule No. ARB/GPMS/H/1971 as amended. According to the records the aircraft had completed 80 flights since that time and had flown a total of approximately 358 hours at the time of the accident.

On 29 August 1976 the pilot in command at the time of the accident experienced a significant vibration in the subject aircraft during a final approach to land. The licensed aircraft maintenance engineer who normally carried out maintenance work on the aircraft diagnosed that one of the hydraulic drag dampers on the main rotor was sticking. The complete set of three dampers was replaced by a set of functioning dampers from another aircraft and this particular vibration problem did not recur. The owners and the engineer concerned agreed, however, to invite a vibration specialist to assist them in identifying sources of residual vibration and to attempt to reduce this.

The work was carried out on the day of the accident by the engineer, the pilot involved in the accident and the vibration specialist using commercial track and balance instrumentation which consisted of vibration sensors fitted to the aircraft and stroboscopic examination equipment. The power supply for the above equipment was taken from the aircraft electrical system by means of a temporary connection in the cabin. Observations were made during ground runs, hover and forward flight and adjustments were made to the main rotor mass balance by changing the balance weights on all three blades before the initial flight and, subsequently, by adjustments to the blade tracking. The final adjustments were followed by a further flight during which satisfactory vibration levels and blade tracking were obtained. This further flight immediately preceded the flight on which the accident occurred.

1.6.4 *Maintenance history*

Immediately prior to its re-certification in 1976 the aircraft had lain in storage for more than six years. As part of the work for re-certification a 'Check 5' inspection to the Approved Maintenance Schedule was agreed by the CAA to be necessary together with an internal examination of the main gear box. The General Purpose Maintenance Schedule, Helicopter (ARB/GPMS/H/1971), as amended, and the Brantly 305 Maintenance and Overhaul Manual were used as the approved reference documents for this work which was completed in May 1976.

The following items, the location of which are indicated in Appendix I, were considered to have maintenance aspects relevant to the investigation:

(a) Freewheel Clutch. The freewheel clutch was noted in the component record compiled by the licensed engineer concerned as being due for overhaul at 600 hrs. However, both the component history card and a revision to the Maintenance and Overhaul Manual dated 26 October 1967 cite 300 hours as the overhaul time. The Brantly 305 Maintenance and Overhaul Manual supplied to the engineer shortly before the work was undertaken did not contain the revision and scheduled the clutch for replacement at 500 hours. The clutch was not replaced or overhauled at the time of the above re-certification.

(b) Flapping Hinge bearings. The three main rotor blades are each articulated at two points. The main flapping hinges are at the blade roots but the drag hinges are at approximately one third span with a limited degree of flap available also at that station. Each main flapping hinge is contained in a clevis to which is attached a torsion strap which takes the main centrifugal loads and permits blade angle change. The main flapping hinges comprise two needle roller bearings, known as clevis bearings, in each clevis. The aircraft manufacturer has recognised that the performance of the bearings is directly related to the lubrication of their loaded surfaces which are at the sides closest to the hub centre. It has been found that these surfaces may not receive adequate lubrication because the lubricant tends to centrifuge towards the unloaded surfaces furthest from the hub centre. The bearings have been the subject of a number of service instructions.

The bearings installed in G-ATLO at the time of the accident were of an authorised part number. The maintenance records show that these bearings were originally installed on 5 February 1968 at 236.05 aircraft hours. The records also show that on 15 September 1968 at 271.05 hours, they were re-orientated through 180 degrees in accordance with AD 68.1.7. Instructions describing this procedure were contained in Service Kit 305-67-5 and an amendment to the Maintenance and Overhaul Manual dated 18 December 1967. According to these instructions the bearings should be re-orientated after 50 hours of service to place fresh bearing surfaces and rollers on the inboard, loaded, side of the bearing. After a further 50 hours the bearings should be discarded.

During the pre-certification maintenance action at 299.15 aircraft hours the bearings were not changed but were rotated thus placing, in the loaded sector, surfaces which already had 35 hours recorded usage. AD 76-08-06, which requires the bearings to be checked for roughness and re-states the time limits specified in Service Kit 305-67-5, was noted as being complied with. At 346.35 hours a maintenance Certificate of Release was issued which contained a note: 'Check 1 insp. and clevis rotation at 370 hours, 23.25 hours left'. No further maintenance action on the bearings is recorded up to the time of the accident.

(c) Main transmission gearbox attachment bolts

In undertaking the main transmission gearbox inspection at the time of its last certification the four main transmission gearbox attachment bolts were removed and were used again when the gearbox was re-installed. The main transmission gearbox fitted to the aircraft at the time of the accident was not the gearbox originally fitted to the aircraft, a replacement gearbox having been fitted in February 1968. During the life of the aircraft both gearboxes had been removed and refitted on a number of occasions but there is no evidence, either from the records or elsewhere, that the main gearbox attachment bolts had ever been renewed.

The 500 hour interval inspection in the Brantly 305 Maintenance and Overhaul Manual requires the main transmission to be removed and new mounting bolts to be used on re-installation. This is the only reference in the Manual to the time at which the bolts should be renewed. However, a Special Transmission Inspection in the same Manual requires that – "Until rescinded as a result of service experience" an internal inspection of the transmission will be carried out at 300 hours intervals. The purpose of this Special Inspection is to determine the internal condition of the transmission and, in defining the sequence of operations to be carried out under this inspection, the Manual cross-refers to those parts of the 500 hour interval inspection relating to disassembly and to the inspection of the gear teeth. The Special Inspection does not specifically call for replacement of the attachment bolts when the transmission is re-installed.

Item 35 of the Check 5 (Airframe – Structures Section) of the ARB/GPMS/H/1971, the Check 5 of this Schedule having been the basis of the last re-certification (See 1.6.4 para 1) called for the following work to be undertaken – "Remove at each main structural joint and primary load carrying brackets sufficient attachment bolts to determine serviceability of bolts and housings. Crack test attachment bolts and associated castings and brackets".

A Foreword to the ARB/GPMS/H/1971 authorises a degree of flexibility in its use in that it states that the Schedule is intended to permit Engineers and Organisations to use their discretion in the selection of specific inspection items and work activities which, in their engineering judgment, are necessary to preserve the airworthiness of the helicopter. At the same time it emphasises the need for careful assessment of the work requirements and in particular, calls for due consideration to be given to the age, condition, previous recorded history, hours flown and elapsed time since the requirements were last satisfied. Although provision is made for items in any Check, or any part thereof, to be waived, provided a record of such is made in the appropriate Log Book, the need is emphasised for due regard to be given to recommendations issued by Manufacturers either in Maintenance, Overhaul and Repair Manuals, Maintenance Schedules, Technical Instructions, Service Bulletins or other publications. Mandatory Instructions issued by the CAA or the Airworthiness Authority of the country of origin of the helicopter cannot be waived.

The ARB/GPMS/H/1971 contains a requirement for records to be kept of the hours/time accumulated on items required to be scrapped, retired or overhauled at specific periods. The components record for G-ATLO, raised by the engineer concerned at the time of the last recertification and showing the hours accumulated and times at which overhaul or retirement of components would become due, did not contain a reference to any retirement life for the main transmission gearbox attachment bolts.

The CAA have confirmed that the attachment bolts do not, in their view, have a scrap life because such a life is not laid down either in the Type Certificate Data Sheet, Maintenance Manual or Manufacturer's Service Bulletins relating to the aircraft. Furthermore the CAA has stated that the note in the Maintenance and Overhaul Manual requiring the bolts to be renewed at the 500 hour interval inspection is not to be interpreted as defining a 500 hour scrap life for the bolts. However, the CAA has recorded its view that for reasons of good engineering practice in this specific application, the bolts are required to be renewed at each and every occasion on which the transmission is dismantled and that this requirement overrides the need for removal and crack detection under Item 5.1.35 of the ARB/GPMS/H/1971.

The engineer concerned has stated that he did not renew the bolts at the time the gearbox was last installed because the aircraft had completed only 299 hours at the time and he considered that the bolts should normally be replaced at 500 hours.

The engineer also stated that the bolts were visually inspected and crack-tested by a dye penetrant method before refitting. No entry relating to this examination of the bolts, to the fact that they were re-installed or to the time when the engineer considered they would be due for replacement was entered in the maintenance records.

1.7 Meteorological information

An aftercast of the weather situation in the vicinity was prepared by the Meteorological Office at Gloucester as follows:

Date and time:	1 October 1976 1536 hrs
Surface wind:	110 deg 10-15 kt
Weather:	Occasional showers, isolated thunderstorms
Surface visibility:	20 km but 6 km in showers or thunderstorms 3/8 cu sc 2,500 ft tops 12,000 ft but 7/8 cu cb 1,200 ft tops 25,000 ft in showers and thunderstorms
Inference:	Depression centred over Devon with unstable South easterly airstream over the West Midlands
Zero deg C Isotherm:	7,500 ft
Conditions of light:	Full daylight

The weather is not considered pertinent to the accident.

1.8 Aids to navigation

Not applicable

1.9 Communications

Not applicable

1.10 Aerodrome and ground facilities

Not applicable

1.11 Flight recorder

Not required and none fitted

1.12 Wreckage

1.12.1 Wreckage examination on site

The aircraft crashed into a dense thicket of small trees and bushes separating two fields. The direction of impact through the trees was between 112^{o} and 115^{o} magnetic.

There were numerous and substantial main rotor blade strikes to the starboard (advancing blade) side of the impact path for approximately half the distance of the aircraft's passage through the trees at a height of between 3 to 6 feet above ground level. These strikes yielded consistent evidence of aircraft impact angles corresponding to 20 degrees "nose up" and 35 degrees "starboard side down" along the track.

The main rotor blades were found in several fragments and the aircraft's undercarriage, one tailplane and the rotor head with the main gearbox were found separated from the fuselage but lying in its immediate vicinity. The fuselage lay on its starboard side and had been destroyed by fire. Its condition as found suggested that it had been basically intact but distorted before the fire occurred. The tail rotor was present and intact. Two main rotor tips were recovered but the third was not found. The main rotor trailing edges which are foam filled with a metal outer skin, were found in a large number of fragments in the immediate vicinity of the main wreckage.

1.12.2 Further examination of wreckage

The wreckage was removed from the site for further examination. A detailed examination of the remains of the aircraft was carried out and, in particular, the power unit, transmission and main rotor system were examined for possible defects and for sources of abnormal vibration.

The power unit was badly damaged by fire, to the extent that much of the fuel system and engine auxiliaries had been consumed. Post-crash examination of the electrical system, which was also severely fire damaged did not prove productive. A detailed strip examination of the engine revealed no evidence of a pre-crash defect. The engine mountings, though badly fire damaged, showed no sign of pre-crash failure. An examination of the main gearbox revealed no internal damage. The nature of the damage to the flexible coupling, the freewheel clutch and the centrifugal clutch, which form the transmission train between the engine and the gearbox was consistent with the high speed detachment of the gearbox during the crash. The ball bearing locating the end of the gearbox input shaft into the end of the engine crankshaft had severe hammering on its inner race which indicated that it had been subject to heavy and reversing axial loads. The stub shaft was found to feature a modified extension (see section 1.17). The chrome surface of the shaft where it locates in the bearing was discoloured due to overheating.

Though the main rotor blades had sustained substantial damage and the outboard blades were fractured in several places the damage was all consistent with the final impact. Several inches of leading edge spar were missing from each blade and trailing edge tip formers were recovered from only two blades. However, the metal skinning of the foam filled trailing edge was found for almost all of each blade and, in particular, for each blade tip.

The inboard and outboard flapping stops on the main rotor showed no impact damage.

At the drag hinge the movement of each blade is controlled through the action of two dampers, one compressed rubber damper and one hydraulic damper, acting in parallel. Impact marks showed that the blades had hit their forward stops and in an extreme and violent rearward movement had damaged the rubber dampers in compression. The attachments of all the hydraulic dampers to the outboard blades had fractured and an examination of these dampers showed that they were all damaged internally in a manner consistent with a violent extension. The examination also revealed that a modification to prevent hydraulic locking had been incorporated and that there was no evidence to indicate pre-crash malfunction of the dampers.

The flapping hinge bearings were examined and on all but one the grease in the loaded sector was found to be contaminated with corrosion products. In one blade the grease in the leading bearing was severely contaminated with corrosion products and was stiff and rough to turn. When the bearing was cleaned the roughness virtually disappeared. The trailing bearing on this blade showed some tendency to stick at one position.

In the flapping hinge from another blade the bearing inner races were not in the orientation recorded in the log book entry but were turned at about 90 degrees to it. In one of the two bearings the surface presented for load bearing had corrosion present. The rusted surface showed signs of having been "worked" since the corrosion formed.

The rotor hub retaining nut was tight, properly locked and showed no signs of having moved since installation. However, when the hub was removed, the conical nut on the mast on which the hub rests was found to be severely fretted on the bearing surface and this applied also to the corresponding surface on the hub.

The main rotor control mechanism had sustained some damage but there was no evidence of pre-crash failure or disconnection.

On this model of helicopter the main gearbox is attached, at the forward end, to reinforced points on the cabin rear bulkhead and, at the rear, to a welded tube frame. The rear support frame members had all failed in one or more places and exhibited brittle fractures. The front port mounting point had been torn from the bulkhead in a manner which indicated that the gearbox, to which it was attached, had detached in a clockwise direction as viewed from above. The front starboard mounting point was still in position and had received no distortion damage at the time of gearbox detachment. Examination showed that the attachment bolt at this point had failed at the gearbox/attachment lug position approximately in the plane of the gearbox lug face. Examination of the fractured surfaces of the bolt at this position showed that they bore indications of fatigue. Although subjected to a degree of bending none of the remaining three mounting bolts had failed.

1.12.3 *Examination of main gearbox starboard front attachment bolt*

Metallurgical examination of the main gearbox starboard front attachment bolt confirmed that fatigue was present in the fracture and that it had progressed through 90% of the cross-section. The fracture occurred in the shank of the bolt remote from the threaded portion and within the bore of the gearbox flange. A corrosion pit was present at the origin of the fracture. The fatigue exhibited a smooth acceleration in the spacing of the striations to the final instantaneous fracture which was of a tensile nature. The fatigue markings were of a coarse, banded type and their uniform appearance suggested that they were associated with 'stop-start' transitions in the loading pattern and probably representative of individual flights. A count of the fatigue bands suggested that 90% of the fatigue growth was achieved in 20 to 30 loading cycles. No certainty could be applied to the number of cycles in the first 10% of growth but it was estimated that they amounted to no more than 200 and possibly much less. It was not possible to determine whether the fatigue had been present during the aircraft's long period of storage. A hardness test on the half of the bolt which had not been fire damaged yielded a tensile strength of 74 tons PSI which is within the required specification. The contact face between the gearbox flange and its airframe mounting point showed evidence of hammering which was absent from the other gearbox attachment points.

1.13 Medical and pathological information

Not applicable

1.14 Fire

Fire broke out in the fuselage area of the aircraft shortly after impact and rapidly developed. Four appliances from the civil fire brigade reached the scene approximately 16 minutes after the accident occurred and extinguished the fire in the aircraft and surrounding woods.

1.15 Survival aspects

The accident was survivable. All occupants were wearing lap straps which, in the case of the pilot and the observer, were still intact after the accident. The pilot, although only slightly injured, was knocked unconscious in the impact.

1.16 Tests and research

None

1.17 Additional information

The Brantly 305 was known, during its main period of operation from 1965 to 1970, occasionally to suffer in flight from a violent vertical vibration associated with the main rotor. It appears that no specific cause of this behaviour was found but in instances where the aircraft was not damaged it was sometimes possible to eliminate it by attention to defects in control rod end wear, rotor balance and tracking, engine rough running, drag damper malfunction and flapping hinge bearing roughness. It was considered that the type had a propensity for this sort of behaviour due to some main rotor aero-mechanical interconnection, a situation exacerbated by any of the defects mentioned above. Additionally it was considered that the engine mounting possibly allowed sufficient vertical movement for the stub on the gearbox input shaft to disengage partially from the locating ball bearing at the end of the crankshaft and thereby introduce vibration due to shaft whirl. To combat this possibility the United Kingdom agents introduced a modification in 1967 which extended the stub shaft to ensure it full retention in the bearing. This modification carried United Kingdom Air Registration Board approval.

2. Analysis

When the aircraft crashed into the wood it appears to have been steeply banked to the right with considerable nose-up pitch. Its forward speed could not be estimated accurately but it is not considered to have been high because the fuselage received little damage other than removal of its undercarriage and the main gearbox. The impact evidence is reasonably consistent, therefore, with witness evidence that the aircraft was turning during its descent and with the passengers' impressions that the pilot had been able to effect a degree of flare before the final impact.

Though a fatigued bolt was found in a critical location on the main gearbox attachment this did not provide immediate evidence that it constituted the primary cause of the accident.

Three hypotheses were initially examined to explain the final failure of the bolt. Firstly it could be postulated that the already extensively fatigued bolt was exploited to final failure by the ground impact forces. As a second alternative the final failure may be considered to have occurred in the air following rapid fatigue progression due to abnormal loading of some kind earlier in the subject flight. The third possibility examined was that the failure occurred in the air following steady progression of the fatigue under normal flight loads.

Consideration of all the evidence surrounding the bolt failure, however, rules out the first possibility and, for the following reasons, leaves no doubt that the bolt failed during flight. The fatigue fracture itself had progressed through 90% of the cross-sectional area of the bolt and had the appearance of a smoothly accelerating process into the final brittle failure. Thus the extent of the fatigue and the rear-equality of the areas of the last fatigue step and the final fracture indicate that no abnormal load, over that producing the fatigue, was required to cause the final separation. The hammering marks on the gearbox flange around the location of the failed bolt also provided evidence that the bolt separated whilst the gearbox was still being retained by the other bolts. Additionally the absence of distortion on the gearbox starboard front mounting point and the nature of failure of the other three attachments during the impact sequence supported this evidence. It remained clear, therefore, that the above evidence was not consistent with the bolt failing at the time the aircraft struck the ground.

It could be postulated that the final fatigue was the result, and not the initiation, of the violent vibration experienced during the accident flight. The possibility was considered, therefore, that the fatigue was accelerated to failure during this final vibration which may have had some other cause. Although the aircraft type has a history of vibration problems examination of the wreckage did not produce evidence to indicate that severe vibration preceded the bolt failure on the final flight. Examination of the power unit produced no evidence to suggest that rough running may have occurred during the accident flight although this defect has been considered a factor in some previous instances where heavy vibration had developed on the same type of aircraft. In other instances of vibration on the type a single identifiable mechanical cause was not found but the behaviour has sometimes been ascribed to an accumulation of small effects such as control rod end wear, poor rotor tracking, flapping hinge bearing roughness, drag damper malfunction and the possibility of misalighnment or disengagement of the engine crankshaft from it locating stub on the main gearbox input shaft. Of these possible defects no evidence could be found to confirm that any particular one contributed to the problems on the accident flight. The main flapping hinge bearings had exceeded their approved lives and showed signs of deterioration with corrosion present. The poor condition of the bearings in one particular blade could have been a source of vibration and may have made the aircraft susceptible to abnormal vibration. The fretting on the lower seating face of the main rotor hub also indicated, for whatever reason, an unusual amount of movement in the hub at some stage and a consequent high level of repetitive loadings. The damage found on the gearbox stub shaft and the associated small bearing was also evidence that considerable relative movement and misalignment between the gearbox and enging shafts occurred at some stage. However, this movement may more readily be explained by displacement of the gearbox due to the broken bolt rather than indicating that the modified shaft extension was ineffective in some other vibration situation.

Whilst, therefore, the possibility cannot be entirely dismissed that a massive vibration associated with the main rotor accelerated the already existing fatigue in the subject bolt to failure, it is also clear that the final failure of the bolt could have produced the massive vibration with its attendant effects and, for the following reasons, seems the more likely possibility. The main rotor control rods pass immediately in front of the cabin rear bulkhead to which the front of the gearbox is attached. Any movement between the gearbox and the bulkhead would result in control inputs to the main rotor. The forces produced by rotor pitch changes, the fuselage mass and the reduced fuselage stiffness would result in further control inputs and conditions would then exist for either a runaway in control or a massive vibration. Additionally the gearbox may well have become misaligned with the engine crankshaft and a degree of whirl could have developed between the two shafts thus accentuating the out of balance conditions. Thus, whilst it cannot be stated with certainty that the known vertical vibration phenomenon did not occur during the accident flight it appears highly probable, considering both the above features and the characteristics of the fractured bolt, that the primary cause of the severe vibration and subsequent loss of control was the final failure of the main gearbox starboard front mounting bolt.

Tests showed that the bolt met the material strength requirements and that some other factor must have led to the initiation of fatigue. The corrosion pit at the fatigue origin may well have been the result of the long period of storage and appears to have been the main initiating factor. If the initiation occurred as a result of storage the fatigue progressed to final failure in only 57 hours of operation involving 80 flight cycles. On the other hand the possibility that the initiation commenced before the aircraft went into its last storage is indicated by the fact that 10% of the fatigue growth may have taken place over as much as 200 cycles.

The Brantly Maintenance and Overhaul Manual calls for new mounting bolts to be used on re-installation of the main transmission gearbox at the 500 hour interval inspection. The CAA has stated that the main transmission attachment bolts have no scrap life but that the bolts should be renewed at each and every time the transmission is dismantled.

The absence of a retirement life for the bolts on the component record of G-ATLO accords with this although no specific instructions calling for their renewal at every occasion on which the gearbox is dismantled are contained in the maintenance documentation for the aircraft. Such renewal would depend, therefore, upon the opinion of the engineer responsible for the work at the time that this action would constitute good engineering practice.

The engineer concerned with the last re-certification work on the aircraft considered that the bolts had a life of 500 hours, having based this view on the requirement in the Maintenance and Overhaul Manual to replace them at the 500 hour Check. He stated that he did not replace the bolts at the time of the last re-certification because the aircraft had only flown 299 hours up to that time. From his examination of the bolts he considered them to be serviceable, anticipating that they would be due for renewal after a further approximate 200 hours operation. However, as the aircraft was being certificated for the first time in the General Purpose Category and, from that time onwards, would be maintained to the ARB/GPMS/H/1971, the Check in the new schedule after a further 200 hours would not call for an examination of the bolts. On this basis, therefore, it would have been prudent for the engineer to have made an entry in the records at the time of re-certification indicating that the bolts had not been changed and also proper for him to indicate his view that the bolts would be due for replacement at the next 200 hour Check.

The Operator has stated that he considers that the main transmission gearbox attachment bolts should have been changed at the time of the last re-certification because, in his opinion, the Special Transmission Inspection was due at about that time and by cross-reference of this Inspection in the Maintenance and Overhaul Manual to the 500 hour Check, he considers that a requirement to renew the bolts was inferred. This inference can be challenged, however, in so far as the cross-reference applies only to particular paragraphs of the 500 hour Check. In any case, the 300 hour Special Transmission Inspection was not due on the gearbox then installed because of an earlier change of that unit.

As the aircraft had been in storage for a very long period prior to re-certification, and as the evidence of corrosion on the failed bolt indicates, it would have constituted good engineering practice to have replaced the subject bolts at the time of the last certification of the aircraft. However, both the absence of a special instruction and the latitude for discretion contained with the maintenance instructions then applicable did not appear to make renewal of the bolts mandatory.

Although the cause of the failure of the main transmission gearbox attachment bolt remains in no doubt, it is apparent from the immediately preceding paragraphs that the maintenance instructions for the subject bolts are open to more than one interpretation. It is considered, therefore, that a clarification of the maintenance requirements for these attachment bolts is now shown to be necessary so that possible future misinterpretation will be avoided.

Examination of the maintenance records showed that the free wheel clutch had exceeded its overhaul life. It appears that the documentation held by the licensed engineer concerned had not been fully revised in this respect but no evidence was found to suggest that the extended operation of the clutch contributed in any way to the accident.

During the pre-certification maintenance action at 299.15 aircraft hours the flapping hinge clevis bearings were rotated, thus placing in the loaded sector surfaces which already had 35 hours recorded usage. At 314.15 aircraft hours, the loaded surfaces of the bearings reached the maximum permitted 50 hours service in that condition and the aircraft should not have been operated, therefore, without further maintenance action on the bearings. There is no record of such action having been taken. At the time of the accident the loaded surfaces of the bearings had, according to the records, totalled an in-service time of approximately 94 hours. It was not possible to assess the degree, if any, to which the condition of the bearings had contributed to the vibration history of the aircraft.

There was no permanent provision in the aircraft for connecting the vibration sensing and stroboscopic equipment and its electrical supply was obtained by means of a temporary connection in the cabin. The installation could be deemed therefore to constitute a modification. CAA approval had not been sought prior to installation of the equipment but a certificate of compliance for the work had been entered in the aircraft records. There was no evidence to suggest that the installation contributed in any way to the accident.

3. Conclusions

(a) Findings

(i) The pilot was properly licensed and adequately experienced to carry out the flight.

(ii) A Certificate of Airworthiness had been issued in respect of the aircraft and was current at the time of the accident.

(iii) The aircraft was loaded below the maximum permissible weight and its centre of gravity was within the prescribed limits.

(iv) The maintenance of the aircraft had been undertaken on the basis of an Approved Maintenance Schedule but the associated airworthiness reference material specific to the aircraft type had not been complied with in the following respects:-

(a) The freewheel clutch had exceeded its overhaul life

(b) The main rotor flapping hinge bearings had exceeded their approved lives.

(v) The accident occurred following severe main rotor related vibration during flight after which it was not possible for the pilot to maintain full control of the aircraft.

(vi) The main rotor gearbox forward starboard attachment bolt failed in flight and this failure was the probable cause of the severe vibration.

(vii) The bolt failed as a result of fatigue which had its origin in a corrosion pit in the shank of the bolt.

(b) Cause

The accident was caused by a loss of control resulting from severe vibration which probably developed when a main rotor gearbox attachment bolt failed in flight. The gearbox attachment bolt failed as a result of fatigue.

4. Safety Recommendations

It is recommended that:-

4.1 An instruction be issued defining when the main transmission gearbox attachment bolts on the Brantly 305 should be renewed.

4.2 Consideration be given to inviting Operators and Engineers to report to the Airworthiness Authority upon any areas of maintenance instructions where it appears that parts of approved maintenance schedules are seen to cross-refer either within the same schedule or to other approved maintenance schedules in a manner which presents ambiguity, or which may lead to possible misinterpretation, in order that some guidance or instructions may be given to enable Operators and Engineers to determine the overriding requirement in such cases.

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Accidents Investigation Branch
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