Aircraft Type and Registration:	Agusta A109E, G-HIMJ	
No & Type of Engines:	2 Pratt & Whitney Canada PW206C turboshaft engines	
Year of Manufacture:	2003	
Date & Time (UTC):	14 March 2004 at 1600 hrs	
Location:	Colney Park Heliport, Leeds Bradford Airport, Leeds	
Type of Flight:	Private	
Persons on Board:	Crew - 1	Passengers - 1
Injuries:	Crew - None	Passengers - None
Nature of Damage:	All three landing gear units badly damaged. Wrinkling of the aircraft structure. Tail rotor drive shaft failure. Damage to the tail rotor drive shaft tunnel and engine drive shaft. Failure of four bolts securing lower attachment lug bracket of main rotor gearbox aft left brace assembly	
Commander's Licence:	Private Pilot's Licence	
Commander's Age:	54 years	
Commander's Flying Experience:	2,958 hours (of which 614 were on type) Last 90 days - 46 hours Last 28 days - 26 hours	
Information Source:	AAIB Field Investigation	

History of flight

The pilot reported that the aircraft lifted into the hover and made a normal departure into wind towards the west. Shortly after takeoff, at approximately 300 feet, a whining noise became audible. This was followed by a bang. The pilot realised that something serious had happened, but he did not know what this was, since all indications were normal and he still had control of the helicopter. He therefore elected to land immediately. He selected a nearby field and when flaring the helicopter near the ground, and applying the collective lever, the fuselage began to spin about the rotor axis. Upon landing, the helicopter stayed upright, although it was spinning and it rotated two or three times before stopping.

It was evident that the tail rotor and its control system remained intact but its drive system was no longer functional.

Significant aircraft features

The aircraft has a conventional layout with a main rotor gearbox mounted on a horizontal plate above the cabin section. The plate normally carries main rotor system torque reaction loads, whilst lift loads and moments applied by the rotor head are transmitted by four brace assemblies attaching the upper part of the gearbox to the top of the cabin structure. A drive shaft takes power from the front of each engine to respectively the left and right sides of the combiner gearing forming the rear of the main-rotor gearbox. These engine output drive shafts pass through holes in a titanium alloy bulkhead to reach the freewheels and input pinions of the combiner gearbox. The tail rotor drive shaft exits aft from the rear of the combiner gearing, between the input drive shafts.

The two forward main rotor gearbox brace assemblies slope upwards and aft from their forward attachments on the cabin roof and are angled at approximately 30 degrees to the longitudinal axis of the aircraft when viewed in side elevation. The rear brace assemblies have axes nearer the vertical when so viewed. The lower ends of both forward and rear brace assemblies are mounted close the edges of the cabin roof structure and slope inwards to their attachments near the top of the gearbox when viewed in front or rear elevation. (See Figure 1.)

The lower end of each brace assembly terminates in a fork fitting through which passes a bolt, also passing through a ball-joint within the lug of a steel attachment bracket. Each of these brackets incorporates four bolt holes which enable the former to be bolted to the main cabin roof structure through similarly spaced holes in a matching abutment bracket integral with the upper fuselage structure. Each abutment bracket is orientated in such a way that once the steel attachment bolts are installed, their axes lie approximately parallel with the axis of the corresponding brace assembly. In this way, tensile loads in any of the brace assemblies should result in evenly distributed purely tensile loading of each of the four corresponding attachment bracket securing bolts.

G-HIMJ was a sub-variant of the A109E design which, amongst other differences, utilised modified forward main rotor gearbox brace assemblies, incorporating vibration dampers, in place of the plain rigid brace assemblies hitherto employed. Whilst in service, the dampers in G-HIMJ were found to have become defective and the braces were temporarily replaced by components of the rigid design, owing to a supply problem with damper equipped brace assemblies. This was a standard procedure should damper equipped brace assemblies not be available. Subsequently, replacement brace assemblies, incorporating dampers, were fitted and these were in use at the time of the accident.

The tail rotor drive shaft consists of a tubular aluminium alloy forward section with Bendix couplings and a splined connection on the forward end, allowing slight relative longitudinal movement. This section is splined to the rear of the rotor brake unit, itself mounted on the output shaft as it exits the rear of the combiner section of the main rotor gearbox. The shaft then passes through a structural tunnel of sheet titanium alloy, positioned between the two engines. The tunnel has a constant cross-section over most of its length, but increases in cross sectional area at its forward end.

Aircraft history

The aircraft was nearly 12 months old at the time of the accident, having flown 276 hours and recorded 327 landings. The records showed that only routine maintenance had been carried out involving no dismantling or re-assembly of significant structure.

Post accident examination

Examination of the aircraft revealed that the forward section of the tail-rotor drive shaft had failed at a point some 6 inches aft of the rotor brake. Damage to the No 1 engine output drive shaft indicated that it had come into rotational contact with part of the adjacent bulkhead structure. Further examination revealed severe heat damage and cracking of the titanium alloy tail-rotor shaft tunnel at the junction between the tapered front section and the parallel-sided mid section. The lower attachment of the rear left brace assembly securing the main rotor gearbox to the aircraft structure had failed. The failure took the form of fracture of the four bolts connecting the steel attachment bracket to the aircraft structure.

The aircraft systems were partly dismantled to enable both ends of the failed forward tail-rotor drive shaft to be removed and to assist access to the lower attachments of both the rear brace assemblies of the main rotor gearbox.

Detailed examination

The tail rotor drive shaft had evidence of overheating at the point of failure and rotational scoring marks in the same area. Most of the fracture faces appeared to have deformed in a soft state, apparently as a result of coming into repeated mutual contact after failure. The short forward section of the shaft incorporated only one small fracture face which appeared not to have suffered further smearing and damage after failure. This face had the characteristic fibrous appearance of fractures which have occurred on light alloy structural and mechanical components which have failed under load when the components are known to have been subjected to fire or severe overheat at the time of failure. In addition, the internal painted surface showed evidence of heat discolouration. There was also some evidence consistent with torsional deformation of the shaft occurring whilst hot.

The lower attachment bolts of the rear left brace assembly were examined. The nuts and the fractured end faces of the separated threaded portions of the bolts remaining within the nuts revealed evidence of fatigue in all four fracture faces. Similar fatigue failure evidence was present on the corresponding fracture faces of the bolt shanks. The latter remained within the brackets after failure so their pre-impact locations could be determined. Two adjacent pairs of bolts had fatigue evidence across at least 95% of each of their cross-sections whilst the remaining two each had fatigue evidence across some 65%.

The corresponding four bolts in the attachment of the rear right brace assembly were examined. A torque wrench was used to assess the value required to begin further tightening of the bolts in question. One bolt was tightened until approximately the correct minimum torque value of 100 lbf inches was reached and required ¾ of a turn to achieve that torque figure. This bolt also required an initial 51 lbf inches torque to initiate rotation whereas the other three bolts rotated when peak torque values of less than 40 lbf inches were applied.

Non destructive examination of the bolts from the lower attachment of the rear right brace assembly was carried out. No evidence of cracking was found. Metrology carried out on both the failed and intact bolts confirmed that all were within the manufacturer's dimensional tolerances.

Further metalurgical analysis

An exercise in fatigue striation counting was carried out on the fracture faces of the bolts. This identified evidence of a number of fatigue cycles considerably fewer than the total number of flight cycles recorded on the aircraft since new.

Airworthiness follow-up action

The manufacturer stated that they had previously been informed that one attachment bolt had failed on the corresponding bracket of another 109E aircraft some time before the accident. The cause of the failure was not fully investigated by the operator and it was assumed at the time that this was a one-off event. After the accident to G-HIMJ, however, the manufacturer issued a Bolletino to all operators of relevant types which required an initial inspection, within five operating hours, to confirm that no bolts securing the brace assembly attachment brackets had failed. The Bolletino then required sufficient dismantling, within 25 hours operation, to enable the torque settings of all such bolts to be checked.

As a result of this Bolletino 12 loose bolts were identified affecting five aircraft: these bolts were related to one forward and five rear attachment brackets. When these returns were analysed, however, no particular pattern of build sequences or service use could be identified.

Design history of bracket

The 109 Series aircraft was originally developed with the lower attachment brackets for the gearbox brace assembly manufactured from aluminium alloy. It is understood from conversations with manufacturer's personnel that failure of one of these brackets occurred some years ago, in flight, whilst they were operating such a machine. The aircraft was reportedly able to remain airborne and under control for a period after the failure. It appears that the tail rotor drive was not affected on this occasion. Examination of the failed component reportedly indicated that it had suffered a fatigue fracture. As a result of this finding, it was realised that the fatigue spectrum of the bracket was insufficiently understood. The component was therefore replaced both on in-service examples of the type and on newly built aircraft by a geometrically similar component manufactured from a steel alloy. Being a material of considerably greater ultimate tensile strength (UTS) it is very much less susceptible to fatigue cracking under the same loading conditions compared with a geometrically similar aluminium alloy component.

It is understood that the rear left hand bracket normally carries the highest peak load of the four brace assembly lower brackets securing the main rotor gearbox.

Fatigue behaviour of bolted joints

The attachments between the brace assemblies and the airframe utilise bolted joints uniting the steel lug brackets and the main structure. This is atypical of traditional aeronautical practice in that the bolts are loaded in tension rather than shear. The presence of a ball joint within the attachment lug at the lower end of each brace assembly, combined with the orientation of the joint face, ensures that only axial loading is applied to the bolt group.

To achieve a satisfactory fatigue life in any such bolts requires them to be tightened to a level which creates within them a tensile stress exceeding that to which they would be subjected as a result of carrying only their working load applied via the brace assembly. Under such conditions, the elasticity of the two mating faces loaded in compression ensures that the tensile load in the bolts remains unchanged through a large range of varying tensile loads in the brace. The information for assembly of such a joint would thus be expected to require a specified bolt torque value which ensured significantly higher static bolt tension when assembled than the figure due solely to any repeated flight loads. Without such tension, the joint would have a severely limited fatigue life. This would not only result from the unsatisfactory joint characteristics but may also have been exacerbated by a modified load spectrum as a consequence of a changed dynamic response resulting from the altered stiffness of the rotor system locating structure.

Analysis

The loss of tail-rotor drive was clearly the result of the rotating drive shaft coming into contact with the titanium tunnel through which it passed. This created frictional heating evident in the form of cracking and discolouration of the titanium tunnel as well as the hot failure evidence visible on the shaft. This contact resulted from the failure of the rear left brace assembly attachment which allowed greater than normal displacement of the upper end of the gearbox under load. This permitted sufficient misalignment of both the tail-rotor drive shaft and the engine output shafts to bring the former and one of the latter into contact with fixed titanium alloy structure.

Some of the load normally carried by the failed brace assembly attachment is assumed to have been transferred to the remaining three brace assemblies and some to have transferred into a bending moment applied to the torque plate attaching the underside of the main rotor gearbox to the cabin roof structure. Use of the less rigid forward brace assemblies, incorporating dampers, permitted greater displacement of the gearbox and hence greater misalignment of the engine input and tail-rotor drive shafts connected to the gearbox on G-HIMJ than would be the case had the rigid brace assemblies been in use. In this way, the immediate consequence (loss of tail rotor drive) on the incident to G-HIMJ apparently differed from the more benign effects of the bracket failure experienced on an earlier 109 Series aircraft being operated by the manufacturer.

Examination of the bolt fractures made it clear that one pair of bolts on the failed attachment fractured almost entirely in fatigue, whilst the remaining pair initially continued to carry load. Thereafter, the remaining pair of bolts apparently alone supported the load previously shared with the failed bolts and a load, moreover, further accentuated in magnitude and accompanied by a major bending element, owing to the offset of the bolt reaction from the load axis of the brace. These remaining bolts appear to have finally failed in overload after fatigue cracks had propagated approximately half way across the section, a failure sequence consistent with most of the fatigue damage and all the overload failure occurring in this bolt pair after fracture of the first pair of bolts.

The unusually short in-service life of this failed joint contrasts with the reported generally trouble-free nature of these joints on other 109E aircraft. There was no obvious cause for this early failure. The presence of lower than normal torque settings on the bolts of the corresponding right brace bracket attachment is, however, significant. If one or more of the bolts of the failed attachment had similarly been installed with low torque and hence low pre-load, it is possible that a loading cycle could have been applied in service to one or more of the bolt group. This contrasts with the assumed design condition of bolts pre-loaded to a level at which a steady tension force is retained therein, as a result of elastic deformation of the joint faces, regardless of the presence or absence of tensile flight loading in the brace.

Since the rear left brace assembly is subjected to the highest design loading of the four braces, the bolted joint of its lower bracket (the failed joint) would be likely to suffer the onset of fatigue damage before the corresponding bolts of the rear starboard joint, if broadly similar and insufficient installation torque values had been applied to both bolt groups.

The evidence of fatigue failure in this low-hours machine, combined with the low number of recorded flight cycles, thus strongly suggests that the failed bolts must have been incorrectly torque tightened. There is no record of the joint being dismantled during the short service life of the aircraft and the extensive dismantling of mechanical components required to gain access to this area virtually precludes the possibility of such activity occurring without it being recorded.

It is understood that correct setting of the torque values on the bolts in question, at installation, is sufficient to ensure that appropriate torque, and hence tension, remains in those bolts throughout their service lives. It must therefore be assumed that the low torque values noted in the rear right brace assembly attachment bolt group existed from manufacture, or that they were lower than the specified values at installation. These low torque values indicate that an assembly quality failure occurred when this aircraft was manufactured and in the absence of any evidence of a pre-existing defect in the failed left side joint, it must be assumed that a similar assembly error occurred in that position causing the observed fatigue damage.

The level of cyclic loading occurring once per flight, if distributed equally between the group of incorrectly torque tightened bolts, would have been of insufficient magnitude to create rapid fatigue damage, which suggests that those bolt torques and hence pre-load values varied between the four bolts. This inequality, taken in conjunction with the relatively high stiffness of the bracket, would result in not only the static preloads but the peak values of the cyclic operating loads being very unevenly distributed between the four bolts in the joint.

It would thus be expected that a high proportion of the peak repeated loading initially applied to the joint would be reacted by one bolt. Under such circumstances the cyclic loading on that bolt would be high in relation to the UTS of the material and thus able to rapidly initiate and sustain cumulative fatigue damage. Thereafter, either shortly before or immediately after failure of the first bolt, a second, adjacent bolt would sustain an excessively high cyclic loading as a result of the cyclic loading of the brace. This would produce conditions conducive to rapid fatigue initiation and progression in that bolt. Once the second bolt failed, very high tensile and some bending stresses would be present in the remaining pair, now offset from the load axis, and subjected to consequent mechanical advantage. This would lead to rapid failure, a significant part of the cross-section failing in overload. This was the observed condition of the fracture faces of two adjacent failed bolts.

The important function of the failed joint emphasises the need to ensure that the fatigue life can be predicted (or exceeds the life of the remainder of the airframe). The orientation of the bolted joint, in relation to load direction, dictates that this can only be realised with robust control of bolt assembly torque values achieved during initial build and on any subsequent refitting of the bolted joint. The manufacturer stated, however, that an evaluation of the lower right attachment residual strength, in the event of failure of the lower left attachment, would still provide a positive margin of safety. This evaluation was valid for the design manoeuvre case.

Conclusion

The evidence is consistent with incorrect torque tightening of the bolts of the failed joint at manufacture of the aircraft. The installed bolt tension is of critical importance to the integrity of attachment between the lower brackets of the brace assemblies and the cabin structure of the 109 Series aircraft.

Following this accident Agusta have reviewed the assembly process and issued revised guidance to the production team. In addition, the production documents have been revised to include identification of the torque wrench used, the calibration expiration date and the torque values applied to the bolts.

Figure 1