

ACCIDENT

Aircraft Type and Registration:	McDonnell Douglas Helicopters Hughes 369E, G-KSWI	
No & Type of Engines:	1 Rolls Royce Allison C250-C20B turboshaft engine	
Year of Manufacture:	1986	
Date & Time (UTC):	19 June 2011 at 1317 hrs	
Location:	Glastonbury, Somerset	
Type of Flight:	Private	
Persons on Board:	Crew - 1	Passengers - None
Injuries:	Crew - 1 (Serious)	Passengers - N/A
Nature of Damage:	The aircraft was destroyed	
Commander's Licence:	Private Pilot's Licence	
Commander's Age:	57 years	
Commander's Flying Experience:	730 hours (of which 720 were on type) Last 90 days - 20 hours Last 28 days - 10 hours	
Information Source:	AAIB Field Investigation	

Synopsis

While flying in the cruise at an altitude of 2,200 ft amsl, it is probable that the helicopter sustained a mechanical failure that resulted in the loss of pitch control to one of the tail rotor blades. During the subsequent attempt to land in a field, the airspeed reduced to the point where directional control of the helicopter seems to have been insufficient to maintain heading. At a height of approximately 50 ft, the helicopter yawed rapidly to the right before the rotation ceased and it developed a high rate of descent. The helicopter struck the ground heavily and was destroyed. The pilot survived but sustained serious injuries. There was no fire.

The investigation established the presence of fatigue cracks emanating from corrosion pits on the tail rotor

blade pitch horn on one blade, which led to its failure. Also, the associated tail rotor pitch link had failed. The sequence of the two failures could not be established but either could explain the helicopter's behaviour before it crashed. Neither the failed section of this tail rotor blade pitch horn nor the associated pitch link were recovered from the accident site.

Four Safety Recommendations are made.

History of the flight

The pilot flew the helicopter from his home in Cornwall to Draycot Airfield, near Swindon. Two hours after arriving at Draycot the pilot was seen to complete his checks prior to departing westwards, at

around 1250 hrs, on the return flight back to Cornwall. Witnesses to the departure say the helicopter looked and sounded normal. Weather conditions at the time were good. Thirty five miles to the west, the Bristol Airport actual weather was reported as: wind 14 kt from the southwest, visibility greater than 10 km and a temperature of 15°C. At 1255 hrs, whilst flying at an altitude of 2,200 ft, the pilot established communications with Bristol ATC who provided him with a Basic Service. At 1317 hrs, Bristol ATC called G-KSWI but there was no reply; about the same time the radar return from G-KSWI indicated that the helicopter was descending. The radar return then disappeared when the helicopter was in the vicinity of Glastonbury Tor.

At approximately 1315 hrs, witnesses in the area of Glastonbury saw a helicopter descending in a westerly direction to the south of Glastonbury Tor. Many witnesses described the helicopter as making a loud “clunking sound”, which they considered to be abnormal. The helicopter was seen to descend to low level, where it flew an orbit to the right and then at an estimated height of about 50 ft it started to spin, with the nose rotating to the right. Witnesses reported that the noise from the helicopter then reduced and it appeared to stop spinning, before “dropping” to the ground. The helicopter was severely damaged but there was no fire. The emergency services were quickly on the scene and airlifted the pilot, who had sustained serious injuries, to hospital.

Witnesses

A teacher was playing with her young children on a school tennis court (see Figure 3) when she heard the noise of a helicopter, which was descending towards her house 40 m away. She reported that the helicopter flew at low level, almost over her house,

before it started to fly around an adjacent field, which contained a herd of cows. It had almost completed a full orbit to the right, approximately 100 m from where she was standing, when it began to spin clockwise. It completed at least two rotations before the noise reduced and it “dropped” to the ground. She immediately ran into the house to alert her husband and the emergency services.

The teacher’s husband was inside the house when he heard a noise which he described as “like something being stuck in a Hoover but a lot louder.” He ran outside and went towards the wreckage where he found the pilot, who was unconscious, hanging from his seatbelt partly out of the left side of the helicopter. There was a strong smell of fuel.

There were a number of other witnesses to the accident and their accounts were consistent with the teacher’s statement. The only difference was the number of rotations the helicopter made before it “dropped” to the ground, with reports varying between two and six.

Meteorology

The weather around the accident site was under the influence of a transient ridge of high pressure which maintained a westerly flow over the area. Cells of convective cloud had developed over southern England and close to the Glastonbury area which produced light showers. Visibility was generally between 26 and 40 km, reducing to 7 km in these light showers. The surface wind was from between 250°M and 270°M at a strength of 12 to 15 kt.

Recorded flight data

The helicopter was equipped with a FlymapL GPS, which is able to record a flight log of GPS time, position, altitude, track and groundspeed, for any flight

that exceeds a user defined groundspeed, which in this case was set to 40 kt¹. Data was successfully recovered for the accident flight, together with a number of previous flights. Radar data was also recovered from the Clee Hill radar head, positioned about 70 nm to the north of the accident track; however, coverage was lost as the helicopter descended below 1,500 ft when line-of-sight contact was lost. Figure 1 compares the GPS and radar recorded data for the last two minutes of

the flight. The radar groundspeed is an average speed based on the straight-line distance covered between each consecutive radar contact, every eight seconds, and the radar pressure altitude is based on 1013 mb with ±50 ft accuracy.

From the GPS it was determined that, on the morning of the accident, a flight of 55 minutes duration was made. The accident flight commenced

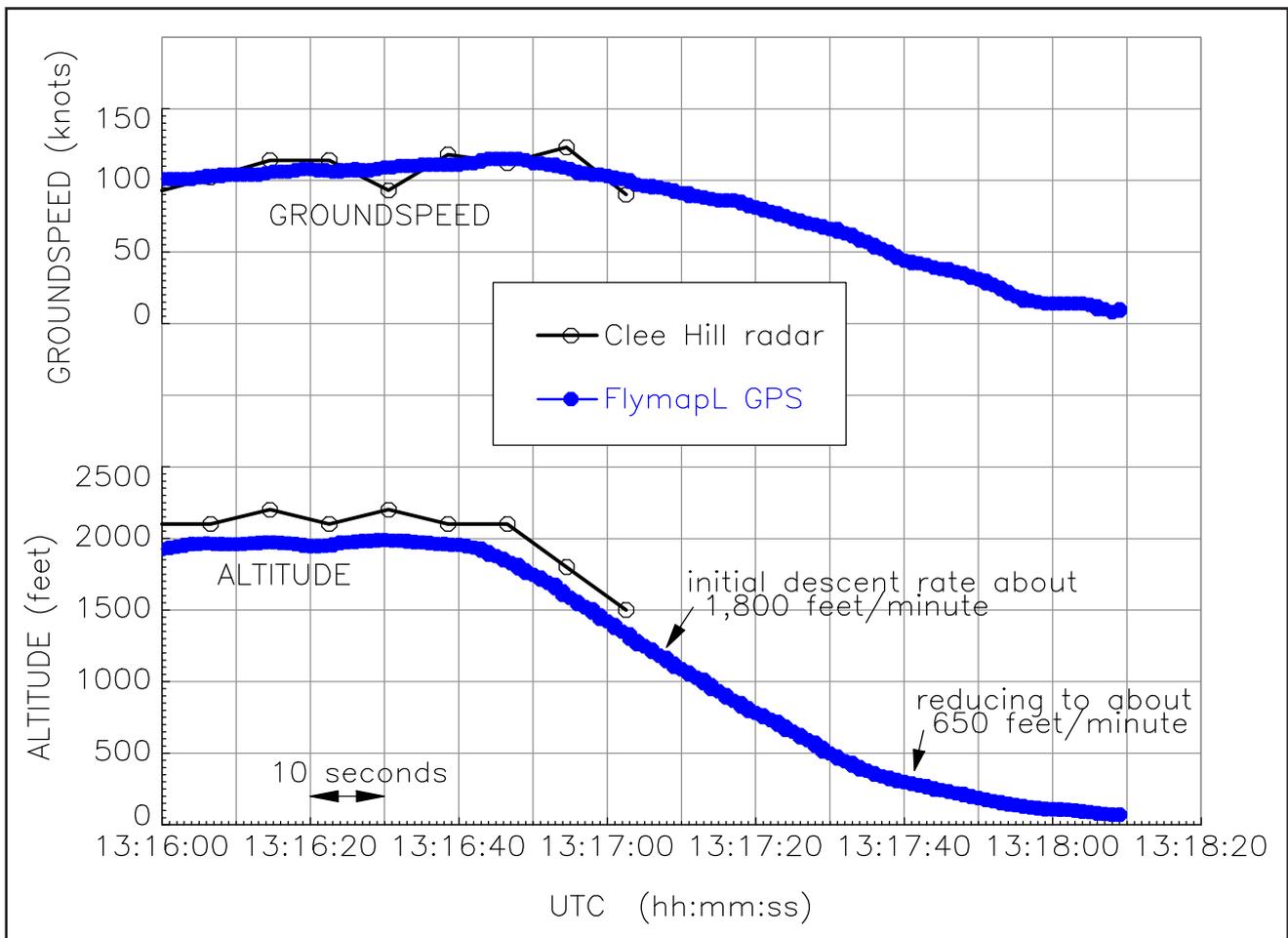


Figure 1
Recorded GPS and radar data for the final descent

Footnote

¹ The user can pre-select either a groundspeed of 2, 10, 20 or 40 kt. When the groundspeed falls below the selected value, the GPS starts writing the flight log to its non-volatile memory. This writing process takes a finite amount of time, during which additional data points are logged and recorded. For the accident flight this amounted to an additional 25 seconds worth of data below 40 kt.

2 hours 25 minutes later. Due to the logging logic, the actual time of the flight would have been slightly longer than the 55 minutes recorded and the period the helicopter spent on the ground without the engine and rotors turning would have been slightly less than 2 hours 25 minutes.

For the accident flight, 25 minutes of flight data was recorded, starting at 1253:11 hrs. Again, the actual flight time would have been longer.

Approximately 90 seconds before the end of the log, the helicopter started to descend. The first part of the descent lasted about 50 seconds and the rate of descent (ROD) averaged about 1,800 ft/minute, during which G-KSWI decelerated from 110 kt to approximately 60 kt groundspeed. During the final part of the descent, the ROD averaged about 650 ft/minute with the groundspeed reducing to below 30 kt 18 seconds before the last logged point. During the last three seconds of the log, the groundspeed reduced from 10 kt to 8 kt, before increasing to a final value of 10 kt. Figures 2 and 3 show the final part of the aircraft's GPS track to the accident site.



Figure 2

Radar and GPS tracks for the last 90 seconds of flight



Figure 3

GPS track for the last 35 seconds of flight

Accident site

The helicopter came to rest on a heading of 060°M in a grass field approximately 170 m x 160 m (Figure 4). Low voltage overhead power cables ran along the eastern and southern edges of the field, with ‘visibility’ markers attached at regular intervals along the cables.

There were a number of main rotor blade strike marks on the ground between the 6 and 11 ‘o’ clock position relative to the helicopter. The tail section had separated

from the fuselage and was found with the ‘blue²’ tail rotor blade resting on the ground (see Figure 5). The section of the tail cone still attached to the fuselage had bent approximately 90° in an anti-clockwise direction. The rear section of the left skid had broken away from the helicopter and the shattered transparency from the left side of the cockpit was scattered approximately 10° to 20° to the right of the aircraft. Debris from the tail cone, drive shaft and tail rotor pitch control rods

Footnote

² The two tail rotor blades were annotated as ‘blue’ and ‘green’.

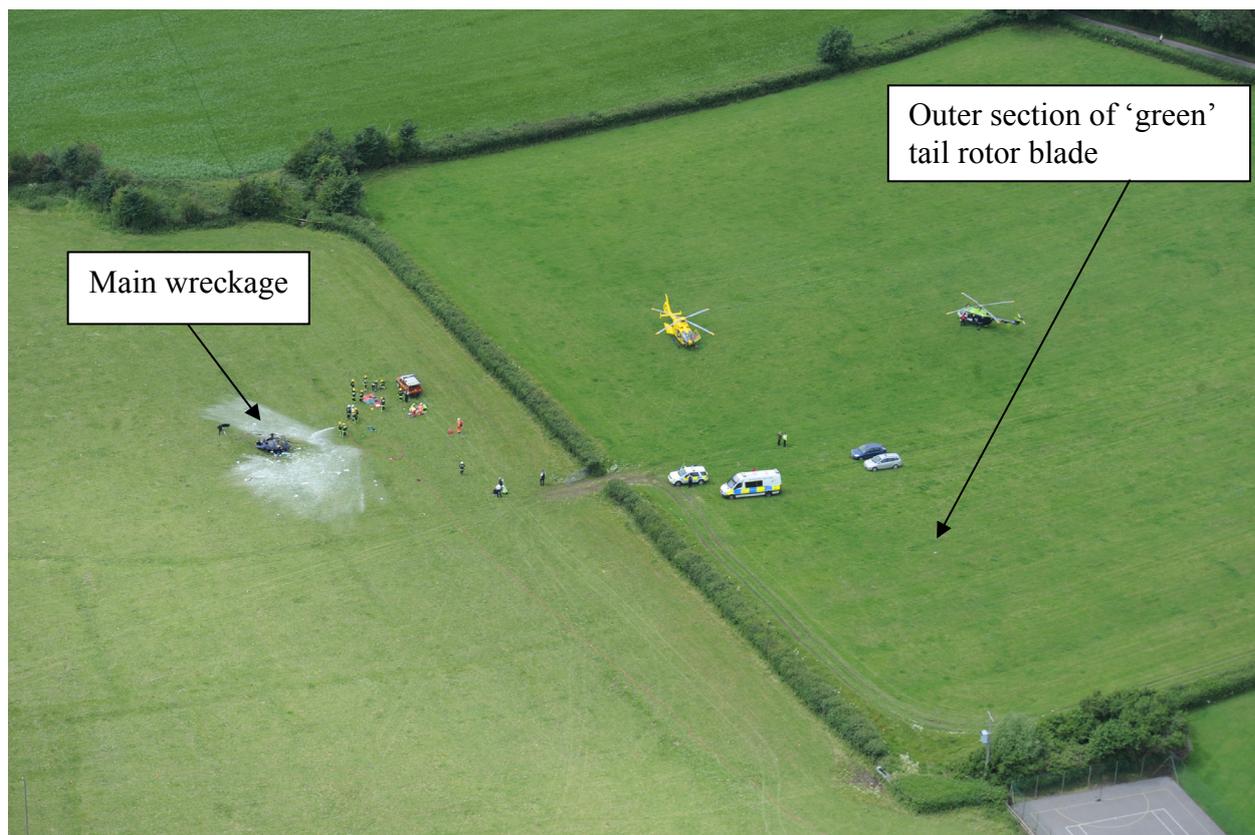


Figure 4
Accident site

were found up to 30 m to the right of the helicopter across an arc of 160°. The outer section of the ‘green’ tail rotor blade was found approximately 90 m away and 40° to the right of the helicopter (see Figure 4). The grass in the area of the engine exhaust had been burnt and there was a strong smell of aviation fuel around the aircraft.

The ground marks indicated that the helicopter initially struck the ground on the rear section of its left skid with little or no forward speed. The helicopter then appears to have rotated slightly in an anti-clockwise direction before the right skid firmly contacted the ground. Ground marks indicated that, at the same time as the skids made contact with the surface, the main rotor blades struck the ground and the tail section detached.

Description of the helicopter

The MD 369E is a five-seat helicopter, primarily constructed from aluminium alloy. It has a five-blade fully articulated main rotor system, which rotates anti-clockwise when viewed from above, and a two-bladed semi-rigid tail rotor. The pilot, who is provided with a four-point harness, sits on a deformable seat base and normally flies the helicopter from the left seat. The helicopter is equipped with a skid-type landing gear.

The helicopter can be fitted with dual flying controls. A throttle twist grip is fitted to each of the collective levers. Rolling the throttle outboard, past a detent, engages a latch in the head of the collective lever and puts the engine gas producer (N_1) in the ground idle



Figure 5

Helicopter wreckage, as found

position. Rolling the throttle fully outboard causes the speed of the engine power turbine (N_2) to increase to 100%. The governor will now maintain the main rotor speed within prescribed limits by automatically adjusting the engine power.

The flying controls are manually powered, with the cyclic, collective and tail rotor pedals controls connected to the main and tail rotor assemblies by a series of control tubes passing through a number of bell cranks. An electrical cyclic trim actuator, operated by a switch on top of the cyclic control, allows the pilot to trim out the flight loads.

The tail rotor blades, which are secured to the hub assembly by a torsion strap, consist of an aerofoil and a

root section that incorporates a pitch horn. A pitch link connects the pitch horn to the pitch control assembly which, in turn, is connected through the control system to the pedals in the cockpit. In the event of a disconnect in the tail rotor control system, the torsion strap will rotate the blade to a preset pitch. The rotor assembly is balanced by weights which can be attached to either the tip of the aerofoil section of the blade or the bolt which secures the pitch link to the pitch horn. (See Figure 6.)

The tail rotor blades fitted to the accident aircraft were identified with colour markings as being either the 'green' or 'blue' blade. While both blades were produced by the same manufacturer, they were of slightly different design with the 'blue' blade incorporating a pocket in the pitch horn.

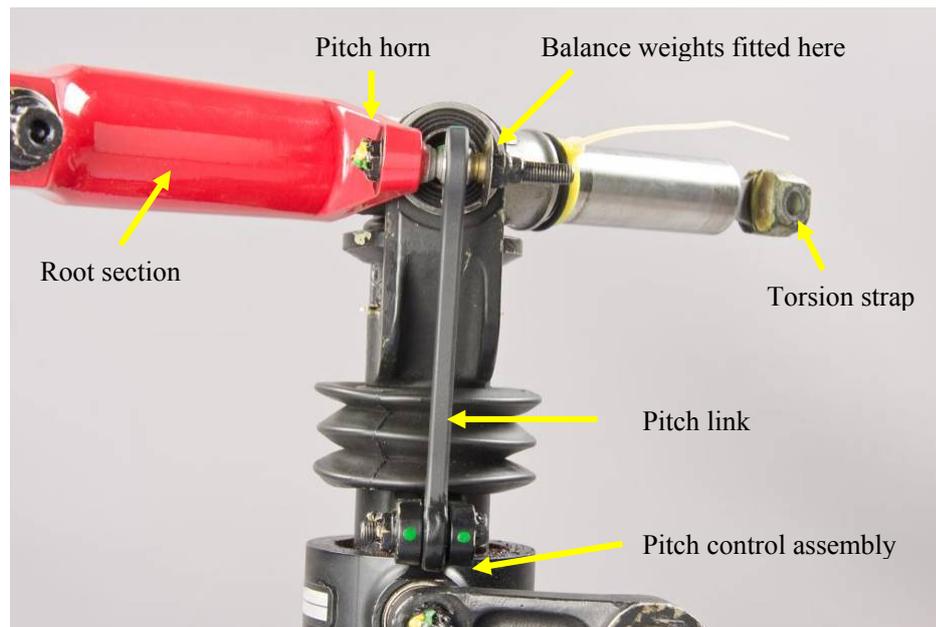


Figure 6

Significant features on the tail rotor assembly

The normal operating speed of the main rotor (N_R) is 487 to 492 rpm which, with a gearing of 5.965:1, gives a tail rotor speed range of 2,904 to 2,935 rpm. The flight manual advises that a minimum rate of descent, with power-off, is achieved at an indicated airspeed of 60 kt and 410 N_R and that the maximum range in autorotation is attained at 80 kt and 410 N_R .

On-site examination of the wreckage

General

The left skid tube had fractured just aft of the forward saddle mount, with the forward supporting strut having twisted and bent forward; the rear section of the broken skid had been forced into the underside of the helicopter. The left side of the cabin had been extensively deformed and there was deformation to all the major structural frames and floor, consistent with a relatively high speed vertical impact. The aircraft skin below the cabin had been torn and forced upwards, puncturing the flexible fuel tank linings in several places. The transparencies in both doors on the left side of the aircraft and the

windscreen in front of the pilot had broken into a number of pieces. The pilot's harness, which had been released by the emergency services, was intact. His seat base had deformed. Overall, the left side of the helicopter had sustained more damage than the right side.

Main rotor blades and transmission

The transmission decking had deformed and the transmission supporting frames had buckled. All five main rotor blades had been damaged and paint transfer from the tail cone to the blades indicated that three of the blades struck the tail cone with increasing severity. The damage to the blades was consistent with them having struck the ground and the aircraft structure, while still being driven by the engine.

All five droop stops had been damaged; three were split, and two were flattened. The aircraft manufacturer advised the investigation that this damage was consistent with the helicopter sustaining a high vertical impact with the ground. There was damage to the upper shoe of the hub

that was indicative of extensive blade flapping; however, it was not possible to establish if this flapping occurred before the helicopter struck the ground.

The controls between the rotor head, cyclic stick and collective lever were intact, with no evidence of a control restriction having occurred prior to ground impact.

Engine

The engine had moved downwards, relative to the airframe, and the lower part of the combustion chamber had contacted the ground. Both lower engine mounts, the engine transmission drive shaft and the fuel supply pipe fitting at the engine driven fuel pump filter had fractured and failed. The collective head on the pilot's collective lever had detached and the engine control linkage had fractured at the bell cranks situated beneath the front seats and the engine firewall. It was not possible to establish the position of the throttle at the time of the accident.

Tail rotor section

The tail section had broken away from the helicopter as a result of the tail cone having been struck by the main rotor blades. The only other damage to the tail section was on the left horizontal stabiliser, which was consistent with the aircraft having landed heavily.

The tail rotor drive shaft had broken into five sections and the pitch control rod into four sections as a result of having been struck by the main rotor blades. The tail rotor gearbox was full of oil, its magnetic chip detector was clean and the gearbox turned freely. Control continuity was established between the pedals in the cockpit and the tail rotor pitch links; there was no evidence of a disconnection or control restriction prior to the rotor blades severing the tail cone.

The 'green' tail rotor blade had failed approximately 33 cm from the centre-line of the tail gearbox output shaft and was found approximately 90 m from the aircraft. Although it was not possible to identify any ground marks from the blade striking the ground, the helicopter and blade manufacturer advised the investigation that the damage to the blade and the position at which it failed was consistent with it having struck the ground. They also stated that it is not unusual for tail rotor blades to strike the ground without leaving any recognisable ground marks. The pitch link to the 'green' blade was intact.

The 'blue' tail rotor blade had a bend approximately 40 cm from the centre of the tail gearbox output shaft, such that the outer section of the blade had bent away from the tail pylon. This damage was consistent with the blade tip having made contact with the ground after the drive to the tail rotor gearbox had been lost.

The outer portion of the pitch horn had broken away from the 'blue' blade. The 'blue' pitch link was also missing, though its bearing was still bolted to the pitch control assembly. Despite three extensive searches, covering an area 150 m around the wreckage and 300 m along the final ground track of the helicopter, the parts were not recovered.

Relevant maintenance

On 17 June 2009, at 3,056 airframe hours, corrosion was rectified around the pitch horn on the 'green' tail rotor blade.

On 12 March 2010, at 3,153.4 airframe hours, the pilot reported that both pedals were '*very stiff*' to operate. The maintenance organisation reported that the restriction was caused by the swelling of the bushes (bearings) inside the root section of both tail rotor blades. The

bushes were replaced and the helicopter flew without any further reports of stiff pedals.

On 28 May 2010, at 3,181.3 airframe hours, both tail rotor blade pitch links were replaced with new items during the annual inspection. Documentation showed that the links had been purchased from the helicopter manufacturer and both had the part number 369D21723-13. The maintenance organisation, on the advice of the blade manufacturer, removed corrosion and restored the finish on the aerofoil sections of both tail rotor blades. No work was carried out on the blade roots.

On 4 May 2011, at 3,290.6 airframe hours, the helicopter completed its Annual / 100 hour inspection during which corrosion was discovered on the aerofoil section of both tail rotor blades and their pitch links. There were no reports of corrosion in the area of the blade roots or pitch horns. The blades were returned to the blade manufacturer to have the corrosion rectified and the erosion strip on the blue blade replaced. The

maintenance organisation rectified the corrosion and restored the surface finish on both pitch links. Following the fitting of the blades, the tail rotor was balanced and, from the work sheets, it was established that 23.7 g of weights³ had been attached to the pitch horn on the 'blue' blade. No balance weights had been added to the pitch horn on the 'green' blade.

Detailed examination of the tail rotor assembly

Apart from the damage to the tail rotor blades and the missing pitch link, there was no other significant damage to the tail rotor assembly. See Figure 7.

The bearing from the missing 'blue' pitch link, which connects the pitch link to the pitch control assembly fork end, had been retained by the securing bolt. Apart from some slight play, the bearing was found to be in relatively good condition and was free to rotate with no evidence of it having seized. There was some mechanical damage and missing paint at the end of the fork end that was believed to have occurred during the accident sequence.



Figure 7
Tail rotor assembly

Footnote

³ A maximum of 26.91 g of balance weights are allowed to be added to the pitch horn.

There was also some mechanical damage on the pitch control assembly that had been caused by the blade hub knocking against it.

Examination of the ‘blue’ tail rotor blade pitch horn

Significant features on fracture surface

The fracture surface on the ‘blue’ tail rotor blade pitch horn was examined by metallurgists at QinetiQ, using high magnification optical devices and a Scanning Electron Microscope (SEM). The following six significant features, which are highlighted in Figure 8, were identified:

- A fatigue crack, identified as ‘Crack A’, which started at a corrosion pit on the lip of the pocket. The fracture surface in this area was flat with visible striations, from which it was possible to identify the direction of propagation that is highlighted by the blue arrows. The corrosion pit was 0.146 mm deep and 0.153 mm wide at the widest point.
- A fatigue crack, identified as ‘Crack B’, which started at a corrosion pit on the lip of the pocket. The fracture surface in this area

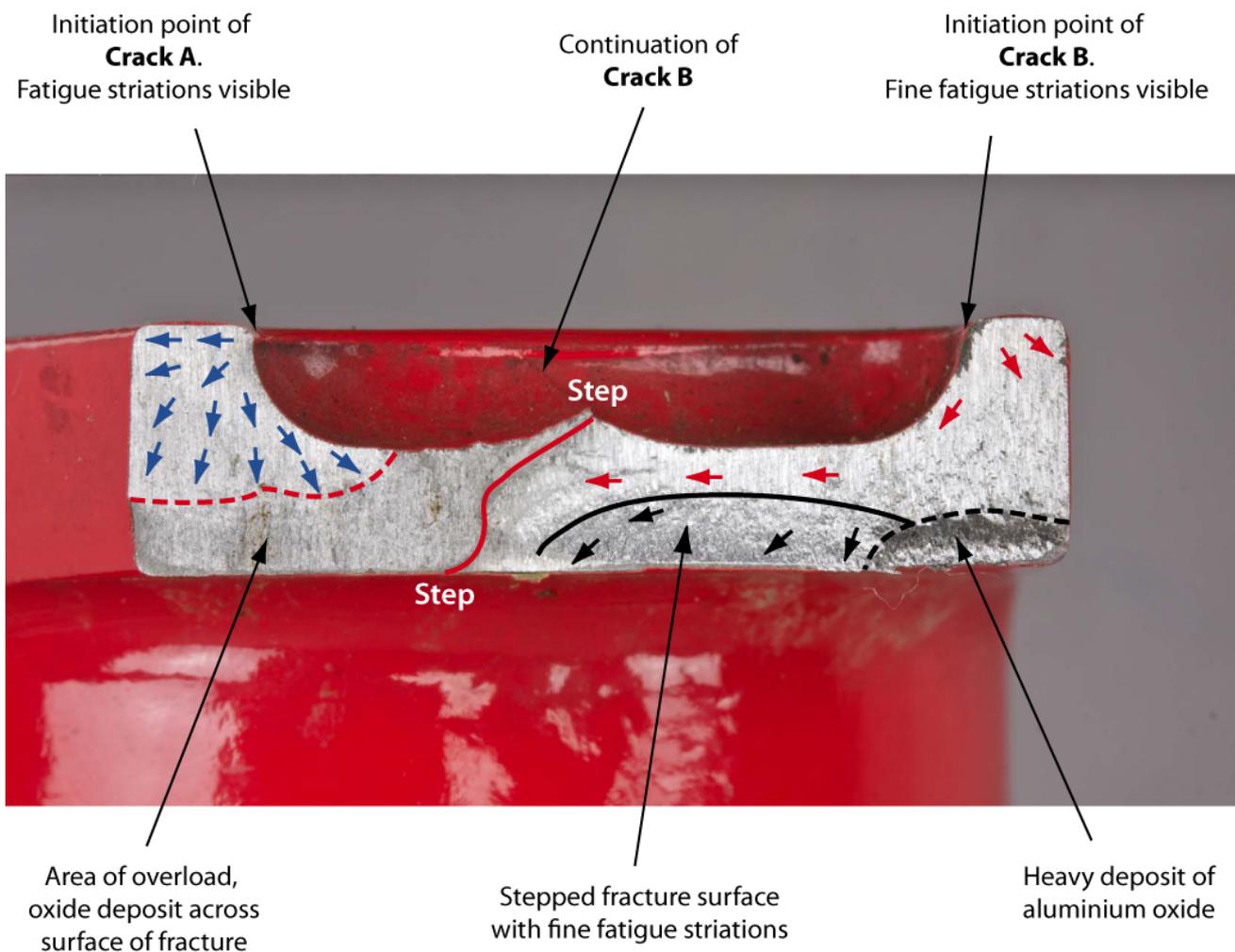


Figure 8
Significant features on the fracture face of the pitch horn

was flat, with visible signs of heavy and fine striations which were difficult to resolve. The direction of propagation is highlighted by the red arrows. The corrosion pit was 0.487 mm deep and extended 1.8 mm around the radius into the pocket and 0.9 mm along the external surface.

- A step, which ran across the fracture face and is marked by the solid red line.
- An area that had failed in overload and was covered in an oxide deposit. When examined by the AAIB, within four hours of the accident, this section had the same bright appearance as the remainder of the fracture; therefore, the oxidisation must have occurred in the days following the accident. This area is bounded by the dashed and solid red lines.
- A rough, stepped, fractured surface with fine striations associated with fatigue cracking. Due to the nature of the surface it was difficult to resolve the striation marks with any great confidence. This area is bounded by the solid black line and the direction of propagation is highlighted by the black arrows.
- A dark area that was identified as a heavy deposit of aluminium oxide, approximately 5 mm long and up to 1.7 mm deep, which appeared to run from an area of mechanical damage; there was no evidence of any fatigue cracking in this area. The blade manufacturer advised the investigation that this section of the blade root is normally subject to a compressive load and it is not

unusual to see damage to the surface finish in this area as a result of slippage of the tools used to tighten the pitch link attachment bolt. This corrosion was present four hours after the accident and would have been present prior to the failure of the pitch horn. This area is bounded by the dashed black line.

Estimation of time for crack to propagate to its critical crack length

There are three stages in the initiation and growth of a fatigue crack: initiation and formation of the crack; crack propagation; the crack reaches its critical length and failure occurs. Initiation and formation of the crack can take significantly longer than the time for the crack to propagate and reach its critical crack length. However, there was insufficient evidence to determine, with any great confidence, the time taken for the initiation and formation of either of the cracks identified in the fracture surface.

Crack propagation and the time to failure can be determined by counting the number of striations on the failed surface and considering the frequency of the cyclic loads that, in this case, were applied to the pitch horn. However, while striations were identified across a number of areas of the fracture surface, apart from those associated with 'Crack A', they were very fine and in places difficult to resolve. It was, therefore, not possible in these areas to count the number of striations with any great confidence.

The striations associated with 'Crack A' were more clearly defined in the region 1.47 mm to 6.15 mm from the initiation point; it was also noted that some of the striations in this region were stronger than others. The total number of striations in the region between 1.47 mm and 6.15 mm was calculated as 52,300, which,

by extrapolation back to 0.13 mm from the initiation point, gave an estimated 109,209 striations. It was considered that, at most, the pitch horn would have experienced two cyclic loads per revolution of the tail rotor. Assuming that during the cruise the helicopter was being flown at the upper limit of the normal operating speed for the main rotor (492 rpm), then the tail rotor would have been rotating at 2,935 rpm, which would give a time of 18.6 minutes for the crack to propagate and reach its critical length. The time taken for the crack to have initiated and formed would need to be added to this time in order to achieve the total time from crack initiation to failure.

The number of strong striations was calculated as 2,461 in the region between 1.47 mm and 6.15 mm, which by extrapolation back to 0.13 mm from the initiation point, gave an estimated 8,713 striations. At two loading cycles per tail rotor revolution this would give a crack propagation time of 1.48 minutes.

QinetiQ have seen this combination of strong and weak striations in fatigue cracks on helicopter main rotor blades; the strong striations being caused by blade loading and the weaker striations by vibration. It is, therefore, possible that the strong striations were caused by the changing loads on the blade as it rotated. The weaker striations may have been caused by vibration or the loads from the flaying, broken pitch link which might still have been attached to the pitch control assembly.

Despite the presence of thousands of striations, there was no evidence of any beach-marks on the fracture face. Beach-marks can contain thousands of striations and form when the load is removed for a sufficient period to allow corrosion products to form. The lack of beach-marks suggests that crack growth was rapid and probably occurred during the accident flight.

Condition of the surface of the pitch horn

QinetiQ established that the pitch horn had been manufactured from 7075 aluminium alloy and the properties conformed to the heat treatment specified in the blade manufacturer's drawing number 500P3120.

The paint layer was removed with a chemical paint stripper and a more detailed inspection of the pitch horn was carried out; abrasive products were not used in order to ensure that the process did not damage the surface of the metal or remove any products of corrosion. A photograph of the pitch horn after the paint had been removed is at Figure 9.

The examination revealed evidence of corrosion up to several hundred microns deep, in several places, including around the lip of the pocket and the bush: this corrosion had not been evident before the paint had been removed. There was evidence that the surface of the pitch horn had been shot peened and a chromate treatment, such as Alodine 1200, had been applied to provide protection against corrosion. However, the Alodine had been mechanically abraded away along the edges of the pitch horn, inner bush and part of the lip of the pocket. The dimples from the shot peening also appeared to have been abraded away in these areas. Optical micrographs of polished micro-sections, that had been etched with Keller's reagent, and strain maps, using Electron Backscatter Diffraction, were produced. These indicated that, where there was visible evidence of dimpling, on average the shot peening had affected the microstructure up to a depth of 0.075 mm. Where the dimpling had been abraded away the effect of the shot peening had reduced to a depth of around 0.04 mm.

It could also be seen from the SEM image, reproduced at Figure 10, that the surface of the pitch horn was covered with scratch marks that appear to have

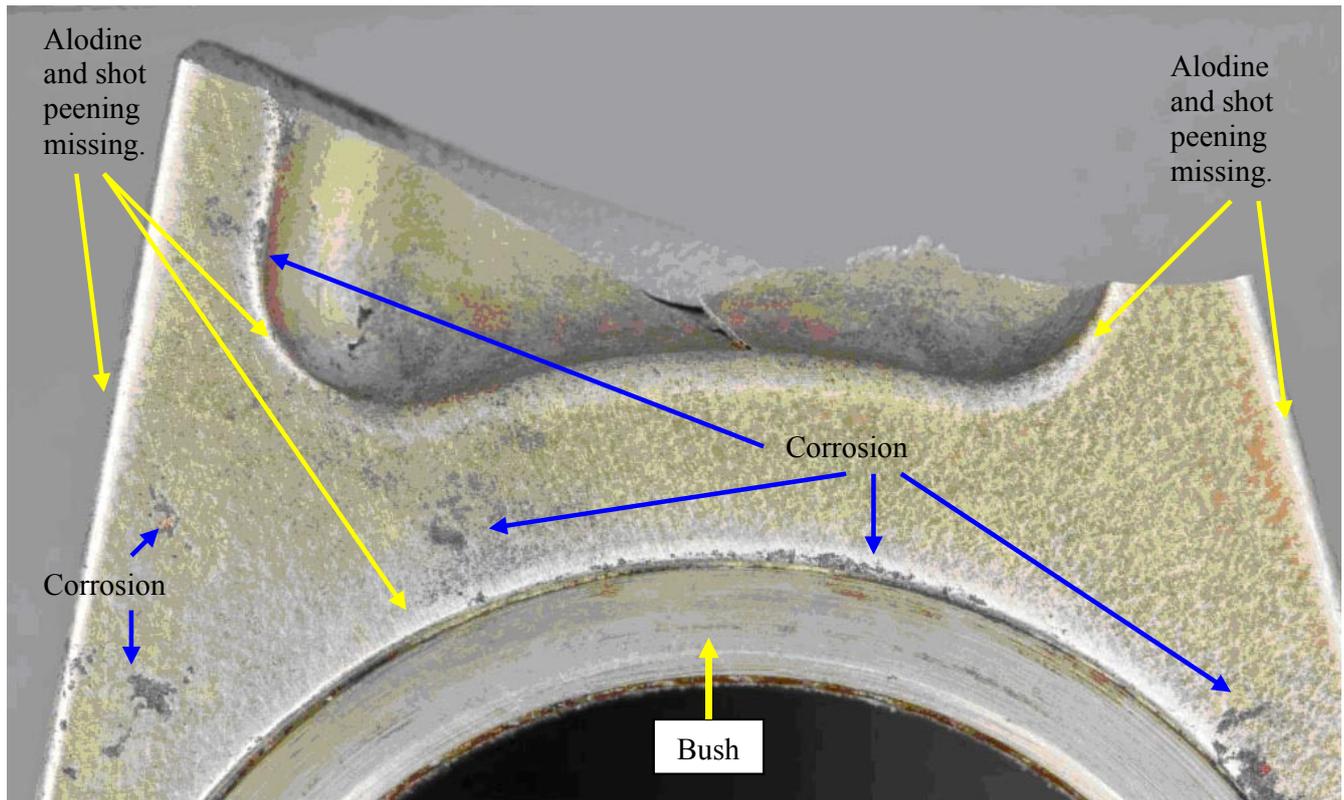


Figure 9

Pitch horn with paint removed

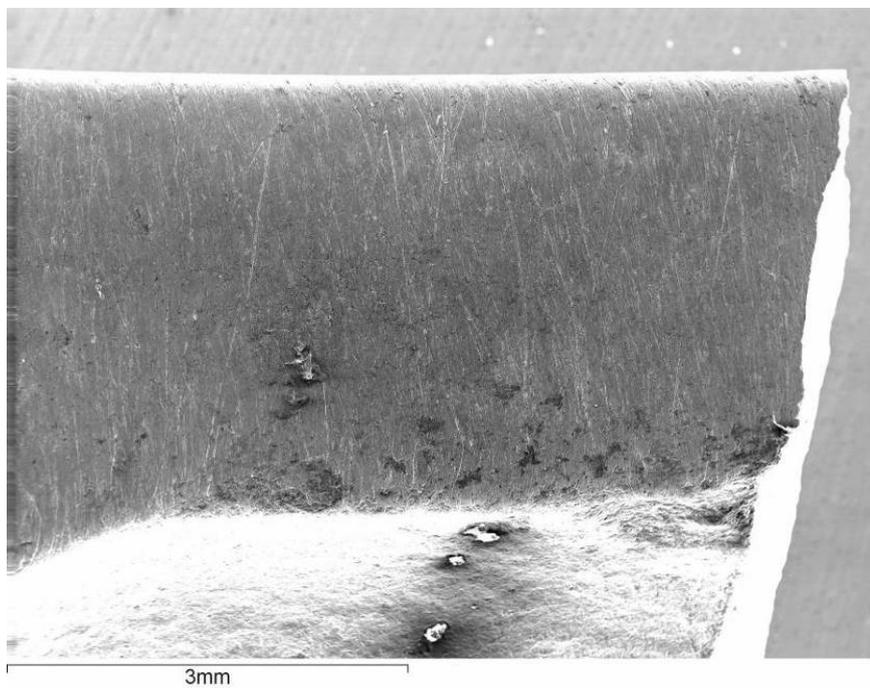


Figure 10

SEM image showing scratch marks on the surface of the pitch horn

penetrated through the Alodine surface finish. The sharp edges of these scratches could act as stress concentrators. The scratches were consistent with the surface having been rubbed with an abrasive material.

There was also a burr around the lip of the pocket where, in places, the metal appeared to have folded over on itself: it was from this area that 'Cracks A' and 'B' appear to have originated. See Figure 11. The blade manufacturer advised the investigation that the burr can be formed during the shot peening process.

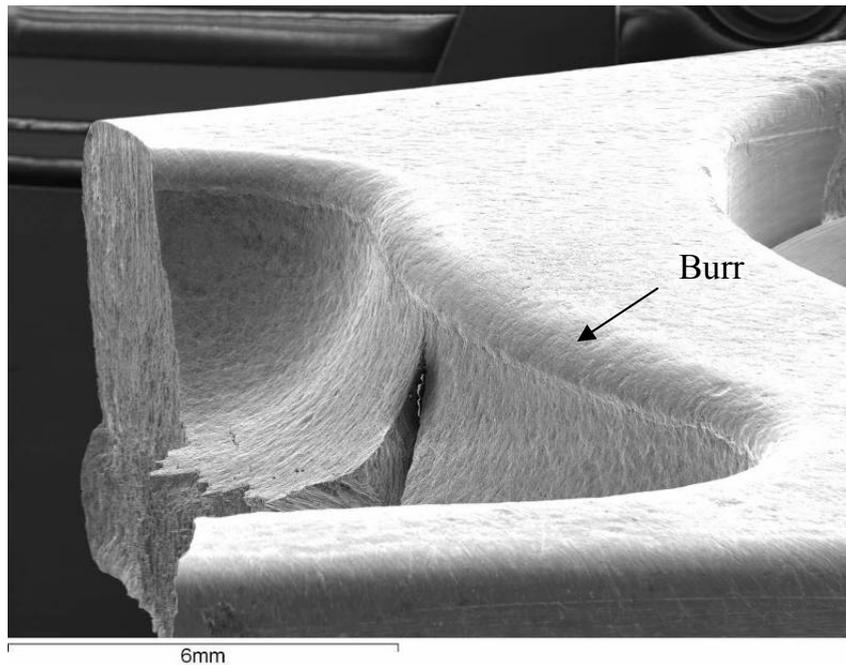


Figure 11

SEM image showing burr around the edge of the pocket

Examination of ‘green’ tail rotor blade pitch horn

The paint was chemically removed from the ‘green’ tail rotor blade (serial number B391) which was then examined using high magnification optical devices. With the exception of one small area, see Figure 12, there was evidence that all of the pitch horn, including the edges and radius had been shot peened. The surface was free of scratches and apart from the radius at the end of the pitch horn it was evenly covered in Alodine. There were a number of areas of corrosion, particularly around the bush and the insert for the pitch link securing bolt. In addition there was an area of mechanical damage that was covered with an intact layer of paint.

There was also a small repaired section where the Alodine and the dimples from the shot peening had been rubbed away leaving scratches in the surface. The aircraft documentation recorded that corrosion had been removed, in the field, on 17 June 2009 at 3,053.5 airframe hours. This was prior to the blade

being returned to the blade manufacturer during the annual maintenance in May 2011 for corrosion to be removed from the aerofoil section.

Detailed examination of the ‘green’ tail rotor blade pitch link

QinetiQ established that the pitch link had been manufactured from 2024 aluminium alloy in the T4 heat treatment condition and contained 4.1% copper. The drawing specifies that the pitch link can be manufactured from one of three aluminium alloys; 2014, 2024 or 7075. The link had the part number 369D21723-13 painted on the side, which is the same part number listed in the purchase order for the links the maintenance organisation fitted during the annual inspection, completed on 28 May 2010. The bearing from the blue pitch link, that remained attached to the pitch control assembly, had the part number 369A7951-57 which is specified in the Illustrated Parts Catalogue (IPC) as the bearing for use in the pitch link with the part number 369D21723-13.

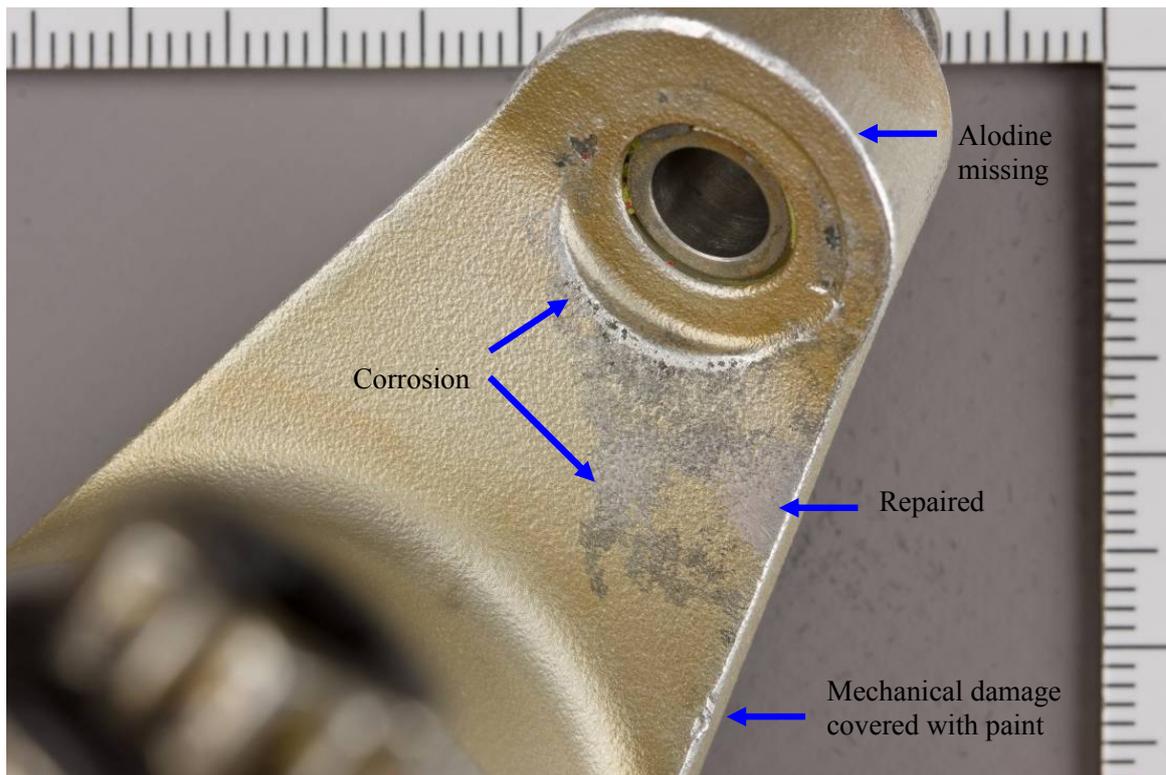


Figure 12

Damage on 'green' blade pitch horn

There was slight play in the bearing on the 'green' blade pitch link, which had been connected to the tail rotor pitch control assembly, and evidence that the paint in this area had been 'touched up.' The maintenance organisation that replaced the pitch link at the last annual maintenance informed the investigation that in their experience, wear normally occurs in the bearing which is connected to the pitch horn, whereas they rarely find wear in the other bearing. Their normal practice, if the bearing wear is within limits, is to rotate the link so that the bearing with the wear is then connected to the pitch control assembly. This practice is consistent with the advice given in the Aircraft Maintenance Manual (Chap 64-30-00).

The paint was chemically removed and the link was examined using high magnification optical devices and a SEM. In the areas where the paint had been 'touched up',

there was no evidence of the Alodine surface finish and the tops of the dimples, caused by the shot peening, had been removed by a coarse abrasive that had left scratches in the surface. Corrosion products were present in this area. (See Figure 13.)

The maintenance organisation advised the investigation that they followed the instructions in the Aircraft Maintenance Manual (Chap 64-20-10, Table 502) and the Corrosion Control Manual which states with regard to the removal of corrosion in the pitch link:

'abrasive blend and polish minor surface defects.....using 400 grit silicon carbide abrasive paper.'

The mechanic who carried out the work stated that he removed the paint and corrosion by lightly buffing



Figure 13

Section of 'green' pitch link with paint removed

the pitch link with 600 grade followed by 1500 grade silicon carbide paper. He then cleaned the surface with Alumiprep 33 before applying Alodine 1201 with a paper cloth and brush. Finally, he applied two coats of zinc chromate primer and a top coat of matt black paint. The technical sheet for Alumiprep 33 states that it is a phosphoric acid based cleaner, brightener and pre-paint conditioner that should not be used on high copper bearing aluminium alloys. The manufacturer of Alumiprep 33 advised the investigation:

'... this product due to its composition is not capable of dissolving copper or copper oxide. This might theoretically lead to relatively higher copper concentration at the Al surface, when used on 2000 series aluminium. High copper concentrations at the surface may reduce the corrosion resistance of the material.'

A corrosion expert within QinetiQ confirmed that Alumiprep 33 should not be used on 2000 series

aluminium as it leaves concentrations of copper rich precipitates. As copper is less reactive than aluminium, a galvanic cell could be established that leaves the surface susceptible to subsequent corrosion; moreover, the copper rich surface may reduce the effectiveness of the Alodine treatment.

Previous failures of the tail rotor blade pitch link

The helicopter manufacturer reviewed the service history and accident history of tail rotor pitch links. The review included available data for the 369A (OH-6A), 369H, 369HE, 369HS, 369HM, 369D, 369E and 369FF model helicopters and found only one previous report of a tail rotor pitch link having failed. This occurred in 1997 and the helicopter landed safely without sustaining any further damage. The cause of the failure was fatigue, attributed to extensive damage to the pitch link bore caused by movement of the bearing in the bore. Wear and damage to the bearing and pitch link was visibly apparent.

Previous failures of tail rotor blades

Background

The design standard of the 'blue' tail rotor blade, which has a pocket machined into the pitch horn to reduce weight, was introduced onto the Hughes 369 series of helicopters by a Supplemental Type Certificate in March 1999. In 2001, the manufacturer of the blade became the Original Equipment Manufacturer (OEM) supplier of tail rotor blades to the helicopter manufacturer. At the time of the accident approximately 1,100 of these blades had been delivered, of which the manufacturer estimated that there were around 200 still in service.

Failures prior to 2004

In 2002, there were two reported occurrences of the pitch horn fracturing on blades with a pocket in the pitch horn. Initially, this was thought to have been caused by overload failure. The following year there were a further two occurrences. These were found to have been caused by fatigue cracking that had started in the area of the pocket. A review by the blade manufacturer of the evidence from the earlier occurrences concluded that they may also have failed as a result of fatigue cracking.

As the helicopters involved in the occurrences prior to 2004 landed safely, no formal reports into the circumstances of the incidents were produced. However, the helicopter manufacturer was able to provide the investigation with the pilot's report and some simple field notes for an incident on 26 February 2003 involving a MD369 helicopter, registration N234RF.

The pilot reported that the helicopter was being flown at 115 kt and had just commenced a right turn when he heard a "loud snap" accompanied by very strong medium frequency vibrations. He established that the

helicopter would not respond to full left pedal input and assumed that he had had a tail rotor control malfunction. The vibration was sufficiently strong to "unlatch and pop open" the passenger door. The pilot flew the helicopter for a further five minutes, with full left pedal applied, during which he descended at an airspeed of approximately 70 kt. He performed a running landing, touching down at an airspeed of approximately 30 kt, using the throttle to maintain directional control during the landing and subsequent ground run.

The field notes for this accident included a photograph of the failed tail rotor blade pitch horn fracture surface, annotated with the significant features (see Figure 14). The features were very similar to those seen on the fracture face of the tail rotor blade from G-KSWI, with two fatigue cracks emanating from the lip of the pocket.

There was also a sketch and photographs of the pitch link which, with the failed part of the pitch horn, had remained connected to the pitch control assembly. The sketch showed that the pitch link had bent sideways and that there was a gap between the bearing and the recess in which it was fitted. This indicated that the metal in the pitch link around the bearing had undergone plastic deformation. A note on the sketch stated that the pitch link had made contact with the bell crank, which is part of the pitch control assembly.

Action by the FAA

As a result of these findings, the blade manufacturer reviewed the stresses in the area of the pockets. This resulted in the FAA issuing an Emergency Airworthiness Directive (2003-08-51), in April 2003, to reduce the service life of these blades from 5,140 hours to 400 hours. In June 2003, the FAA issued an alternative means of compliance that allowed the original life to be restored.

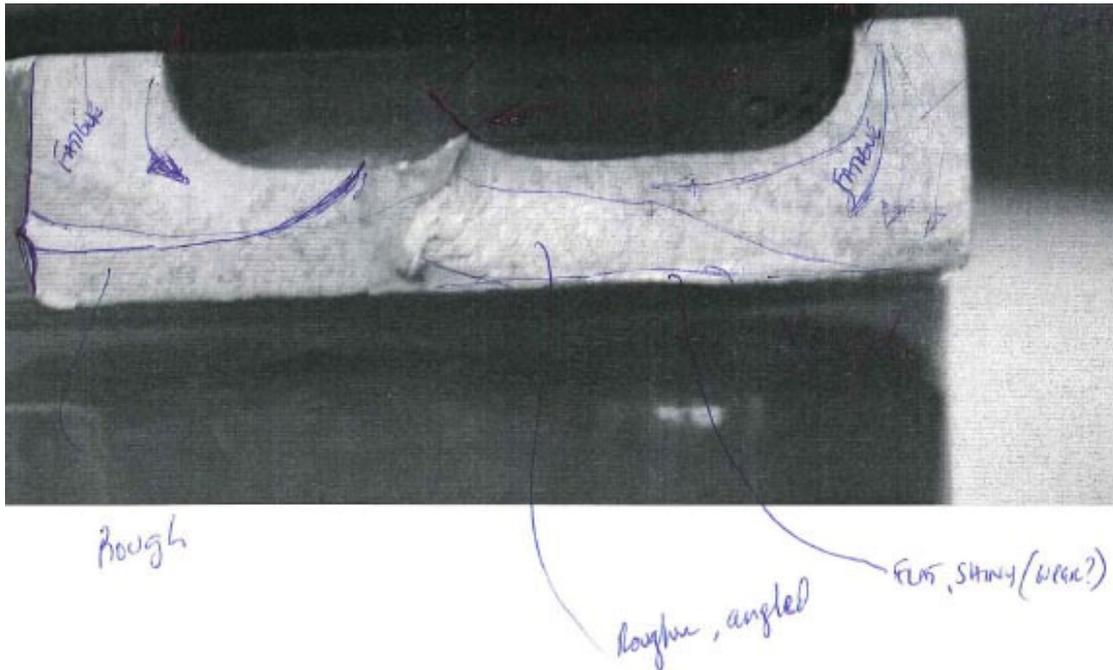


Figure 14

Fracture face of pitch horn that failed on N234RF in February 2003

This required blades that had zero time to be shot peened in the area of the pitch horn and identified with the letter 'M' painted on the blade root fitting. Blades in-service were returned to the manufacturer to have the paint removed around the area of the pitch horn, which then had to undergo an eddy current inspection to ensure that it was free of cracks. On blades that passed this check the inside edge of the pocket was machined to provide a corner radius of 0.254 mm (0.01"). The surface was then shot peened and the surface finished restored. These blades were identified with the letter 'I'. In addition, the design of newly manufactured blades was changed so that they no longer incorporated the pocket.

Failures after 2004

The investigation was advised of a further failure of a tail rotor pitch horn that occurred on 24 February 2010. On that occasion an experienced helicopter pilot, whilst flying in the cruise, identified that he had had a tail rotor control malfunction and successfully performed a running

landing without damaging the helicopter. A metallurgy report concluded that the failure of the tail rotor blade occurred as a result of corrosion fatigue cracking which initiated at two, or possibly more, locations around the lip of the pocket. The examination also identified extensive corrosion that had caused blistering and flaking of the paint finish. A photograph of the fracture face is at Figure 15.

Finite Element Analysis of the pitch horn

The blade manufacturer produced a Finite Element Analysis (FEA) model of the tail rotor blade using Abaqus/CAE, version 6.11-1, software. The analysis considered the probable maximum loads with the pitch link intact and the pitch link having failed at the pitch control assembly. The modelling did not include any compressive stresses resulting from the shot peening or any additional loads resulting from increased friction in the bushes (bearings) fitted inside the root of the tail rotor blades.

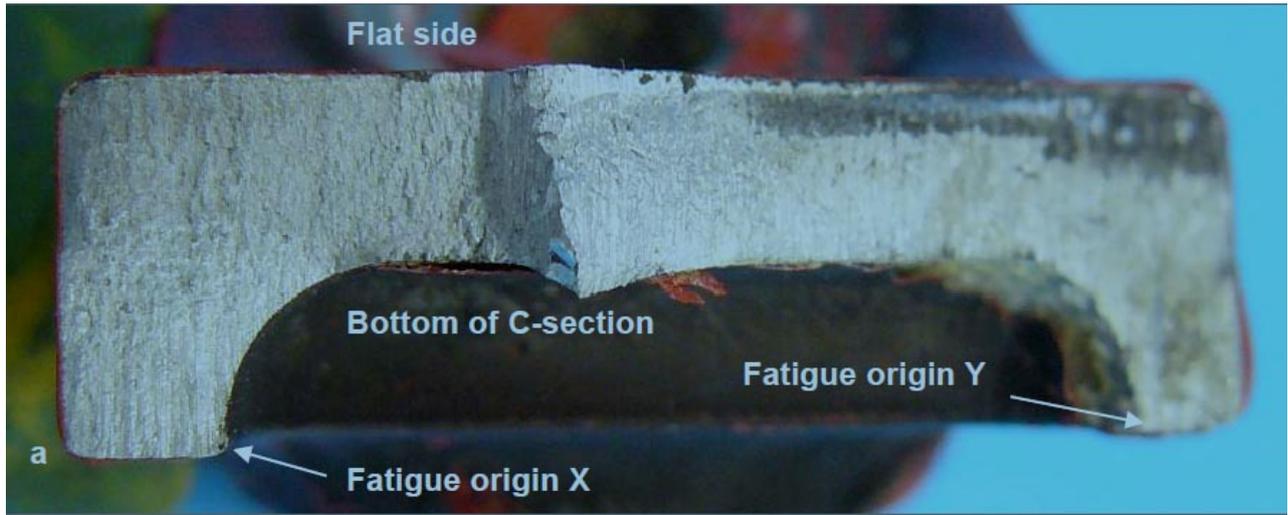


Figure 15

Fracture face of pitch horn that failed on the 24 February 2010

For the normal maximum loading condition, Figure 16a shows that the maximum stress occurs in the outboard upper corner of the pocket. As the pitch link loads are reversible, the maximum stress would be mirrored. In Figure 16b it can be seen that, for the failed pitch link condition, the maximum stress occurs at all four corners of the pocket with the highest stress at the inboard corners.

the fatigue cracks, highlighted in Figures 9, 14 and 15, emanated from points close to the inboard corners.

History of the ‘blue’ tail rotor blade

The ‘blue’ tail rotor blade, on which the pitch horn failed, had a data plate that identified it as part number 500P3100-101MT and serial number A881. The letter ‘M’ was painted on the root section of the blade which,

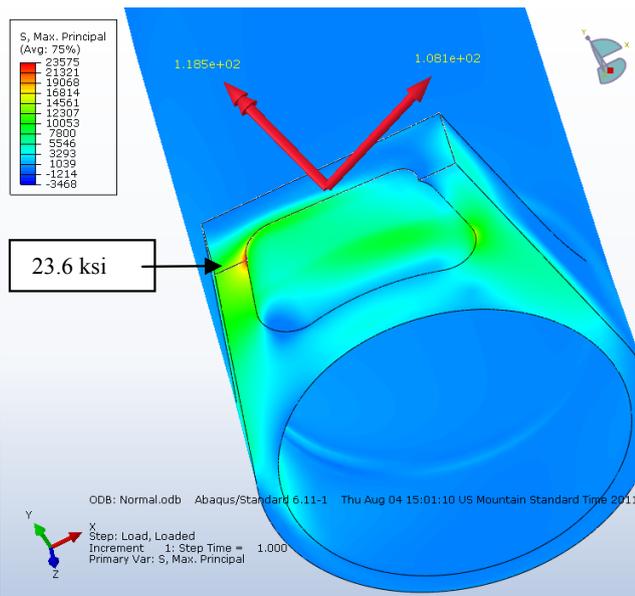


Figure 16a

Maximum stress - normal loading

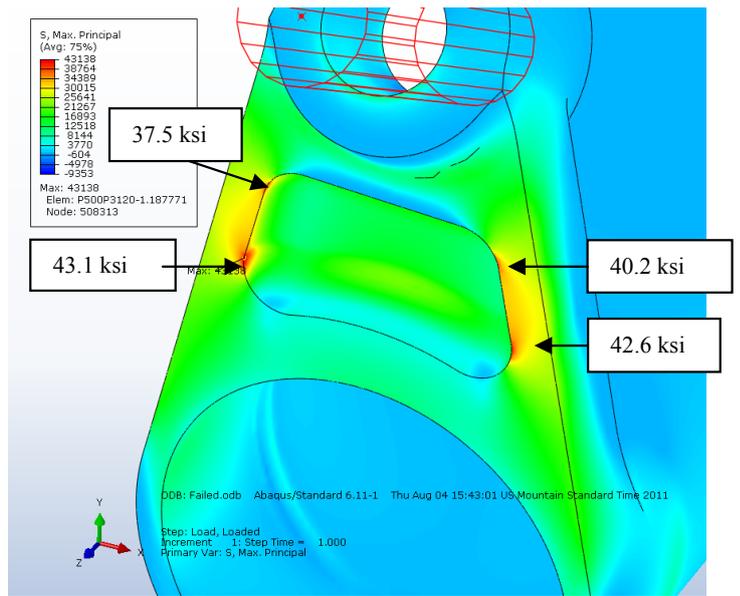


Figure 16b

Maximum stress - failed pitch link

with the letters at the end of the part number, signified that the area around the pitch horn had been shot peened before the blade had entered service.

The blade was manufactured in April 2003 and returned to the blade manufacturer in August 2006 (life used 150 hours) for two Service Bulletins to be embodied; these were not related to the pitch horn. It was next returned to the blade manufacturer in April 2011 (life used 761 hrs) to replace the erosion strip and rectify corrosion which had been found on the blade skin and the tip of the blade during the annual / 100 hour inspection. The blade manufacturer advised the investigation that the paint finish was removed from the aerofoil section prior to removing all visible signs of corrosion in the blade skins and at the tip of the blade. As there was no visible evidence of corrosion in the blade root and pitch horn, it was only considered necessary to lightly abrade the paint in this area, to provide a keyed surface prior to the painting of the entire blade.

Damage on other tail rotor blades fitted to MD 369 helicopters

During the investigation, the organisation that maintained G-KSWI informed the investigation of a set of tail rotor blades from another MD 369 helicopter that were being returned to the blade manufacturer to rectify corrosion in the area of the pitch horn pockets. There was also evidence of mechanical damage and chips in the paint work (see Figure 17). These blades had been fitted to the helicopter since 2004, had flown 262 hours and had never been returned to the blade manufacturer.

The paint finish was removed from the pitch horn on one of the blades using a chemical paint stripper in order not to damage the metal surface or remove any of the corrosion products. The majority of the corrosion discovered in the area of the pocket had been visible through the paint layer as blisters. However, there were patches that were not detectable with the



Figure 17

Corrosion in the area of the blade pocket

paint in place. The surface showed evidence of having been shot peened and appeared to have been treated with Alodine. There was no evidence of any abrasion or scratch marks on the areas that had been exposed during the examination.

The blade manufacturer advised the investigation that it is common to see evidence of corrosion and mechanical damage, due to tool usage, in the area of the pitch horn. The normal practice when the blades are returned to the blade manufacturer is to remove the corrosion mechanically and restore the shot peening, chromate conversion coating (Alodine 1200) and paint finish.

Removal of corrosion

The instructions on the removal of corrosion are specified in the manufacturer's Component Overhaul Manual (CSP-COM-5) and the Corrosion Control Manual (CSP-A-3). While the corrosion Control Manual has an issue date of 9 February 1981, it is still an active document listed on the manufacturer's website, with a latest revision date of 30 July 1993.

Pitch link

The repair of the pitch links is specified in Table 502 in Chapter 64-20-10 of the Component Overhaul Manual. The maximum repairable limit is '*minor surface defects*', for which the corrective action is to '*Abrasive blend and polish minor surface defects.*' Chapter 4-14 in the Corrosion Control Manual provides advice on removing corrosion from aluminium and aluminium alloy surfaces and, in addition to the use of 400 grit silicon carbide abrasive paper, it also describes the use of Triacid etch solution to clean the surface. The maintenance organisation that removed the corrosion from the pitch links on the accident aircraft used Alumiprep 33 as an alternative surface cleaner to Triacid, which they advised is not commercially

available in the UK. The investigation could find no advice in the helicopter maintenance manuals regarding the restoration of shot peening following rework of the pitch links.

Tail rotor blades

With regard to the root fitting on the tail rotor blades, Table 501 in Chap 64-20-10 of the Component Overhaul Manual states:

'Cracks, scratches, nicks and gouges....Defect depth must not exceed 0.020 inch (0.508 mm) after rework...Abrasive blend and polish allowable defect...'

The metallurgy established that the effective depth of shot peening was around 0.075 mm, which is considerably less than the depth to which damage can be blended out. The investigation could find no advice in the helicopter or blade manufacturer maintenance manuals regarding the need to restore shot peening following rework of the tail rotor blade pitch horns.

Procedures following a loss of thrust from the tail rotor

The Flight Manual contains advice on the actions to take in the event of a tail rotor (anti-torque) failure. It states:

- '- The nose of the aircraft will turn right with power application. The nose of the aircraft will turn left with power reduction.*
- Anti-torque failure, fixed tail rotor pitch setting. The procedure to follow is: 'Adjust power to maintain 50 to 60 kt airspeed. Perform a shallow approach and running landing to a suitable area, touching down*

into wind at a speed between effective translational lift and 30 kt.....

- *Complete loss of thrust in the hover...is normally indicated by an uncommanded right turn.' The procedure to follow is: 'Place the twistgrip in the ground idle position and perform a hovering autorotation'. There is also a warning to 'reduce altitude to 12 ft or less prior to placing the twistgrip in the ground idle position and performing a hovering autorotation'.*

Pilot's continuation training

The pilot last flew his Licence Proficiency Skill check on 6 November 2010. The examiner reported that they practised a tail rotor failure in the hover, but did not practise a loss of tail rotor control in forward flight. The examiner considered it unlikely that the pilot would practise emergencies, such as a loss of tail rotor control, when he was not flying with him.

Medical information

The pilot held a current JAA Class 2 medical with the limitation that he was required to wear spectacles. A pair of spectacles was recovered from the wreckage.

During the accident sequence the pilot sustained fractures to his pelvis, lower back and ribs as well as extensive head injuries and several large cuts and puncture wounds to his body. The pilot previously flew helicopters wearing a helmet but around eight months prior to the accident he started to use a noise cancelling headset.

The pilot spent several months in hospital and was not able to recall any details of the events immediately prior to the accident.

Analysis

General

As far as could be established, the pilot was flying a direct route back to Cornwall. The flight appeared to progress satisfactorily until the helicopter was in the vicinity of Glastonbury, when, without making any radio calls, the pilot commenced a rapid descent and appeared to position the helicopter for a landing in a field. The description of the helicopter starting to spin is consistent with the airspeed becoming too low for the pilot to maintain directional control, following a loss of tail rotor authority.

Examination of the accident site and wreckage indicated that the helicopter struck the ground with a high rate of descent and low forward speed. The tail section was severed by the main rotor blades after the helicopter struck the ground and the distribution of the light wreckage indicated that the fuselage had been rotating anti-clockwise after initial contact with the ground. While this is the opposite direction to that experienced during a tail rotor failure, it was consistent with the direction the fuselage would rotate if the tail cone was struck by the main rotor blades. The burnt grass and damage to the helicopter caused by the blade strikes suggested that the engine had been running at a low power setting. It would have stopped after the main fuel feed pipe was severed during the impact. Due to the damage to the pilot's collective lever and engine control linkages it was not possible to establish the position of the throttle. Apart from the failure of the pitch horn and pitch link on the 'blue' blade, the investigation could identify no other reason why the pilot would suffer a reduction in yaw control.

Loss of pitch control to one tail rotor blade would result in it being moved by the torsion strap to a pre-set position, with the result that the maximum thrust available from the tail rotor would be reduced. It is likely that this

would have resulted in sudden severe vibration which the pilot would probably have felt through his pedals and the airframe, as experienced by a pilot in a previous event. Given that the pilot made no emergency call to ATC, and made no apparent attempt to divert to another airfield, it is likely that the problem commenced shortly before the helicopter's rapid descent. He was probably then pre-occupied with selecting a suitable field in which to undertake a precautionary landing.

Initially, the helicopter was being flown at an airspeed of around 120 kt when the tail fin would have provided the majority of the thrust required to counter the torque reaction from the main rotor. It is, therefore, possible that the pilot was unaware that he had lost pitch control of one of the tail rotor blades. The collective lever would have been lowered during the descent, which would have reduced the engine power and the torque reaction from the main rotor. During the last 37 seconds the pilot reduced the rate of descent, which would have required an increase in engine power, increasing the torque reaction from the main rotor. The flight manual recommends that, in the event of the tail rotor assuming a fixed pitch setting, the pilot should adjust power to maintain an airspeed of 50 to 60 kt, perform a shallow approach and carry out a running landing between effective translational lift and 30 kt. The witnesses stated that the pilot appeared to orbit the field and the GPS data shows that the helicopter ground speed reduced to around 10 kt.

The evidence indicated that one tail rotor blade had assumed a fixed pitch setting and that the pilot reduced the helicopter's airspeed to the point at which there was insufficient thrust available from the tail rotor to counter the torque reaction from the main rotor. As a result, the helicopter started spinning to the right. Once the helicopter had started rotating at low airspeed, the pilot would have needed to reduce torque, by rolling the

throttle off, to stop the rotation, and use the collective lever to arrest the subsequent descent and cushion the touchdown. Witness reports of the engine noise reducing and the helicopter rotation stopping, before it rapidly descended, suggest that the pilot moved the throttle towards the ground idle position. The damage to the helicopter was consistent with the engine having been at a low power setting.

Aircraft handling

In the five previously reported failures of the tail rotor pitch horn, the pilots were able to land the helicopters with no further damage. On this occasion, it seems likely that the pilot, who had not recently practised loss of tail rotor control emergency procedures, was uncertain of the nature of the malfunction until the aircraft started to spin and he, probably, realised that he had suffered a loss of tail rotor thrust.

Failures that could have resulted in a loss of yaw control

The investigation identified two failures that could have resulted in the loss of pitch control of the 'blue' tail rotor blade; failure of the pitch horn and failure of the pitch link.

There were two fatigue cracks and an area of overload on the fracture surface of the 'blue' blade pitch horn where the FEA established that the maximum stress occurs. Assuming a blade loading of two cycles per tail rotor revolution, counting all the strong and weak striations in 'Crack A' would give an estimated minimum time of 18.5 minutes for the crack to have propagated and failed. However, if it is assumed that the strong striations are formed by the cyclic blade loading and the weaker striations by high frequency vibration, or the flaying of a broken pitch link if it failed first, then the minimum time for the crack to propagate and fail is 1.5 minutes. The time

between the start of the descent and the loss of GPS data was 1.5 minutes. Therefore, it is possible that the pitch link could have failed first, at the pitch control assembly, and the increased load on the pitch horn might then have been sufficient to cause 'Crack A' and 'Crack B' to grow and for the pitch horn to eventually fail in overload.

In summary, failure of the pitch horn could have resulted in the failure of the pitch link; similarly, failure of the pitch link could have resulted in failure of the pitch horn. However, without being able to recover the missing pitch link, and severed part of the pitch horn, it was not possible to establish the failure sequence.

Stiff yaw pedals

The increased force that the pilot had to apply through the yaw pedals, as a result of the swelling of the bushes (bearings) in the root section of the tail rotor blades, would have increased the static and cyclic loads applied to the pitch horn. However, the report of 'stiff pedals' occurred over one year and approximately 150 flying hours prior to the accident and both pitch links were replaced after this fault had been rectified. Following the accident, both tail rotor blades were found to be free to rotate on their spindles and the investigation could identify no evidence to indicate that the increased loads resulting from the 'stiff pedals' contributed to the eventual failure of either the pitch horn or pitch link.

Tail rotor pitch horn

The fatigue cracks that resulted in the previous failures of the pitch horn are reported to have originated in the area of the pocket, where the FEA undertaken by the blade manufacturer showed that the maximum stress occurs. The investigation identified a burr on the accident blade that ran along the edge of the pocket; the blade manufacturer believed that this could be caused by the shot peening process. The alternative means of compliance

with Emergency Airworthiness Directive 2003-08-51 required the edges of the pockets to be machined, to introduce a 2.29 mm (0.090") radius, on blades that have been in-service prior to them being shot peened: these blades were subsequently identified with the letter 'I'. Zero time blades, such as the one involved in this accident and marked with the letter 'M', did not require the pockets to be machined. The manufacturer advised that approximately 200 blades were manufactured to the 'M' configuration. It is not known how many 'M' configuration blades remain in-service.

It was also established that the blade manufacturer regularly receives blades that have corrosion and mechanical damage on the pitch horn. However, they only remove the paint around the pitch horn if there are visible signs of mechanical damage or paint blistering. Examination of three tail rotor blades (both blades removed from G-KSWI and one blade removed from another helicopter) established that not all the corrosion is visible through the surface finish.

The fatigue cracks on the tail rotor blade pitch horn appear to have formed in the same areas as in the previous occurrences. Also, it is not possible to check if there is corrosion on the blade root and pitch horn without first removing the surface finish. Therefore, the following Safety Recommendation is made:

Safety Recommendation 2011-100

It is recommended that the Federal Aviation Administration review Helicopter Technology Company's service life and approved maintenance programme, with regards to the inspection for corrosion, for tail rotor blades fitted to the MD 369 series of helicopters that have a pocket in the pitch horn (Part number 500P3100-101), to ensure their continued airworthiness.

The investigation established that, at some point, the pitch horn on the 'blue' tail rotor blade had been rubbed down with an abrasive material which removed some of the Alodine and left sharp scratches that appeared to penetrate through the Alodine layer across most of the surface. The shot peened surface was also missing in a number of areas covered in scratch marks. Similar damage was seen on a small area of the 'green' blade that had been repaired in the field. There was no documented evidence of the surface finish on the 'blue' blade having been disturbed other than when it was returned to the blade manufacturer 15 flying hours prior to the accident, when the paint around the root was lightly abraded. The blade manufacturer was adamant that their processes would not cause such damage, a position which was supported by the fact that the 'green' blade, which had been returned to the manufacturer at the same time for similar work to be carried, displayed none of this damage. The scratches would have compromised the corrosion protection and, with the lack of shot peening, would have made the blade pitch horn more susceptible to fatigue cracking. Given that there might be blades in service where the effectiveness of the shot peening on the tail rotor pitch horn has been compromised, the following Safety Recommendation is made:

Safety Recommendation 2011-101

It is recommended that the Federal Aviation Administration requires that Helicopter Technology Company ensures that there is an effective layer of shot peening on the pitch horns of in service tail rotor blades (Part number 500P3100-101) fitted to MD 369 helicopters.

Tail rotor pitch link

During the previous reported failure of the pitch link and five failures of the pitch horn, the pitch link

remained attached to the pitch control assembly. In this accident, it would appear that the pitch link detached in flight, leaving no evidence as to how it failed.

The 'green' and 'blue' tail rotor pitch links had been replaced with new items just over one year and approximately 124 flying hours prior to the accident. They were both removed and inspected during the last annual inspection, when corrosion had been removed from both links using 600 and 1500 grade abrasive paper, and had flown on G-KSWI for a further 15 hours prior to the accident. While the maintenance organisation appears to have followed the advice in the Corrosion Manual, the use of 600 grade silicon carbide paper on the 'green' pitch link introduced sharp edges and compromised the effectiveness of the shot peening, leaving the link more susceptible to corrosion and fatigue cracking. There was also no requirement in the maintenance and overhaul manuals to restore material properties following the rectification of corrosion. The following Safety Recommendation is made:

Safety Recommendation 2011-102

It is recommended that the Federal Aviation Administration requires that MD Helicopters ensures that an effective layer of shot peening is maintained on the pitch links fitted to MD 369 helicopters.

Maintenance, Overhaul and Corrosion Manuals

The MD 369 Maintenance, Overhaul and Corrosion Manuals contain no advice on the need to ensure that the surface properties, such as shot peening, are restored following the rectification of corrosion on tail rotor blades and pitch links. Therefore, the following Safety Recommendation is made:

Safety Recommendation 2011-103

It is recommended that MD Helicopters, in consultation with Helicopter Technology Company, updates the advice in the MD 369 helicopter Maintenance, Overhaul and Corrosion Manuals, with regard to the removal of corrosion and restoration of the surface finish and material properties on the tail rotor blades and pitch links, to ensure that the information is appropriate.