Robinson R22 Beta, G-IORG

AAIB Bulletin No: 9/2003	Ref: EW/C2002/05/04	Category: 2.3
Aircraft Type and Registration:	Robinson R22 Beta, G-IORG	
No & Type of Engines:	1 Lycoming O-320-B2C piston engine	
Year of Manufacture:	1991	
Date & Time (UTC):	14 May 2002 at 0855 hrs	
Location:	Sywell Aerodrome, Northamptonshire	
Type of Flight:	Aerial work	
Persons on Board:	Crew - 1	Passengers - 1
Injuries:	Crew - None	Passengers - None
Nature of Damage:	Main rotor blade cracked	
Commander's Licence:	Commercial Pilot's Licence	
Commander's Age:	57 years	
Commander's Flying Experience:	13,500 hours (of which 8,000 were on type)	
	Last 90 days - 250 hours	
	Last 28 days - 64 hours	
Information Source:	AAIB Field Investigation	

Synopsis

Over the period of two consecutive flights, the pilot noticed the onset of an increasing level of vibration, which he assessed as coming from the main rotor head. Post flight examination revealed the presence of an extensive crack in one of the main rotor blade root end fittings. Subsequent detailed examination revealed this to be a crack that had been produced by a fatigue mechanism, and that the blade was extremely close to the point of catastrophic failure. The origin of the crack was determined to be in an area where two regions of scoring/abrasion/material excavation damage, occasioned during an adhesive clean-up process during manufacture, had removed the shot peened surface treatment. This position was also an area of likely stress concentration due to a section change of the forged aluminium alloy root fitting. Similar damage has been identified on other root fittings removed from service, although no other known occurrences of cracking in this region have been identified. Such damage is not open to inspection on a serviceable blade. There was no known history of previous blade damage, or of any extreme flying duties performed by the helicopter. The blade had a Total Time of 747 hours out of a planned life of 2,200 hours.

History of the flight

G-IORG was used primarily as a camera platform for photographic survey work, flown usually by a freelance commercial pilot with the owner operating the camera from the left hand seat. On the morning of the incident the helicopter was flown by this pilot from his home to Maxey, Cambridgeshire, where he picked up the owner as his passenger. Weather conditions were fine with good visibility and a westerly wind. The plan was to carry out a photographic flight, the intended destination being Retford (Gamston) Airport, Nottinghamshire.

During the previous flight the pilot had noticed some vibration which had not been present on earlier flights; this vibration was again noticeable on his flight over to Maxey. When he subsequently took off with the owner on board, the vibration was considerably worse and he decided to abandon the proposed photographic flight and fly instead to Sywell Aerodrome, Northampton, where the helicopter was maintained, so that the cause of the vibration could be investigated.

The pilot decided to use a reduced power setting of 20 inches Hg for the flight to Sywell. Some 20 minutes into the flight the vibration became markedly worse but with Sywell now in sight the pilot decided to continue to there. He made a radio call to advise that he had a problem with vibration and once across the airfield boundary transited to the maintenance facility at low level as a precaution. The pilot described the vibration as being of low frequency and from the rotor head. He commented that he had experienced similar amounts of vibration in other types of helicopter and was not therefore unduly concerned.

After landing, a test pilot from the maintenance organisation went out to carry out an assessment. During his preliminary walk-round inspection, oil contamination was noted around the rotor head area. After climbing up to investigate the source of this oil, he was examining the spindle bearing oil retention boots for damage which from experience he considered a likely source, when he saw a large crack in one of the main rotor blades close to the root end, see Figure 1. The test pilot was appalled at the extent of the crack, and immediately quarantined the aircraft pending the AAIB investigation.

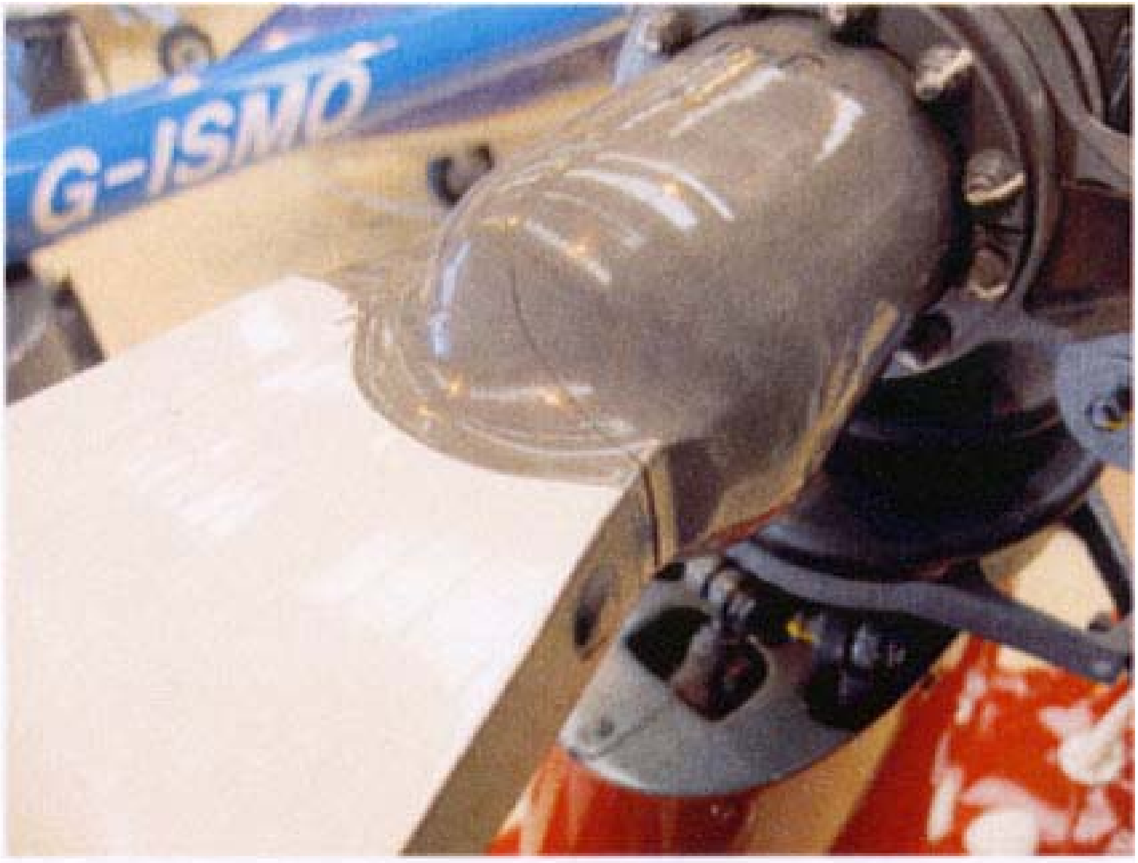


Figure 1
Crack in root fitting of main rotor blade

Operational information

The Pilot's Operating Handbook (POH) contains the following information for pilots under the heading '*Safety Tips*'

> *'A change in the sound or vibration of the helicopter may indicate an impending failure of a critical component. Make a safe landing and thoroughly inspect aircraft before flight is resumed.'*

The pilot flying the aircraft at the time of the incident states that when considering possible reasons for the vibration, he had not considered that it might be due to a cracked blade.

Aircraft information

G-IORG, airframe serial N° 1679, was built by the Robinson Helicopter Company in Torrance, USA, in 1991 and registered initially in Finland as OH-HRU. In 2000, after having accumulated some 1,992 airframe hours, the helicopter underwent a full overhaul (to zero hours) at an approved agency in the UK. Whilst undergoing this major overhaul, the aircraft was purchased for use as an aerial photography platform and placed on the UK register as G-IORG. Thereafter, it was used primarily for aerial photographic survey work and flown by regular free-lance pilots, with the owner operating a hand held still-camera from the passenger seat. It was occasionally flown for private use by the owner.

The helicopter was fitted with an 'aftermarket' side door panel in the passenger's door, which could slide open to provide a clear line of sight for the camera. Whilst filming, the helicopter was typically flown at a speed of about 60 kts and at a height of between 1,000 and 1,500 feet agl; turns to the left were used to give an unobstructed view of the ground, usually with a bank angle of around 30°. Dual controls were not fitted for these flights.

The owner and pilot reported that the helicopter was normally parked outdoors, at either the owner's or pilot's home, or at some other convenient location having regard to the planned survey schedule. The main rotor blades were usually tied down fore and aft.

At the time of the incident, the aircraft had accumulated a total of 747 flying hours since overhaul, giving a total hours since manufacture of 2,747. The main rotor blades installed at the time of the incident were manufactured by the Robinson Helicopter Company and delivered as a matched pair in February 2000, when they were installed on G-IORG during the course of the major overhaul. The cracked blade had thus had accumulated a total of approximately 747 flight hours at the time of the incident on 14 May 2002, out of a planned life of 2,200 hrs.

Relevant maintenance history

The helicopter's maintenance records show no significant maintenance activity relating to the main rotor during the 747 flight hours accumulated on the UK Register until approximately 8.6 hrs prior to the incident. Then, a vibration was noted during a routine test flight carried out as part of an annual maintenance inspection. This vibration was felt through the airframe (as opposed to the controls), and became apparent only after the aircraft had lifted off into the hover; during engine start and rotor run-up, vibration levels had been normal. Although noticeable, the level of vibration in the hover had not been severe and the test pilot had elected to continue the planned flight. During this, the vibration persisted whilst manoeuvring at low airspeeds but reduced once the aircraft had climbed and accelerated to circuit speeds. Upon completion of the test flight, which was satisfactory in all other respects, the aircraft was returned to the hangar for work to correct the rotor vibration.

Initial attempts to cure the vibration by making adjustments to blade track and balance were not successful. The blades were therefore removed and subjected to visual inspections for possible damage. Checks on the condition of the spindle bearings, pitch links, torque links, and teeter hinge friction were also carried out. No damage, excessive wear, or any other abnormality was found; the teeter hinge friction was within prescribed limits, and manual manipulation of the spindle bearings suggested that they were in very good condition.

After reinstallation of the blades, further attempts to balance and track the rotor were made but, when these failed to effect a cure, a decision was made to carry out a blade-shift. This entailed the removal of a shim from one side of the teeter hinge bolt and its reinstallation on the opposing side, effecting a 0.015" chordwise displacement of the rotor assembly and a corresponding chordwise shift in its centre of mass. Subsequent balance and tracking adjustments were successful in reducing the vibration to imperceptible levels and, upon completion of the remaining formalities and paperwork, the aircraft was returned to the owner's pilot on 2 April. The owner confirms that after its return from the annual inspection, the aircraft had felt smoother than at any time since he purchased it.

During the first four sorties following the annual inspection, extending over some five flight hours, the aircraft had remained exceptionally smooth. However, during the next flight the pilot noted the onset of a slight rotor vibration, albeit at a level which caused him no particular concern. On the subsequent flights these vibrations became more severe, culminating in the diversion to Sywell on 14 May.

The aircraft's technical log contained no relevant entries prior to the annual inspection test flight and neither the owner nor any of his pilots had reported any concerns regarding the rotor system.

Work carried out on the helicopter as part of the annual inspection, prior to the test flight when the vibration had first been noticed, had been essentially routine, but had included rectification of the following defects of potential significance in terms of the main rotor system:

- Inspection for blade root cracks in accordance with SL-53 [spar bolt hole location]

- Clutch operating switch replaced (original found to be intermittent)
- Rotor brake microswitch replaced
- Vee belts (main drive) replaced
- Governor switch cover on collective lever re-attached.

Blade construction

The R22 main rotor blade is of conventional bonded metal construction, comprising top and bottom skins of aluminium alloy with an aluminium honeycomb core, a hollow 'D' section leading edge spar of stainless steel, and a forged aluminium alloy root-fitting, Figure 2. The root fitting incorporates a short stub section which is bonded inside the blade proper, and two fingers which extend further into the blade. The larger of these fingers abuts the rear face of the hollow spar, to which it is fixed by a combination of adhesive bonding and a series of bolts which engage a strip of captive nuts located inside the spar. A smaller finger at the trailing edge of the root fitting is bonded directly to the skins. Aluminium alloy doublers bonded externally to the skins (omitted for clarity from Figure 2) provide local stiffening and reinforcement of the skins in the vicinity of their connection to the root fitting. An aluminium alloy cap is bonded to cover the end of the hollow spar and an adhesive fillet blends the root end of this cap to the root fitting.

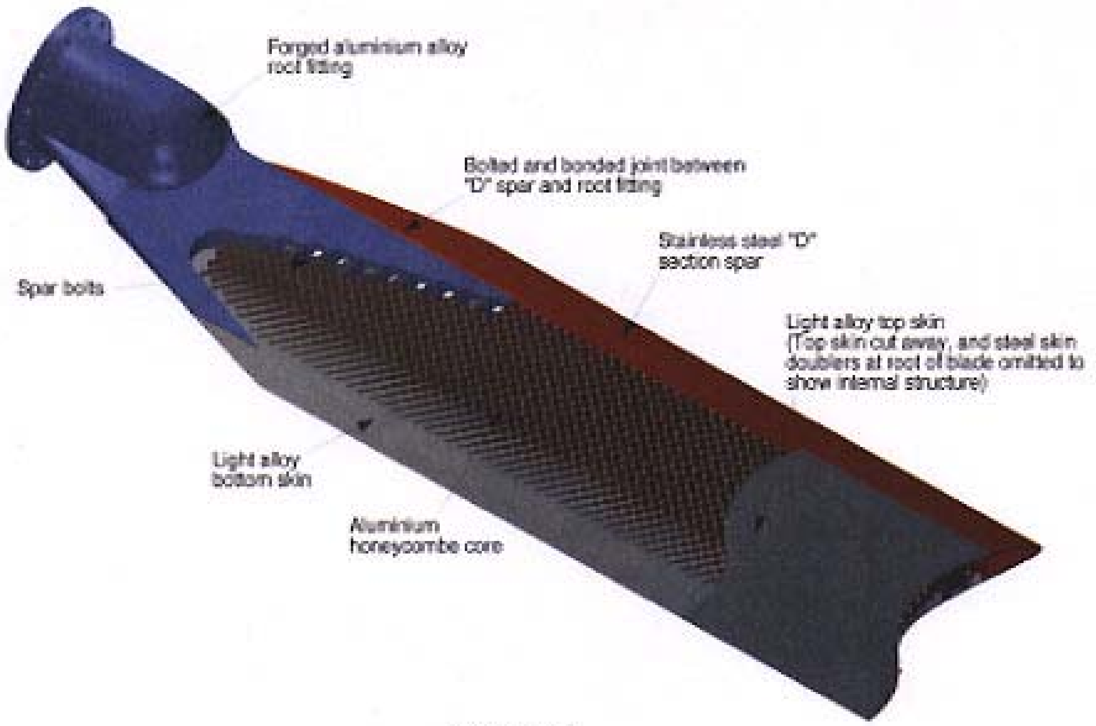


Figure 2a
R22 Main rotor blade construction details

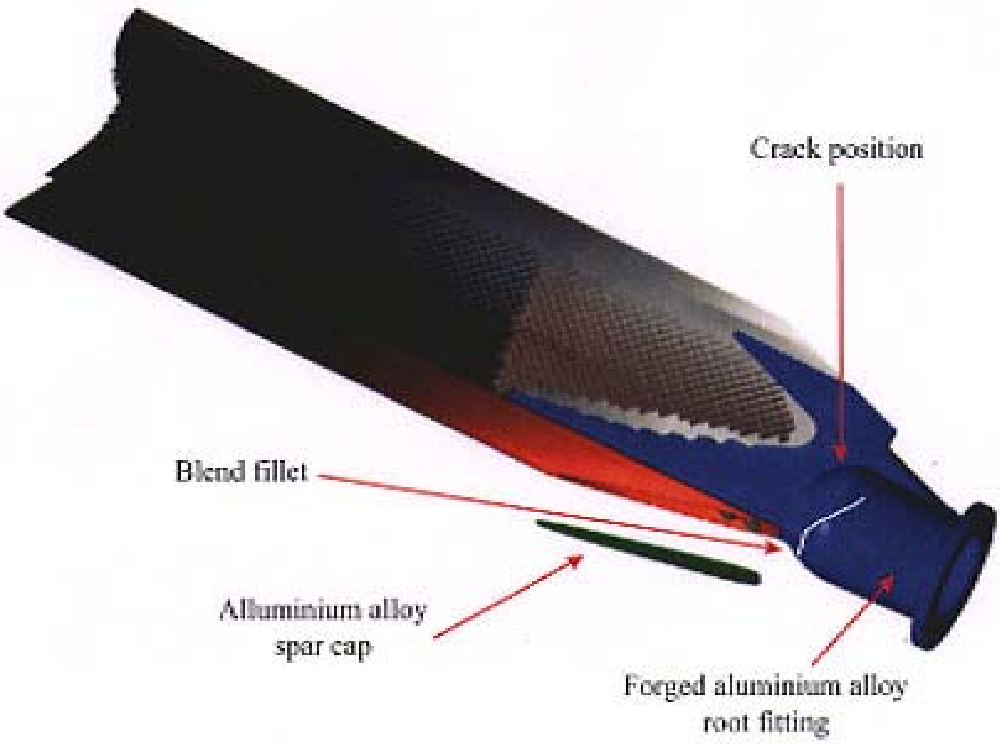


Figure 2b
R22 Main Rotor Blade construction details

Failed Main rotor blade P/N° A016-Z, S/N° 12110C

The crack, which was confined to the blade root fitting, had apparently initiated at the leading edge and propagated back towards the trailing edge, turning progressively inboard as the crack developed, Figure 3.

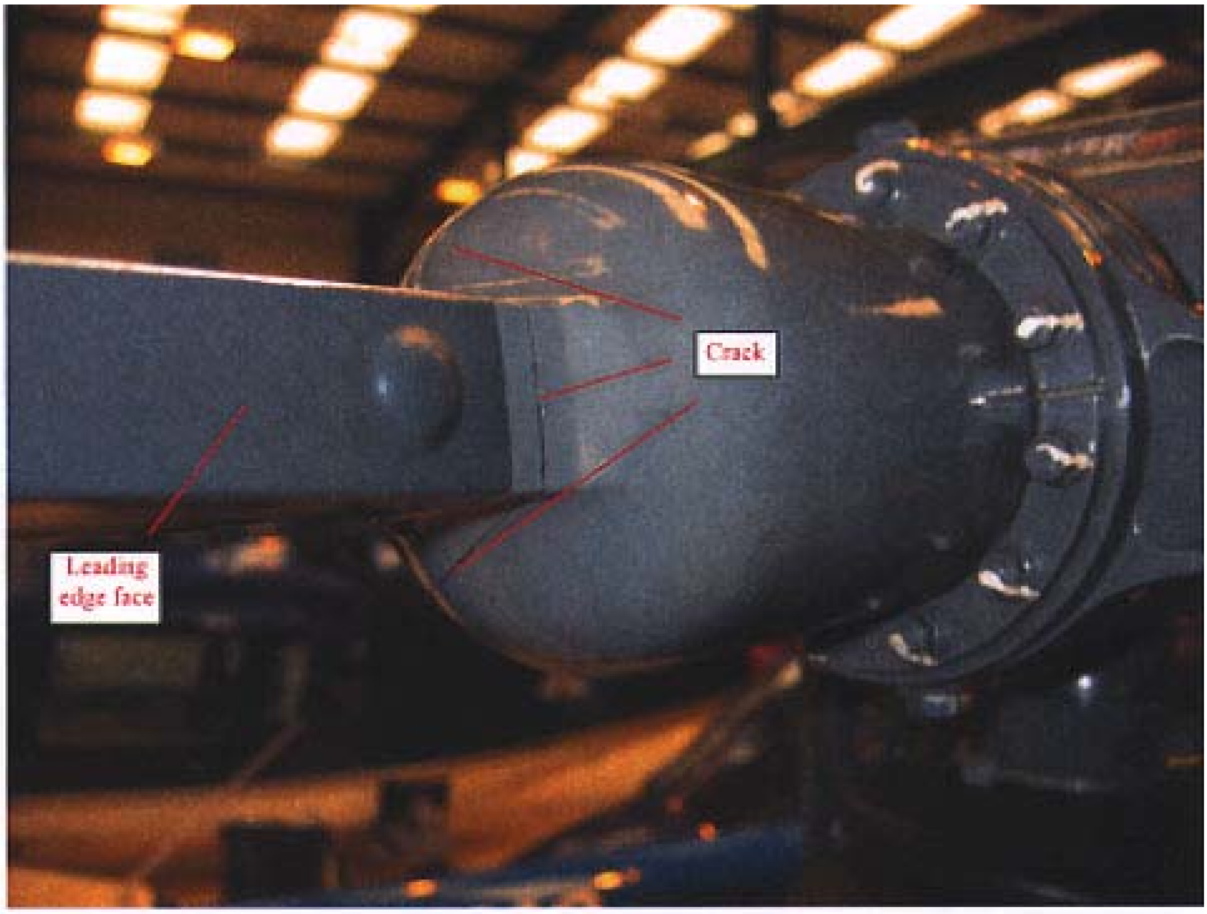


Figure 3
Main rotor blade in situ, showing the crack extending from the leading edge face across both top and bottom surfaces of root fitting

In total, the crack had extended a distance of approximately 9 cm beyond the leading edge on the lower surface of the fitting and approximately 7 cm on the upper surface, Figures 4 and 5. The crack had evidently penetrated through the full thickness of the forging over a significant proportion of the cross-section, and had opened up sufficiently to produce a discernible step-like discontinuity across the fracture at the leading edge. Localised distortions were also visible in the surface of the forging in the vicinity of the crack fronts, particularly on the upper half of the fitting (Figure 5), suggested that the crack, which was considered at this time to probably be associated with a fatigue mechanism, had reached an advanced stage involving low cycle rapid growth and that the blade had been very close to catastrophic failure.

Examination of the crack fissure in the leading edge with the aid of a hand magnifying glass revealed numerous microscopic cracks in the paint adjoining both sides of the fracture. The character, position, and orientation of these cracks in relation to the main crack appeared consistent with them having been caused by abnormally high strain. This was either as a consequence of main crack growth or as a result of some high strain event(s) occurring before the main crack had begun to propagate significantly; it was not possible to establish which of these was the more likely cause.

Figure 4
Undersurface crack (length approximately 9 cm)

Figure 5
Topsurface crack (length approximately 7 cm)

Figures 4 and 5
Extent of crack growth through top and bottom surfaces of root fitting
(pen tip identifies ends of crack)

Examination of R22 blades on other serviceable aircraft in the hangar, revealed a range of fine cracks in the paint on the leading edge of their root fittings in the relevant area. Subsequently, the majority of another operator's R22 fleet was found to exhibit surface cracking in this same area, both on the leading edge and also extending around the lower corner onto the underside. In general, the cracks visible on these serviceable blades appeared to be finer and tighter than those adjacent to the main fracture on G-IORG.

After consultation with the manufacturer, the cracked blade was removed from G-IORG. A detailed visual inspection of the blade revealed no evidence of damage, corrosion, disbonding or deformation. The tracking adjustment tab was not deformed or damaged and was set to a position similar to those observed on other R22 blades installed on serviceable aircraft. Overall, the blade appeared to be in excellent condition.

Manual manipulation of the spindle bearing at this stage, with the blade in its 'as removed' state and with the oil retention boot (which was intact and undamaged) still in place, revealed considerable stiffness and roughness in the spindle bearings. Extraction of the spindle bearing assembly from the root fitting, using manufacturer-supplied service tooling, revealed that the crack extended completely through the walls of the cavity on both sides of the blade. The bearing cavity was largely devoid of the oil, which would normally fill it, but any oil originally inside the cavity was likely to have been driven out through the crack under centrifugal loading. This was almost certainly the source of the oil contamination of the rotor head noted when the aircraft had landed immediately after the incident. After disassembly of the bearing stack at a later stage in the investigation, microscopic examination of the individual bearing raceways showed that the stiffness had been caused by an accumulation normal wear products which had oxidised and become attached to the surface of the race. This was considered almost certainly due to a combination of the loss of oil and the consequent entry of air/moisture into the bearing cavity. The underlying condition of all of the balls and tracks making up the spindle bearing assembly was good, and no evidence was found of any brinelling or any other feature indicative of excessive centrifugal loading of the blade.

Opposing main rotor blade

The opposing blade displayed no evidence of cracking comparable to that seen on the failed blade, but microscopic surface cracking in the paint at the leading edge could just be discerned in the relevant area of the root fitting. These fine cracks bore some similarity to the fine secondary paint cracks noted on the failed blade, but they were fewer in number and much tighter. Overall, the character of these cracks was consistent more with paint shrinkage due to ageing than to strain, but the possibility of the latter could not be totally ruled out without recourse to invasive and potentially damaging measures to penetrate and remove the adhesive, filler, and paint materials in the affected area. Subsequently, at later stage in the investigation, procedures were developed which enabled these covering materials to be removed safely, and it was shown that the cracks did not extend into the metal.

After removal from the aircraft, the blade was inspected in detail. In contrast to the cracked blade, the spindle bearing cavity in the intact blade was completely filled with oil and the bearings themselves operated smoothly. No evidence symptomatic of excessive vibration or other abnormality was found, and subsequent re-examination at the manufacturer's facility revealed no dimensional or other anomalies which could have contributed to, or been associated with, the fatigue crack in the failed blade. Examination of the blade root fitting in the area of interest using eddy current and X-ray techniques revealed no evidence of cracking.

Rotor head, engine and airframe

The aircraft was in exceptionally good condition overall. However, given the potential implications of an overload of the blade system as a possible causal or contributory factor in the initiation of the fatigue crack, checks were carried out on the rotor system, and on the aircraft as a whole, to look for evidence of any condition capable of inducing a sustained, or transitory, abnormal loading of the blade roots.

All component parts of the rotor head were in good condition. The flapping hinge bolts had been torque tightened to give the correct amount of bolt stretch, and the teeter hinge friction was within the prescribed limits. The rotor chord balance weights were securely attached: these were removed, and found to weigh 184 grams, including the bolt and nut. All pitch link bearings were in good condition. The swash plate scissor-link bearings displayed some looseness and wear, but were no worse than the equivalent bearings installed on other R22 aircraft examined for comparative purposes. It is a known feature of the R22 that rotor vibration is more likely to result from stiff new scissor-link bearings than from well used bearings exhibiting a relatively large amount of slack.

A limited inspection of the engine was carried out in situ. Removal of the exhaust valve springs and caps from all four cylinders to inspect for signs of valve overtravel, revealed no evidence of any significant engine overspeed, this being one possible cause of a main rotor overspeed. The engine and gearbox mountings were intact and in good condition, and the rotor mast run-out was within limits. The drive belts were in good condition, and the clutch mechanism was correctly rigged, disengaged correctly, and appeared to be in good condition.

In summary, no evidence was found of any condition affecting the airframe, engine or transmission capable of causing any abnormal loading of the main rotor the main rotor system.

Detailed examination of the fracture

The inboard section of the blade, containing the whole of the cracked root fitting, was excised from the blade and taken to an independent laboratory where it was subject to detailed metallurgical examination under AAIB supervision.

External surface features

Preliminary optical microscopy of the leading edge surface confirmed the presence of the numerous secondary cracks adjoining the main crack noted previously, Figures 6a and 6b. Their appearance was consistent with abnormally high strain in the surface material, but it was not apparent whether they had proceeded, or were symptomatic of, the main crack.

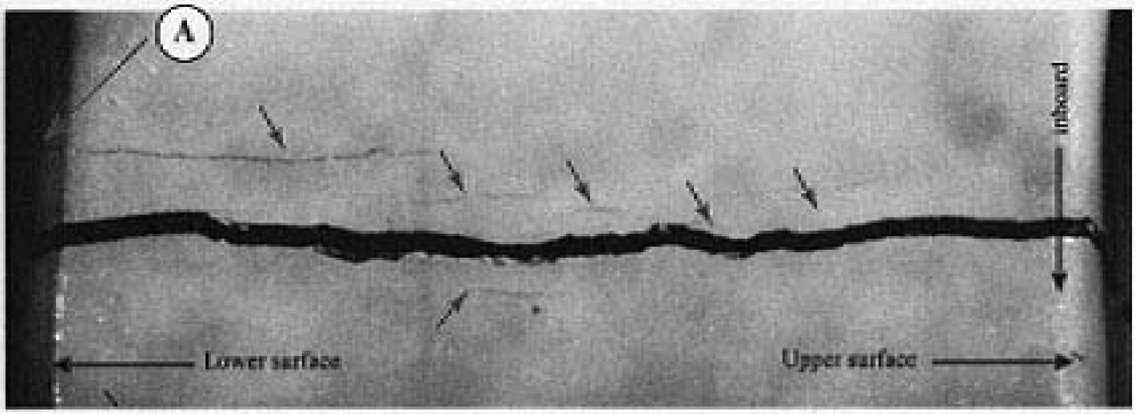


Figure 6a
Composite photograph - view onto leading edge face

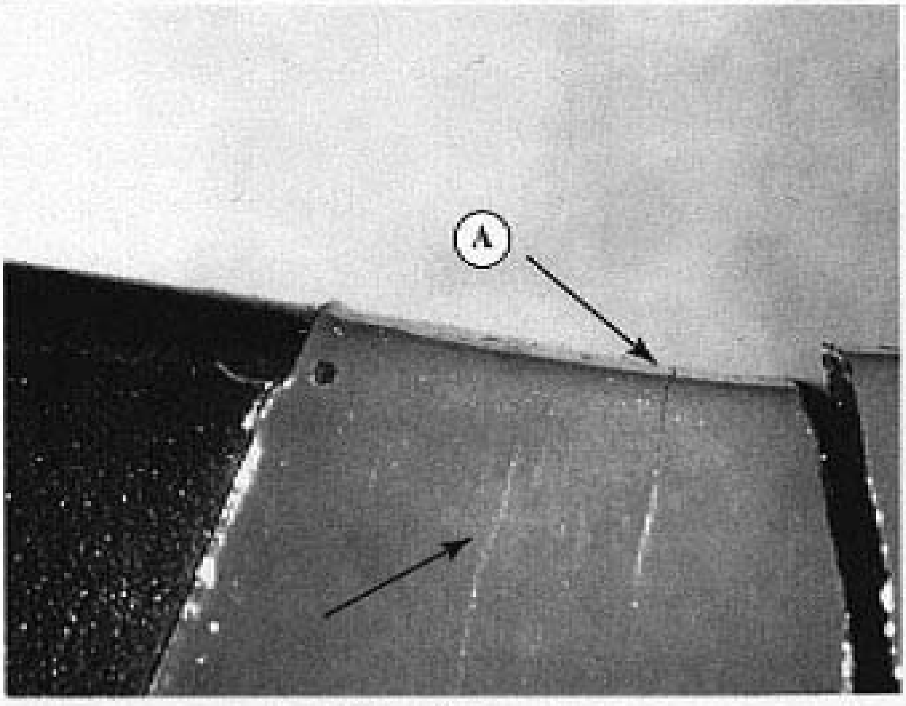


Figure 6b
View onto lower face

Figure 6a and 6b
Secondary cracking in the filler/paint of the leading edge and lower surfaces.
Note: Crack labelled 'A' extends around lower corner onto lower surface

Without recourse to sectioning and other invasive techniques, it was not possible to determine visually whether the secondary cracks visible on the exterior of the fitting were confined to the adhesive/filler/paint layers or whether they extended into the metal of the fitting proper. Examination of the relevant parts of the blade root using eddy current techniques revealed no evidence of secondary cracking in the root fitting; however, the presence of the filler reduced the sensitivity of the technique.

Fatigue fracture details

In order to expose the fracture faces for examination, a cut was made from the trailing edge side of the fitting to intercept a point just beyond the furthest extent of the visible fatigue crack, and the pieces cleanly pulled apart so as to avoid bruising of the fracture faces. The outboard half of the fracture thus exposed is shown in Figure 7. (Note the thick layer of adhesive/filler/paint on the leading edge surface.)

Figure 7
Outboard half of fracture

Fracture Origin

The fracture originated at the leading edge lower corner of the section in the region identified in Figure 7, and shown in detail in Figure 8. There was no single-point origin; rather, the crack initiated over an approximately 1 mm length of surface coinciding with a facet adjacent to the bottom corner of the leading edge; this was consistent with stress-induced cracking. No evidence was found of any corrosion or pitting of the surface in the origin region; nor of any inclusions or other metallurgical defects within the material.

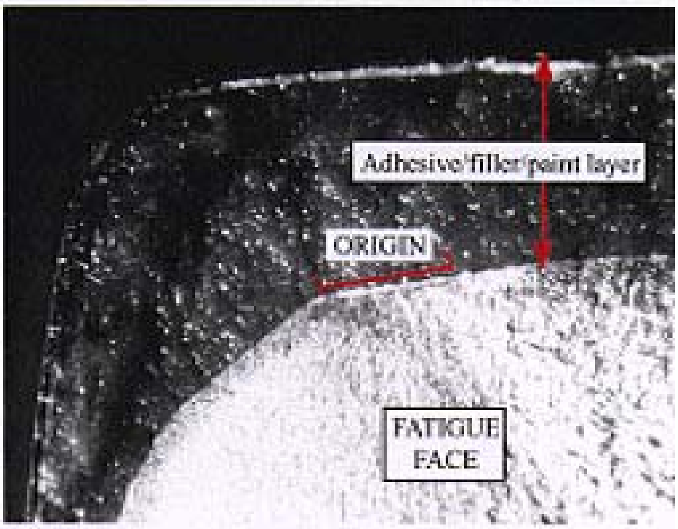


Figure 8
Close up of origin region

Fracture growth characteristics

The crack had propagated across the leading edge of the root fitting and into the hub section, and extended over a total distance of the order of 80 mm. Development occurred in stages, producing an irregular series of conchoidal beach marks over much of the surface. This indicated interruptions to crack growth and these were interspersed with groups of finely spaced fatigue striations, the result of individual load cycles. The finer striation spacing was initially of the order of 0.1 to 0.5 µm but was more typically 1 to 2 µm across the fracture. This spacing was overlaid by at least one other series of spacings up to 20 µm. Within this overall pattern, three distinct and sequential regions of growth were apparent.

Stage 1 growth region.

This extended from the origin region out as far as the first measurable striations, a propagation distance of approximately 10 mm. Since it was not possible to identify individual striations within this region, the actual growth distance per load cycle (striation spacing) could not be determined directly.

Stage 2 growth region

This extended from approximately from 10 mm to approximately 50 mm of the crack length, and exhibited clearly defined beach marks of variable width and spacing. Within these bands, individual striations were visible having a spacing mainly in the range 1 µm to 2 µm; some striations finer than 1 µm were also visible, but in other parts of the region individual striations were not discernible.

Stage 3 growth region

This extended from approximately 50 mm crack length to the crack front at, approximately, 80 mm. Initially in this region, fatigue growth predominated but with small intervals of ductile rupture towards the crack front. Ductile fracture characteristics began to dominate nearer the crack front and these areas were separated by smaller and smaller bands of fatigue. However, the fracture had remained a progressive process, albeit an accelerating one.

Further investigations

Sections

The outboard segment of the failed root fitting was cut to produce spanwise sections normal to the chordwise axis, in a plane set approximately 4 mm back from the lower corner of the leading edge. The etched and mounted sections thus produced showed not only the grain structure of the fitting proper, but also the inboard end of the stainless steel spar and its aluminium cover plate, together with the various layers of primer, adhesive, filler, and paint.

It was confirmed that the secondary surface cracks (at least, those visible in the sections) did not penetrate through the filler layer into the metal of the fitting.

Under high magnification, noticeable differences were apparent in the sub-surface microstructure of the root fitting in the region beneath the spar, compared with that evident beyond the end of the spar, inboard of the adhesive fillet. Specifically:

- a thin anodic layer was visible in the region beneath the spar/adhesive fillet, but absent inboard of the fillet, except over a small band approximately 0.5 mm wide located approximately mid way between the end of the spar and the fracture, and
- deformation of the grain structure consistent with that produced by shot peening was visible in the region beneath the spar/adhesive fillet, but absent inboard of the fillet; except in the narrow 0.5 mm band where the anodic layer was present (see above).

Where evidence of grain deformation characteristic of shot peening was present, the visible depth of the deformation layer was typically 0.002". Figure 9 shows the mounted and etched section, labelled to show the features referred to above.

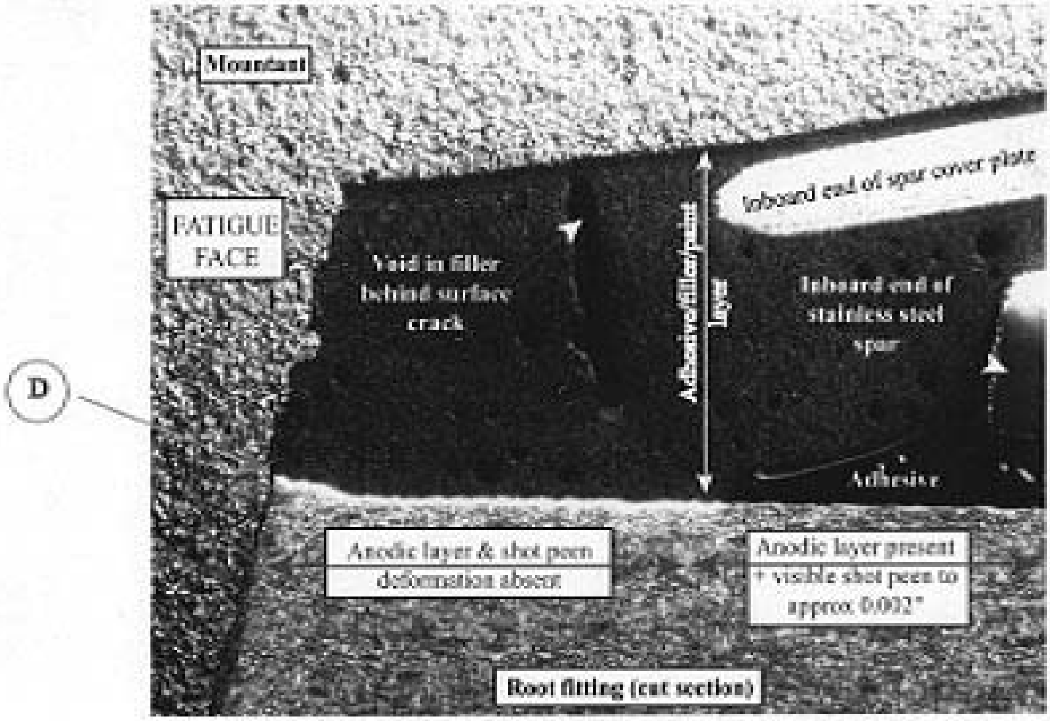


Figure 9
Mounted and etched spanwise section, cut normal to chordwise axis in a plane approximately 4 mm back from the lower corner of the leading edge
(Arrow 'D' shows the view direction of the oblique macro photograph shown in Figure 10)

The absence of both the anodic layer and deformation (shot peened) layer on the surface of the fitting inboard of the spar, and the concave cross-sectional shape of the adhesive fillet at the end of the spar, suggested that some form of post-bond surface clean-up had taken place during manufacture. This had scoured the exposed surface sufficiently to have removed these layers inboard of the spar.

Material properties

Hardness tests carried out on the mounted sections produced results consistent with the material being 7075 aluminium alloy in the T73 condition, as specified by the manufacturer.

Surface finish

In light of the evidence, which suggested that the leading edge face of the blade root might have been subject to an abrasive process during manufacture sufficient to remove the anodic and shot peened layer inboard of the spar, attempts were made at an early stage in the investigation to examine the surface finish of the root fitting in the vicinity of the origin. These attempts were severely hampered by the presence the adhesive primer, adhesive, filler and paint layers, of which the adhesive materials proved extremely tenacious and therefore difficult to remove. Because of the significant risk that removal of these overlaying materials could result in obliteration, or further damage, to the surface of the fitting, initial attempts to expose the surface were highly circumspect. These attempts were confined to a very small area close to the fatigue origin on the excised segment from the lower leading corner of the fitting outboard of the fracture, ie, from the element containing the outboard fracture face which had been used to provide the etched and mounted sections referred to previously.

Figure 10 shows the region thus exposed, looking obliquely down in a spanwise direction onto the fracture and leading edge faces. The exposed leading edge surface was characterised by a series of closely spaced scores, running transversely across the leading edge and parallel to the plane of the fracture, which appeared to have been produced by an abrasive process. These features are highlighted in Figure 10.

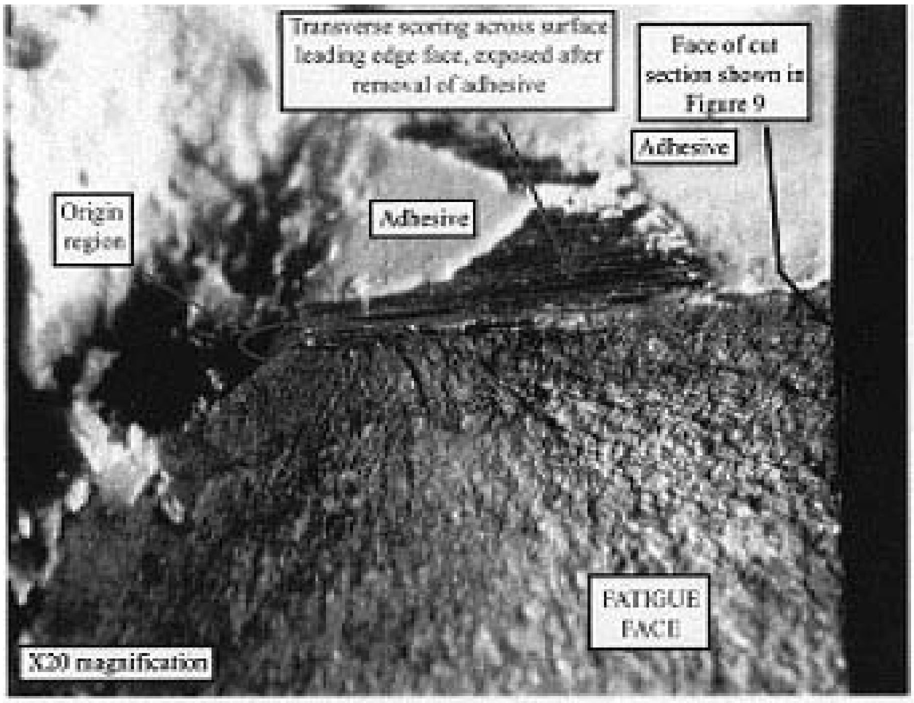


Figure 10
Transverse surface scoring of leading edge face in vacinity of origin, exposed after localised removal of filler/adhesive layers
(Oblique view, looking spanwise onto outboard fracture face and leading edge surface - in direction of arrow 'D' in Figure 9)

The overall surface condition of the cracked root fitting could not be assessed properly without the expenditure of considerable effort to remove the overlaying spar cover plate, skin doublers, adhesive, filler, and paint layers; all without damaging the underlying surfaces. After much experimentation, it was found that following some initial careful chipping to remove the bulk filler materials, the underlying, and much more tenacious adhesive layers, could be removed safely using pegwood sticks mounted in a high speed rotary tool to carefully grind away the non-metallic material. This was aided by occasional soaking of the working area with acetone.

With the root fitting surfaces thus exposed, it was apparent that the scoring observed previously, in the small area of the leading edge exposed originally, Figure 9, was actually part of a much more widespread region of transverse scoring. Two distinct transverse bands of abrasion were present on the leading edge, each approximately 2.5 mm in width, and separated by a spanwise distance of approximately 7.5 mm. Each band extended around the chamfers at the upper and lower corners of the leading edge and continued, for a short distance, chordwise, back across the upper and lower surfaces respectively, see Figures 11a, 11b and 11c. The fatigue crack origin coincided with an area of particularly aggressive scoring around the lower leading edge corner, which formed part of the more outboard of the two bands of abrasion. Similar, though slightly less intensive scoring, was evident on the upper corner.

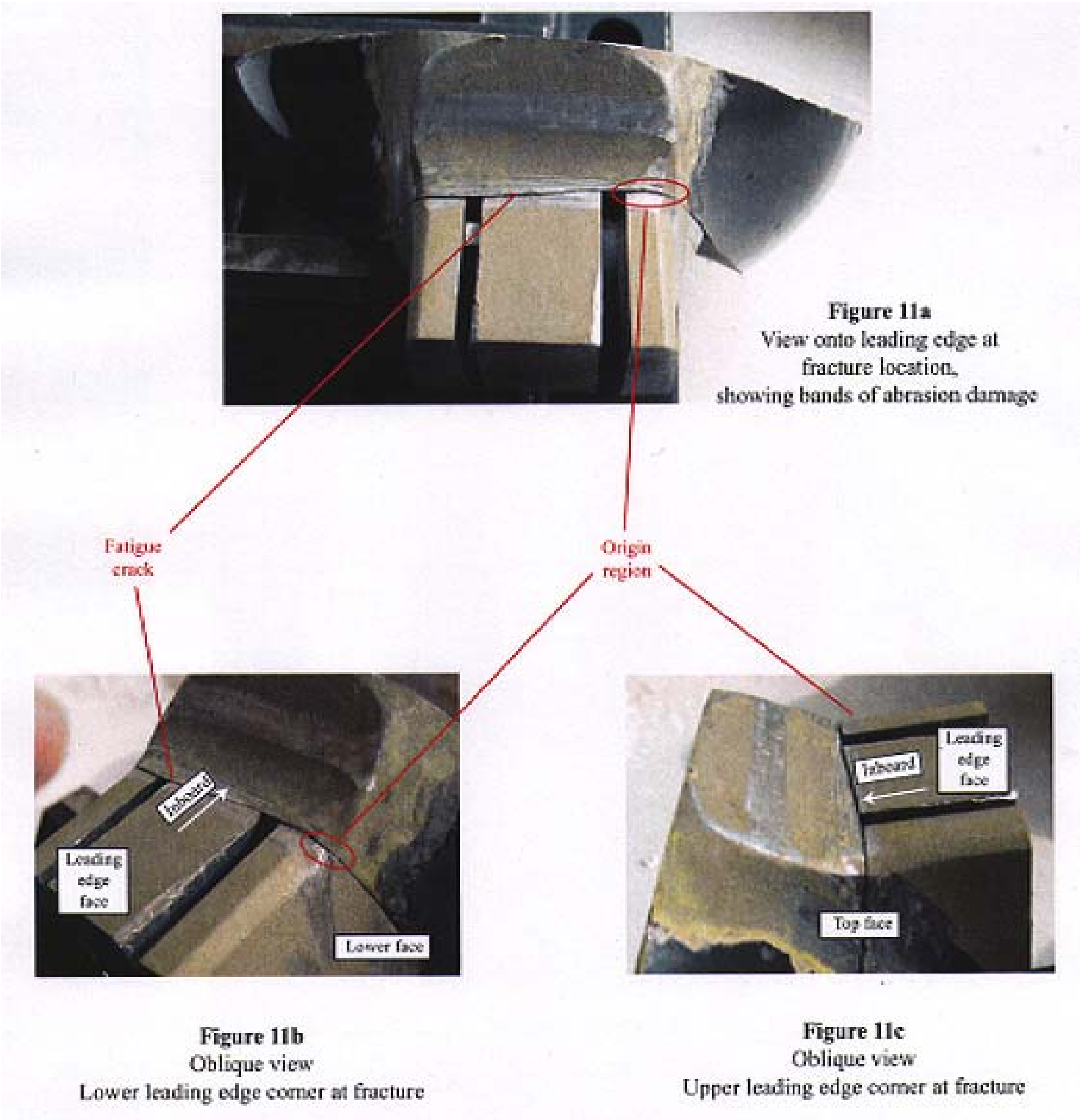


Figure 11a
View onto leading edge at fracture location, showing bands of abrasion damage

Figure 11b
Oblique view
Lower leading edge corner at fracture

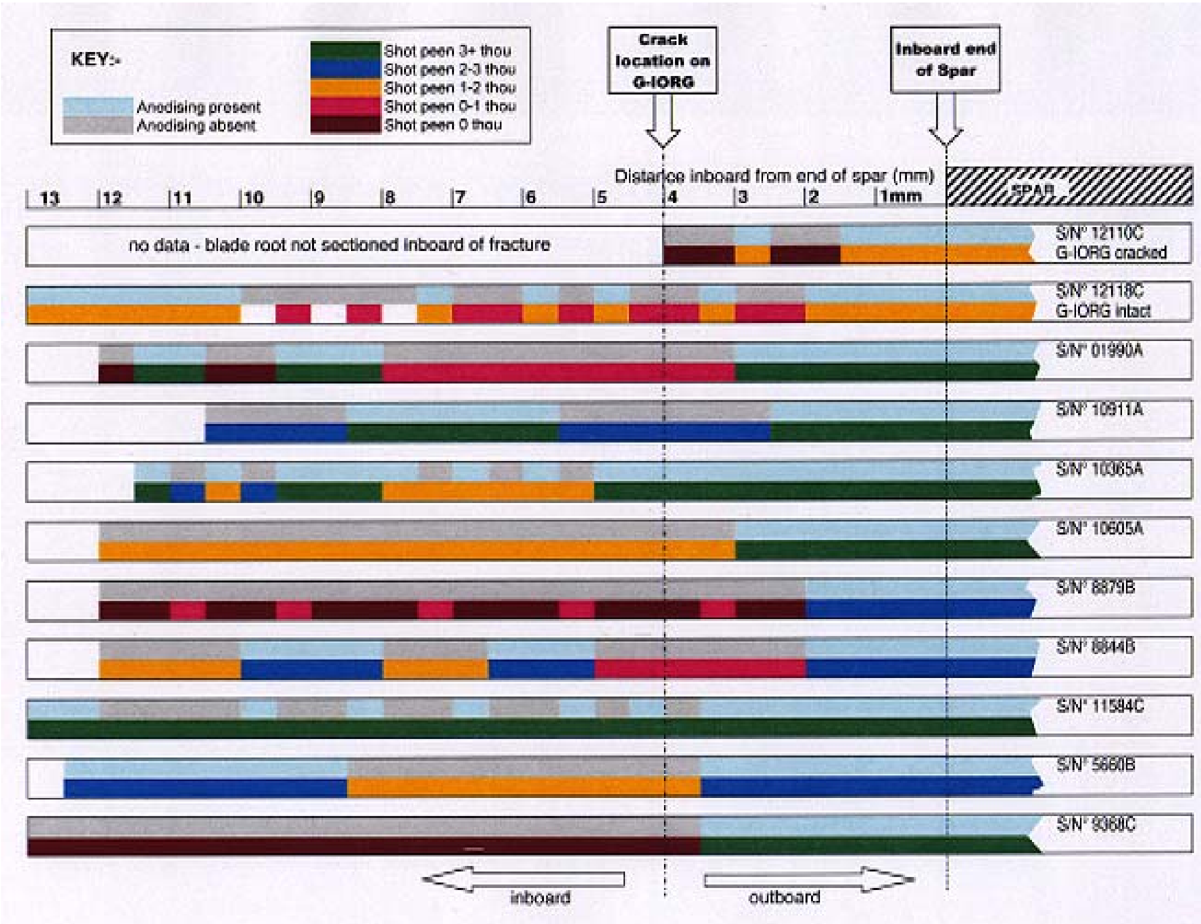
Figure 11c
Oblique view
Upper leading edge corner at fracture

Comparisons with other R22 root fittings

With considerable assistance from a major R22 operator and service centre for R22 helicopters in the UK, a total of nine sample R22 main rotor blade root fittings were obtained for comparative examination. Most of these blades were time-expired, others had been scrapped due to damage in service; all blade roots were undamaged. The intact, opposing, blade from G-IORG was also examined for comparative purposes.

Distribution of anodic and shot peened layers in the leading edge surfaces

The root fittings of the sample blades were cut to excise the region corresponding to the cracked part of the blade root from G-IORG, and the elements thus obtained were sectioned at locations similar to those used on the cracked blade. These were mounted, polished and etched to allow the distribution of any anodic layer, and the presence and thickness of any shot peened layer, to be determined for each. The results obtained are presented in chart form at Figure 12, which shows the spanwise distributions of anodic and visible shot peen layers for each of the sample blades, and also for the cracked and intact blades from G-IORG. The data clearly shows that the surface layers are intact beneath the spar and for a short distance, typically up to about 3 mm, inboard of the spar end. Further inboard, however, varying amounts of surface have been removed, consistent with an abrasive process during manufacture to remove excess adhesive after bonding, but prior to the application of filler.



Spanwise distributions of anodic and shot peen deformation layers for G-IORG blades.
and 9 other R22 blade root samples
Data based on examinations of polished and etched spanwise sections taken normal to the clockwise axis, in a plane set approximately 4 mm from the lower corner of the leading edge

Surface condition - other blades

Removal of the filler and adhesive layers from the opposing intact blade from both G-IORG, and the range of sample blades revealed a broadly consistent pattern of transverse abrasion across the leading edge face just inboard of the spar, and around the corners. This was similar to that noted on the cracked blade from G-IORG. Figure 13 shows the opposing blade from G-IORG, and Figure 14 shows these characteristics on the other sample blades examined by the AAIB.

Figure 13
Surface condition of intact blade from G-IORG
(Note: 'dimple' bruise marks are post-incident damage, from initial attempts to remove filler/adhesive)

Material properties

Hardness determinations (Vickers hardness - HV10) were carried out on the microsections from both the failed blade (S/N° 12110C) and a comparison blade (S/N° 9368C). The results gave equivalent average values of 145HB and 147HB respectively. These values are consistent with the aluminium alloy 7075 in the T73 condition.

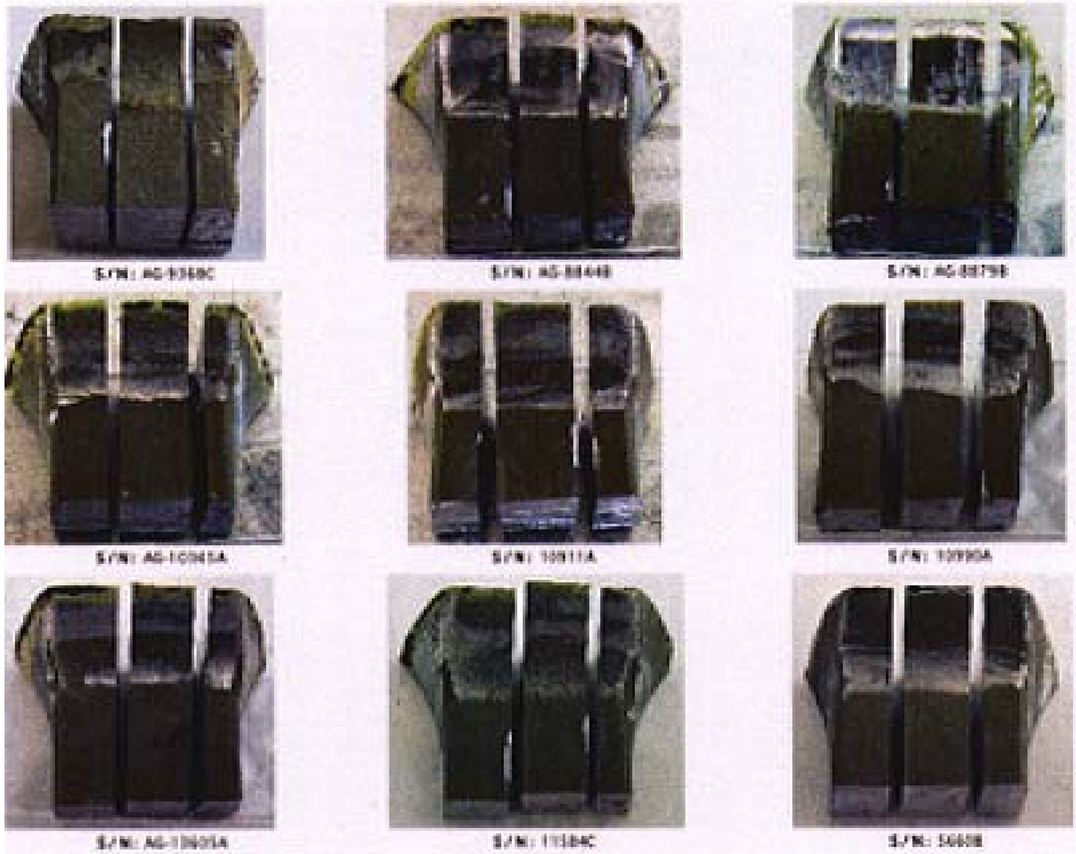


Figure 14
Surface condition of sample blades obtained by AAIB for comparision with the blades from G-IORG

Additional information

Main blade S/N° 12110C

The manufacturer's production records show that the cracked blade was one of a batch of 10 blades which completed assembly on 16 February 2000. It was the 12,110th R22 main rotor blade manufactured, and the 787th blade produced to revision 'AI' standard. The C suffix in the serial number shows that it was bonded in the factory's 'C' assembly fixture.

Except for specialist processes, such as initial forging and shot peening of the root fitting, the main rotor blades are manufactured entirely by the Robinson Helicopter Company. After completion, the blades are graded on the basis of their mass and blade twist criteria, and are usually supplied to the customer as a matched pair.

Root fitting

The root fitting is manufactured from 7075 aluminium alloy, solution heat treated and aged to the T73 condition. After forging, the fitting is dimensionally checked before being initially machined to produce the attachment flange profile, associated bolt holes and to rough-bore the spindle bearing cavity. It is then shot peened prior to finish-machining the flange faces and the bore of the bearing cavity. After cleaning, the fitting is acid etched and anodized before being coated with an adhesive primer and baked for 30 minutes at 250°F.

Bonded blade assembly

The blade is essentially a bonded sandwich assembly, supplemented by bolts at the connection between the spar and the root fitting, and at the attachments of the tip weights to the spar. All adhesive is applied in the form of a 'pre-preg' tape, comprising an open mesh matrix which supports a thin layer of adhesive within peel-off protective outer tapes on each side. After being first cut to the required shape, the tape is applied in the manner of a double-sided adhesive tape, serving both to locate and maintain the positions of the mating parts during their preliminary assembly. It also provides the final bond between these parts after the assembly has been cured at elevated temperature. Prior to assembly, all components are subject to preparative surface treatments including the application of adhesive primer, after which all processes are carried out in 'clean-room' conditions.

The assembly sequence is as follows:

1. The spar is firstly bolted to the root fitting with an interlayer of pre-preg adhesive tape, and the bolts torqued up. These bolts remain in situ thereafter except for the innermost pair, which, during the assembly stage, also locate a special tooling block to evenly distribute the clamping forces during the final curing process; these are removed subsequently, and replaced by shorter bolts. The tip weights are also bolted to the spar at this stage.

2. With the spar assembled to the root fitting, pre-preg adhesive tape is applied to the skins. The spar/root fitting sub-assembly and the honeycomb core are then laid into position against one of the skins, and the second skin placed in position to complete the section.

3. The external skin doublers are added to the external skins in the root area, and the whole package placed in a bonding fixture which, when closed, clamps the blade under constant pressure between heated patterns shaped to control both the aerofoil section and blade-twist during the curing process.

4. The blade assembly is cured in the fixture for one hour at a temperature of 250°-300°F, during which process the adhesive softens and the cross-sectional profile shrinks accordingly.

5. The bonding fixture is designed to accommodate the shrinkage due to adhesive softening, and maintains an even pressure to hold both the correct blade profile and twist; however, as a consequence, excess adhesive is squeezed from the free edges of the bonded joints. At the inboard end of the blade, in particular, both at the skins where the doublers add to the amount of adhesive present and also at the inboard end of the bolted joint between the spar and the root fitting, significant amounts of excess adhesive will usually exude out onto the root fitting.

6. When cured, and after the assembly has cooled, the blade is removed from the fixture and moved out of the clean room for post-assembly inspection, clean up, and preparation for final finishing. These processes included:

- The removal of sacrificial regions of blade from the tip and the root trailing edge corner. The material thus removed served as coupons for testing bond peel strength: a quality assurance measure which provides information about the overall quality of the bond process for that blade.

- The trimming back of excess material from the wrap-around skin doublers at the root end of the blade, to match the spar run-out taper.

- Removal of excess adhesive generally, and from the root fitting in particular. (The latter process typically involves the use of an abrasive processes which includes a small grit-loaded rubber grinding wheel to cut back and remove adhesive adhering to the domed part of the forging around the inboard edges of the skins. Adhesive is also removed by this process from the leading edge part of the forging immediately inboard of the spar.)

- Removal of the two innermost spar attachment tooling bolts, and the temporary clamping pad, and their replacement with shorter bolts.

- Installation of an aluminium cover plate over the exposed end of the hollow 'D' section spar, where it tapers down at its inboard end.

- Application of epoxy-type filler materials and sealer, to blend out the step between the end of the spar/spar cover-plate, and the root forging, followed by primer and finish paint processes.

Quality control measures - R22 main blade manufacture

Shot peen process

The root fitting is shot peened in accordance with the subcontractor's technical sheet SP-67465, which serves as a process instruction sheet for any given batch of R22 blade roots. This also records the exposure-time, intensity and other pertinent data relating to the batch. The document specifies:

- full masking of the mounting flange face (including the bore); 230 (0.023) diameter) steel shot

- a peen intensity from ".006A to .009A"

- 100% coverage, with 5 minutes exposure in each of four orthogonally opposed positions.

The specification against which technical sheet SP-67465 is written is quoted as RPS-63. This is the aircraft manufacturer's process specification for the .....

'...peening of part surfaces to induce residual compressive stress in the surface layers'.

Revision A of RPS-63, dated July 1993, comprises a 14 page document controlling all aspects of shot peen operations carried out by, or for, Robinson Helicopter Company and lays down guidelines, limits, and recommended practices to be followed. Paragraph 2.1.6 of RPS-63 states,

'Whenever parts are heated after peening as for baking of paints or protective coatings, bonding of components, thermal fitting, or other operations resulting in the heat-up of the part surface, the temperatures employed shall be limited to those listed in table 1'.

Table 1 specifies a maximum temperature of 250°F for aluminium alloy materials.

Process Specification

The process of assembling together the component parts of the main (and tail) rotor blade is detailed in RHC Process Specification RPS-6, issued originally in March 1975. Revision 'W', dated July 1999, the version of the document current at the time of manufacture of the cracked blade on G-IORG, comprises 39 pages of detailed process instructions, of which the following extracts are relevant:-

Relating to the bond cure process...

'5.28.6 Install cover and turn on controller. NOTE: Fixtures shall have root fitting and tip zone controls set to the temperature required to maintain a temperature range of 250° to 300°F for the duration of the cure cycle.'

Relating to the post-cure dressing and cleanup process...

'5.36 Clean up the excess adhesive. Around the root fitting, A934 doublers, A301 doublers, A299 trim tabs, and along spar line remove the minimum adhesive squeeze-out to fair surfaces. Maximum material removal permitted from skins while cleaning off excess adhesive is .004 (See Section 9).'

Section 9 of the document includes...

'9.4 Main and Tail Rotor Root Fitting Damage Limits
All damage on the root fitting shall be blended out using a minimum .10 radius within the following limits:

9.4.1 Main Rotor Blades (Ref. Figure 6) [reproduced in figure 15 above]

9.4.1.6 All other areas of the root fitting may be blended to a maximum depth of 0.060".'

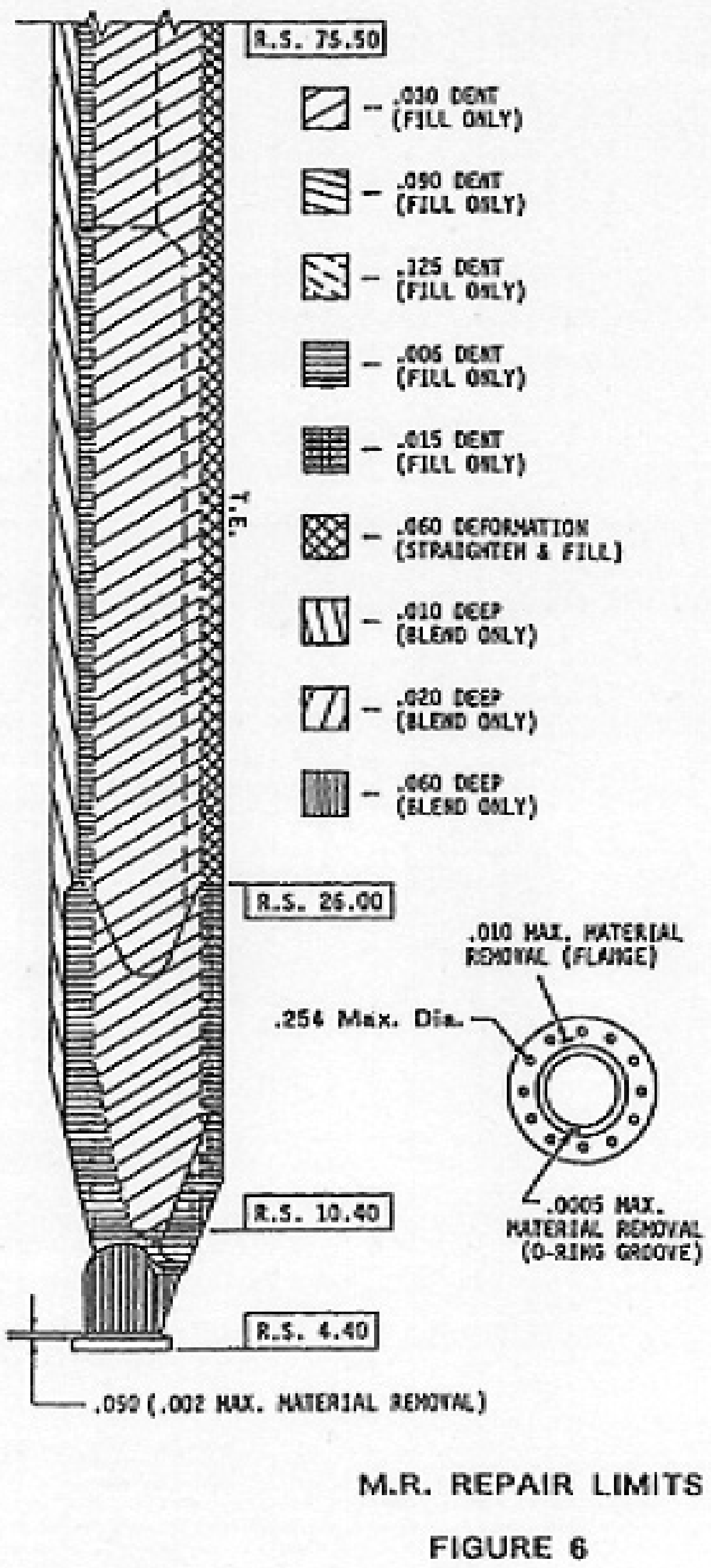


Figure 15
Reproduction of relevant part of Figure 6
from RPS 6 Rev W

The quality control document covering main blade assembly is the Assembly Order sheet, which details each of the assembly/inspection processes to be carried out in sequence order. Each document covers a batch of 10 blades as they move through the assembly shop, and records against each blade:-

- Serial N°
- The blade bond/assembly fixture used (fixture A,B, or C)
- Upon completion of each stage, the date and initials of the operative concerned; or, in the case of an inspection stage, the inspector's stamp and number.

The stages are numbered sequentially with a sequence number, at intervals of 10, ie 10, 20, 30... etc. The instructions covering the post bond trimming and clean-up stages of assembly, sequence numbers 270 through 380, were as follows:- (Note: Inspection items, ie, those covered by an inspector's stamp, are identified thus **[Insp]** in the extract below.)

'270 File trailing edge, root cut off and tip area.

280 Trim corners of skin and doubler and file flush with spar.

290 Clean two inboard bolt holes with solvent to remove all chips, etc. Install NAS1304-5 bolt using adhesive B270-1 and torque to 100 in-lbs. Inspector to check .25 dia. hole, witness torque and check bolt head gap per RPS-6. [Insp]

300 Clean up excess adhesive around trim tabs.

310 Visually inspect trim tab area prior to filler application.[Insp]

320 Fair surfaces of blade at leading edge of trim tab with filler per RPS-6.

330 Clean up excess adhesive around A934-1 and -2 doublers and along skin to spar bond joint.

340 Dimensional inspect. Tool numbers MT050-1 and TA016-2-10, Rev No/ required.[Insp]

350 Sand out all nicks scratches and sand flush any skin material up to .010 above spar, fill all dents up to .010 deep (ref. RPS-6). Inspector to verify.[Insp]

360 Prime all bond lines (ref. RPS-8). Allow to air dry until dry to the touch.

370 Install spar cover and seal bond lines using B270-9.

380 Sand the excess B270-9 adhesive and fair any low areas, pin holes, etc. with filler and spot putty.

390 Vibro-etch per RPS-27, Type 2. NOTE: AUTHORIZED PERSONNEL ONLY.

400 Prepaint inspection.[Insp]

410 Prime per RPS-8 and A002. Apply 2-3 inch wide stripe over bond joints, allow to dry, then - apply (2) full coats.'

Meteorological data

Because the helicopter was parked outside and potentially subject to blade damage caused by adverse weather conditions, historical meteorological data was obtained for all occasions when the wind exceeded 20 kt during for the three months prior to the accident. These data showed that during this period, there were several occasions when the helicopter would have been potentially exposed to gusts in excess of 50 kt, and on many occasions it would have been exposed to winds in excess of 30 kt.

Analysis

Crack location

The fatigue crack was extensive, and the blade had been extremely close to the point of catastrophic failure by the time the aircraft landed at Sywell. Whilst there had been several instances of fatigue failure of R22 main rotor blade root fittings reported prior to the G-IORG failure, in all of these the crack had developed from the bore of the inboard spar attachment bolt hole: an explicable failure mode in an area of known stress concentration. In at least one of these instances, evidence was found suggesting that the blade may have exceeded its permitted service life.

Unlike the spar bolt holes, the inboard hole in particular, the crack on G-IORG occurred in an apparently homogeneous region of the root fitting which contains no particularly abrupt spanwise changes in cross section or, indeed, any other geometric feature which could predispose the fitting to initiation of a fatigue crack at that specific spanwise location. This, together with the fact that this incident appears to have been the first instance of fatigue failure in the root fitting inboard of the spar on the R22 (which is arguably the most widely used helicopter currently in civilian service) suggests that it was likely to have been caused by some factor specific to that blade. For example, damage to, or a defect in, that particular blade, or possibly associated with some environmental factor or flight condition peculiar to G-IORG.

Fatigue fracture characteristics

Propagation period

The fatigue crack had propagated over a total distance of the order of 80 mm. The overall characteristics varied somewhat across the fracture face, but broadly fell into three regimes associated with three sequential stages of propagation: from zero to 10 mm crack length; from 10 mm to 55 mm; and from 55 mm to the final position of the crack front at approximately 80 mm fracture length.

The striation spacing in the Stage 1 regime was too fine to quantify, but if a typical spacing for initial growth is assumed, say of the order of 0.1 µm, then approximately 100,000 load cycles would have been accumulated during the first 10 mm of crack propagation. This is equivalent to approximately 190 minutes of operation at the normal 530 rpm main rotor speed if it is assumed that one load cycle occurs once per revolution of the main rotor.

The striation spacing in the Stage 2 regime was generally of the order of 1 µm. However, in some areas it was as much as 2 µm; elsewhere it was less than 1µm, and in others the spacing was too fine to discriminate accurately. Taken overall, the average spacing during this period of propagation would have been of the order of 1 µm. This corresponds to a period of approximately 85 minutes to propagate from 10 mm to 55 mm.

The Stage 3 regime is characterised by bursts of ductile rupture separated by intervals of fatigue growth. The former becomes increasingly dominant towards the crack front, with a commensurate reduction in width of the intervening bands of fatigue. Assessment of propagation time for this regime is problematic but, using a typical striation spacing within the bands of fatigue of around 1 µm, and the best estimate of the total distance of crack front movement within the bands fatigue, it is estimated that Stage 3 involved a propagation period of approximately 47 minutes: possibly shorter but almost certainly not longer. In the latter stages of this regime, the bands of fatigue separating each ductile burst would have represented typically 70 to 350 load cycles, corresponding to interruptions to the ductile bursts of between 8 seconds and 40 seconds, at 530 rpm rotor speed.

Based on the estimates for each stage outlined above, the total propagation time for the crack is estimated to have been of the order of 5 hrs 22 minutes, at 530 rpm main rotor speed. If it is assumed that a comparable period would have been required for initiation of the crack, then this would suggest a total period of growth, comprising initiation and active propagation, of approximately 10 to 11 hrs. However, it is possible that the initiation period was significantly longer and indeed may have begun immediately after the affected blade entered service. Whilst these estimates should be treated with some caution, it is nevertheless abundantly clear that the total period of propagation, ie the period during which the crack existed physically and was growing, was very small (1.4%) compared with the 747 hrs (100%) already accumulated on the blade in question. Even if the estimate of propagation period were to be doubled or even trebled, the situation would remain fundamentally unchanged.

Correlation of fracture characteristics with aircraft utilisation/significant events

A study of the number and the duration of the flights over the most recent 100 hrs of operation revealed no convincing correlation between the fracture surface characteristics and the pattern of aircraft utilisation.

The potential significance of the vibration problem identified during the annual inspection check flight was considered in some detail, particularly in light of the following:

- Its first appearance approximately 11 hours running time (includes flight time and ground running time) prior to the failure being found: a figure very close to the estimated growth period of 11 to 12 hours.
- Its failure to respond to normal tracking and balance adjustments, ultimately being cured only after a 'head-shift' had been carried out.
- The reported absence of any noticeable vibration prior to the recent annual inspection, 8.6 flying hours prior to the accident.

At a superficial level, the apparent correlation between the 11 hours of run time since the aircraft's return to service and the estimated crack growth time of 10 to 11 hours, together with the reported absence of a prior vibration problem, suggests that some event might have occurred during the annual inspection that may have set in train the initiation process. However, the characteristics of the vibration problem suggest that the crack was present at that time and that its presence was actually the cause of the vibration problems encountered during the annual inspection. The failure to identify any mechanical cause for the vibration, despite extensive investigation and the fact that a 'head-shift' was ultimately required to effect a cure, implies that a fundamental change in the balance of the rotor assembly had already taken place. This, after some 735 satisfactory flight hours, with no changes having being made to the rotor system during that time, suggests strongly that some physical change had taken place in the rotor system over a relatively short period of time immediately prior the annual inspection.

Such a crack in a blade will potentially alter its stiffness characteristics, which in turn will cause it to adopt a different position under load, when compared with the pre-cracked state and/or under static conditions, causing a rotor imbalance when the rotor is running .

From the available evidence, there is little doubt that the vibration identified during the annual inspection resulted from a shift in the centre of mass of the rotor system away from the axis of the rotor mast. This was caused by the root fitting fatigue crack at an early stage in its development. The 'head shift', which apparently cured the vibration, simply moved the centre of mass of the rotor system sufficiently far to cancel its effect, but only temporarily. The vibration reappeared some five flight hours later after further propagation of the crack had resulted in a further shift in the centre of mass under load. It is probable that the vibration was not noticed prior to the annual inspection because the pilot flew the aircraft regularly and consequently did not notice the gradual deterioration, whereas the annual check pilot, who came to the aircraft afresh, detected it immediately.

Potential contributory/causal factors

No external damage was visible and no repairs or re-working of any kind had been carried out on the cracked blade since it had been manufactured. The material properties of the fitting were consistent with those specified and no evidence was found of any metallurgical defects, within the section itself, which could explain the crack. However, significant defects were found on the surface of the fitting proper, on the leading edge face at the crack location and also immediately adjacent to it. This damage was hidden beneath layers of filler and finish-paint applied during the latter stages of blade manufacture; consequently, there was no doubt that these defects had been introduced during manufacture.

Surface manufacturing defects in the cracked blade from G-IORG

The surface damage comprised two transverse bands of surface abrasion across the leading edge face, extending partially around the upper and lower edge corners. These would have created geometric stress concentrations due to:-

i) The surface scoring, at a microscopic level.

ii) The change in surface topography caused by the abrasive excavation of material, forming irregularly shaped valleys. The stress concentrations caused by these excavations will have been superimposed on top of stress concentrations already present in the fitting, at the top and bottom leading edge corners, inherent from the shape of the fitting, the corner profile in particular.

Figure 16 illustrates how the excavated valleys (red boundary) intersect the manufactured chamfer (yellow boundary) to form a particularly complex 3D surface geometry in the crack initiation region. It also shows that the origin of the fatigue fracture lies very close to the point where two separate regions of surface excavation, labelled A & B in Figure 16, conjoin.

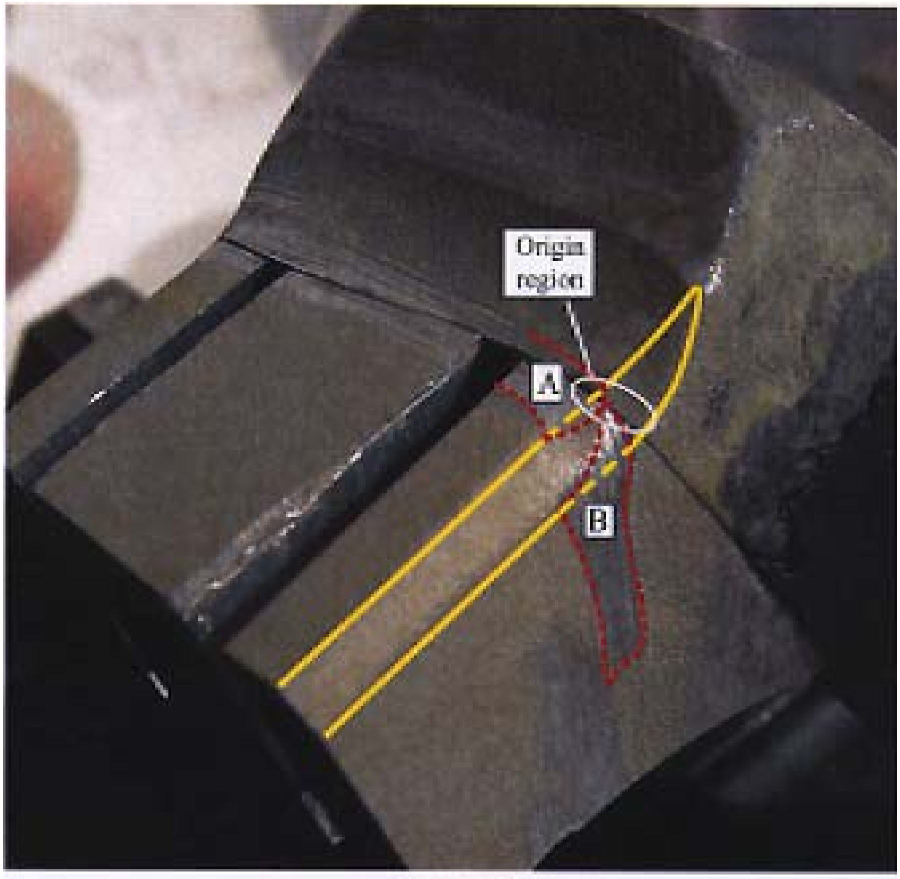


Figure 16
Modifications to the surface topography in the origin region, due to the abrasion damage

In this same area, deep score lines, running around the chamfer, create a series of small but distinct steps, forming a terrace-like surface which further adds to the complexity of the macro surface topography in the region of the crack origin.

Whilst the complex surface geometry thus created was undoubtedly a significant factor contributing to fatigue initiation, and indeed may have been the primary factor, it was not the only factor and, by itself, may not have been sufficient to cause initiation. Other factors of potential importance include:

- The position of the resulting stress concentration in relation to the background working stress level in the region.
- Ablation of the beneficial shot peened surface layer in the affected region.

It is beyond the scope of this report to attempt to quantify the contributions made by each of these factors to the initiation process, but a study of the way these factors apparently combined together in the cracked fitting from G-IORG, compared with the various sample blades, allows inferences to be drawn concerning their relative significance.

In broad terms, all ten of the sample R22 root fittings examined by the AAIB, as part of this investigation (including the intact blade from G-IORG), exhibited abrasion damage broadly similar to, and in the same general area, as the cracked fitting. These comprised, to a greater or lesser degree, the following:-

- transverse scoring
- surface excavation
- ablation of the shot peened layer

Based on these samples, it appears highly probable that a majority of R22 root fittings will have similar defects.

None of the samples showed any evidence of crack initiation and, whilst the number of flight hours accumulated by some them is unknown, a significant number had been taken out of service because they had attained their service life. The fact that none of the samples was cracked, and that no prior instance of cracking in this location had occurred previously despite the probability that a majority of R22 blades will have one or more defects similar to the cracked (G-IORG) and sample blades, suggests that the mere presence of these factors per se did not initiate the crack. Rather, it suggests that the critical factor was the manner in which they came together, ie that on the cracked blade, they combined together in a particularly damaging way, as follows:

i) The combination of excavated volume, added to surface scoring and the position of these features in relation to the existing geometry (chamfer, etc), gave rise to a particularly complex surface topography, creating correspondingly large stress concentrations.

ii) The most significant stress concentration occurred at a critical spanwise position, ie, just inboard of the spar, where the cross-sectional form of the fitting did not benefit from the increase in section-area as the fitting changes shape from an aerofoil to a hemispherical form. Figure 17 shows the nominal mean stress profile, based on CF tensile loading, acting upon sectional area-changes across the affected region. From this it can be seen that the stress concentration produced around the conjoined/stepped region is located outboard of the region where the nominal background stress starts to decrease due to the increase in sectional area.

iii) A thinner than normal overall shot peened layer on the leading edge, and the consequent probability that the abrasion process would remove the peened layer totally in the areas affected, thus depriving the high stress region of the mitigating effect of a fatigue suppressing compressive surface layer.

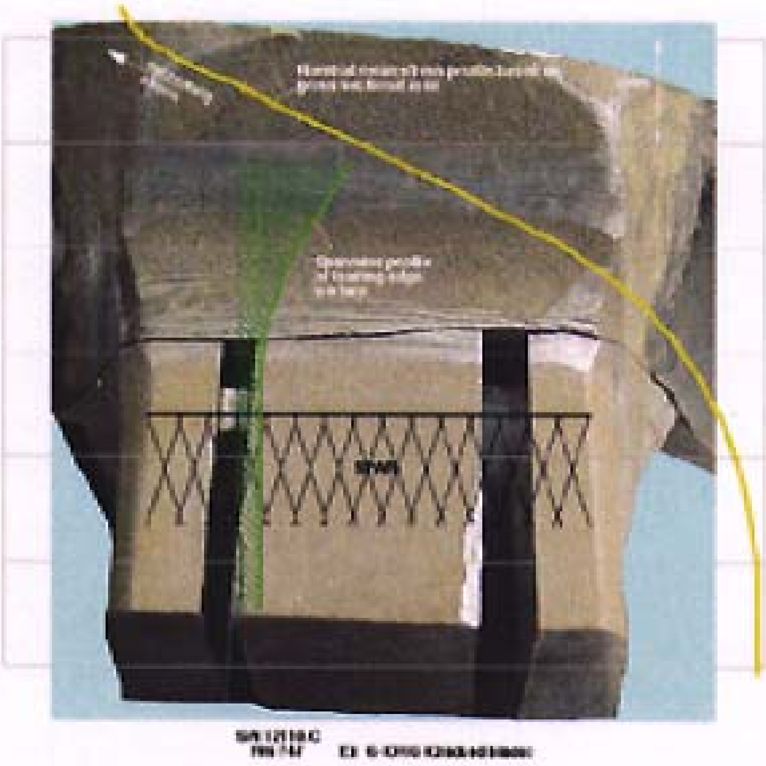


Figure 17
Correlation between surface damage and mean stress profile (yellow) and surface profile (green)

The contribution to crack initiation caused by the removal of the shot peened layer is particularly difficult to assess quantitatively, but it is believed to be significant that some 67% of the nine sample blade roots, ie, not including the two blades from G-IORG, exhibited an undisturbed shot peened layer which visibly extended to a depth in excess of 0.003"; the remaining 33% of the samples were in the range 0.002" to 0.003" deep. In contrast, the visible shot peened layer depth on both the cracked blade and the intact opposing blade from G-IORG was in the range 0.001" to 0.002". It is believed to be significant that both of the fittings from G-IORG, which the manufacturer's records show were from the same batch, had significantly thinner original peened layers than the norm. This would have made these fittings, potentially, more vulnerable to total ablation of the shot peened layer during the post-bond clean up process and, thus, more vulnerable to fatigue crack initiation than the population of blades at large.

Potential in-flight events

So far as can be determined, the cracked blade had been operated normally for approaching 750 hrs, some 30% of its permitted life, before fatigue initiation began. This raised the possibility that some notable event in the aircraft's recent history might have triggered fatigue initiation. For example:

i) An abnormally high flight manoeuvre-load

ii) Some event causing direct physical changes in the blade sufficient to predispose it to fatigue cracking thereafter

iii) Some change to the aircraft, remote from the blade, which altered the loading environment experienced by the blade.

No evidence was found to suggest that the aircraft had been subjected to any abnormal or unusually vigorous manoeuvring, or any change in its normal pattern of flight, during its recent history. Indeed, in order to provide a stable photographic platform for the role in which the aircraft was used most of the time, it had to be flown in a steady and predictable manner. Steep turns, abrupt pitching or rolling manoeuvres, or significant or prolonged side slipping, were neither required nor reportedly used.

The possibility that the blade had been subject to an overspeed, sufficient to raise the centrifugal loading significantly above normal levels, was considered in some detail. This was done so particularly in light of the fine secondary cracks noted in the paint and filler adjoining the main fracture, which could have been associated with such an event. The main rotor speed governor switch on G-IORG, located on the end of the collective lever, was protected by a cylindrical shroud, which would have reduced the likelihood of an inadvertent powered overspeed during run-up on the ground should the governor switch have been knocked off accidentally. The documentation covering the recent annual inspection records re-attachment of this shroud. This raises a question as to whether it might have been detached prior to the annual inspection, thereby making a possible powered overspeed potentially more likely. However, there was no entry in the technical log reporting an overspeed, and no evidence was found to suggest that such an event had occurred. Unlike many R22 helicopters, G-IORG was not was not used routinely for pilot training and, consequently, would not have been exposed regularly to the risk of a rotor overspeed during autorotation. So far as could be established, no autorotations had been carried out during the period of relevance. Finally, examination of the spindle bearings revealed none of the characteristic damage symptomatic of significant main rotor overspeed on the R22.

In summary, no evidence was found to suggest that any abnormal event during flight had been a factor in the crack initiation.

Potential blade damage events

The possibility of an engine start with the clutch engaged was considered as a potential damaging event, particularly as that the clutch operating switch was found to be intermittent and replaced during the annual inspection. Such an event could induce potentially damaging transient loads in the main rotor blades, in-plane, as the engine fires up. However, G-IORG was fitted with an interlock system to prevent engine starter engagement with the clutch engaged. This uses the clutch transport system microswitches to sense when the clutch is engaged and should have inhibited starter operation, regardless of how the clutch came to be engaged, or the condition of the clutch switch per se. On the available evidence, therefore, an engine start with the clutch engaged is not likely to have been a factor.

Maladjustment of the drive belts, leaving the belts excessively tight, could potentially generate sufficient drag to turn the main rotor with the clutch mechanism mechanically disengaged, ie, with the belts nominally loosened. Such a condition is more likely immediately after the installation of new belts, before they have had a chance to stretch and properly 'bed in'. In fact, as the drive belts were found to be worn and were replaced during the annual inspection, it is therefore unlikely that they would have an excessively tight condition during the critical period prior to the inspection. Since, for the reasons given earlier in this analysis, the fatigue crack was almost certainly present prior to the annual inspection, ie whilst the belts were old and in a relatively stretched state, such an event is not likely to have occurred during the period of interest.

A sudden stoppage of the main rotor was also considered resulting from, for example, a blade striking an obstruction. Because the very strong stainless steel 'D' section spar forms the leading edge of the blade, it was considered possible for significant blade strikes to have occurred without leaving visible evidence. However, the exchange of momentum in this circumstance will occur predominantly at the impact site on the blade, and a strike sufficient to induce damaging loads at in the blade root area would likely require the blade to be driven under considerable power at the time of arrest. Such an event is unlikely to pass unnoticed, or unrecorded, and would almost certainly result in damage elsewhere on the blade. The absence of such damage, therefore, tends to rule out such a blade strike having occurred. The R22 rotor brake is not sufficiently powerful to induce a sudden stoppage and, in any event, would tend to load the trailing edges of the blades in tension and not the leading edge regions.

The meteorological record suggested that during the period between the recent inspection and the time of the incident there were several occasions on which the helicopter would have been exposed to gusts in excess of 50 kt, and many occasions on which winds were in excess of 30 kt. Whilst it was not possible to establish, in any meaningful way, the extent to which the aircraft may have been vulnerable to damage from high winds whilst parked, notwithstanding the fact that the blades were reportedly always tied down on such occasions, there was certainly no evidence to suggest that the blade did suffer any wind damage. Inquiries made as to the aircraft's location, during the period it was at the maintenance facility undergoing the annual inspection, were similarly inconclusive but the maintenance company's policy is to not leave R22 aircraft parked outside overnight. So far as can be established, the aircraft was brought in shortly after its arrival for the annual inspection in the early evening of 19 April 2002, and remained inside the hanger thereafter except for its brief excursions outside for flight testing.

In summary, there is no evidence to suggest that the blade suffered any physical damage in the critical period leading up to the annual inspection.

'Secondary effects' as a potential cause of fatigue initiation

Any engine, transmission, or airframe condition potentially capable of inducing secondary loading of the blade sufficient to trigger the fatigue initiation would almost certainly have been detected during the annual inspection, and recorded. The absence of such evidence therefore tends to rule out any defect occurring remote from the main rotor as a causal factor.

Interactions between the main rotor blades and turbulence from the window in the non-standard passenger's door, which was usually open during filming, was considered as a possible cause of abnormal blade loading. However, any such effects, if present, would have affected both blades equally and, on balance, was considered unlikely to have been a factor.

Manufacturing quality control

During the bond-cure process, excess adhesive squeezes out in significant quantities from the various joint lines between components. Prior to this investigation, this excess was removed by use of a die grinder fitted with a small grit-loaded rubber wheel. Figure 18 shows a typical blade prior to removal of the excess adhesive, with the areas of the leading edge damaged during grinding operations to remove this excess adhesive identified.

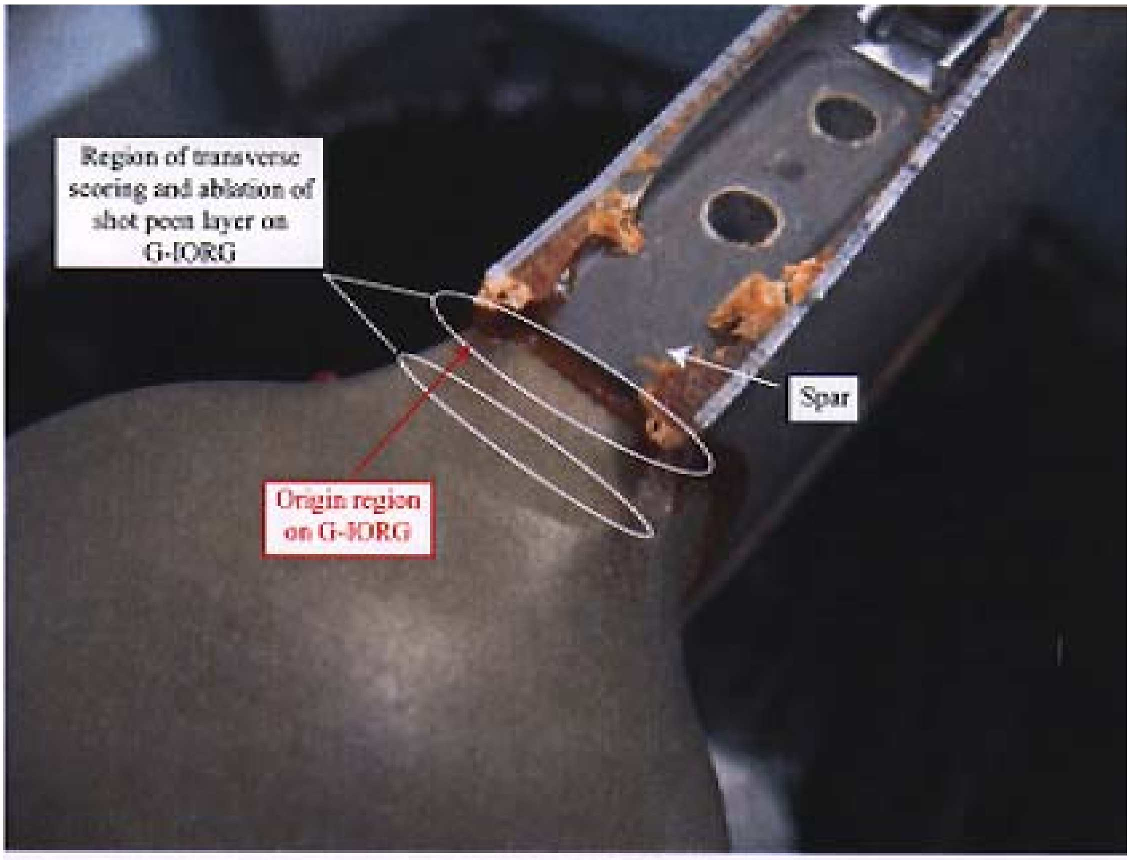


Figure 18
Blade root at inboard end of spar,
before clean-up of excess adhesive and final dressing of blade

Whilst considerable effort had clearly been invested in maintenance of quality control of the bond process, relatively little consideration appears to have been given to controlling the quality of surface finish. The lack of any positive form of control over the use of power grinders on the root fitting, especially for operations which were essentially of a cosmetic nature, was inappropriate, given that this is a stress-critical area where both surface finish and macro geometry are critically important factors in relation to fatigue sensitivity. This shortcoming was compounded by:

- The lack of any specific inspection addressing surface condition of the root fitting following the clean-up operation
- The subsequent application of filler and paint which masks any damage.

The significance of both a good surface finish and the integrity of the shot peened layer was evidently well understood by design staff but, for reasons which remain unclear, were not applied to the blade assembly processes. Not only was the coarse use of grinders indicative of relative ignorance of the importance of a shot peened surface on the part of shop floor staff, but the specified processes do not adequately address the issue, given that material removal up to a depth of 0.060" was permitted (with a minimum radius of 0.1"). This would totally remove any shot peened layer, the depth of which, at most, would be unlikely to exceed 0.005".

Another matter, unrelated to the surface finish, but raising further questions about quality control of the manufacturing process, is the manufacturer's specification document RPS-63, covering shot peen processes, which specifies a maximum temperature of 250°F for aluminium alloy materials which have been peened. This conflicts with the blade bonding process sheet, which requires the bond-cure fixtures to have *'root fitting and tip zone controls set to the temperature required to maintain a temperature range of 250° to 300°F for the duration of the cure cycle'.* (Since this finding, the manufacturer has revised RPS-63 to allow the heating of shot-peened aluminium alloys to 300 F to allow for bonding of shot-peened parts.)

Finally, it would appear that the surface finish issues have existed for some considerable time. Examples of blade root fittings produced solely for use in the R22 main rotor blade fatigue test programme, and which were not therefore subject to subsequent cosmetic filler and paint processes, also exhibited the characteristic grinding damage seen on the sample blades examined and those from G-IORG. Arguably, a possible benefit of this is that this surface damage would have been accounted for to some extent in the fatigue test program, albeit in an uncontrolled, and therefore unquantifiable, manner.

Precursor indications of cracking

A fatigue crack in a main rotor blade, which has grown to the point where it has materially affected blade stiffness, is likely to cause a change in the position of the cracked blade's centre of mass under rotor-running conditions. This is due to abnormal blade flexure and will result in a main rotor vibration. If this vibration is cured solely through normal re-balancing procedures, the vibration is likely to reappear after a relatively short time. This results from further growth of the crack, associated further changes in blade stiffness and the consequent change of position of the centre of mass under loaded conditions. For this reason, any significant main rotor vibration for which no direct cause can be identified, and which reappears shortly after being successfully tuned out by conventional balancing operations, should be viewed, potentially, as a symptom of a fatigue crack in one of the main rotor blades, and appropriate steps taken to investigate further. These comments apply particularly if head shift operations are required to restore balance.

Pilot actions

The increasing vibration level in G-IORG during the final positioning flight to Sywell was assessed by the pilot as being not severe enough to warrant an immediate landing. It is possible that the pilot's previous, and extensive, experience of flying Hiller helicopters in crop spraying operations may have coloured this judgement as to what was an acceptable level of vibration in a R22 helicopter. It is understood that, with the benefit of hindsight and having seen the cracked blade after landing, the pilot did concede that his judgement in the matter had been questionable. Pilots of all rotorcraft should be acutely aware of the potentially catastrophic rate of propagation of fatigue cracks should they develop in a main rotor blade. Once a crack has progressed to the extent that a discernible vibration results, the rate of crack growth is likely to increase dramatically, especially so during the final stages of growth. This is when the vibration produced in the main rotor system will feed back into the cracked blade, raising still further the cyclic stresses which drive the crack. In this circumstance, there will be a significantly risk of catastrophic blade failure occurring within a time frame of minutes, or possibly even seconds. In any situation involving a severe, or indeed any perceptible escalating main rotor vibration, pilots should be advised to interpret these symptoms as indicative of imminent blade failure and land immediately, or as soon as possible should an 'immediate' landing be likely to result in an accident.

Airworthiness

Exchange of information

Throughout the investigation process the aircraft manufacturer, and relevant airworthiness and accident investigation authorities in the USA and UK, were kept informed of all developments in the investigation relating to continued airworthiness of R22 aircraft. They were advised of the AAIB concerns regarding specific airworthiness issues and the following summarises the flow of information from the AAIB.

On the 25 May 2002, the manufacturer was advised of the apparent absence of any identifiable shot peen layer in the origin region of the fatigue crack, and also the presence of transverse scoring in this same area. It was suggested that the manufacturer may wish to investigate whether there had been any change in the personnel involved in the adhesive clean-up process, or any changes in the process used, which might imply a blade batch problem.

At a meeting held at the manufacturer's facility on the 16 June, 2002, attended also by the FAA, attention was drawn to the apparently reduced thickness of the original peen layer beneath the spar compared with that seen on the only other sample examined at that time. The failed section of the root fitting was made available to the manufacturer on extended loan, to facilitate their own investigations. During this meeting, the manufacturer undertook the following actions related to continued airworthiness of the R22 helicopter:-

1. Take appropriate steps as a matter of urgency to inform operators of the potential significance of rapid loss of rotor balance, and the implications of employing head shift operations to effect a cure.

2. Investigate the history of blades undergoing the shot peen process at the manufacturer's subcontractor, and attempt to identify those blades assembled with root fittings processed in the same batch as that of the failed blade.

3. Ensure that the shot peen process is being carried out correctly on blades currently in production.

4. Review and amend the blade clean-up process during manufacture to ensure that the shot peen layer is not compromised by removal of surface material, and that the surface finish is of an acceptable standard.

5. Educate production staff on the importance of surface finish and related issues.

6. Implement inspection procedures, post bond-cure, to check for inappropriate surface attrition and quality of surface finish, prior to any filling and painting operations.

7. Examine samples of blade root fittings from in-service and/or time-expired/damaged blades to extend understanding of the character of, and incidence of, shot peen layer and surface finish discrepancies.

8. Review the options for replacement of blades currently in service should this prove necessary.

On 27 June 2002, a Position Paper was drafted by the AAIB and circulated to the aircraft manufacturer, the FAA, the NTSB, and the UK CAA, outlining the progress of the investigation to date and identifying areas of concern. These included the following provisional assessments relating to cause, and to the issue of continued airworthiness:-

- The fatigue crack initiation was likely to be due to the combined effect of the removal of the shot peen layer and the stress concentration arising from surface scoring at the leading edge.

- It was likely that a significant number of blades in service would contain transverse scores of similar to those seen on the cracked blade.

- The (apparently) unusually thin peen layer, and the consequently higher probability of this layer being removed during the adhesive clean-up process compared with root fittings having a thicker peen layer, could offer a plausible explanation for the absence of prior instances of cracking at this particular location.

- The apparently widespread incidence of ageing cracks in the filler, at the leading edge just inboard of the spar, would tend to reduce any concern which might otherwise be shown by operators regarding any cracking in this area.

- Any attempt to remove the filler to facilitate inspection could potentially result in further scoring of the surface, which could possibly lead to the initiation of fatigue cracks.
- The apparently short time interval between crack initiation and blade failure does not give confidence that a normal inspection regime would detect the embryonic cracks in adequate time.

The AAIB Position Paper also expressed the view that:

i) Priority airworthiness action should be taken to ensure that operators are made aware of the potential significance of sudden changes in main rotor blade balance, and that operators should regard any blades requiring rebalancing (ie repeat balancing after only a short period of operation) as potentially being cracked.

ii) Steps should be taken urgently to identify all those blades built from root forgings sent for shot peen treatment at or around the same time as the cracked blade (S/N° 12110C), and that these blades should be located and removed from service for inspection.

Safety action taken

On 25 June 2002, the manufacturer issued a R22 Safety Alert, stating:-

'UNUSUAL VIBRATION CAN INDICATE A MAIN ROTOR BLADE CRACK

A catastrophic rotor blade fatigue failure can be averted if pilots and mechanics are alert to early indications of a fatigue crack. Although a crack may be internal to blade structure and not visible, it will likely cause a significant increase in rotor vibration several flight hours prior to final failure. If a rotor is smooth after balancing but then goes out of balance again within a few flights it should be considered suspect. Rapidly increasing vibration indicates imminent failure and requires immediate action.

IF MAIN ROTOR VIBRATION INCREASES RAPIDLY OR BECOMES SEVERE DURING A FLIGHT, LAND IMMEDIATELY.

Do not attempt to continue flight to a convenient destination. Have the rotor system thoroughly examined by a qualified mechanic before further flight. If mechanic is not sure whether a crack exists, contact RHC.'

Safety Recommendations

Notwithstanding the fact that the manufacturer has undertaken to implement a series of actions aimed at identifying in service main rotor blades at possible risk of developing fatigue cracks at a similar location to that on the blade from G-IORG, and to minimise the possibility of damage during manufacture of the blade root fitting, the following recommendations are made:

Safety Recommendation 2003-78

It is recommended that the FAA, as the Primary Certificating Authority for the R22 helicopter, require the manufacturer of the R22 helicopter to establish an inspection procedure capable of identifying blades containing cracks originating in the main rotor blade root fitting leading edge region.

Safety Recommendation 2003-79

It is recommended that the FAA require the manufacturer of the R22 helicopter to devise an inspection method which will identify, on in-service blades, the type of root fitting surface abrasion damage found on both a cracked blade and several non-cracked sample blades, that is potentially capable of initiating fatigue cracking. (In devising an appropriate inspection method, due consideration should be given to the beneficial influence of the shot peen layer on the surface of the blade root fitting, and appropriate steps taken to ensure that any procedures used to remove the filler and adhesive layers and expose the metal beneath do not compromise the integrity of the peened layer.)

Safety Recommendation 2003-80

It is recommended that the FAA confirm that the manufacturer of the R22 helicopter has adjusted their manufacturing processes of the main rotor blade, since the discovery of a large crack on an in-service main rotor blade, to preclude abrasion damage of the shot peened surface treatment during the adhesive clean-up process, and ensure that the depth of the shot peened layer on the blade root fitting conforms to the manufacturer's specification.

Conclusions

After landing from a flight during which an increasing level of vibration was experienced, an 80 mm long crack was found in one of the main rotor blade root fittings. The crack was caused by a fatigue process, which had initiated at multiple origins at the lower leading edge corner of this forged aluminium root fitting. No prior instances of fatigue cracking have been known to occur at this location. Crack initiation was attributed to a particularly adverse combination of factors including transverse scoring and significant local erosion of material, including the shot peened layer, from the surface of the forging. This had been caused by hand grinding operations to remove excess adhesive during manufacture. This damage had been superimposed onto an existing leading edge chamfer on the fitting, which created additional stress concentrations. The significance of damaging the surface was apparently not understood by the operatives involved, nor was it detected by inspection before being hidden from view by filler and finish paint layers during subsequent stages of manufacture. The crack initiation period could not be estimated with any confidence but could, as an absolute minimum, have been 5 hrs but equally could have comprised most of the approximately 740 hrs running time accumulated by the blade. The period of crack propagation, however, was estimated to have been approximately 5.3 hrs, based on one load cycle occurring once per revolution of the main rotor.