Australian Government

Australian Transport Safety Bureau

AVIATION SAFETY INVESTIGATION REPORT

200302820

Robinson Helicopter Company

Model R22 Mariner

Bents Basin State Recreation Area, NSW

20 June 2003

Released under the provisions of Section 19CU of Part 2A of the *Air Navigation Act 1920*.

Published by:	Australian Transport Safety Bureau
Postal address:	PO Box 967, Civic Square ACT 2608
Office location:	15 Mort Street, Canberra City, Australian Capital Territory
Telephone:	1800 621 372; from overseas + 61 2 6274 6590
	Accident and serious incident notification: 1800 011 034 (24 hours)
Facsimile:	02 6274 6474; from overseas + 61 2 6274 6474
E-mail:	atsbinfo@atsb.gov.au
Internet:	www.atsb.gov.au

CONTENTS

INTRODUCTION

The Australian Transport Safety Bureau (ATSB) is an operationally independent multi-modal Bureau within the Commonwealth Department of Transport and Regional Services. ATSB investigations are independent of regulatory, operator or other external bodies.

In terms of aviation, the ATSB is responsible for investigating accidents, serious incidents, incidents and safety deficiencies involving civil aircraft operations in Australia, as well as participating in overseas investigations of accidents and serious incidents involving Australian registered aircraft. The ATSB also conducts investigations and studies of the aviation system to identify underlying factors and trends that have the potential to adversely affect safety. A primary concern is the safety of commercial air transport, with particular regard to fare-paying passenger operations.

At the time of the accident the ATSB was conducting aviation investigations in accordance with the provisions of the *Air Navigation Act 1920*, Part 2A. Section 19CA of the Act states that the object of an investigation is to determine the circumstances surrounding any accident, serious incident, incident or safety deficiency to prevent the occurrence of other similar events. The results of these determinations form the basis for safety recommendations and advisory notices, statistical analyses, research, safety studies and ultimately accident prevention programs. As with equivalent overseas organisations, the ATSB has no power to implement its recommendations.

It is not the object of an investigation to determine blame or liability. However, it should be recognised that an investigation report must include factual material of sufficient weight to support the analysis and conclusions reached. That material will at times contain information reflecting on the performance of individuals and organisations, and how their actions may have contributed to the outcomes of the matter under investigation. At all times the ATSB endeavours to balance the use of material that could imply adverse comment, with the need to properly explain what happened, and why, in a fair and unbiased manner.

The 24-hour clock is used in this report to describe the NSW local time of day, Eastern Standard Time (EST), as particular events occurred. Eastern Standard Time was Coordinated Universal Time (UTC) + 10 hours.

EXECUTIVE SUMMARY

On 20 June 2003 at approximately 0840, a Robinson Helicopter Company Model R22 helicopter, registered VH-OHA (OHA), was being used to conduct flying training in the Bankstown training area with an experienced flight instructor and student pilot. The helicopter was observed and heard flying in a normal manner. Witnesses reported subsequently hearing a number of loud bangs and one witness observed what appeared to be a main rotor blade separating from the helicopter. The helicopter descended to the ground in an inverted attitude and both occupants were fatally injured.

Examination of the accident site and helicopter wreckage confirmed that one main rotor blade had failed in-flight. Examination of the helicopter and its systems did not reveal any other abnormality that would have contributed to the loss of the main rotor blade.

The helicopter had recently re-entered service following maintenance which included the fitting of an overhauled engine and the completion of a 100-hourly inspection. The helicopter also underwent maintenance action to rectify a main rotor blade vibration. This maintenance action involved a number of experienced R22 helicopter engineers being consulted about the possible reasons for the main rotor blade vibration. Rectification action was completed in accordance with normal maintenance practices and the manufacturer's maintenance manual. Subsequent examination of the maintenance manual for the R22 helicopter revealed that it did not contain any information in the tracking and balancing section that indicated that a vibration may be the result of a crack in the main rotor blade. The manufacturer had produced other documentation containing this information, but these documents did not formally form part of the maintenance manual.

The helicopter had been manufactured in 1991 and had been imported into Australia in 1996. In the time prior to the accident it had been owned and operated by a number of organisations and individuals, and was operated both commercially and privately.

Following the accident, industry suggestions about the possible under-recording of time in service on the helicopter led the Australian Transport Safety Bureau to concentrate part of the investigation to the recording of time in service of the helicopter. Coincident with this investigation, a seperate investigation of the recording of time in service on the helicopter was conducted by the Australian Civil Aviation Safety Authority (CASA).

Both investigations examined a wide range of documentation and records from numerous sources. The conclusion of both investigations was that the helicopter had not exceeded the mandatory time in service life of 2,200 hours, nor had it exceeded the mandatory calendar time in service life of 12 years. The final time in service of the helicopter was calculated to be 2,055.6 hours and the calendar time in service was 11 years and 8 months.

An examination of the main rotor blade in the ATSB laboratories revealed that it had failed as a result of fatigue crack growth in the blade root fitting at rotor station 10.35. The fatigue crack initiated as a result of localised pitting corrosion in the counterbore of the inboard bolthole. The examination also revealed that while the fatigue failure was in a similar position to two previous main rotor blade failure accidents in Australia, in OHA's case, there was an area of adhesive disbonding between the main rotor blade skin and blade root fitting. This adhesive disbonding meant that the crack in the blade root fitting

did not propagate into the blade skins and so was undetectable using visual means. The two previous failures were linked to under-recording of hours.

The material failure analysis found that the disbonding present on the failed main rotor blade was also present in a number of other main rotor blades that were examined. As a result, the ATSB issued a safety recommendation to the United States Federal Aviation Administration (FAA) and to the Robinson Helicopter Company, seeking that they conduct further testing on main rotor blade root fittings to evaluate the extent of adhesive disbonding in the blade root fitting. This examination was conducted on a total of 51 main rotor blades that had between zero and 2,200 hours time in service. Results of the examination revealed that adhesive disbonding between the spar and root fitting was present in all blades and that the extent of the disbonding was variable.

Subsequent to this accident there was another in-flight failure of a main rotor blade. In February 2004, an R22 helicopter being operated in Israel sustained an in-flight failure of a main rotor blade. This blade had failed as a result of fatigue in the same location as the failure in the Australian accident. The Israeli failure exhibited a similar loss of adhesion and corrosion. Both blades had failed before their mandatory time in service retirement lives and represented a failure of the fatigue fracture control plan. A third failure occurred in New Zealand in November 2004. Preliminary investigations have revealed that the failure may be the result of loadings on the blade that may have exceeded those intended by the manufacturer. The investigation of that accident is continuing.

The manufacturer has issued a safety letter and a service bulletin relating to revised retirement lives for main rotor blades, and has introduced a redesigned main rotor blade into service. The manufacturer indicated that it intends to publish safety alerts and notices on its Internet website as an additional means of bringing safety related information to the notice of owners, operators and maintenance organisations.

The R22 maintenance manual has also been amended by the manufacturer as a result of this investigation. The main rotor blade tracking and balancing section now contains information, which alerts maintenance personnel to the fact that a main rotor blade vibration may be the result of a developing crack.

Safety action taken by the CASA as a result of this accident was to amend an existing airworthiness directive to take into account the findings from the examination of the blade and to introduce additional amendments to the directive, when updated information became available from the manufacturer. They also introduced a discussion paper on the installation of mandatory time in service recorders for helicopters. As at October 2005, CASA was still evaluating the public comments on the discussion paper.

In addition, CASA has drafted a Notice of Proposed Rulemaking (NPRM 0503CS) in which it is proposed to require the retirement of similar main rotor blades by 1 March 2006 on Australian registered Robinson R22 helicopters.

The United States FAA issued a special airworthiness information bulletin and an emergency airworthiness directive.

As a result of several accidents involving main rotor blade failures, the European Aviation Safety Agency, issued an airworthiness directive on 5 July 2005 mandating compliance with the Robinson service bulletin.

The ATSB has contracted research to assess the validity of the usage spectrum assumptions that were used for certification of the Robinson R22 helicopter. A research investigation report on the project is planned to be released in 2006.

1 FACTUAL INFORMATION

1.1 History of the flight

At about 0840 on 20 June 2003, a Robinson Helicopter Company Model R22 helicopter (R22), registered VH-OHA, was being operated to conduct dual flying training in the Bankstown training area. An experienced flight instructor was conducting a student pilot's second flight lesson. The planned lesson was to demonstrate and practise climbing and descending manoeuvres.

A person located to the south of the helicopter's flight path reported seeing and hearing the helicopter flying normally. He reported that, shortly after, there was a sound similar to a car backfiring, followed almost immediately by a louder bang. He also reported seeing what appeared to be a main rotor blade separating from the helicopter, before losing sight of the helicopter as it descended behind trees.

Another person located to the west of the helicopter's flight path also reported seeing and hearing the helicopter flying normally towards her. She reported hearing a loud bang about one minute later. She stated that she then saw the helicopter; that it appeared to be inverted, and that there were items falling from it. The witness then lost sight of the helicopter as it descended behind trees.

The helicopter descended and impacted the ground in an inverted attitude. Both occupants were fatally injured (see Figure 1).

Figure 1: Accident site

The helicopter had recently returned to service following maintenance during which an overhauled engine was fitted, a 100-hourly inspection was completed, and a main rotor blade vibration had been corrected.

1.2 Injuries to persons

Injuries	Crew	Passengers	Others	Total
Fatal	2	-	-	2
Serious	-	-	-	-
Minor	-	-	-	-
None	-	-	-	-

1.3 Damage to the helicopter

The helicopter was destroyed by in-flight break-up and impact forces. Subsequent examination and analysis revealed that the 'blue'[1] main rotor blade, serial number 6249A, had failed in flight (see Figure 2). That failure is further addressed at section 1.16.

Figure 2: Failed 'blue' main rotor blade

1.4 Other damage

Nil.

[1] For ease of identification during maintenance tasks, the helicopter's flight controls have colour-coded marks on the components positioned above the rotating swashplate. The colours used on this helicopter were 'red' (the intact main rotor blade) and 'blue' (the failed main rotor blade).

1.5 Personnel information

The flight instructor held a valid commercial pilot (helicopter) licence, a grade one flight instructor rating, and was endorsed to fly the R22 helicopter. He had recorded in excess of 3,800 hours total time in helicopters, with in excess of 2,700 hours in the R22 type. He held a valid class one medical certificate, was reported to be fit and well rested, and was flying within flight and duty time limitations.

The student was conducting his second flight lesson and held a valid student licence and a valid class one medical certificate. He was reported to be fit and well rested prior to commencing the flight.

1.6 Helicopter information

Manufacturer	Robinson Helicopter Company
Model	R22 Mariner
Serial number	1962M
Registration	VH-OHA
Month and Year of manufacture	October 1991
Maintenance release (A12287)	Valid to 2086.2 hrs or 1 October 2003
Total time in service	1986.2[2]

The helicopter was operating within weight and balance limitations at the time of the accident.

1.6.1 Recent maintenance carried out on the Helicopter

The helicopter was taken by its current owner to a maintenance facility in April 2003 for rectification of a reported rapid onset of a main rotor blade vibration (see section 1.6.2 for additional information on main rotor blade tracking and balancing). The engineer who initially attempted to track and balance the main rotor blade contacted the owner and indicated that the problem was not minor and that the helicopter would require some time in maintenance to rectify the problem. As the maintenance release was due to expire in 3 days, the owner arranged for a 100-hourly inspection to be carried out. The owner indicated that he wished to apply to the Civil Aviation Safety Authority (CASA) for an extension of the operating time on the engine from 2,000 to 2,200 hours time in service. The inspection of the engine revealed that it was not suitable for a life extension, so it was removed for overhaul. As a result, the helicopter remained in maintenance for the next 2 months while the engine was being overhauled.

Following overhaul, the engine was reinstalled in the helicopter and the maintenance effort returned to the remainder of the 100-hourly inspection, and the main rotor blade vibration problem. All airworthiness directives applicable to OHA had been complied with at the time of the accident.

[2] Total time in service was recorded on the helicopter maintenance release and was the figure at the conclusion of the last flight (see section 1.6.6 for more information on total time in service).

1.6.2 Background information on main rotor blade balancing

Vibration readings are obtained using specialised vibration analysis equipment, both on the ground and in flight. Readings are displayed as clock angles from one o'clock to twelve o'clock (or degrees) and in inches per second (IPS) velocity. The helicopter manufacturer's maintenance manual contains a procedure for tracking and balancing the blades when a helicopter main rotor blade develops a vibration considered to be above normal levels.

For Robinson R22 helicopters, vibration readings obtained can be plotted on the *Robinson Helicopter Maintenance Manual – Main Rotor Track and Balance Chart.* That chart provides engineers with span-wise or chord-wise solutions to correct the recorded imbalance. Such changes on the R22 include span-wise blade tip weight changes, chord-wise head shift movements or chord arm weight changes. The *Robinson Helicopter Maintenance Manual* indicates that the maximum allowable IPS reading for a serviceable R22 helicopter is 0.2 IPS.

For ease of identification during maintenance tasks, the helicopter's flight controls have colour-coded marks on the components positioned above the rotating swashplate[3]. The colours used on this helicopter were 'red' (the intact main rotor blade) and 'blue' (the failed main rotor blade).

The vibration charts are generic for the R22 helicopter type and may have to be tailored for an individual helicopter. If the chart is correct for the helicopter, the 'move line' for the vibration corrections should move towards the centre of the chart and a lower IPS reading, as the indicated chart corrections are carried out.

If the 'move line' of the various corrections required by the chart does not move towards the centre of the chart, a one-time correction of the chart can be made. That correction is carried out using a device called a 'clock-angle corrector'. Once the correction has been calculated and applied, all the applicable clock angles on the chart are changed to suit.

1.6.3 Main rotor blade balancing on OHA

The initial vibration readings obtained on OHA were in the order of 3.5 IPS. However, because of the removal of the engine for overhaul, and the 100-hourly inspection, no further work was carried out on the main rotor blade tracking and balancing.

During the 100-hourly inspection, both main rotor blades were removed from the helicopter. The blades were cleaned and the leading edges painted. The main rotor head teeter bearings were also replaced and the hub teeter friction was readjusted to meet the helicopter manufacturer's published specifications.

When the initial ground runs on the helicopter following the reinstallation of the engine and completion of the 100-hourly inspection were undertaken, the equipment that was fitted to the helicopter to measure the main rotor blade vibrations was found to be faulty and was subsequently removed and another unit was fitted. That second unit also displayed errors and was removed from the helicopter. Due to the time taken to source a third unit, a decision was made to check the main rotor blade spindle bearings and to

[3] A swashplate is a disc that is mounted below the main rotor blades on the main rotor mast that allows the transmission of cyclic and collective pitch control movements to the main rotor blades.

drain and replace the spindle bearing oil. Both spindles were also checked for cracks in accordance with CASA Airworthiness Directive (AD) AD/R22/30 and no defects were found.

When the replacement vibration analysis unit arrived, it was fitted to the helicopter and tracking and balancing resumed on the main rotor blades. The initial readings were in the order of 3.99 IPS, which was higher than the readings obtained when the helicopter was first examined for the main rotor blade vibration. The maintenance engineer applied corrections in accordance with the main rotor track and balance chart, but the 'move line' did not appear to be moving toward the centre of the chart. The engineer applied a clock angle correction to the chart, however this also did not result in the 'move line' moving towards the centre of the chart.

After discussing the problem with other maintenance engineers, the rotor head and main rotor blades were removed from the helicopter and statically rebalanced off the helicopter. It was determined that the main rotor assembly was easily balanced in the span-wise direction, but difficulty was experienced in balancing the blades in the chord-wise direction.

The engineers sought advice from the helicopter manufacturer's Australian representative who was located at the airport. He suggested that the engineers remove all the journals and shims from the rotor head and reinstall them in the manufacturer's standard configuration.

Assistance was also sought from another maintenance engineer who had considerable experience with the R22 helicopter type. While the main rotor head and blades were removed from the helicopter, the experienced maintenance engineer examined the blades for any evidence of a problem that may be contributing to the out of balance condition. He also examined the spindles in accordance with AD/R22/30 with a 10x magnifying glass and reported that he did not discover anything that would have contributed to the out of balance condition. The engineer suggested that the main rotor head teeter bolt friction may have been a little high and indicated that high friction can cause balancing problems. The friction on the teeter bolts was reduced to the manufacturer's lowest allowable torque setting. A check of the main rotor mast for straightness was also conducted with no defect found.

The only anomaly found by the engineer was a small area of paint that had been removed from the lower surface of the 'blue' blade. After discussing this with the other engineers, it was found that a small 'bubble' of paint had been found during the 100-hourly inspection, in the area of the 'aero smoothing' filler on the lower surface of the blade adjacent to the skin-to-blade root joint. In accordance with the instructions in the maintenance manual, the engineer removed the paint from the area surrounding the 'bubble' and examined that section for defects. No defects were found, even during examination with a 10x magnifying glass. The area was also 'tap-tested'[4] while the experienced engineer was present. Again no defects were noted. The area was painted with a clear coating so it could be examined during in-service operation.

The main rotor was then refitted to the helicopter and balancing recommenced. The balance appeared to have improved, but was still not within limits. The experienced engineer suggested that the shim and journal stack-up may have been accidentally

[4] Tap-testing involves tapping an area with a small hammer or coin and listening for changes in sound that reveal areas of adhesive disbond.

transposed during reassembly of the main rotor head. The head was again removed, the journals and shims were rotated and the main rotor head refitted to the helicopter. It was subsequently found that there was insufficient clearance between the hub and mast because of the reversed journals and shims, and the head was again removed and the journals and shims refitted. During the reinstallation of the rotor head, the thinnest shim was placed under the side of the bolt head. Balancing was again carried out and an immediate improvement in balance was noted.

With further assistance from other engineers, the balance of the main rotor blades and head was reduced to within the 0.2 IPS required by the manufacturer's maintenance manual and the helicopter was released for service.

1.6.4 Daily Inspection prior to the accident

The helicopter operator reported that they had a procedure whereby a duty pilot conducted daily inspections of the helicopters required for that day's operations and then certified in the maintenance release that the daily inspection had been completed. The pilot on duty on the day of the accident reported that no defects had been found during that inspection. That pilot also reported that the student and instructor completed another daily inspection as part of the flight lesson and did not report any defects before commencing the flight.

1.6.5 Examination of the main rotor head and blades

Following the accident, the main rotor head and both main rotor blades were examined in the ATSB laboratories. The journals, shims and bolts fitted to the head were measured and the main rotor span-wise weights were examined and weighed.

All the journals, shims and span-wise weights were found to be within the limits specified in the helicopter maintenance manual.

The only anomaly found was with a bolt in the main rotor chord weights. The bolt was not of the correct length, and was not of the correct material specification detailed in the maintenance manual. The investigation was unable to determine when this bolt was fitted to the helicopter.

1.6.6 Main rotor blade service information

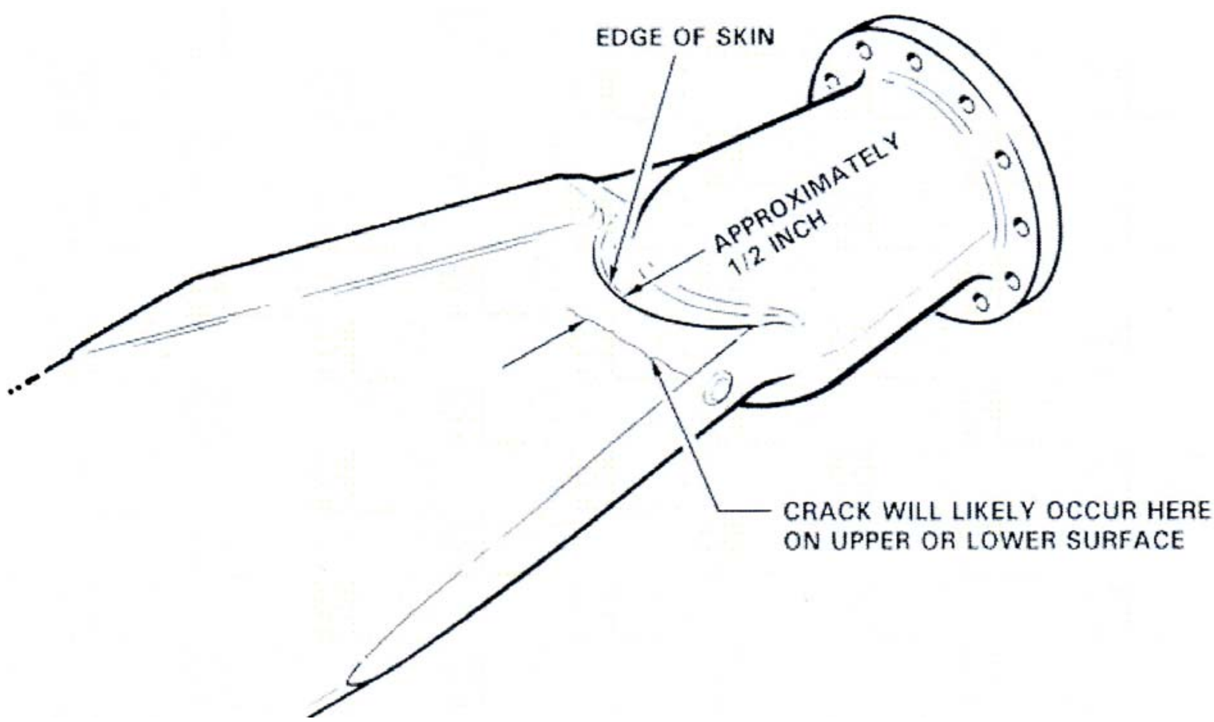
Information on main rotor blade vibration problems was contained in two R22 safety alerts that had been issued by the helicopter manufacturer. The first alert was issued on 21 November 2001 and was titled 'Exceeding Weight or Power Limits Can be Fatal'. A warning in the text of that alert stated:

> If the main rotor does not stay in balance after being adjusted, it could indicate a fatigue crack in progress. Ground the aircraft and perform the inspections described in R22 Service Letter SL-53 before further flight.

Robinson Helicopter Company, R22 Service Letter SL-53, 'Visual Inspection of Main Rotor Blade Root Area', issued 21 Nov. 2001, provides information relating to the potential development of main rotor blade fatigue cracks when the helicopter is operated under conditions where the loads on the main rotor exceed the design limits. It also provides information on potential indicators of blade fatigue cracking, main rotor vibration, and the presence of skin cracks at the location of the inboard bolt hole in the spar-to-root fitting joint.

> R22 SERVICE LETTER SL-53
>
> DATE: 21 Nov 01
>
> TO ALL R22 Owners, Operators and Service Centers
>
> SUBJECT: Visual Inspection of Main Rotor Blade Root Area
>
> BACKGROUND: A main rotor blade fatigue failure could occur if the helicopter is repeatedly flown above its approved gross weight limit or operated above its approved manifold pressure limits. The first indication of a fatigue crack in progress may be a rotor that will not stay balanced after being adjusted. Another indication may be the appearance of a very fine hairline crack appearing in the areas shown in the figure below.
>
> COMPLIANCE PROCEDURE:
>
> 1. Visually examine both the upper and lower surface of each blade in the areas shown with a 10x magnifying glass.
>
> 2. If any indication of a crack is found, immediately ground the aircraft and return the suspect blade to the RHC factory for examination.

Figure 3: Diagram included in Service Letter 53



Robinson Helicopter Company, Safety Notice SN-37, *'Exceeding Approved Limitations Can Be Fatal'*, issued Dec. 2001, provides a further warning of the effects of exceeding power and airspeed limitations on the development of fatigue cracking in main rotor blades. It contains the following warning:

> WARNING
>
> 1) Always operate the aircraft well below its approved Vne (never exceed speed), especially in turbulent wind conditions.
>
> 2) Do not operate the engine above its placarded manifold pressure limits.
>
> 3) Do not load the aircraft above its approved gross weight limit.
>
> 4) The most damaging conditions occur when flying or maneuvering at high airspeeds combined with high power setting.

In response to the detection of a large fatigue crack in the root fitting of a R22 main rotor blade and an Air Accident Investigation Board (AAIB), United Kingdom investigation, the Robinson Helicopter Company issued a R22 Safety Alert on 25 June 2002. The wording of the safety alert was included in the AAIB report as follows:

> 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.

Subsequent to the accident involving OHA, the Robinson Helicopter Company issued Safety Notice SN-39 in July 2003, titled '*Unusual Vibration Can Indicate a Main Rotor Blade Crack*'. This Safety Notice provides advice on the link between main rotor vibrations and the presence of fatigue cracks, without a direct association with excessive operational loads.

> Safety Notice SN-39
>
> Issued: Jul 2003
>
> 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 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. Have the rotor system thoroughly examined by a qualified mechanic before further flight.
>
> If main rotor vibration rapidly increases or becomes severe during flight, make an immediate safe landing. Do not attempt to continue flight to a convenient destination.

Two prior R22 main rotor blade failures (which were the result of under-recording of hours) had been the trigger for the issue of an Australia-unique AD, AD/R22/31. At the time of the accident involving OHA, AD/R22/31 was at amendment 3 status (see Section 4 for subsequent amendment actions taken by CASA as a result of the accident involving OHA). That AD only applied to blades between a specific set of serial numbers. Both the main rotor blades fitted to OHA were outside the serial number range specified in the AD. Consequently, the requirements of AD/R22/31, amendment 3, did not apply to OHA.

The helicopter manufacturer advised the ATSB that safety alerts were issued to address serious safety issues that required immediate attention. The manufacturer indicated that they also issue safety notices, and that they are equally as important as safety alerts. Safety notices are numbered and form part of the Pilot's Operating Handbook (POH). Safety alerts do not form part of the maintenance manual or the POH. Safety alerts and safety notices are only sent to those people who have subscribed to the update service for the POH, and are not routinely sent to owners, operators or maintenance organisations.

The maintenance company that serviced the helicopter during the 100-hourly inspection had current copies of the R22 maintenance manual and the illustrated parts catalogue, and had amendment services for those documents. They did not have a copy of the POH, nor was there a requirement for them to hold a copy. During interview following the accident, the maintenance staff indicated that they had not seen the safety alert that had been issued by the manufacturer on 25 June 2002. They indicated that they had seen SL-53, but considered it applicable only to helicopters that had been operated above the maximum take-off weight.

The R22 maintenance manual contains a main rotor blade track and balance troubleshooting table, and provides guidance to maintenance personnel carrying out main rotor blade vibration analysis. The table contains some of the symptoms and corrections that occur in the track and balance operation of the helicopter.

Symptom 2 of that table – 'Excessive Ship Vibration' indicated various probable causes and corrections. The table did not include any information indicating that vibration may be the result of a cracking blade as detailed in the R22 safety alerts or safety notices.

1.6.7 OHA recorded total time in service

Immediately following the accident, there were suggestions in the helicopter industry that the failure of the main rotor blade was the result of under-recording of flight hours during mustering operations. As a result of this and the two prior failures, the ATSB concentrated part of the investigation effort into the recording of flight hours on the helicopter. CASA also conducted a separate investigation into the recording of flight hours.

1.6.7.1 Total time in service and calendar life – R22 main rotor blades

The *Robinson Helicopter Company - Model R22 Maintenance Manual* contained a section on life-limited components. The section contained a list of 'fatigue life-limited' parts that included the main rotor blades, part number A016-2. The table of parts and their maximum service life was required to be approved by the US Federal Aviation Authority as part of the continuing airworthiness requirements for the R22 helicopter.

The maximum service life listed for the A016-2 main rotor blade was 2,200 hours.

The same section also contained information on the time-in-service recording for the helicopter. The section stated that:

> It is the operator's responsibility to maintain accurate time-in-service records of the airframe and life-limited components. An hourmeter activated by engine oil pressure is standard equipment in the R22 helicopter and is an acceptable means of recording time-in-service.

The section also contained the following information on the use of a collective hour meter[5] when recording time in service:

> The approved overhaul intervals and the fatigue service lives listed in the Airworthiness Limitations Section are based on FAA Advisory Circular 20-95 which assume that 10.5% of the operating time will be in autorotation, runup, or shutdown. Therefore, if an hourmeter activated by the collective control is used to record the time-in-service, the values recorded must be multiplied by 1.12 when determining replacement times for the life-limited components, engine and airframe overhaul periods and other periodic inspection requirements.

The section also imposed a calendar time in service limitation on certain components of the helicopter. The life imposed was 12 years since new, or since last overhaul. OHA had not undergone an overhaul since its manufacture in October 1991. The list of specified components required to be replaced after 12 years included the part number A005-2 main rotor blade and spindle assembly, which included the A016-2 main rotor blades.

1.6.7.2 Previous owner in the USA

Prior to importation into Australia, the helicopter was owned in the United States (US). Documents provided to the ATSB by that owner, through the United States National Transportation Safety Board (NTSB), confirmed that the helicopter had been privately owned and operated. The recorded time in service on the export Certificate of Airworthiness was 639.2 hours.

1.6.7.3 Private owner in Australia

The helicopter arrived in Australia in November 1996 and was placed on the Australian register on 23 December 1996. The total time in service entered into the helicopter logbook was 672.9 hours, which was 33.7 hours greater than that recorded on the export certificate of airworthiness (639.2). The engineer who placed the helicopter on the

[5] A collective hour meter is an hour meter activated by the position of the collective pitch control.

register and completed the logbooks reported that he could not offer any explanation for the error in recording the hours.

The helicopter was first owned in Australia by a private individual who flew the helicopter around Australia. Due to the time that had elapsed since the first Australian owner operated the helicopter, very few records were available to the investigation to confirm the recorded hours on the helicopter.

During that time the helicopter was not operated commercially and it was sold to a flying school on 27 August 1997.

1.6.7.4 Flying school ownership

On 28 August 1997, a collective hour meter was installed in the helicopter at a recorded total time in service of 874.2 hours. The reading on the meter at the time of installation was 0.0 hours.

The owner of the flying school that purchased the helicopter provided documentation relating to the time that the helicopter was operated with that flying school. It consisted of a set of invoices concerning the hire of the helicopter in charter, training or a private-hire role. They commenced on 9 September 1997 and indicated that a total time in service of 273.5 hours had been recorded. The maintenance release entries for the corresponding flights indicated a total time of 256.9 hours. There was an apparent under-recording of time in service of 16.6 hours.

Examination of the documentation provided by the owner of the flying school, and the maintenance releases for the period 9 September 1997 to 22 July 1998 showed that times from the collective hour meter had not been recorded. Those times were only recorded on the maintenance releases from 22 July 1998, the date at which the helicopter was sold to a sales company.

Examination of the times recorded on the maintenance releases from 28 August 1997 (collective hour meter installation) to 22 July 1998 revealed that 294.2 hours had been flown. The first recorded time from the meter on the maintenance release on 22 July 1998 was 268.4 hours. The maintenance releases for this period contained a statement that the ‘TTIS [total time in service] of the helicopter is to be recorded from the collective hour meter and multiplied by 1.12’.

The collective hour meter recorded time of 268.4 multiplied by 1.12 equals 300.6 hours. There was an apparent under-recording of 6.4 hours when the meter recorded time was compared with the time recorded on the maintenance releases.

1.6.7.5 Sales company ownership

The helicopter was sold to a sales company on 22 July 1998 and their policy was to provide helicopters for sale with a fresh 100-hourly inspection. Documentation provided to the ATSB indicated that the time in service at the completion of the 100-hourly inspection was 5.2 hours higher than that last recorded by the flying school.

1.6.7.6 Mustering and aerial survey company ownership

The helicopter was purchased by a mustering and aerial survey company on 1 September 1998. That company also provided available documentation to the ATSB. Due to the elapsed time between owning the helicopter and the time of the accident, complete operating records for the helicopter could not be obtained. Daily flight records for the period 30 July 1999 to 19 October 1999 were provided.

Examination of those flight records indicated that the collective hour meter stop times for each day's flying were recorded. Comparison with the maintenance release revealed that the times recorded matched both the meter and the flight time recorded on the daily flying sheet.

Examination of all maintenance releases for the period 1 September 1998 to 8 August 1999 revealed that the recorded collective hour meter times were continuous, and there was no break in the sequence of recorded times. That covered 863.8 flight hours and nine maintenance releases.

The collective hour meter readings ceased to be recorded on the maintenance release on 18 August 1999. The last recorded meter reading was 873.1 hours. The last recorded TTIS on the maintenance release at that time was 1,836.5 hours. The meter times continued to be recorded on the daily flight record sheets. The last recorded collective hour meter reading on those sheets was 955.7 hours, and was recorded on 19 October 1999. The recorded TTIS on the maintenance release for 19 October 1999 was 1,929.7 hours.

The difference between the two recorded maintenance release times was 93.2 hours. The difference between the two recorded collective hour meter times was 82.6 hours. This time has to be multiplied by 1.12 to record time in service. The adjusted time was 92.5 hours. The difference between the two was 0.7 hours and indicated an apparent over-recording of flight time.

1.6.7.7 Owner at the time of the accident

The helicopter was purchased by the current owner on 5 December 1999. At that time, the maintenance release recorded a TTIS of 1,929.7 hours. The owner reported that the collective hour meter did not function after he took possession of the helicopter. As the helicopter had not been operated commercially, no daily flight records were available. The owner reported that he kept track of the flight times from the instrument panel mounted clock and had entered those times into the maintenance release. During the interview following the accident, the owner reported that he believed that the maintenance personnel had rectified the defect in the collective hour meter installed in the helicopter. An examination of the maintenance worksheets and the helicopter log books did not reveal that any maintenance actions had been conducted on the collective hour meter.

The owner operated a commercial helicopter business, and had other turbine-powered helicopters for use in that company. The owner reported that he sometimes flew the helicopter when he needed to travel between his home and the airport. There were three maintenance releases covering the period of time from that owner's purchase of the helicopter to the accident date, each of which had expired because of the one year time limit, and not because of accruing 100 hours of flight time.

1.6.7.8 Readings on collective hour meter and factory-installed hour meter

The reading on the collective hour meter at the time of the accident was 961.7 hours. The reading on the factory-installed hour meter at the time of the accident was 798.9 hours. There were no entries in the maintenance records to indicate when either the collective hour meter or the factory-installed meter had stopped recording time correctly, nor were there any entries indicating that maintenance had been carried out on either of those recording devices to remedy any defects.

1.6.7.9 Maintenance release notations about flight time recording

There were 18 sequential maintenance releases for OHA, which were examined as part of the investigation. Two maintenance releases contained a requirement that the TTIS of the helicopter was to be recorded by engine hour meter. Those maintenance releases covered the operation of the helicopter before 28 August 1998; the date of the installation of the collective hour meter.

On nine maintenance releases, there was a requirement that the TTIS of the helicopter was to be recorded by the collective hour meter and multiplied by a factor of 1.12. The requirement commenced after the installation of the collective hour meter.

Five maintenance releases contained no requirement as to how TTIS was to be recorded. Those maintenance releases were issued during the period after the installation of the collective hour meter.

Two maintenance releases were issued with no requirement as to how TTIS was to be recorded, but were only issued for the purpose of test flying associated with the introduction of the helicopter to the Australian civil register.

1.6.7.10 Fuel records

Fuel usage can be used to correlate recorded hours on aircraft, provided that accurate fuel usage records are kept. Civil Aviation Regulation (CAR) 220 requires that, when an aircraft is operated commercially, the operator shall keep fuel records to enable the fuel usage of each aircraft to be monitored. This requirement is also repeated in Civil Aviation Order (CAO) 82.1, Appendix 1. There is no such requirement for aircraft operated privately.

No fuel records were made available by any of the commercial operators contacted by the ATSB during this investigation. There was no legal requirement for those records to be kept longer than 12 months.

No fuel usage records were made available by either of the private owners of the helicopter. Accordingly, fuel usage records could not be used to correlate the TTIS of OHA.

1.6.7.11 Helicopter recorded parts usage

In an effort to obtain a supporting picture of the recorded time-in-service hours of OHA, the maintenance records were examined to obtain the recorded parts change on a number of items. Those items ranged from the tail-rotor teeter bearings to the main rotor blade lower pitch change link bearing. Fleet average time-in-service hours for each item was provided by the helicopter manufacturer.

The only anomaly found in the comparison of the recorded parts usage against the fleet average was in the tail-rotor teeter bearings. The fleet average provided by the manufacturer indicated that those bearings lasted approximately 823 hours. The recorded change interval of those items for OHA was approximately 300 hours.

Advice was sought from the helicopter manufacturer's Australian representative and another highly experienced R22 maintenance engineer as to the expected life of tail-rotor teeter bearings in Australian service. Both reported that if the helicopter was operated in the mustering environment, then the life of the bearings could be as low as 100 hours, while operation in a training environment could result in bearing life of approximately 300 hours, with lower time than that being the norm.

Both of those times were consistent with the recorded replacement times for the tail-rotor teetering bearings in the maintenance documentation for OHA.

All other recorded parts usage times were consistent with the fleet average figures provided by the manufacturer.

1.6.7.12 Engine condition at overhaul

The company that overhauled the engine provided details about the condition of the engine at the last overhaul. They reported that the engine did not appear to be in a condition that was consistent with under-reporting of flight hours. The cylinders were replaced as part of the normal overhaul process. Measurements taken from the crankshaft revealed that the bearing journals were still within normal limits, and the crankshaft was cleaned and reinstalled in the overhauled engine.

1.6.7.13 Civil Aviation Safety Authority investigation

CASA conducted its own investigation regarding possible under-recording of TTIS on OHA. That investigation compared pilot log book entries against maintenance release entries and used Airservices AVDATA[6] records for the period April 2001 to March 2003. CASA also included the operation of the helicopter while it was being operated in a mustering role.

The CASA investigation revealed under-reporting of TTIS totalling 49.8 hours while the helicopter was operated in a mustering role. The investigation also revealed the same arithmetic errors in the maintenance releases that the ATSB investigation found.

The CASA investigation also found that there were unrecorded flights on maintenance releases when AVDATA invoices for airways and airport charges had been raised for flights on particular days. That covered 19 flights. The CASA report indicated that they calculated an average flight time over the period covered by the AVDATA records that had been entered into the maintenance releases. That resulted in an average flight time of 0.63 hours per flight being calculated. They therefore increased the TTIS by 12.1 hours to account for those flights.

[6] AVDATA - records of aircraft movements used for charging for the provision of services at an aerodrome.

The CASA report concluded that the adjusted TTIS for OHA at the time of the accident was 2,057 hours.

1.6.7.14 Company invoices and other records

Operators of companies, and pastoral properties where the helicopter was operated in a mustering role, were contacted as part of the ATSB investigation. The majority of the properties and companies had been sold and had changed hands in the period between the use of the helicopter on the property in a mustering role and the date of the accident. Documentation provided by companies or property owners was incomplete and did not allow a complete picture of the operation of the helicopter to be compared with other available information for this period. Other property owners and companies were not able to provide documentation to assist the investigation.

1.6.7.15 Summary of discrepancies and final adjustment to TTIS

The recorded total time in service on the export Certificate of Airworthiness was 33.7 hours less than that recorded in the Australian Logbooks for the helicopter when it was placed on the register. The 33.7 hours represented an over-recording of time in service.

The recorded times from all the maintenance releases that covered the operation of the helicopter were entered into a spreadsheet and summed. That revealed a discrepancy totalling 10.4 hours. One discrepancy totalling 9 hours was discovered when the total time in service was transcribed in error from the end of one maintenance release to the beginning of another. Where collective hour meter times were entered on a maintenance release, the adjustment by the factor of 1.12 was recalculated and resulted in an under-recording of 6.8 hours.

The TTIS of the helicopter was recorded as 1986.2 hours on the maintenance release recovered from the accident site. Based on the departure time from Bankstown and the reported time of the accident from witnesses, approximately 0.7 hours had elapsed. In addition, the maintenance engineers reported that the helicopter was flown for approximately 1 hour during the main rotor blade balancing exercises.

A number of discrepancies in the recording of the TTIS were noted for the time that the helicopter was operated by the flying school. The first was the difference between the hours recorded on the invoices and the hours recorded on the maintenance releases. There was an under-recording of 16.6 hours. The second discrepancy was in the difference between the hours on the maintenance release for the period of installation of the collective hour meter and the meter readings, and totalled 6.4 hours.

Examination of the collective hour meter times for the period during which the helicopter was operated by the mustering company, revealed an over-recording of flight hours of 0.7 hours.

The CASA investigation revealed discrepancies of 49.8 hours when the helicopter was operated by the mustering company. The CASA investigation also adjusted the TTIS by 12.1 hours as a result of discrepancies in recording flights when compared with AVDATA invoices.

A summary of discrepancies in TTIS is included in Table 1.

Table 1: Summary of discrepancies in recorded total time in service

Adjustment to TTIS	Hours
TTIS recorded on maintenance release recovered from accident site	1,986.2
Error in transcribing TTIS from export Certificate	- 33.7
Arithmetic errors in maintenance releases	+ 10.4
Arithmetic errors in calculating 1.12 ratio on maintenance releases	+ 6.8
Adjustment for accident flight time and maintenance flights	+ 1.7
Adjustment resulting from differences in invoices and maintenance release times	+ 16.6
Adjustment for discrepancy in comparing maintenance release hours and collective hour meter hours	+ 6.4
Adjustment of hours when using collective hour meter when used in mustering operations	- 0.7
CASA investigation findings of discrepancies in recording flight time when mustering	+ 49.8
CASA investigation findings of discrepancies in recording flight time when using AVDATA records	+ 12.1
Adjusted TTIS of helicopter at time of accident by ATSB investigation calculations	2,055.6

The adjusted total time in service of 2,055.6 hours was below the retirement life of 2,200 hours for the main rotor blade, specified by the Robinson Helicopter Maintenance Manual and approved by the US Federal Aviation Administration.

1.6.7.16 Calendar time in service

The helicopter first entered service on 1 October 1991. On the day of the accident it had been in service for 11 years, 8 months and 19 days, which was below the 12 years specified by the Robinson Helicopter Maintenance Manual and approved by the US FAA.

1.7 Meteorological information

The weather conditions in the area of the accident were reported as no cloud and light winds, and were consistent with the conditions forecast by the Bureau of Meteorology.

1.8 Aids to navigation

Not applicable.

1.9 Communications

The air traffic control automatic voice recording tapes for the area in which the helicopter had been operating were examined following the accident. Apart from the normal radio broadcasts associated with the departure from Bankstown airport, no other broadcasts from the helicopter could be identified.

1.10 Aerodrome information

Not applicable.

1.11 Flight recorders

The helicopter was not fitted with a flight data recorder or a cockpit voice recorder, nor was either required by regulation.

1.12 Wreckage information

The wreckage trail of the helicopter commenced on flat ground to the west of Bringelly Creek, a tributary of the Nepean River, and extended along a bearing of approximately 240 degrees. The wreckage trail continued up an escarpment and ended on flat ground on the top of the escarpment. One of the main rotor blades could not be located during the initial examination of the wreckage trail and the accident site. The blade was located on the following day after a wider search.

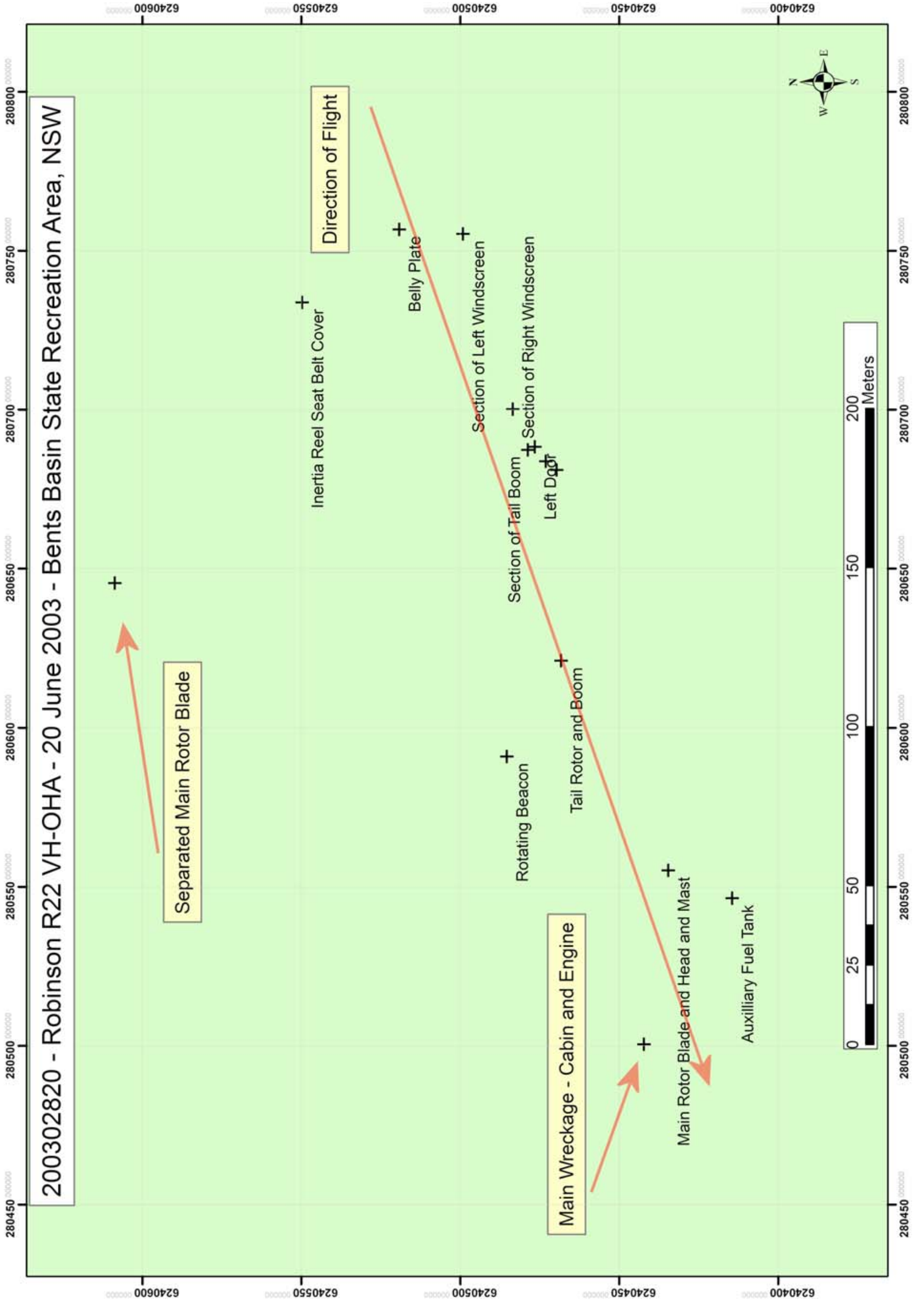
The length of the complete wreckage trail was approximately 700 m, with the cabin and engine located in the most westerly section. The wreckage trail commenced with light paper items from the cockpit. The separated main rotor blade (blue blade) was located approximately 135 m north of the approximate wreckage trail centreline. The other main rotor blade (red blade), which remained attached to the main rotor head, was located approximately 55 m to the east of the main cabin and engine (see Figure 4). All of the components of the helicopter were identified and located within the wreckage trail.

The fuselage impacted the ground in an inverted attitude, and the cockpit area had been crushed to approximately half of its original height. The forward third of the tailboom assembly remained attached to the fuselage. The centre third of the tailboom was in two sections and was located approximately 210 metres from the fuselage. The rear third of the tailboom remained attached to the tailrotor gearbox, which was located approximately 145 m from the fuselage.

The failed ‘blue’ main rotor blade had no damage to either the leading or trailing edge. The ‘red’ main rotor blade had damage to the trailing edge only. There was no evidence that either of the main rotor blades had contacted any part of the helicopter airframe before the failure of the ‘blue’ main rotor blade. The centre third of the tailcone exhibited impact damage and paint transference from the trailing edge of the ‘red’ main rotor blade, that had occurred after the failure of the ‘blue’ main rotor blade.

Examination of the helicopter and engine at the accident site revealed no evidence to indicate that they were not capable of normal operation prior to the failure of the ‘blue’ main rotor blade.

Figure 4: Wreckage Map and Component Location



The seatbelt for the right seat occupant was found unlatched and rescue personnel reported that it had been found in that condition upon their arrival. Subsequent examination of the belt latching mechanism revealed that it had been subjected to an impact with a hard object that is likely to have released the buckle clasp.

1.13 Medical information

There was no evidence that psychological factors or incapacitation affected the performance of either the instructor or the student pilot.

1.14 Fire

There was no evidence of an in-flight or post impact fire.

1.15 Survival aspects

The accident was not survivable.

1.16 Tests and research

Both main rotor blades were recovered from the accident site and were subjected to a detailed material failure examination in the ATSB's laboratory in Canberra.

That examination initially was centred on the fatigue failure and its development. However, as further information was uncovered, it soon became apparent that the accident had wider implications for the global R22 fleet. Additional testing of other R22 main rotor blade root fittings was conducted to ascertain if similar findings in the failed blade were present in other R22 main rotor blades. Following examination of additional blades, the ATSB issued a safety recommendation to the FAA and the manufacturer (see Section 4.2) to conduct testing of a larger sample of main rotor blade root fittings.

Further details of the examination of the failed main rotor blade in OHA, along with the results of the testing following the issuing of the safety recommendation, are contained at Appendix A, *Engineered System Failure Analysis, Main Rotor Blade, Robinson R22, VH-OHA.*

1.17 Organisational information

Not applicable.

1.18 Additional information

1.18.1 Previous Australian accidents – R22 loss of main rotor blade in flight

On 27 May 1990, a Robinson R22 helicopter, registered VH-HBS, crashed in Western Australia following the loss of a main rotor blade in flight. Both persons on board were fatally injured. The accident was investigated by the then Bureau of Air Safety Investigation (see Investigation Report 199000089). During that investigation it was found that the time in service of the failed main rotor blade had exceeded the then mandatory retirement life of 2,000 hours in service because of under-recording of flight time.

In addition, examination of the failed main rotor blade in that accident revealed that it had failed as a result of the initiation of a fatigue crack at rotor station (RS) 10.35, which had been initiated as the result of a manufacturing defect. As a result of the accident investigation, the then Civil Aviation Authority introduced an airworthiness directive (AD) - AD/R22/31 for the R22, which required mandatory inspections of the main rotor blades of R22 helicopters between a specific range of serial numbers (see section 1.6.5).

On 29 July 2000, a Robinson R22 helicopter, registered VH-LDR, crashed in Queensland following the loss of a main rotor blade in flight as a result of the initiation of a fatigue crack at RS 10.35. The pilot was fatally injured and the passenger sustained serious injuries. That accident was investigated by the Australian Transport Safety Bureau (see Investigation Report 200003267), which found that the pilot of the helicopter had been under-recording flight hours. The surviving passenger reported that immediately before the crash, the helicopter developed a severe vibration.

1.18.2 Israeli accident - R22 loss of main rotor blade in flight

On 29 February 2004, a Robinson R22 helicopter crashed in Israel while being operated on a powerline survey. Both occupants of the helicopter were fatally injured. The accident was investigated by the Israeli Ministry of Transport.

The accident investigation revealed that a main rotor blade had failed in flight as a result of the initiation of a fatigue crack, with the location of the crack initiation being at the same locations as the two failures listed above (RS 10.35). The investigation also revealed that 2 days prior to the accident, maintenance work to rectify a main rotor blade vibration had been carried out. The maintenance engineers made numerous changes to the chord weights and a ‘head shift’ to correct the vibration.

The fatigue crack initiator in that accident was reported to be the result of corrosion. Chemical analysis of the corrosion products indicated the presence of chlorine and other elements, however the analysis did not support the presence of seawater as being the initiating corrosive agent. The crack propagation in that accident also did not extend into either the upper or lower skin until it had reached the critical size when failure was imminent.

The investigation also revealed that there were areas of adhesive disbonding between the spar and root fitting present on the failed main rotor blade. There were also areas of adhesive disbonding present in the skin to root fitting area. These findings were consistent with those found by the ATSB during preliminary testing of the failed ‘blue’ main rotor blade on OHA, a number of other R22 main rotor blades, and were the foundation for issuing Safety Recommendation R20030186 (see section 4.1 for more information about R20030186).

The recorded time in service of the failed blade in that accident was 1,490 hours, with the blades accumulating approximately 11.8 years time in service.

1.18.3 New Zealand accident - R22 loss of main rotor blade in flight

On 27 November 2004, a Robinson R22 helicopter, registered ZK-HWP, was destroyed following the loss of a main rotor blade in flight while conducting aerial agricultural spraying operations. The pilot survived the accident and reported that he noticed an increase in main rotor vibration during the previous flight, which was described as feeling like a blade out of track. At that time, the pilot reported that he had conducted a post-flight visual inspection of both main rotor blades, and that no defects could be seen in either blade. On the next flight, shortly after becoming airborne, a main rotor blade separated from the rotor hub at the blade root. The failure was as a result of the initiation of a fatigue crack at RS 10.35.

On 29 July 2004, the New Zealand Civil Aviation Authority had issued Airworthiness Directive AD – DCA/R22/40A that required operators to ensure that all pilots were aware that an increase in vibration may be a sign of imminent main rotor blade failure. It required that in the event of vibration, the helicopter was to be landed immediately and that the blade undergo mandatory inspections before any further flight.

Preliminary investigations have revealed that the failure may be the result of loadings on the blade that may have exceeded those intended by the manufacturer. The investigation into that accident is continuing.

1.18.4 NTSB Special Investigation Report – Robinson Helicopter Company R22 Loss of Main Rotor Control Accidents

Following a number of accidents involving R22 helicopters in which loss of main rotor control was considered to be a factor, the US National Transportation Safety Board (NTSB) conducted a special investigation into that phenomenon. They published a special investigation report on 2 April 1996 (NTSB/SIR-96/03), in which they included a number of findings and safety recommendations. In that report, the NTSB defined loss of control as:

- loss of main rotor control;
- structural failure of the main rotor blade that did not involve preexisting fatigue of rotor blade materials; or
- loss of aircraft control or collision with terrain for unknown reasons, in the absence of structural failure, encounter with instrument meteorological conditions, or pilot impairment from drugs or alcohol.

Material failure analysis of the failed ‘blue’ main rotor blade from OHA revealed that loss of main rotor control was not a factor in this accident.

2 ANALYSIS

2.1 Introduction

The accident involving Robinson R22 helicopter, VH-OHA, occurred when the 'blue' main rotor blade, serial number 6249A, fractured in the region of the transition from the blade airfoil section to the blade root fitting at rotor station 10.35. The complete separation of the blade airfoil section resulted in the creation of an immediate and catastrophic loss of control combined with severe out-of-balance forces on the helicopter.

Blade fracture occurred as a result of the initiation of a fatigue crack in the counterbore of the inboard bolt hole in the blade spar to root fitting joint. Subsequent growth of the fatigue crack reduced the strength of the blade to a level that allowed fracture to occur under normal flight loads. Increased stress magnitude through adhesive disbonding, the presence of a corrosive environment and the development of corrosion pits at the inboard bolthole acted to reduce the time to fatigue crack initiation.

A detailed analysis of the main rotor blade fracture is contained at Appendix A, *Engineered System Failure Analysis, Main Rotor Blade, Robinson R22, VH-OHA.*

2.2 Main rotor blade service information

The helicopter manufacturer issued safety information in the form of safety alerts and safety notices. This service information is only sent to those people who have subscribed to the update service for the Pilot Operating Handbook. Consequently, owners, operators or maintenance organisations who do not subscribe to that service, may not be in possession of information that may be critical to safety of flight.

In addition, the helicopter manufacturer's R22 maintenance manual did not contain information indicating that excessive vibration may be the result of a crack in the main rotor blade.

2.3 Flight crew and weather

There was nothing found during the investigation to suggest that flight control input by the student pilot or the flight instructor contributed to the loss of the main rotor blade in flight. The flight exercise of climbing and descending did not involve a high level of risk, nor did it require the conduct of flight manoeuvres that would have imposed anything other than normal operational stresses on the main rotor blades. The weather conditions were reported to have been as forecast and did not contribute to the accident.

2.4 Regulatory action

The United States Federal Aviation Administration have not mandated the requirements of Robinson Helicopter Company Service Letter 54. It has, however, been mandated by the European Aviation Safety Agency.

3 CONCLUSIONS

3.1 Findings

3.1.1 Helicopter

1. The main rotor blades fitted to the helicopter were the original main rotor blades fitted to the helicopter at the time of manufacture.

2. The ATSB investigation calculated that the helicopter had accumulated 2,055.6 hours of recorded flight time at the time of the accident.

3. The helicopter had been in service for 11 years, 8 months and 19 days at the time of the accident.

4. Maintenance to rectify main rotor blade vibration had been carried out prior to the flight during which the accident occurred.

5. There was no evidence that indicated that either the helicopter or engine was incapable of normal operation prior to the failure of the ‘blue’ main rotor blade in flight.

6. At the time of the accident, all required maintenance had been carried out, including applicable airworthiness directives relating to the helicopter.

7. The information relating to vibration as an indicator of a crack in a main rotor blade was not contained in documentation held by the maintenance organisation.

8. Prior to failure of the ‘blue’ main rotor blade, the helicopter was not subject to a loss of main rotor control.

3.1.2 Flight crew

9. The student pilot and the flight instructor were both correctly licensed and endorsed to conduct the flight.

10. The student pilot and the flight instructor were reported to have been medically fit, and adequately rested prior to commencing the flight.

3.1.3 Main rotor blade failure analysis findings

11. R22 main rotor blade, P/N A019-2 Rev. AD, S/N 6249A fractured as a result of fatigue crack growth in the blade root fitting at rotor station (RS) 10.35. Fatigue crack initiation occurred at three sites in the counterbore of the inboard bolt hole. The site of primary fatigue crack initiation was associated with localised corrosion in the bolt hole counterbore.

12. Blade fracture occurred within the ‘safe life’ operational period, specified at the time of the accident.

13. In addition to the fatigue cracking in the blade root fitting, progressive disbonding had occurred in the adhesive joint between the blade spar and blade root fitting, and in the adhesive joints between the upper and lower blade skins and the blade root fitting.

14. Disbonding of the adhesive joints between the upper and lower blade skins and the blade root fitting prevented fatigue crack growth in the blade root fitting from being transferred to the blade skins, and so prevented the detection of cracking in the blade root fitting through a visual inspection of the outer surface of the blade skins.

15. The disbonding of the adhesive joint between the blade spar and blade root fitting, from the inboard end of the spar to the inboard bolt hole had two effects: the magnitude of stress in the bolt hole counterbore was increased, and a path was created for moisture and, in particular, moisture containing chloride salts, to come into contact with the bolt hole counterbore surface.

16. The effects of increased stress magnitude, the presence of a corrosive environment and corrosion pits reduced the number of loading cycles to fatigue crack initiation. A decrease in the number of loading cycles to fatigue crack initiation resulted in a decrease in the time to fatigue crack initiation.

17. Increased stress levels at this location decreased the time to fatigue failure.

18. It is evident that a critical factor in accelerating the onset of fatigue crack initiation is the progressive disbonding of the adhesive joint between the blade spar and blade root fitting to the inboard bolt hole. Examination of other main rotor blades, in particular those that were paired with the failed blades, revealed that adhesive disbonding had not extended to the inboard bolt hole despite being exposed to similar environmental and operating conditions. Fatigue cracking had not developed in these blades.

3.2 Significant factors

1. Main rotor blade vibration was evident during flights preceding the accident flight.

2. Main rotor blade, p/n A016-2, s/n 6249 (blue), fractured during flight following the propagation of a fatigue crack in the blade root fitting.

3. Disbonding of the adhesive joints between the upper and lower blade skins and the blade root fitting prevented the detection of the fatigue crack in the blade root fitting through a visual inspection of the outer surface of the blade.

4. Following the main rotor blade failure, the helicopter was rendered uncontrollable and impacted the ground.

4 SAFETY ACTION

4.1 Safety action taken by the ATSB

Following preliminary investigation of the failed ‘blue’ main rotor blade, the ATSB issued Air Safety Recommendation R20030186 to both the manufacturer (Robinson Helicopter Company) and the Federal Aviation Administration. Prior to the issuing of this recommendation the FAA, Robinson Helicopter Company, the National Transportation Safety Board and the Australian Civil Aviation Safety Authority were comprehensively briefed on the preliminary findings of the failed blade. The text of that recommendation was:

> The Australian Transport Safety Bureau recommends that the United States Federal Aviation Administration, in conjunction with the manufacturer of the helicopter, the Robinson Helicopter Company, conduct a review of a representative sample of main rotor blade root fittings to establish the integrity of the adhesive bond in the spar to root fitting joint. The review should establish the extent of the loss of adhesion and the extent to which corrosion has infiltrated in the region of the inboard bolt hole of the blade root fitting. If possible, where disbonding is discovered, the operating history and in-service flight spectrum of the helicopter and the environmental conditions under which it operated should also be assessed.
>
> When completed, the results of the review should be forwarded to the ATSB for analysis as part of the ongoing accident investigation.

The FAA responded on 8 March 2004 with the following letter:

> We offer the following response to the subject recommendation from the Australian Transport Safety Bureau (ATSB) regarding the failure of a main rotor blade (MRB) on a Robinson Helicopter Company (RHC) Model R22 helicopter:
>
> The FAA released a Special Airworthiness Information Bulletin (SAIB) SW-04-36 on December 17, 2003, that recommends actions that can be taken by the operator to maintain the airworthiness of the MRBs. One of the recommended actions in the SAIB is to accomplish the inspections contained in RHC Service Letter SL-21 A, "Main rotor blade sealant and filler cracking," dated May 31, 2002. The service letter details inspections for cracks of the blade root area without damaging the root fitting. The FAA has determined that more in-depth inspections of the root fittings could result in damage to the blade and increase the possibility of cracks after the blade is returned to service.
>
> The MRB inspections by the National Transportation Board (NTSB) have not resulted in an effective inspection or test to detect adhesive failure. The NTSB investigation is not complete, and additional steps remain to be taken. Metallurgical testing will be conducted on the unidentified substances found on the blades. The analysis will also look at those areas where the metal had an unusual corrosion like appearance. For the blades where the adhesive failure extended to the bolt hole and where the white crystalline deposits were found near the bore, the examination will include sectioning the root fittings through the inboard bolt hole to look for evidence of corrosion pitting and/or cracks. Blade root fitting specimens will also be sent to the adhesive manufacturer for evaluation of the adhesive.

RHC plans an on-going sampling of returned MRBs to look at the spar to root fitting joint so that a wider sampling of blades may be generated. Emphasis will be placed on examining the R44 blades (the problem area is completely enclosed on the R44 blade design) to determine if the design differences between the Model R44 and R22 blades impact this adhesive failure phenomenon. This is particularly important since the new R22 blade design for the Model R22, which has been FAA approved, incorporates the closed root fitting design from the R44 blade.

RHC has also modified the MRB to allow cracks to propagate more slowly, such that if the blade is flown beyond the service life and a crack develops, the operator will have more advanced warning prior to failure. The root fitting of the R22 MRB was redesigned to be more robust in the root fitting area. The new blades, part number (P/N) A016-4, will be the only new blades available and will eventually replace all of the older style blades. The FAA has also been working with RHC to determine if a new calendar life limit, in addition to the total in service time, for blades, P/N A016-1 and -2 is necessary.

Additionally, RHC has corrected the manufacturing procedure that allowed the mechanics to cause scratches during adhesive cleanup. Any R22 MRB produced after November 2001 should be free of these abrasion scratches. The depth of the shot peen layer is not in question, and the quality control system at RHC has been reviewed and determined to be acceptable.

Also, the FAA has asked RHC to make a study of cattle mustering maneuvres [sic] to determine if they are causing high stresses that would cause premature fatigue failures. So far, it appears that these maneuvres [sic] should not cause excessive failures. There is interest in gathering information on different methods of mustering to determine if some operations are more severe than others.

Civil Aviation Safety Authority (CASA) has indicated previously that some operators have been under reporting MRB operational hours. We understand that the three accidents, which precipitated the CASA AD's, involved helicopters used in cattle mustering operations. All three aircraft experienced blade failure due to a fatigue crack in the first bolt hole outboard of the blade root, which is the location of a normal fatigue crack, due to operation after the 2200-hour retirement limit. Evidence shows that operators in Australia in the past have routinely flown beyond retirement life limits. Two of the three accident ships have records indicating they were operated past the life limit, and the third ship has circumstantial evidence that indicates that the blades were in service past the life limits. To help resolve the problem, RHC published Safety Notice SN-37 in the R22 Pilot Operating Handbook, which warns pilots not to exceed published service lives.

We do not plan any further action and recommend that this recommendation be closed.

Following receipt of this letter, the ATSB responded with the following letter:

I refer to the recent FAA response to ATSB Recommendation R20030186, which was issued on 17 September 2003 following a main rotor blade failure on a Robinson Helicopter Company model R22 helicopter. The FAA has assigned FAA recommendation number 03.233 to the response.

The background to the recommendation clearly indicates that the initiation point of the fatigue failure was from corrosion and not fatigue that would be induced by exceeding published retirement time-in-service. While previous failures have been associated with exceedances of retirement times, and there may be industry rumours and anecdotal evidence that this is a widespread practice within the Australian Mustering industry, detailed investigation of this accident to date has not revealed any evidence of retirement life exceedance.

The Special Airworthiness Information Bulletin (SAIB) issued by the FAA on 17 December 2003 does not provide any additional information for operators to assist them in determining the airworthiness of the main rotor blades. In the subject main rotor blade, which failed in this accident, none of the information provided in the SAIB, nor the RHC service letters referenced in it, would have allowed maintenance personnel to detect the crack.

The recent accident in Israel in which an R22 helicopter sustained what appears to be an identical failure of the main rotor blade, only highlights that the mechanism of failure is not confined to helicopters operated in a mustering environment. The Israeli helicopter was previously operated in the USA before it was exported to Israel. Preliminary information provided to the ATSB is that the main rotor blade had only been in service for approximately 2/3 of its published service life.

The common thread between the Israeli accident and the Australian accident is that immediately prior to the accident, both helicopters were inspected, by maintenance personnel, for main rotor blade vibration. While RHC Safety Notice SN-39 addresses main rotor vibration, this vibration is now known to be at the end of the process of structural deterioration and may not provide sufficient time for a pilot to land the helicopter before the final fracture of the blade.

While the newly designed RHC main rotor blade may address the problem, it has only just entered production and it will take some time for all present A016-2 blades to be retired from the worldwide fleet.

The ATSB has classified the response to R20030186 as OPEN as it does not adequately address the problem highlighted in the recommendation.

The ATSB urges the FAA to again closely look at the problem of adhesive disbonding in the R22 main rotor blade root fitting and to what extent the disbonding permits corrosive compounds to enter this critical area.

The ATSB also urges the FAA to investigate methods of detecting cracks in the root fitting prior to failure.

The ATSB looks forward to your earliest reply.

The following reply from the FAA was received on 26 May 2004.

In response to the Australian Transport Safety Bureau (ATSB) letter of March 17, 2004, we have reviewed our memorandum of January 29, 2004, on the subject recommendation. The ATSB letter referred to the recent accident in Israel involving failure of a main rotor blade (MRB). The ATSB letter requested that we further look at adhesive disbonding in the MRB root fitting and methods for detecting cracks in the root fitting before failure.

> Since our previous response, we have issued Emergency Airworthiness Directive (AD) 2004-06-52 on March 18, 2004 (copy attached). That AD was prompted by accidents in Australia and Israel involving failure of the MRB. The accident investigations revealed that cracked blades result in an increase in helicopter vibration. Following a track-and-balance of the blades, the vibrations return to normal for a short time and then slowly increase again until blade failure occurs. This prompted us to mandate replacing the blades if an abnormal increase in vibration occurs within 5 hours total time-in-service (TTIS) after the last track-and-balance. This AD also adds a retirement life of 10 years to the current 2,200 hours TTIS retirement life.
>
> Neither the FAA nor the manufacturer has identified an effective inspection or test to detect a crack or adhesive disbonding in the root fitting area before failure of the MRB. The FAA is taking part in investigating the failure of the MRB involved in the Israel accident. However, we believe the actions taken in the Emergency AD address the unsafe condition. If ATSB has suggestions for an inspection method, we would welcome any ideas and would evaluate the method with the manufacturer before putting it into practice.

The ATSB has classified the response to R20030186 as CLOSED-ACCEPTED.

The ATSB has contracted a research investigation project monitoring the usage spectrum of a Robinson R22 helicopter used for cattle mustering in the north of Australia. The project will then compare the actual usage spectrum with the assumed usage spectrum that was used for certification, to assess the validity of some of the structural and maintenance control assumptions on which the airframe integrity is based. The testing will continue through the 2005 mustering season, and a report on the project is planned to be released in 2006.

4.2 Safety action taken by the Australian Civil Aviation Safety Authority

4.2.1 Amendments to Airworthiness Directive AD/R22/31

At the time of the accident, CASA AD/R22/31 was at amendment 3 status. On 27 June 2003, CASA issued amendment 4 to that AD. It required a visual inspection of the inboard end of each main rotor blade using a 10x magnifying glass, and an eddy current inspection of the inboard bolt hole on each main rotor blade. That AD applied to all R22 helicopters regardless of main rotor blade serial number. It also required the installation of a placard in the cockpit indicating that in the event of main rotor blade vibration increasing during flight that the pilot was to land the helicopter immediately.

On 25 July 2003, CASA issued amendment 5 to the AD. The amendment allowed further time for maintenance data to be provided by the manufacturer and allowed time for maintainers to undergo training in the eddy current inspections called for in the AD.

On 8 August 2003, CASA issued amendment 6 to the AD. The amendment removed the requirement for the eddy current inspection on the advice of the helicopter manufacturer. That advice indicated that problems may be introduced by repeated removal and re-installation of the inboard bolt in the field. It introduced a distinction between those blades that had been used in aerial mustering and those which had not. For those blades that had been used in aerial mustering, a 1500 hour time in service retirement life was introduced. For those blades that had passed the 1500 hour limit, they were to be retired from service by a specified date.

On 19 August 2003, amendment 7 to the AD was issued by CASA. The amendment increased the compliance time for operators and owners to comply with the AD as a result of being remotely located from suitable maintenance facilities.

On 18 September 2003 amendment 8 to the AD was issued which introduced a retirement schedule based on calendar time and operating time in service. The distinction between those blades that had been used in mustering and those that had not, was removed.

On 19 March 2004, CASA issued amendment 9 to the AD, in response to FAA Emergency AD [2004-06-52]. That emergency AD was issued following the crash of a R22 helicopter in Israel (see section 1.18) and the accident involving OHA. The amendment to the CASA AD deleted the retirement schedule which had been introduced in amendment 8 to the AD. It was replaced with the revised main rotor blade retirement times contained in the FAA Emergency AD.

On 22 March 2004, CASA issued amendment 10 to the AD. The amendment clarified some of the wording of the compliance section of the AD.

On 14 October 2004, CASA issued amendment 11 to the AD. The amendment clarified the applicability of the AD (see the CASA website at http://www.casa.gov.au/).

4.2.2 Release of Discussion Paper

The Australian helicopter mustering industry and the Standard Consultative Committee of CASA requested that CASA investigate the mandatory fitment of tamper resistant time recording devices to single engine helicopters. This request was the result of rumours that circulated shortly after the accident involving OHA, the two previous accidents in Australia that involved under-recording of time in service (see section 1.18.1) and other evidence that some operators of helicopters are not recording time in service hours accurately. The discussion paper sought responses from the industry on the concept of fitting tamper-resistant recording devices to single engine helicopters.

The public comment period on the Discussion Paper closed on 25 February 2005. As at October 2005, CASA evaluation of public comments was continuing.

4.2.3 Publication of Notice of Proposed Rule Making

CASA has drafted a Notice of Proposed Rule Making (NPRM 0503CS) in which it proposed to introduce a unique Australian Airworthiness Directive to mandate the retirement of A016-2 main rotor blades by 1 March 2006.

4.3 Safety action taken by the US Federal Aviation Administration

The US FAA issued Special Airworthiness Information Bulletin SW-04-36 on 17 December 2003. The Bulletin asked operators to:

- read Robinson Helicopter Company (RHC) Safety Notices SN-39 and SN-37
- read RHC Service Letter SL-53

- ensure that the helicopter was not operated beyond the part retirement lives currently published in the R22 Maintenance Manual.

- perform the inspections in SL-21A to inspect for cracks in the main rotor blade sealant and filler.

Finally it contained information that the FAA believed that the failure was associated with operation of the blade past its mandatory retirement life.

The FAA issued emergency airworthiness directive AD 2004-06-52 on 18 March 2004. That AD introduced a requirement to track and balance all A016-2 main rotor blades that were older than 5 years or had greater than 1,000 hours TTIS, within 10 flight hours or 30 days. It required that blades that had undergone this track and balance and subsequently developed a vibration within 5 hours TTIS to be retired from service. It also revised the calendar TTIS of the A016-2 main rotor blade to 10 years.

4.4 Safety action taken by European Aviation Safety Agency

As a result of several accidents involving part number A016-2 main rotor blade failures in Robinson R22 helicopters, the European Aviation Safety Agency, issued Airworthiness Directive 2005-0019 on 5 July 2005. This Airworthiness Directive mandates compliance with Robinson Helicopter Company Service Bulletin SB-94, requiring all part number A016-2 main rotor blades to be replaced with part number A016-4 before 1 December 2005.

4.5 Safety action taken by the Robinson Helicopter Company

On 25 march 2004, the Robinson Helicopter Company issued Service Letter (SL) 54 in response to the issuing of FAA AD 2004-06-52. That SL revised the life-limited parts list calendar time limit for the A016-2 main rotor blades to 10 years. The SL also contained the following warning:

> Blades are more susceptible to corrosion if the helicopter is normally parked outside in humid climates, particularly in tropical or coastal areas. RHC strongly recommends that A016-1 & -2 blades in these areas be removed from service as soon as possible and prior to 10 years.

A revised SL-54 was issued by the manufacturer on 31 March 2004. It contained essentially the same information; however the text of the warning had been modified to state:

> Blades are more susceptible to corrosion if the helicopter is normally parked outside in humid climates, particularly in tropical or coastal areas. RHC strongly recommends that A016-1 & -2 blades operated in these corrosive areas be removed from service as soon as possible and prior to 5 years time-in-service.

Another revised SL-54 was issued on 17 June 2004. That SL contained amendments to the costs involved in replacing the blades only. The warning above remained the same.

A Service Bulletin (SB), SB-94 was issued on 14 December 2004. This SB required that all A016-2 main rotor blades with a serial number prior to 12000 be retired from service by 1 July 2005 and all remaining A016-2 main rotor blades to be retired from service by 1 December 2005.

As at June 2005, the FAA had not issued an AD that mandated the information contained in SB-94. SB-94 is not mandatory in Australia without accompanying AD action from CASA.

The manufacturer has introduced a redesigned main rotor blade into service. It has also indicated that it intends to publish safety alerts and notices on its internet website as an additional means of bringing safety related information to the notice of owners, operators and maintenance organisations.

The manufacturer has amended the R22 maintenance manual as a result of this investigation. The main rotor blade tracking and balancing section now contains information alerting maintenance personnel to the fact that a main rotor blade vibration may be the result of a developing crack.

APPENDIX A: MAIN ROTOR BLADE FAILURE ANALYSIS REPORT

ENGINEERED SYSTEM FAILURE ANALYSIS REPORT

MAIN ROTOR BLADE FRACTURE
ROBINSON R22, VH-OHA
OCCURRENCE 200302820

Dr Arjen Romeyn
Principal Failure Analyst – Engineered Systems

2 November 2005

SUMMARY

The accident involving Robinson R22 helicopter, VH-OHA, occurred when one of the two main rotor blades fractured in the region of the transition from the blade airfoil section to the blade root fitting. The complete separation of the blade airfoil section resulted in the creation of an immediate and catastrophic loss of control combined with severe out-of-balance forces on the helicopter.

Blade fracture occurred as a result of the initiation of a fatigue crack in the counterbore of the inboard bolthole in the blade spar to root fitting joint. Subsequent growth of the fatigue crack reduced the strength of the blade to a value that allowed fracture to occur under normal flight loads (loads normally encountered during operation within the helicopter's design envelope).

The threat of main rotor blade fatigue failure in Robinson R22 helicopters, along with many other critical components in aircraft structures and machines subjected to alternating stresses during operation, is managed by defining a safe life operating time period. The safe life is the period of operational time prior to the initiation of fatigue cracking and it is the key element in the fracture control plan for single load path, flight critical components, for which there is no structural redundancy. Minimisation of the threat of main rotor blade fatigue failure is dependent on the accurate prediction of the period of operational time prior to the initiation of fatigue cracking and the ability of safety factors to accommodate any uncertainties in prediction.

The life of Robinson R22 main rotor blades is limited, ultimately, by the development of fatigue cracks. It is known, from the development testing of blade part number A016-2, that the critical location for fatigue crack initiation is the inboard bolthole in the joint between the blade spar and blade root fitting (an adhesively bonded and bolted joint).

Operational experience has shown that blade fracture resulting from fatigue crack initiation at the inboard bolthole (blade spar to blade root fitting joint) of R22 main rotor blades is rare, however, in the case of part number A016-2 blades; it is not an isolated occurrence. Fatigue cracking at the inboard bolthole has resulted in blade fracture occurrences at times before and following the accident involving VH-OHA. Two Australian accidents, before the VH-OHA accident, occurred when the specified blade retirement life was exceeded. Two International (Israel and New Zealand) accidents, following the VH-OHA accident, occurred at operational times within the specified safe life period.

In the case of the fracture of the blade fitted to VH-OHA, rigorous investigation of several sources of helicopter operating records established that blade fracture occurred within the safe life period specified at the time of the accident.

Blade fracture during operation represents a failure of the fracture control plan. There are two critical features of the fracture control plan that require analysis in order to determine why the main rotor blade fractured during operation. Firstly, it is necessary to determine if the fracture occurred within the specified retirement time or whether

the specified retirement time had been exceeded. Secondly, it is necessary to determine if there was an excessive variation in fatigue crack initiation time.

Fatigue is the process of crack initiation, incremental growth and final fracture which has its origins in the mechanism by which a material accommodates the effects of localised alternating stresses. Fatigue is dependent on the number and magnitude of stress cycles. It is not primarily dependent on operational time.

Because the process of fatigue cracking (crack initiation, crack growth, and final fracture) is dependent on the number and magnitude of alternating stress cycles it is affected by: the number and magnitude of loads applied to the structure during each period of operation, the nature of load transfer in a complex structure, the presence of stress concentrating features, environmental interactions during operation and storage, and the material in the blade structure. Variation in these factors can decrease the operational time to fatigue fracture to a point where fracture occurs prior to the specified operational retirement time. An analysis of variation in these factors is required to determine why the blade fractured within the specified safe life.

It is a feature of each occurrence involving the inflight fracture of a main rotor blade (part number A016-2) that the remaining blade fitted to the helicopter showed no evidence of fatigue crack initiation at the location of the inboard bolthole (blade spar to blade root fitting joint), despite being subjected to the same number and magnitude of alternating loads during each flight, the same operating environment and the same storage environment. This observation indicates that blade to blade variations may be more important than variations between individual operations in eroding safety margins and allowing fatigue initiation, crack growth, and final fracture to occur within the specified safe life.

A detailed analysis of the fractured blade and other blade fracture events has identified that changes in the blade structure, in the vicinity of the inboard bolthole of the blade spar to blade root fitting joint, do occur during operation. A critical change, in the case of blades that have fractured, is the progressive growth of a region of adhesive disbond from the end of the blade spar, extending between the blade spar and blade root fitting to, and beyond, the inboard bolthole. Examination of other intact blades, in particular, the blades paired with the failed blades, and 59 blades from a variety of operating environments around the world and a variety of operational times, revealed that while disbond growth from the end of the blade spar had occurred in almost each case, disbonding had not progressed to the inboard bolthole.

It is evident that disbond growth between the blade spar and root fitting to the inboard bolthole is a critical factor, which has an effect on the magnitude of alternating stresses at the bolthole.

Research into the behaviour of aircraft structural joints has revealed that in the case of joints that combine adhesive bonding and bolts, load transfer occurs through the adhesive. An effect of load transfer through the adhesive is the reduction in the stress concentration effect created by the presence of bolt holes. Adhesive disbonding in the region surrounding the inboard bolthole in R22 main rotor blades will restore the stress concentration effect of the bolthole. The increase in local stresses created by the stress concentration effect will result in a marked increase in the stress magnitude in the counterbore region of the bolthole and will have a consequent effect of decreasing the number of stress cycles to fatigue crack initiation in the root fitting.

It is evident from detailed microscopic examination that disbond growth involves a process of cracking in the adhesive and progressive separation at the spar/adhesive interface in response to alternating stresses developed during operation. This process of structural change through disbond growth is, in essence, a process of fatigue. The rate of adhesive joint cracking/disbond growth will be affected by the magnitude of the local alternating stresses, the presence of stress concentrating features, the number of stress cycles, the frequency of stress cycles and the effects of the environment (temperature, absorption of moisture) on the polymeric material used as the adhesive.

The nature of load transfer in adhesive joints creates a stress concentration at the end of the joint (end of the spar). This part of the adhesive joint is most sensitive to variations in joint detail that create additional local stress concentrations. Features such as a sharp corner on the spar end, lack of grit blasting surface modification on the spar end surface, and voids in the adhesive near the spar end were evident in an examination of several blades, including the blades from VH-OHA. Increases in local stresses at the end of the adhesive joint through the combined effects of joint stress concentration, stress levels created by helicopter operation, in particular, blade in-plane bending loads and the presence of joint detail stress concentrators, will favour disbond growth.

Research into the behaviour of aircraft structural joints has also revealed that disbond growth in adhesively bonded and bolted joints may be arrested if the bolt adjacent to the disbond can effectively transfer load. Variations in the ability of bolts installed in the inboard bolthole of the spar to root fitting joint may determine whether disbonding is arrested or allowed to proceed to the bolthole.

Disbond growth to the inboard bolthole has one other consequence when a helicopter is operated in a moist environment, in particular, an environment where the moisture contains chloride salts. These environments can cause pitting in the aluminium alloy used in the blade root fitting. The process of pitting is a function of calendar time once the corrosive environment has been able to come into contact with the aluminium alloy. The effect of pitting on the performance of the blade root fitting is most pronounced when the location of the pits coincides with the regions of highest local stress – the counterbore region of the inboard bolthole when disbonding has extended to the bolthole. Pitting in the high stress region of the counterbore will reduce the operational time to fatigue crack initiation. It was evident that pitting corrosion, caused by moisture and chloride salts, contributed to the initiation of fatigue cracking in the inboard bolthole of the fractured blade from VH-OHA.

Identification of each factor that contributes to the initiation and rate of growth of cracking/disbonding in the adhesive and fatigue cracking in the root fitting at the inboard bolthole, provides an opportunity to control or limit these factors so that blade fracture, through fatigue crack initiation and crack growth in the blade root fitting, does not occur within the specified safe life of R22 main rotor blades.

The reliability of visual inspection of the outer surface of blades as a means to detect underlying cracks in blade root fittings is dependent on the process of fatigue crack growth from the fitting across an adhesive bond. Adhesive bond strength is a dominant factor in determining whether cracking extends across the bond or whether disbonding between the root fitting and skin occurs. For the case of the fractured blade from VH-OHA, the adhesive bond strength was such that disbonding between the root fitting and blade skins had occurred. Cracking in the root fitting had not extended into the blade skins; therefore, visual inspection of the outer surface of the blade could not detect the underlying crack in the root fitting.

1 INTRODUCTION

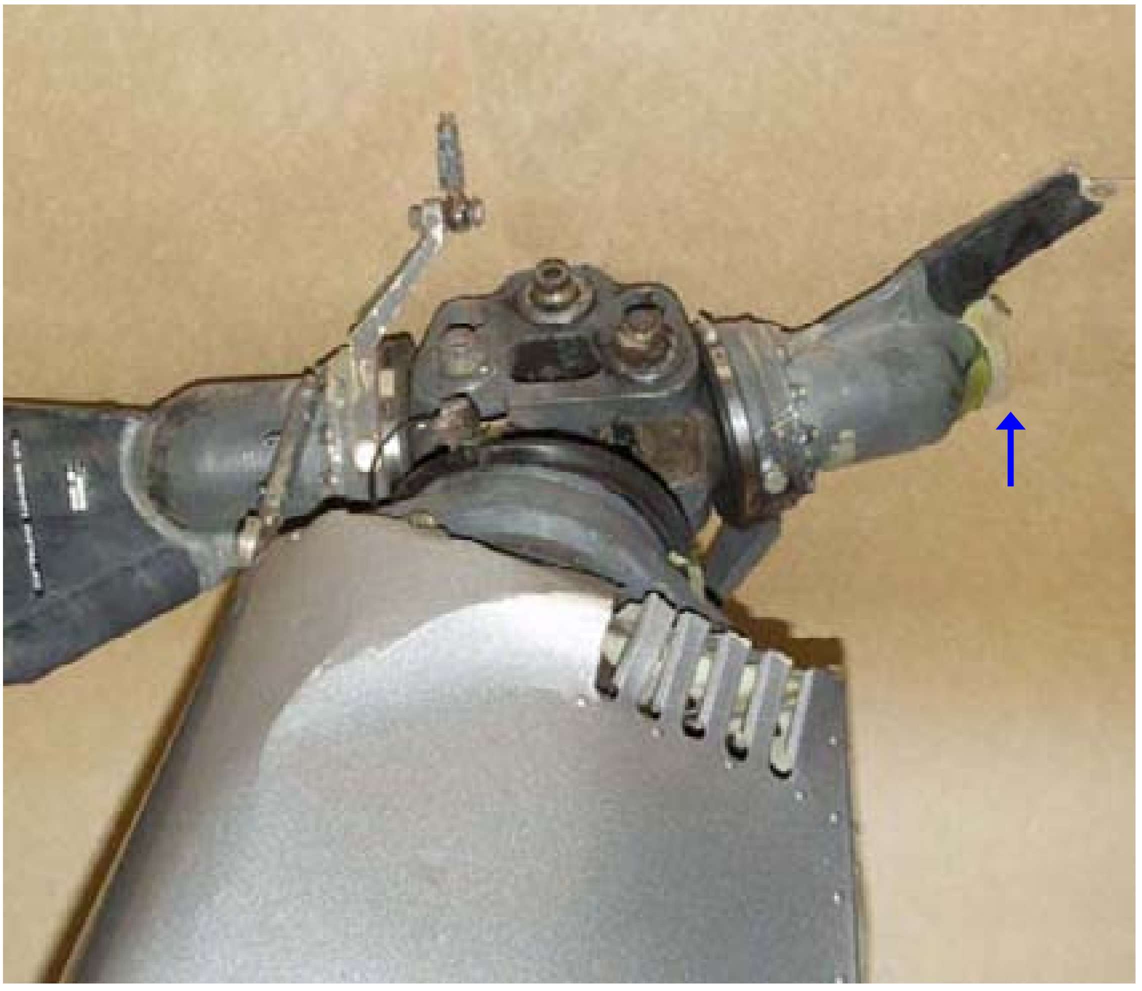
A Robinson R22 Mariner helicopter, registration VH-OHA, was involved in a fatal accident, 13km of NW Camden Airport, while being used for flying instruction.

An examination of the wreckage revealed that one of the two main rotor blades had separated from the helicopter during flight. The separated blade was found some distance from the helicopter wreckage.

The root fitting of the blade, an aluminium alloy forging that accommodates the blade spindle and bearings, had fractured at the innermost bolthole of the root fitting to blade spar joint, see figures 1 and 2.

Reports from witnesses established that the blade had fractured during forward flight.

Figure 1: The recovered main rotor head assembly, VH-OHA



The fracture location (arrowed) is located at rotor station (RS) 10.35, the location of the inboard bolthole in the fitting to spar joint.

2 PHYSICAL EVIDENCE

Both main rotor blades from the helicopter were recovered for detailed examination.

Figure 2: The fractured main rotor blade, VH-OHA

The lower surface of the main rotor blade (painted black) is shown in this figure. The upper surface of the blade is painted white.

2.1 Main Rotor Blade Identification

Main rotor blades are identified by a part number and a serial number. These numbers are recorded on a decal attached to the lower blade surface near the blade root. The fractured blade was identified as p/n A016-2, rev. AD, s/n 6249A, see figure 3. The other main rotor blade fitted to the helicopter at the time of the accident was identified as p/n A016-2, rev. AD, s/n 6283A, see figure 3. The serial numbers of the blades were cross checked with the serial number engraved on a normally enclosed part of the blade structure.

Figure 3: Blade identification decals on the main rotor blades from VH-OHA

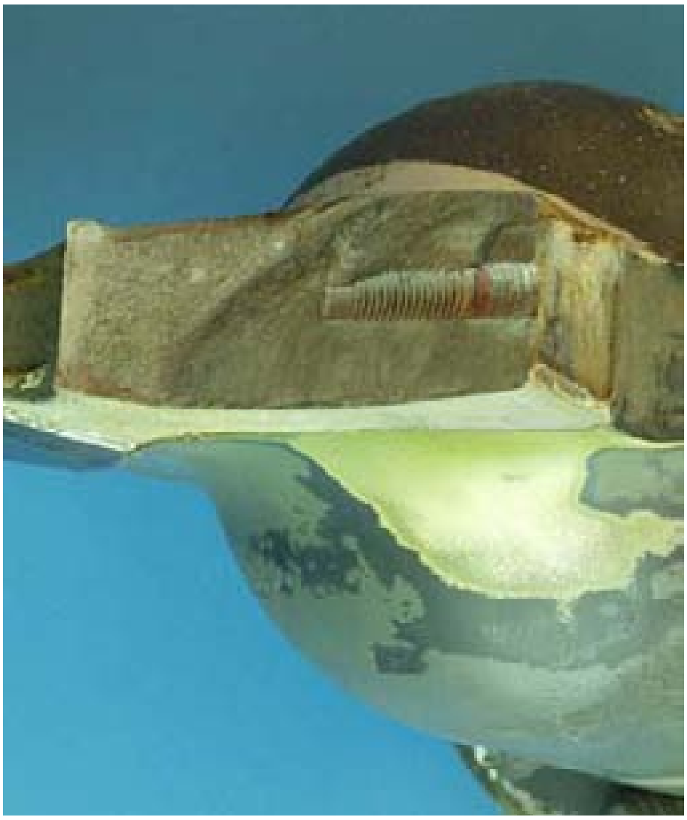
a) fractured blade

b) intact blade

2.2 Classification of Fracture Mechanism

The fracture surface features indicate that fracture occurred as a result of the progressive growth of a crack from the inboard bolthole. The mechanism of crack growth was established to be fatigue, that is, crack initiation and progressive crack extension in response to the development of a number of repeated localised alternating stresses during operation. The features of the fracture are shown in figures 4 and 5.

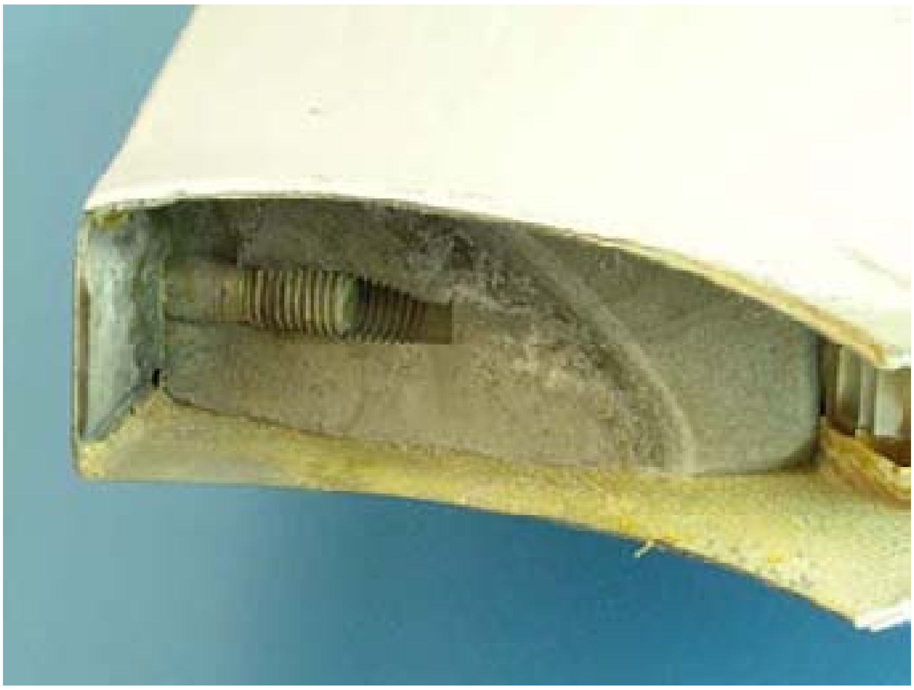
Figure 4: Fracture features, blade s/n 6249A – as recovered



a) The fitting side, or inboard side, of the fracture

b) The blade side, or outboard side, of the fracture

Figure 5: Detailed views of the blade side of the fracture, as received



a) Looking down from the upper surface of the blade

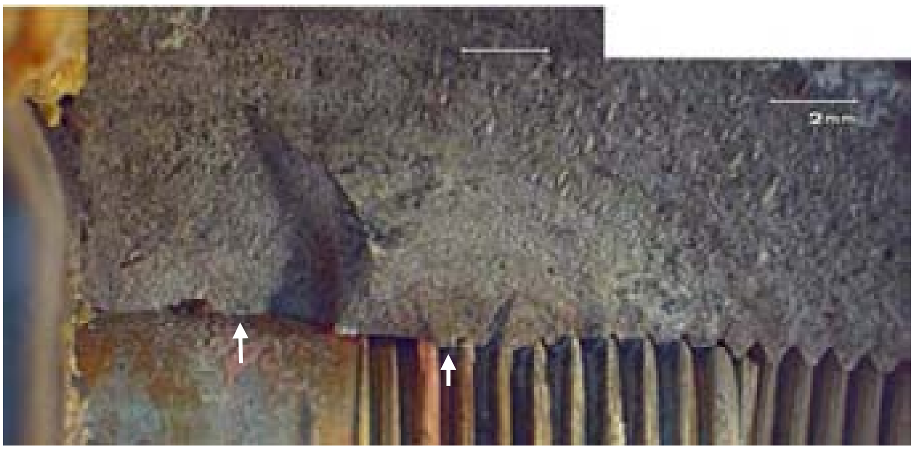
b) Looking up from the lower surface of the blade

2.3 Fatigue Crack Initiation Sites

Fatigue cracking initiated at three sites within the bolthole. All three sites lay in a plane normal to the blade spanwise axis, see figures 6 and 7. One site was located on the lower side of the bolthole counterbore region. The remaining two initiation sites were located on the upper side of the bolthole, one in the counterbore region and one in the threaded region.

It is a normal feature of the manufacture of these main rotor blades (part number A016-2) that the two blind threaded holes at the inboard end of the spar to fitting joint are counterbored to accommodate the increased diameter of the bolt shank. The counterboring process does not remove the entire thread form. A remnant of the thread root remains in the counterbored region.

Figure 6: The sites of fatigue crack initiation, blade s/n 6249A



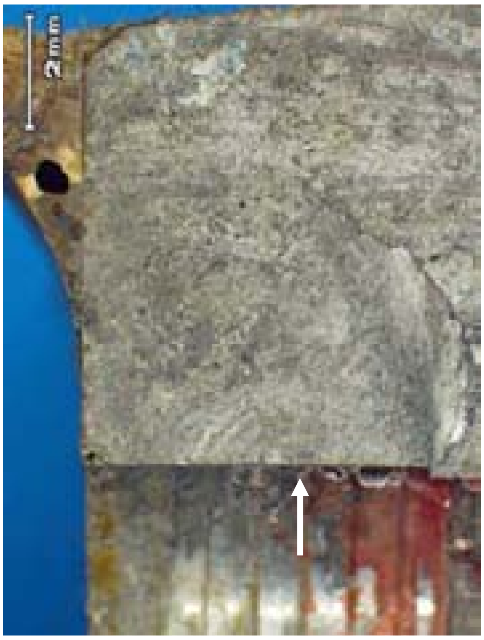
a) Upper side of the bolthole with the bolt in situ. The sites of fatigue crack initiation are arrowed

b) Lower side of the bolthole with the bolt in situ. The site of fatigue crack initiation is arrowed

It is evident that moisture had penetrated the fatigue crack and reacted with the crack surfaces. Following removal of the bolt it was evident that localised corrosion had occurred in the counterbore, see figures 8 and 9.

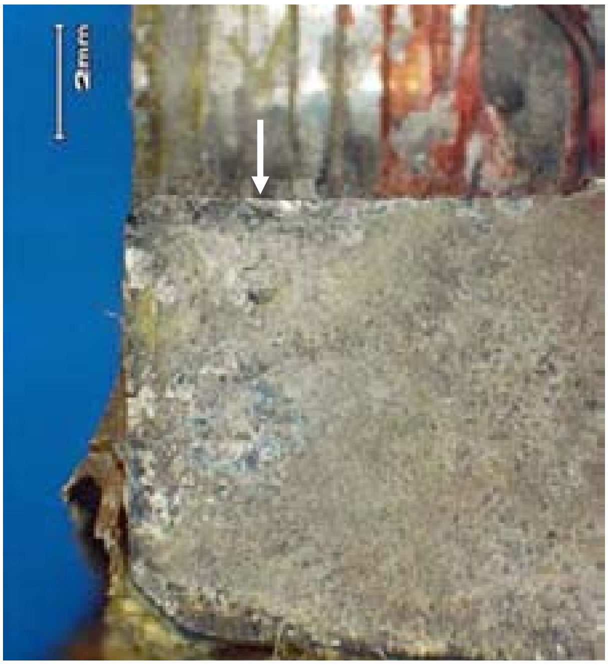
The nature of crack progression markings indicate that fatigue crack initiation occurred first at the lower side of the bolthole, secondly at the thread on the upper side of the bolthole and thirdly at the counterbore on the upper side of the bolthole.

Figure 7: Detailed views of the fatigue crack initiation sites following removal of the bolt and cleaning of the fracture surface



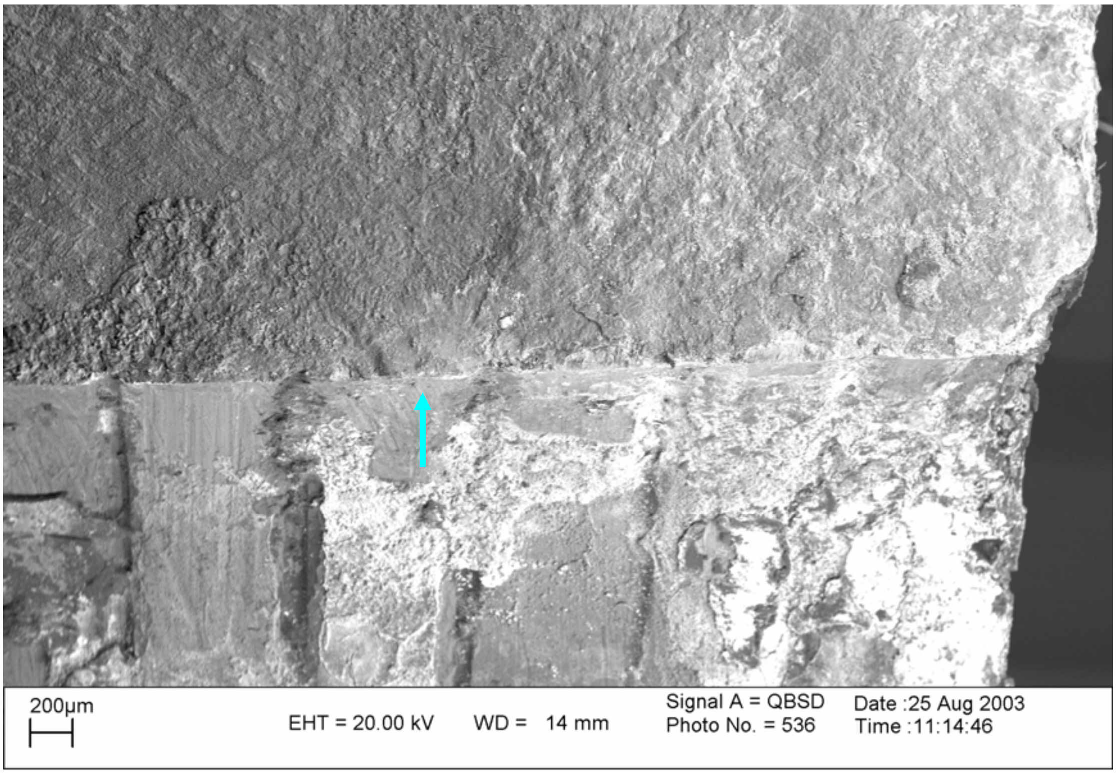
a) Third initiation site, crack growth from this site was influenced by the presence of the crack growing from second initiation site

b) Second initiation site



c) First initiation site

Figure 8: Scanning electron micrograph of the first initiation site



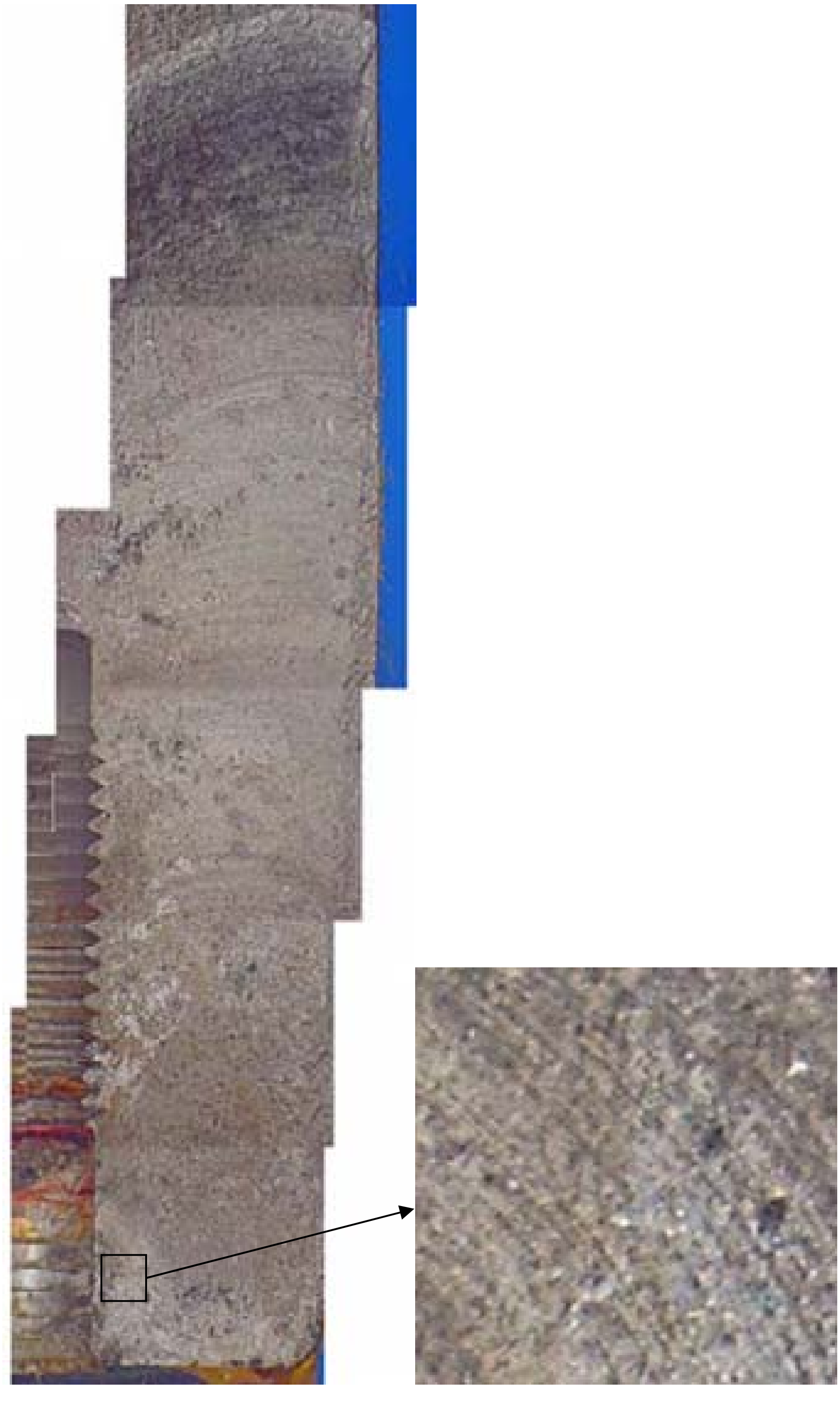
Fatigue crack initiation was associated with localised corrosion. It was not associated with the remnant of the thread.

Figure 9: Extended focus light microscope image of the first initiation site

2.4 Fatigue Crack Growth

A key feature of the fatigue crack progression markings is the visually apparent banding. These bands can be seen in the photomontage of the fracture, see figure 10.

Figure 10: Photomontage of the fatigue fracture (lower half)



Photographed with oblique illumination and processed digitally with a differential contrast filter. The image to the right is a photographic enlargement (approximately 12X)

Fatigue crack progression mark bands indicate that crack growth has occurred as a result of the repeated application of blocks of variable amplitude loads. For the case of aircraft operation, blocks of variable amplitude loads are associated with each flight cycle. One major loading cycle is associated with the generation of lift forces during flight. Superimposed on this load cycle are other loading cycles associated with manoeuvres and the operation of mechanisms. For a helicopter main rotor, the major load cycle is developed with the start of blade rotation, through takeoff and flight, to the stoppage of blade rotation. Inflight alternating loads on the main rotor are developed through manoeuvres and the alternating drag forces on the blades as the blades advance and retreat with each rotation of the rotor during flight.

Although the fine features of the fatigue fracture surface had been masked by the products of the reaction of water and the aluminium alloy fitting, at least 150 bands could be discerned over the length of crack progression. On the basis that each band is likely to be a result of a flight cycle, it appears that crack growth has occurred over a period of at least 150 flights. An estimation of the time of crack growth would require a detailed knowledge of the duration of each flight or major load cycle.

2.4.1 Fatigue crack growth in the blade skin

No fatigue cracking had occurred in either the upper or lower blade skins adjacent to the cracking in the fitting. Paint had been removed from the lower surface of the blade, in the region of the inboard bolthole, during maintenance just prior to the final flight in an attempt to determine if a crack was present in the skin, see figure 11.

This behaviour differed from prior, similar, blade fractures in which fatigue cracking had extended into the blade skin. In place of crack propagation through the adhesive and into the skin, disbonding had occurred between the fitting and skin.

Figure 11: The location and extent of paint removal, blade s/n 6249A

2.4.2 Adhesive disbonding

A feature of the failure of the main rotor blade (s/n 6249A) is the failure of the adhesive bond (disbonding) between various parts of the joint. Regions of disbonding had occurred between the end of the spar and the root fitting and at the inboard ends of the upper and lower blade skins, see figures 12 and 13.

Figure 12: The extent of disbonding between the spar and root fitting, blade s/n 6249A

a) The spar and blade skins were removed mechanically from the remnant of the root fitting attached to the blade. Regions of adhesive fractured during this process can be discerned by colour (light cream) and texture (rough) from regions of inservice bond failure (dark honey, generally smooth surface)

b) Disbonding extends beyond the inboard bolthole

Figure 13: The extent of disbonding between the blade skins and root fitting, blade s/n 6249A

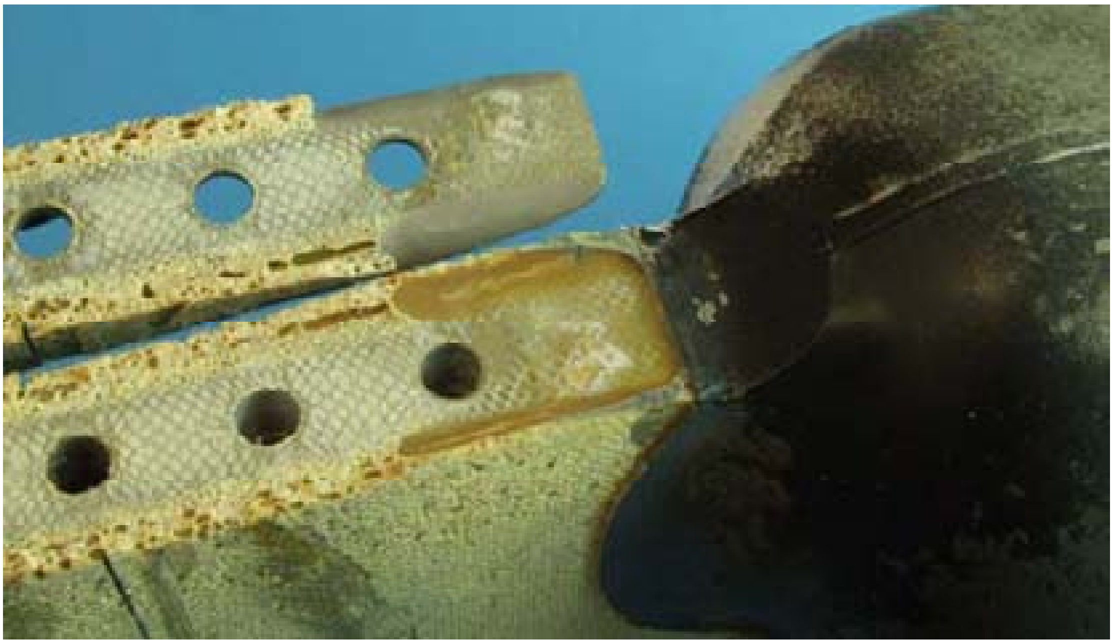
a) Disbonding extends from the inboard edge of the skin to the fracture. The paint film on the root fitting was badly weathered

b) Disbonding extends from the inboard edge of the skin to the fracture

2.4.3 Examination of the intact blade, VH-OHA

The intact blade from VH-OHA was stripped down to reveal the condition of adhesive bonding between the spar, skin and fitting.

Figure 14: Extent of adhesive disbond in blade s/n 6283A (VH-OHA)

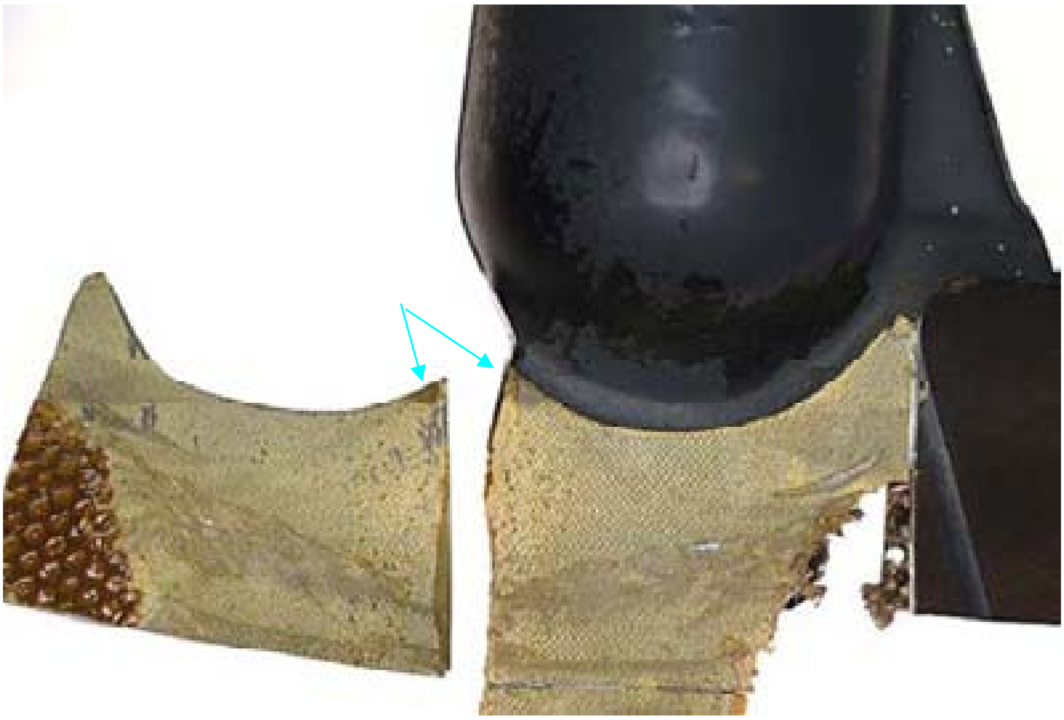


a) Disbonding has extended from the end of the spar to the edge of the inboard bolthole

b) Detailed view of the region between the inboard bolthole and spar end

Figure 15: The nature of the adhesive bond between the upper and lower blade skins, blade s/n 6283A

a)



b) Only a small region of adhesive disbonding was present, arrowed

2.5 Recorded Evidence

The accident investigation team established the operational time of the helicopter and main rotor blades by examining of a variety of operational and maintenance records. Both blades had been fitted to the helicopter at the time of its manufacture, 11 years, 9 months, prior to the accident. It was established that the blades had not exceeded the specified retirement time of 2200 hours. The total operational time of the blades, since new, was found to be 2053.3 hours.

No electronically recorded data, which would assist in the analysis of the main rotor blade fracture, was available.

2.6 Reported Evidence

Reports of main rotor vibration were made in the time prior to the accident and maintenance action addressing the main rotor vibration was carried out prior to the accident flight. This action included checking for the presence of cracks in the skin by the removal of paint from the lower blade skin and balancing the main rotor.

3 EVALUATION

It is important to evaluate the main rotor blade failure against the performance of all parts of the blade failure prevention system. This system is developed and maintained through the phases of design, manufacture, operation and maintenance:

- Design issues centre on the prediction of operating stresses, failure stresses, retirement times and safety factors
- Manufacturing issues centre on the control of variation in blade structure between individual blades
- Operational issues centre on controlling operational stresses so that the design limits are not exceeded
- Maintenance issues centre on monitoring the blade structure for evidence of deterioration, retiring blades at the specified time and restoring those parts of the main rotor system that can be replaced or adjusted, following wear or repair.

3.1 Design Background

Helicopter main rotor components, along with the other dynamic components, are designed on the basis of their response to alternating loads. A major part of the development of a helicopter is the analysis and development testing to predict the development of fatigue cracking in these components.

The fracture of a main rotor blade during operation is a catastrophic event. Because no operational or structural redundancy can be provided, the threat of blade fracture to safe operation is managed by a safe life approach.

The safe life approach to failure prevention in main rotor blades is based on retirement after a prescribed period of operation, a period usually measured in hours of operation. The retirement time is based on the period of operation prior to the initiation of fatigue cracking.

Because of the need to operate under demanding alternating loading conditions while being light in weight, fatigue cracking will develop in Robinson R22, p/n A016-2, main rotor blades with continued normal operation well beyond the specified retirement time. The failure of these blades during development testing occurs at the inboard bolthole.

The nature of helicopter operations, in both the low speed and high speed regimes, leads to a degree of uncertainty in the prediction of main rotor operational loads[1]. The virtual six degrees of freedom of manoeuvrability at low speed complicates the definition of a flight envelope and creates an extremely complex loading environment through interactions with the environment (terrain, earth boundary layer turbulence,

[1] D P Schrage 'A Review of Rotorcraft Structural Integrity Airworthiness Approaches and Issues', Proceedings of the FAA-NASA Symposium on the Continued Airworthiness of Aircraft Structures, Atlanta Georgia August 28-30, 1996, DOT/FAA/AR-97/2, II, July 1997

wake induced from obstacles, etc.). While in the relatively high speed flight regime the aerodynamic load gradient across the main rotor disc covers the entire subsonic, and a substantial part of the transonic regimes. Stalling on the retreating blade side of the main rotor disc and compressibility on the advancing side produces extreme complications for the prediction of alternating loads.

Uncertainty in the prediction of loads combined with variations in blade structure requires the use of large factors of safety in the fracture control plan. The success of the safe life approach relies on predicting operational stresses and structural response and limiting their variation so the bounds of the safety factor are not exceeded.

3.2 Fracture Control Plan

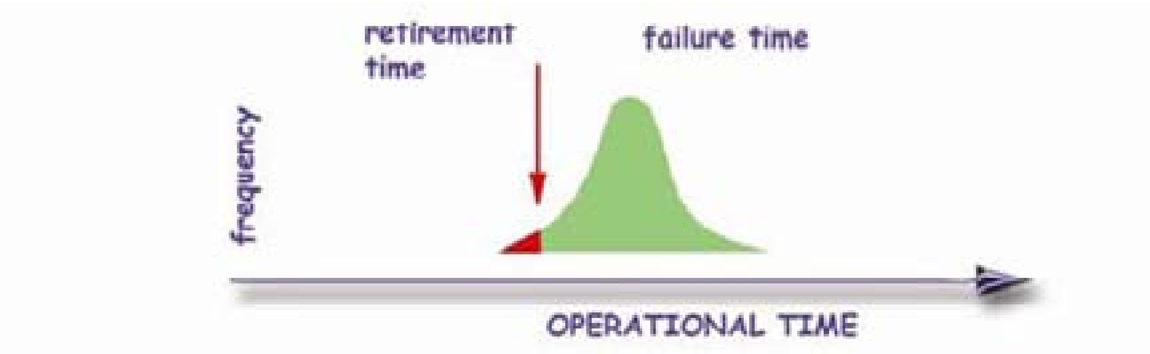
The essence of maintaining structural integrity through the use of a safe life approach is ensuring that components are retired from service before their strength is affected by the development of cracks or other deterioration. The safe life is defined in terms of an operational time.

Because of variations in the response of complex structures to operational loads there will always be a distribution of structural failure times. Fracture control by component retirement must account for variations in failure time by applying an adequate safety factor. The relationship between retirement time and the distribution of failure time is shown schematically in figure 16.

Figure 16: The desired relationship between retirement time and failure time

Deviations from the specified retirement time and/or variations in the structural response of a component that reduce the failure times from those established during design, development and certification testing will increase the probability of structural failure. For a population of main rotor blades, the probability of failure is defined by the overlap of the retirement time with the tail of the failure time distribution. This effect is shown schematically in figure 17.

Figure 17: The relationship between retirement time and failure time that results in the failure of a number of components of a particular configuration



It is important to be aware that the erosion of safety factors will, initially, result in a small number of failures in a population of blades (all blades of the same design). However, it is in the nature of random variations in a population (such as the distribution of failure times for a particular main rotor blade configuration) that the identity of which blade will fail cannot be determined beforehand.

Variations in the response of main rotor blade structures arise from two sources: variations in stresses developed during operation and variations in the strength of the structure (the stress required to cause failure). Structural failure occurs when the operational stress exceeds the structural failure stress. The probability of failure can be defined as the overlap of the distributions of operational stress and failure stress (strength). This concept is shown schematically in figure 18.

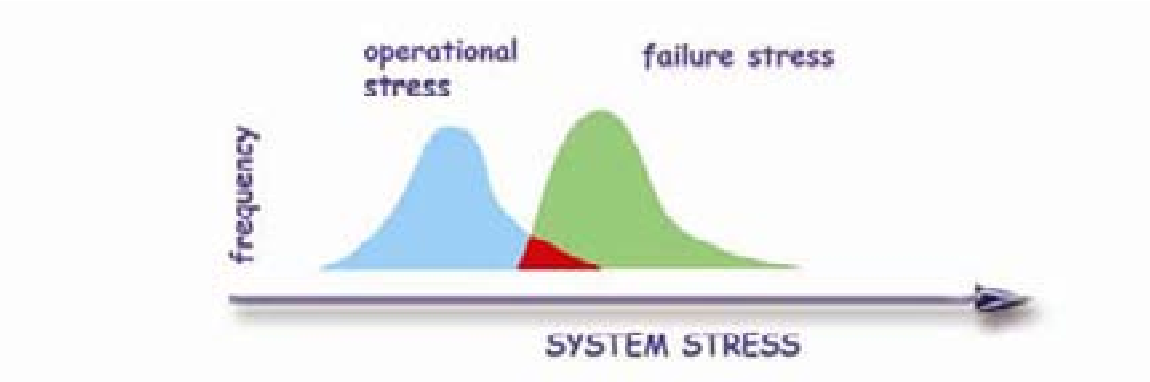
Figure 18: Schematic illustration of the probability of failure under conditions of variable operational stress and failure stress – no failure condition

If the distributions of operational stress and failure stress (system strength) do not overlap the probability of failure is 0. For the case of the main rotor blades, the blade failure stress during normal operation is determined by fatigue crack initiation and growth.

Figure 19: Schematic illustration of the probability of failure under conditions of variable operational stress and failure stress - failure condition

The area of overlap (coloured red) represents the probability of system failure (blade failure)

Figure 20: Schematic illustration of the probability of failure under conditions of variable operational stress and failure stress, failure condition – the effect of skewed distribution



Changes in the nature of distributions, for example skewing the distribution toward the lower end of the failure stress distribution will increase the probability of failure – everything else being equal. Similarly skewing the operational stress distribution toward the higher end of the distribution will also increase the probability of failure.

Structural failure through the initiation and growth of a fatigue crack introduces another level of complexity into the prediction of structural response through its dependency on several interacting factors and time-varying processes:

- The relationship between the magnitude and frequency of alternating stresses, operational loads and time
- The development of structural deterioration through usage and environmental interactions
- The presence or creation of stress concentrating features.

4 OTHER SIMILAR FAILURES OF R22 MAIN ROTOR BLADES

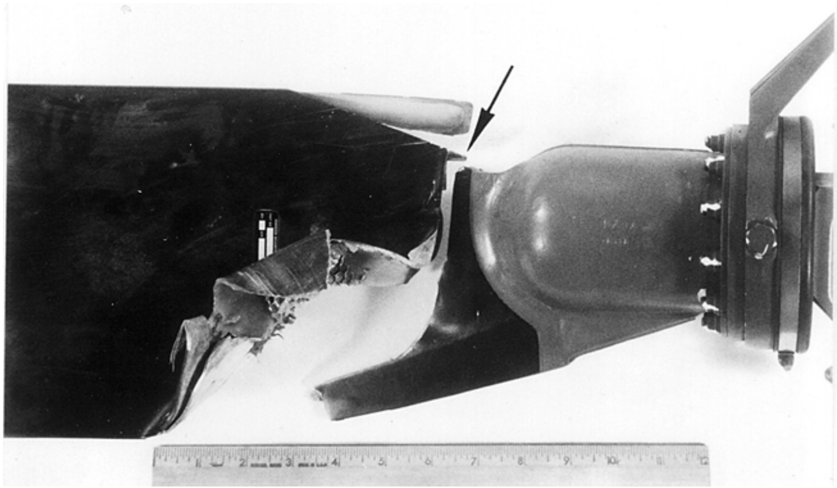
4.1 R22, N9065D, 1 Sep 1981

National Transportation Safety Board, United States of America, investigation number NYC 81-F-A079

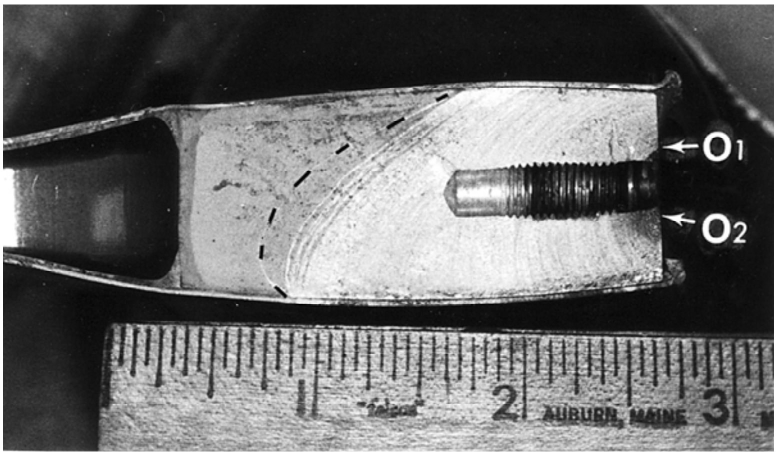
Blade part number: A016-1, revision V   Blade serial number: 370

Time since new: 690 hours

Figure 21: Photographs from NTSB Metallurgist's Factual Report No. 82-32 April 7, 1982

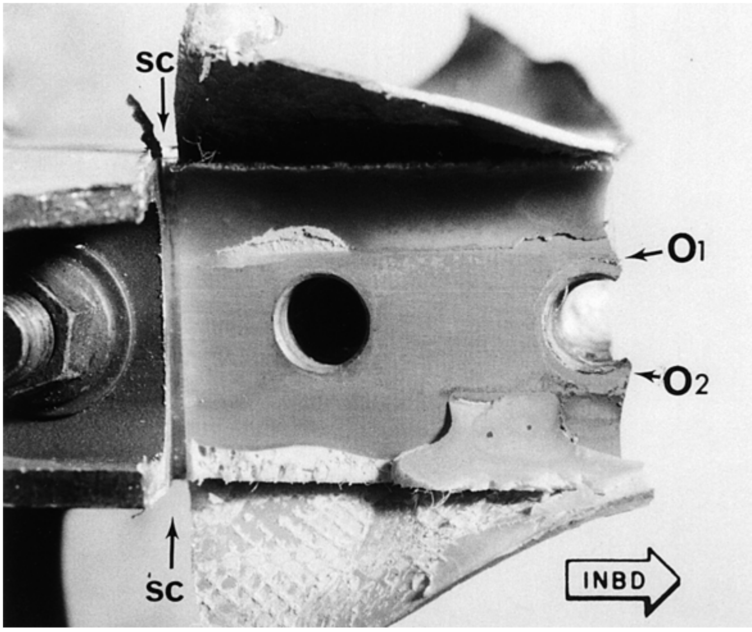


a) Fracture location



b) The root fitting fractured as a result of fatigue crack propagation from the inboard bolthole. In this case fatigue cracking initiated at the interface between the spar and the edge of the fitting (indicated by O_1 and O_2)

Figure 21: continued



c) The spars of blades manufactured to the configuration of A016-1 were not adhesively bonded to the fitting

Investigation of this blade failure revealed that fatigue cracking initiated from two region of fretting damage on the edge of the root fitting.

As a result of this failure the blade was redesigned to the A016-2 part number configuration. The redesigned blade was extensively tested to establish a new life limit.

4.2 R22, VH-HBS, 28 May 1990

Australian Investigation, Occurrence number 199000089

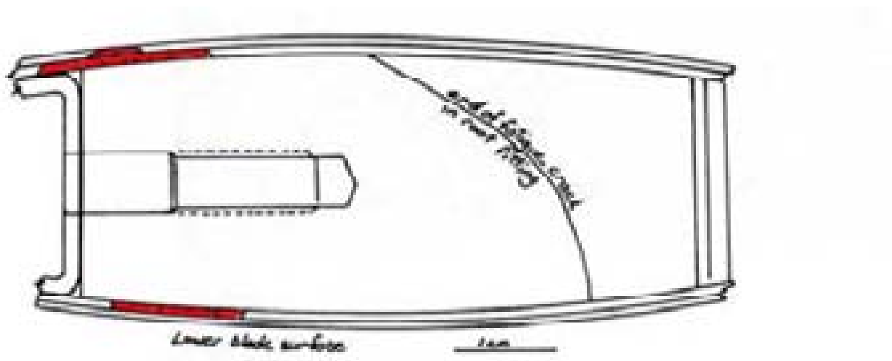
Blade part number: A016-2, revision AB   Blade serial number: 2961

Time since new: in excess of 2257 hours

Figure 22: Photographs from Failure Analysis Report, Civil Aviation Authority Australia Report X10-90

a) Fracture location

b) The blade side of the fracture



c) The regions coloured red show the extent of fatigue crack growth in the blades skins

Figure 22: continued

d) The root fitting side of the fracture, as recovered

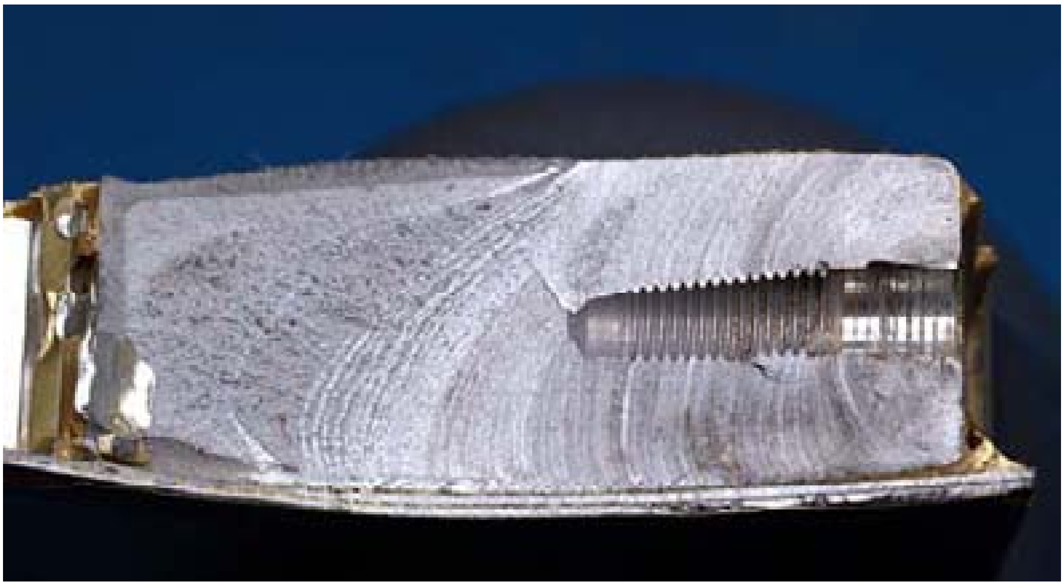
4.3 R22, VH-LDR, 29 July 2000

Australian investigation occurrence number 200003267

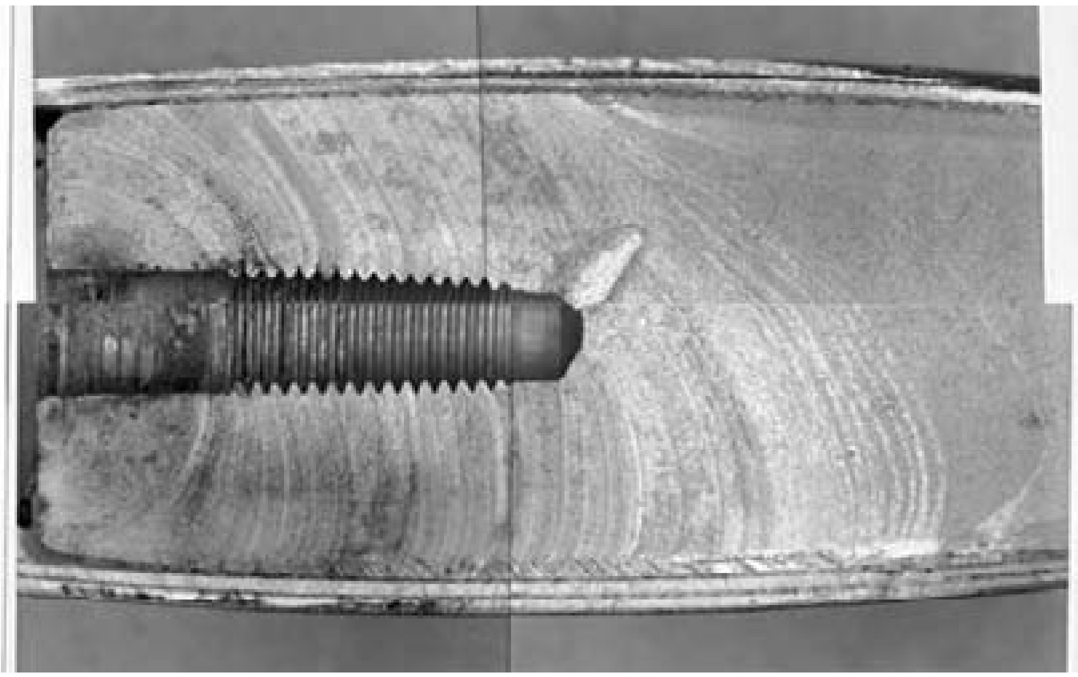
Blade part number: A016-2, revision AG, Blade serial number: 9278B

Time since new: investigation concluded that the time in service greatly exceeded the blade retirement time

Figure 23: Photographs of the fractured main rotor blade

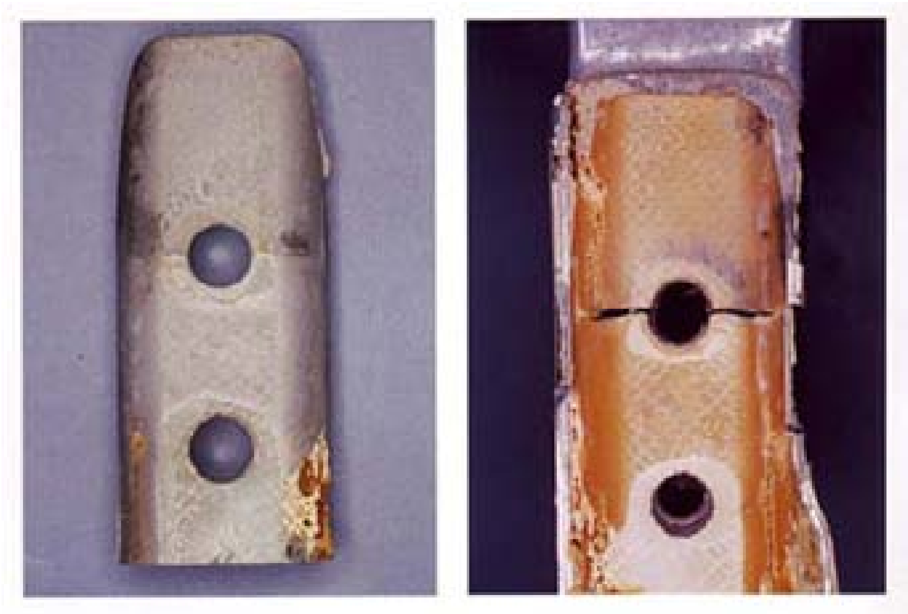


a) The root fitting side of the fracture

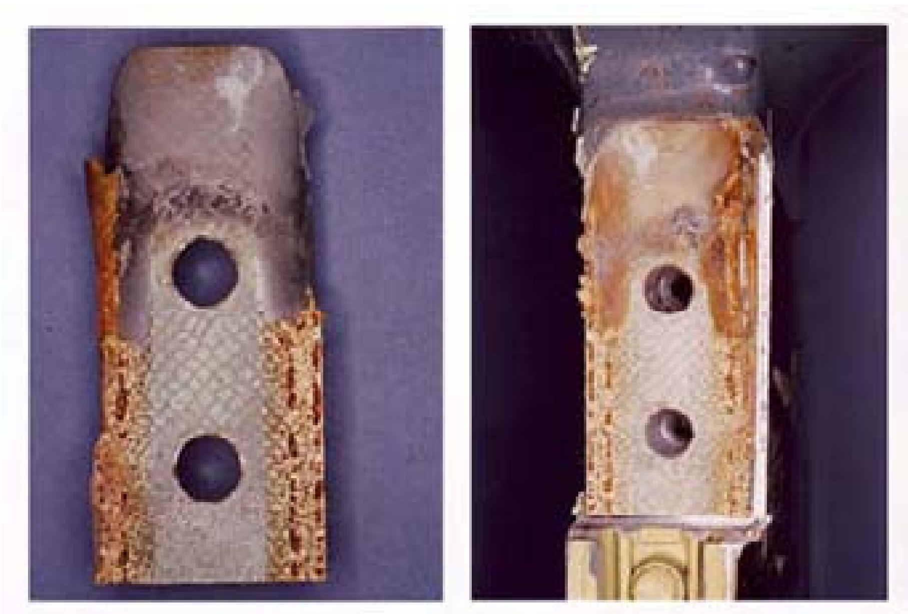


b) The blade side of the fracture

Figure 24: Photographs showing the extent of disbonding between the spar and fitting for both blades fitted to the helicopter (VH-LDR)



a) Fractured blade 9278B



b) Intact blade, s/n 8382B

4.4 R22, 4X-BCM, 29 Feb 2004

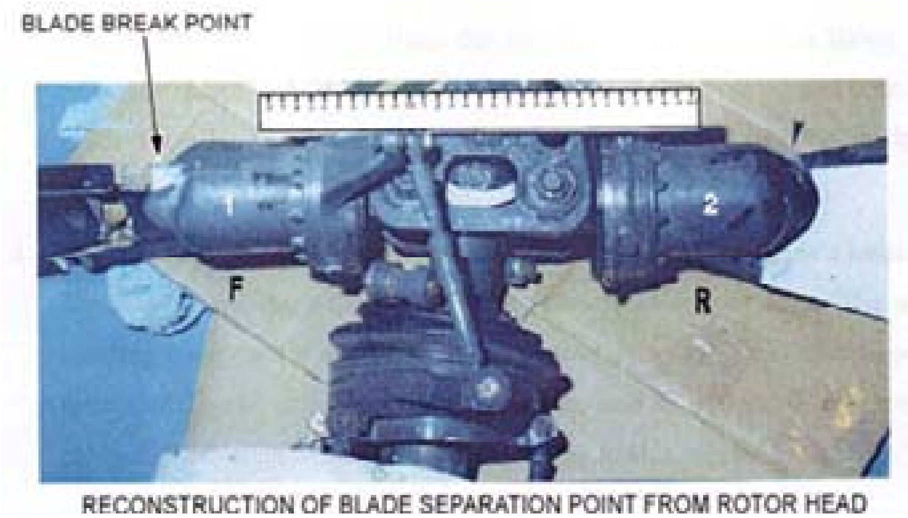
The State of Israel Ministry of Transportation investigation, Acc No 6-04

Blade part number: A016-2, revision AE   Blade serial number: 7055A

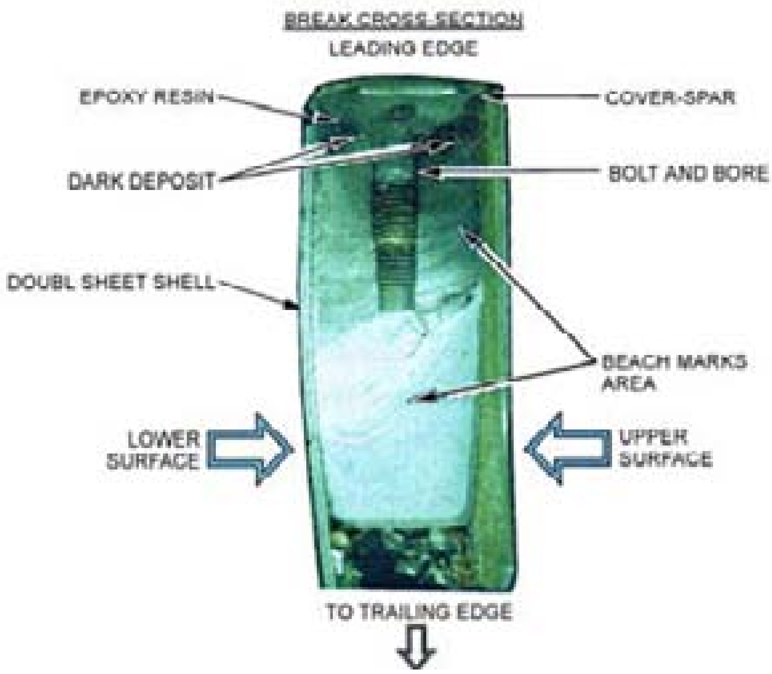
Time since new: approximately 1490 hours, 11.8 years

Fracture occurred during forward flight, power line survey.

Figure 25: Photographs from the accident investigation report



a) Fracture location



b) Fracture surface

Figure 25: continued

c) The root fitting side of the fracture, as recovered

4.5 R22, ZK-HWP, Dec 2004

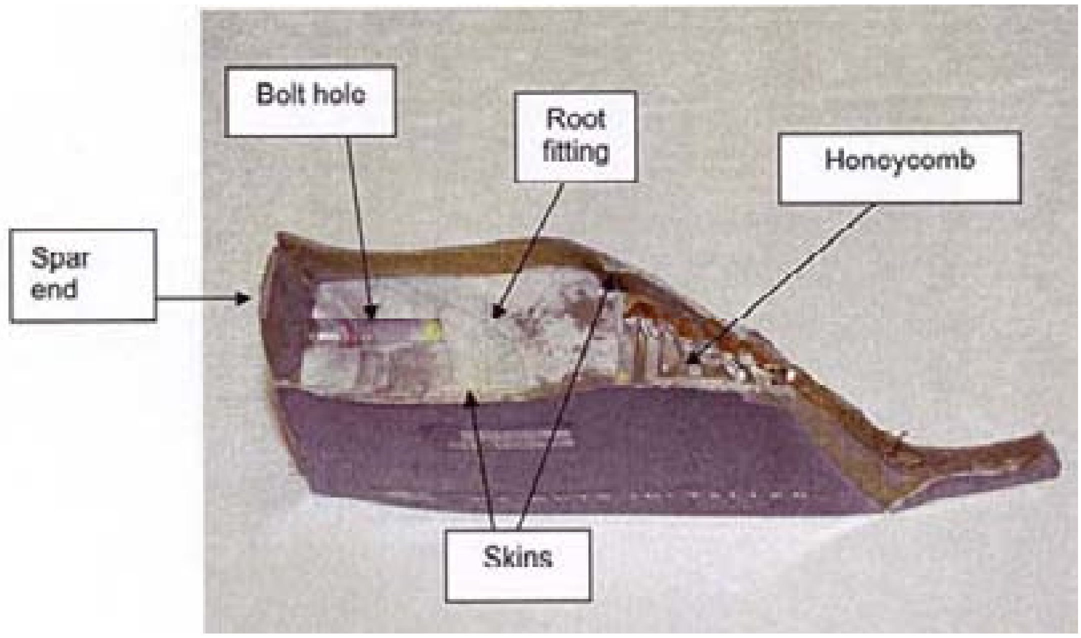
New Zealand Civil Aviation Authority investigation

Blade part number: A016-2, revision AI Blade serial number: 13443A

Time since new: approximately 700 hours, 2.5 years

Fracture occurred just after takeoff, close to the ground.

Figure 26: Photographs from the Specialist Report on the blade failure



a) The blade side of the fracture

b) "The arrows indicate the location of the fracture origins on both sides of the bolt hole very close to the leading edge"

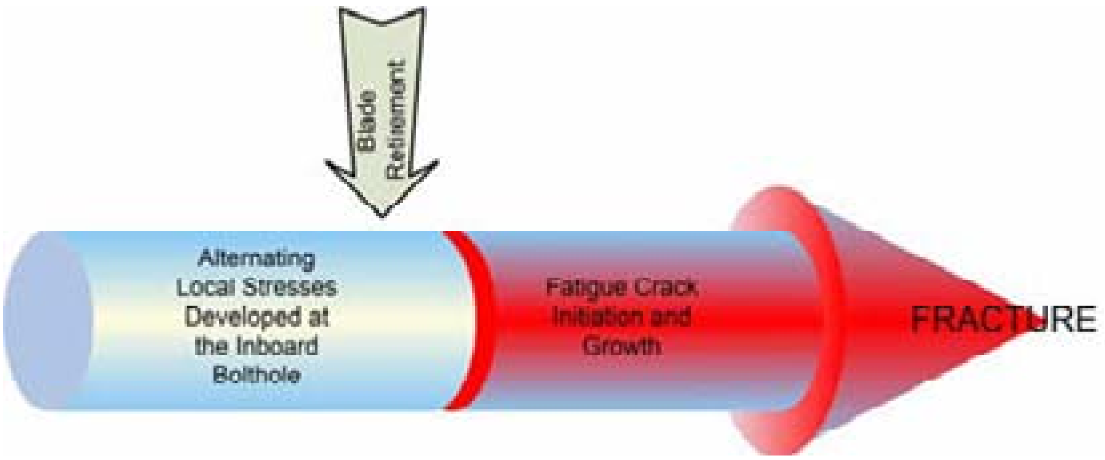
5 ANALYSIS

All sources of variation from the design case need to be explored in order to determine why the main rotor blade fracture control plan failed. Variations may be a simple case of the specified retirement time being exceeded or be associated with complex, time-varying processes that determine the initiation of fatigue cracking.

5.1 Variability in Retirement Time

At the time of the accident involving VH-OHA, two retirement times were specified for A016-2 main rotor blades – 2200 hours in operation or 12 calendar years since installation. The limit on operational hours is associated directly with the process of fatigue cracking while the limit on calendar time is used to avoid general deterioration (eg breakdown of corrosion protection schemes) that occurs with exposure to an operating environment.

Figure 27: Schematic timeline showing the relationship between blade retirement time and fatigue crack initiation time



An extensive investigation of operational and maintenance records was carried out to establish the operational life of the main rotor blades fitted to VH-OHA. The results indicated that the fractured blade had been in service for 2053.3 hours and 11 years, 9 months.

In comparison, investigations of previous blade failures (VH-HBS, VH-LDR) indicated that the operational retirement time had been exceeded.

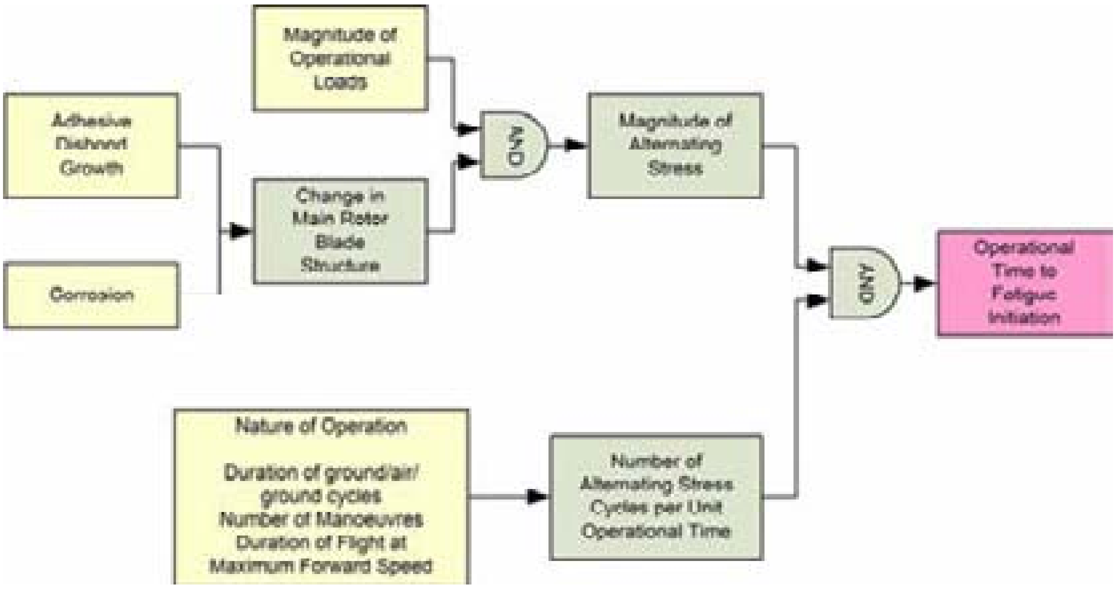
However, two instances of blade fracture have occurred at times within the specified retirement time (Israel, 4X-BCM; New Zealand, ZK-HWP).

5.2 Variation in Fatigue Crack Initiation Time

Fatigue crack initiation is dependent on the magnitude and number of alternating stress cycles imposed on a component. There is an inverse relationship between the magnitude of the alternating stress cycle and the number of stress cycles. An increase in stress cycle magnitude results in a decrease in the number of stress cycles to crack initiation. Fatigue crack initiation is not directly dependent on operational time.

Various factors, some time varying, may affect the magnitude of alternating stresses in a component and other factors may affect the number of alternating stress cycles imposed on a component per unit operational time, see figure 28.

Figure 28: The relationship between factors that may influence fatigue initiation and operational time to fatigue initiation.



The alternating stresses created in a main rotor blade, at the inboard bolthole, have their origin in the loads imposed on the blade during the operation of the helicopter. Three types of blade loading are significant, axial loading, out-of-plane of the rotor, bending and, in-plane of the rotor, bending.

Axial loads on a main rotor blade are created by the rotation of the main rotor assembly and are related to the centrifugal forces created by rotation. The magnitude of this load will vary with main rotor revolutions per minute (rpm). A major alternating stress cycle is created within the blade each time the rotor is accelerated from rest to its operating rpm and then decelerated to rest at the end of the flight cycle. This stress cycle is commonly referred to as the ground-air-ground cycle.

Out-of-plane bending loads on a main rotor blade are created by the lift forces generated by the rotation of the main rotor and result in upward bending of the outer sections of the main rotor blades (main rotor coning). The magnitude of this load will vary with the magnitude of the lift forces on the rotor blade which in turn is a function of operational load factor (helicopter weight plus manoeuvre load factors). A major stress is created with each flight plus each manoeuvre during a flight.

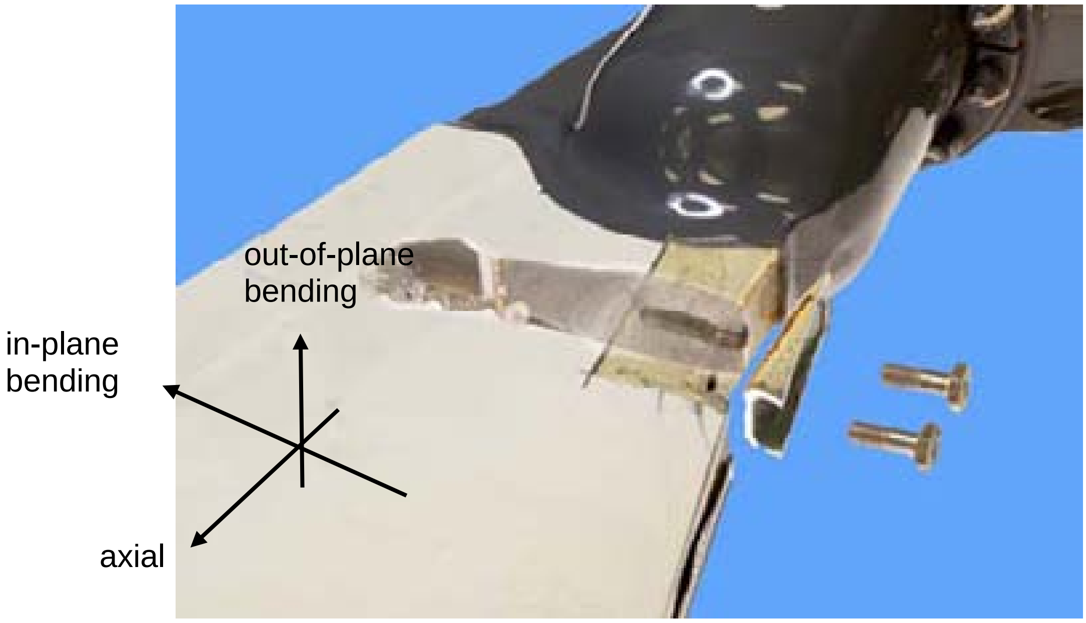
In-plane bending loads on a main rotor are created by the drag forces generated by main rotor rotation and helicopter flight. The magnitude of drag loads will vary with blade angle of attack and relative airspeed over the blade. Stress cycles may occur several times in each revolution of the main rotor because of the effects of advancing and retreating blades.

Each loading condition will affect the distribution of stress in the root fitting at the location of the inboard bolthole. Loading along the axis of the blade will result in a uniform distribution of stress in the fitting. Blade bending loads will result in the creation of stress gradients. Out-of-plane bending from lift forces will result in higher stresses in the lower section of the blade (the side closest to the lifting surface) while in-plane bending from drag forces will result in higher stresses in the section of the fitting close to the blade leading edge.

It is important to note that, for the two bladed rotor system used in the Robinson R22 main rotor design, the drag forces created during helicopter operation are accommodated by the blade root and hub structures. No lead-lag hinges are provided. The blade root structure is required to be designed so that the stresses resulting from drag forces do not result in fracture during the operational life of the blade[2].

A number of observations of the nature of fatigue crack initiation and growth in R22 main rotor blades indicate that stress gradients are present in the root fitting at the location of the inboard bolthole. The asymmetry of the crack front with respect to the chordwise axis of the blade - greater crack extension in the lower half of the blade – is consistent with higher stresses in the lower half of the blade. Fatigue crack initiation in the counterbore region close to the forward edge of the fitting as opposed to crack initiation from sites further down the threaded hole may also be consistent with the presence of higher stresses near the blade leading edge.

Figure 29: Composite photograph showing the orientation of the plane of fatigue cracking with respect to the blade axes and the direction of blade loads



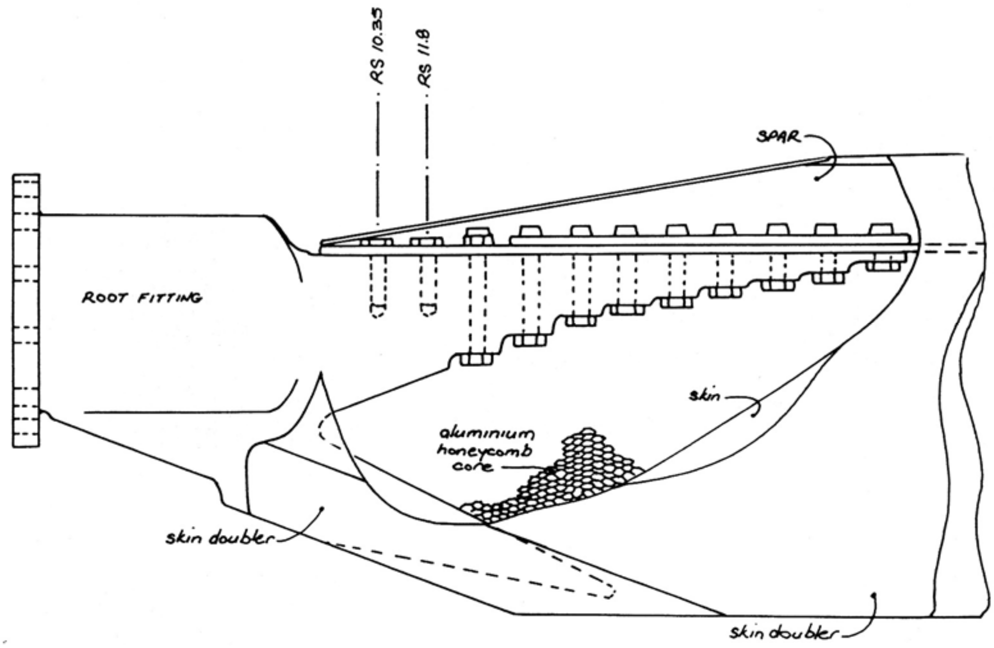
---

[2] R W Prouty, 'Practical Helicopter Aerodynamics', reprints of 'Aerodynamics' columns that have appeared in Rotor & Wing International magazine from 1979 to 1982

5.2.1 Change in blade structure at the spar to fitting joint

The initiation and growth of the fatigue crack that resulted in blade fracture occurred within the joint between the blade spar and root fitting. This joint is formed during manufacture by bolting the spar to the fitting with a series of eleven bolts/studs. In addition to the bolts, the spar and the fitting are adhesively bonded to each other and the overlapping blade skins.

Figure 30: Schematic illustration of the blade spar to root fitting joint



All of the loads imposed on the main rotor blades during operation are transferred to their root fittings through the joint between the spar and fitting. The nature of load transfer, stress gradients and stress concentration at the location of the inboard bolthole depend on the nature of the joint in the vicinity of the bolthole.

The joint between the blade spar and root fitting is not simple. An analysis of load transfer, stress gradients and stress concentration requires an understanding of the features of a bolted joint, the features of an adhesively bonded joint and the features of an adhesively bonded and bolted joint.

Bolted Joint

In a bolted joint, loads are transferred through each bolt bearing against the two components of the joint. If the clamping force of the bolts is sufficient to create a friction force between the mating surfaces of the two components, load will be transferred over the area of surface contact (this type of load transfer is known as bypass load transfer).

The site of the inboard bolthole is the location of highest stress for the root fitting in the spar to fitting joint. At this point all of the loads from the main rotor blade have been transferred to the root fitting. In addition to the joint load transfer effects, the bolthole creates a physical stress concentrator. Two conditions determine the effect of the bolthole as a stress concentrator. Firstly, if the bolt is preloaded to an extent that generates a high frictional force between the spar and the surrounds of the bolthole then the stress concentration effect of the hole is minimised. Secondly, if there is no preload in the bolt and consequently no frictional force between the spar and the surrounds of the bolthole then the hole will act as a stress concentrating feature.

Adhesively Bonded Joint

The inclusion of a structural adhesive between all of the surfaces in the spar to root fitting joint, changes the nature of load transfer from the spar to fitting. In this case the joint displays the features of an adhesive joint. Loads are transferred by shear through the adhesive layer between all of the joint surfaces. It is in the nature of adhesive joints that the stress distribution is not uniform throughout the joint. The regions of highest stress are at the ends of the joint. The magnitude of stress concentration at the ends of an adhesively bonded joint is highly dependent on the local stiffness of the adherends. Design strategies employed to minimise the stress concentrations at the end of adhesively bonded joints are based on controlling local stiffness through tapering the adherends[3]

When adhesively bonded joints are subjected to alternating stresses, regions of disbonding may develop at the joint edge and extend, progressively with repeated stress cycles, into the joint. Disbond formation and growth will be affected by the magnitude of stress concentration at the joint edge and the effects of the environment.

The magnitude of stress concentration at the end of the spar will be a function of the magnitude of the applied loads and the detailed geometry at the spar end. In addition to the effects of tapering on local stiffness, disbond growth may also be influenced by other smaller geometric features such as, voids in the adhesive, initial small regions of disbond and sharp edges etc.

Moisture absorption and higher temperatures can affect the strength of the adhesive polymer. The interfaces between the adhesive and the metal adherends are particularly important features in determining the strength of the joint and its resistance to disbond growth. Moisture absorption and penetration is known to have a detrimental effect on the strength of metal/adhesive interfaces[4].

Bonded and Bolted Joints

Extensive analysis of aircraft structural joints has been undertaken by Hart-Smith[5]. On the issue of joints that are assembled by both bolting and adhesive bonding, it was concluded that bonding and bolting do not work together in transferring load through the joint. The bonded load path is, generally, much stiffer than the load path through the bolts. The bolts are useful as assembly aids but remain essentially unloaded while the bond is intact. If disbonding to the first bolt in a joint occurs, only this bolt will be

---

[3] W S Johnson, L M Butkus, 'Designing for the Durability of Bonded Structures', Proceedings of the FAA-NASA Symposium on the Continued Airworthiness of Aircraft Structures, DOT/FAA/AR-97/2, I, July 1997, p149

[4] Nak-Ho Sung, "Moisture Effects on Adhesive Joints" , Engineered Materials Handbook, Vol 3, p622, ASM International, 1991

[5] L J Hart-Smith, 'An Engineer's Viewpoint on Design and Analysis of Aircraft Structural Joints', International Conference on Aircraft Damage Assessment and Repair, Melbourne Australia August 26-28, 1991, Douglas Aircraft Company Paper MDC 91K0067

fully loaded. Of significance to this investigation is the observation that fully effective load transfer through the bolt adjacent to the disbond protects the remaining adhesive bond from disbonding.

Observations of disbonding

Extensive disbonding had occurred in both main rotor blades from VH-OHA and in other previous failures of A016-2 blades (VH-HBS and VH-LDR) and the subsequent blade failures in Israel and New Zealand. In each case, disbonding in the failed blade extended to and past the inboard bolthole, while in the case of the intact blade of each blade pairing, disbonding did not extend to the bolthole.

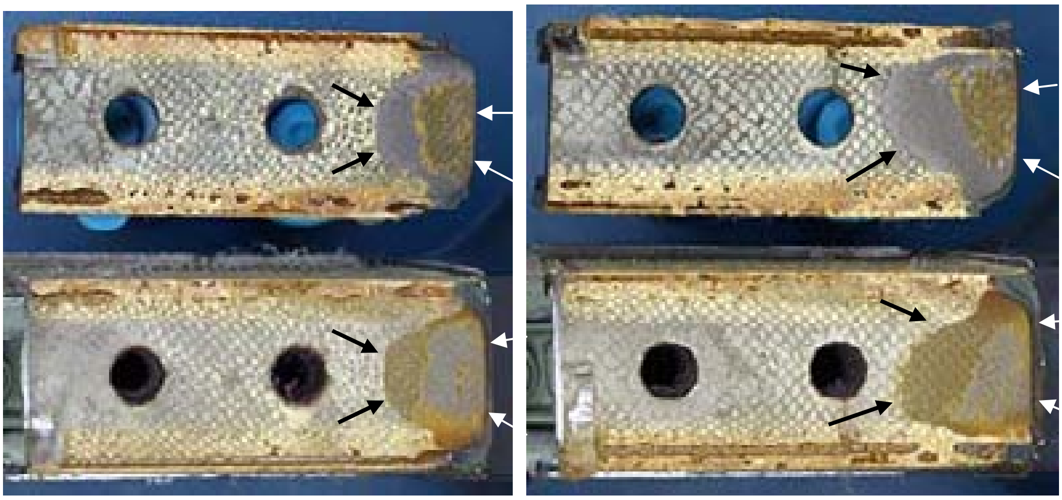
5.3 Adhesive Disbonding Survey

In order to examine the effect of operating environment (eg temperature, humidity, flight profile) on the development of disbonding in the adhesive joint between the inboard end of the blade spar and blade root fitting, a number of blades were stripped down.

5.3.1 Sample from Australian operators

Initially, 10 blades from a variety of Australian operators were examined. The blades had been retired from operation for various reasons and represented a range of operating times and climatic regions. The intact spar/fitting joints were stripped down to expose the nature of adhesion between the spar and fitting, and both blade skins (upper and lower) and the fitting. The strip-down protocol is attached in Appendix A.

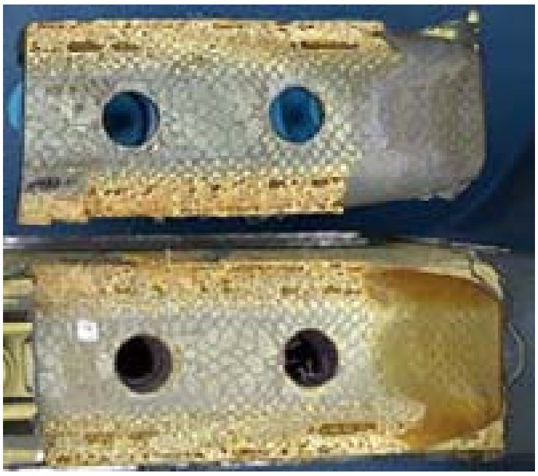
Figure 31: The extent of disbonding between the root fitting and inboard end of the blade spar – helicopters used in cattle mustering operations



a) VH-UXF, 12616B VH-UXF, 12587B

Regions of disbond are characterised by the darker, honey brown colour and regions were the metallic surface of the spar is exposed. Regions were the intact bond was fractured during the strip down process are characterised by a light cream colour and a rougher appearance. The cross-hatched appearance of the adhesive is an effect of the scrim cloth used to create the adhesive film. The regions of adhesive disbond are arrowed.

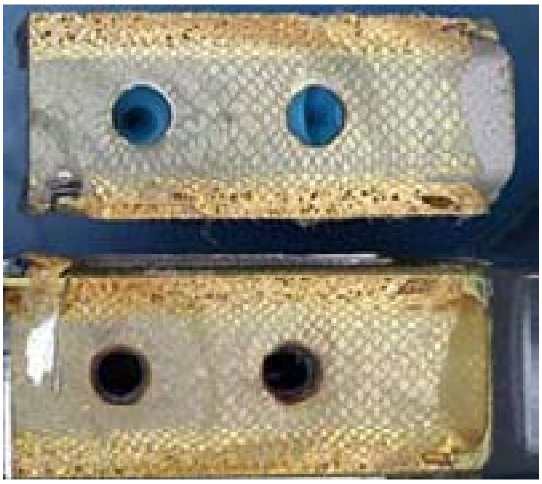
Figure 31: continued

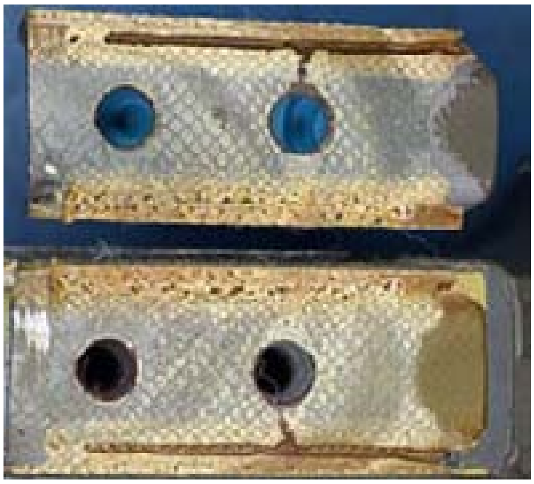


b) VH-LOT s/n 8411B

VH-LOT s/n 8414B (operational time

approximately 1700 hours)

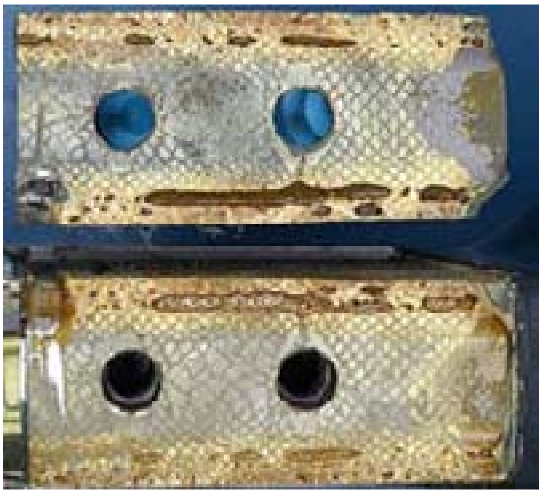




c) VH-HCF s/n11303A

VH-NWJ, s/n 4480, 1400 hours TIS

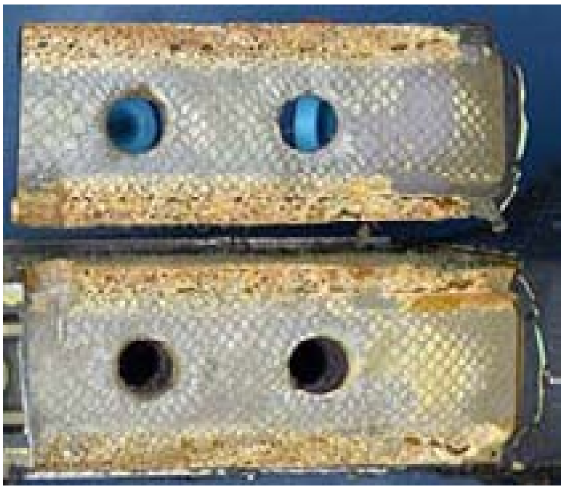
Figure 32: The extent of disbonding between the root fitting and inboard end of the blade spar – helicopters not used in cattle mustering operations



VH-HVX s/n74810 1693.4 hours TIS

VH-HCO, s/n 11277B, 203 hours TIS

Figure 33: The extent of disbonding between the root fitting and inboard end of the blade spar – helicopter operational details unknown



s/n 8595B

s/n 8603B

blades 8595B and 8603B were fitted to the same helicopter, registration unknown

5.3.2 Sample from worldwide operators

In response to ATSB Safety Recommendation R20030186, Robinson Helicopter Co. examined a sample of 51 blades under the supervision of the US National Transportation Safety Board (NTSB) to determine if regions of adhesive disbond, similar to that observed in the Australian sample, were present in blades from other operating environments. The examination protocol was provided by the ATSB, see Appendix A.

The main rotor blades examined were drawn from blades returned to the factory following retirement or damage. This sample covered operations in many parts of the world and many types of operation. It also covered a wide range of operating and calendar times: from 12 years to less than 6 months, and from 2265 hours to zero hours (blade damaged in shipment). Refer appendix B.

It was reported that almost all blades exhibited an area of adhesive disbond extending from the end of the spar toward the bolthole. It was concluded that there was no clear correlation between the calendar time or service time and extent of disbond. Interestingly, the two blades (s/n 14008B and s/n 14011B) that had been damaged in shipment (zero operational time) exhibited a small region of disbonding (approximately 0.5 mm).

5.4 Processes of Structural Deterioration

At the start of operation, two things happen in the main rotor blades of R22 helicopters, alternating stresses are created at the inboard bolthole in the blade root fitting and alternating stresses are created at the end of the spar to fitting adhesive joint. With continued operation these alternating stresses can result in crack initiation, crack growth and disbond growth. Each process of structural deterioration is dependent on the magnitude and number of stress cycles combined with the effects of the operating environment. In practice, these processes of structural deterioration are limited by retiring the blades after a specified operational period.

In the initial stage of operation, these two processes will proceed independently. However, if disbond growth extends to the inboard bolthole, the magnitude of the alternating stresses at the bolthole will increase because of the change in load transfer around the bolthole. This increase in alternating stress magnitude will decrease the time to fatigue crack initiation and growth. In addition, corrosive materials in the operating environment will be able to affect a critical region of the blade.

It is apparent from the evaluation of several accidents involving blade fracture that disbond growth to the inboard bolthole is the key process leading to fatigue crack initiation, growth and final fracture during operation. Without disbond growth to the inboard bolthole, blades paired with the fractured blades have not developed fatigue cracks despite being subjected to the same operating loads and the same operating environment.

The factors that determine whether disbonding will occur during operation and those that determine the rate of disbond growth to the inboard bolthole can be grouped into three categories; initial conditions, operating conditions and mitigating conditions.

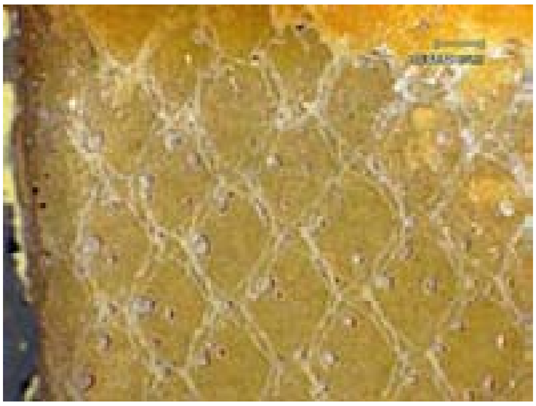
The identification of the variables that affect disbonding provides an opportunity to control or limit these variables and restore the blade structural safety margin.

5.4.1 Initial conditions

In addition to the possibility that a small region of disbond may be present from manufacture, as illustrated by the disbond discovered in blades that had not been exposed to operational loads (blades s/n 14008B and s/n 14011B), other variations in the adhesive layer near the spar end were discovered. In cases where there was extensive disbonding, numerous voids were present in the adhesive layer, see figure 34.

Figure 34: Examples of bondline voids at the end of the spar/fitting adhesive joint

VH-LDR, s/n 9278B



VH-LOT, s/n 8414B

Note; the cross-hatched feature is the scrim adhesive carrier cloth.

Flaws in the bondline, particularly those in the highly stressed region at the end of the adhesive joint, will act as sites of stress concentration and may act as sites of crack initiation under conditions of alternating stress[6]

5.4.2 Operating conditions

Close examination of the regions of disbonding between the spar and root fitting of fractured blades, and those stripped down during the Australian survey, revealed that crack growth in the adhesive from the end of the spar had occurred. While crack growth occurred close to the underside of the spar, adhesive remained bonded to both the spar and fitting, see figures 35 and 36.

[6] 'Fatigue and Fracture', E Sancaktar, 'Adhesives and Sealants', Engineered Materials Handbook, ASM International, 1990, USA

Figure 35: The distribution of adhesive material bonded to the spar, VH-OHA

a) VH-OHA fractured blade, adhesive bonded to the spar, at the spar end is arrowed

b) VH-OHA intact blade, adhesive bonded to the spar, at the spar end is arrowed

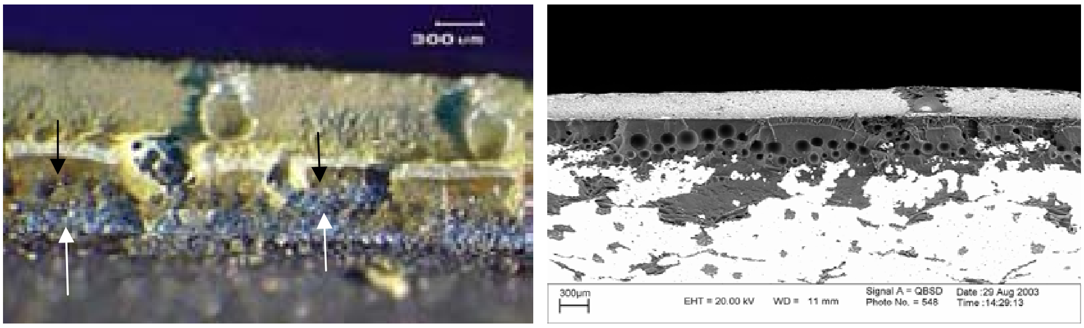
Figure 36: The distribution of adhesive material bonded to the spar, VH-LDR

a) VH-LDR fractured blade, adhesive bonded to the spar, at the spar end is arrowed

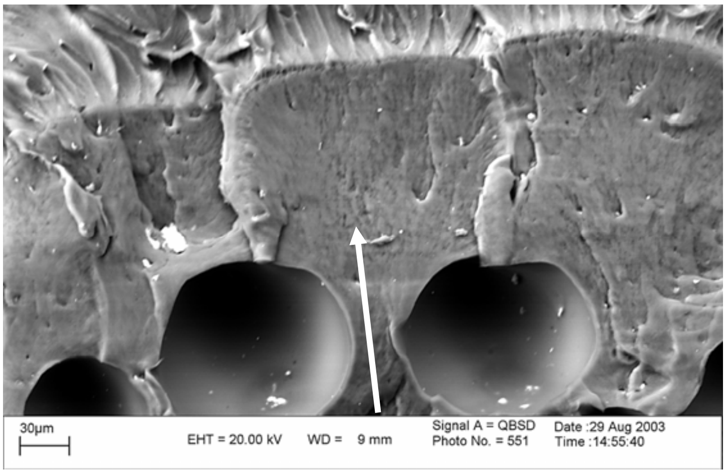
b) VH-LDR intact blade, adhesive bonded to the spar, at the spar end is arrowed

Additional evidence of crack growth under conditions of alternating stress (fatigue) was found in the adhesive fillet at the end of the spar, see figure 37. The crack surface features in this region are not subjected to relative movements that obliterate fine detail as is the case with crack growth into the spar/fitting joint.

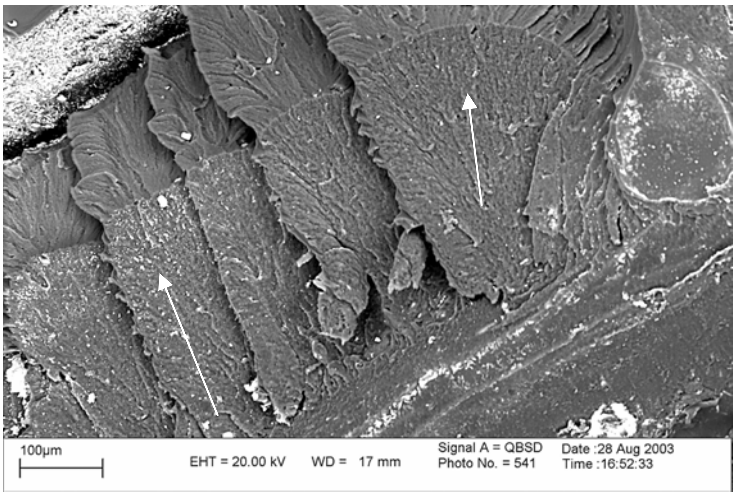
Figure 37: Examples of fatigue cracking in the adhesive fillet at the end of the adhesive joint



a) VH-HCO, s/n 11277B, the extent of fatigue cracking is arrowed (light micrograph at left, scanning electron micrograph at right)



b) VH-HCO, s/n 11277B, the direction of fatigue crack growth is arrowed



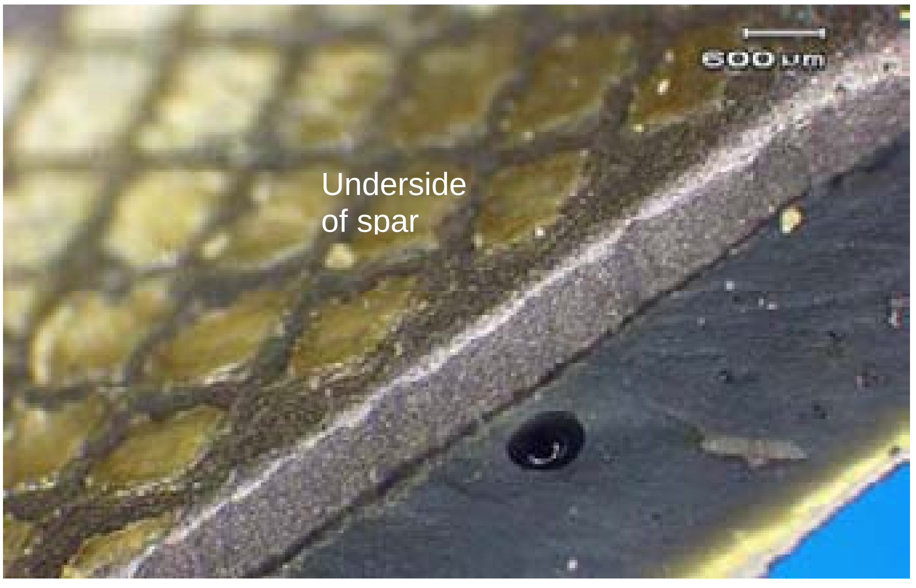
c) VH-LOT, s/n 8411B, the direction of fatigue crack growth is arrowed

Fatigue cracking in the adhesive bonding material will be affected by the magnitude and frequency of alternating stresses created by operational loads. It will also be affected by the moisture absorption and the presence of stress concentrating features in the joint.

During the examination of fractured blades and those surveyed for disbonding variation were observed in the geometric features and surface condition at the end of blade spars. An example of some of these features, sharp edges, and incomplete grit blasting coverage, is shown in figure 38.

Figure 38: Examples of the geometric and surface finish variations at the spar end

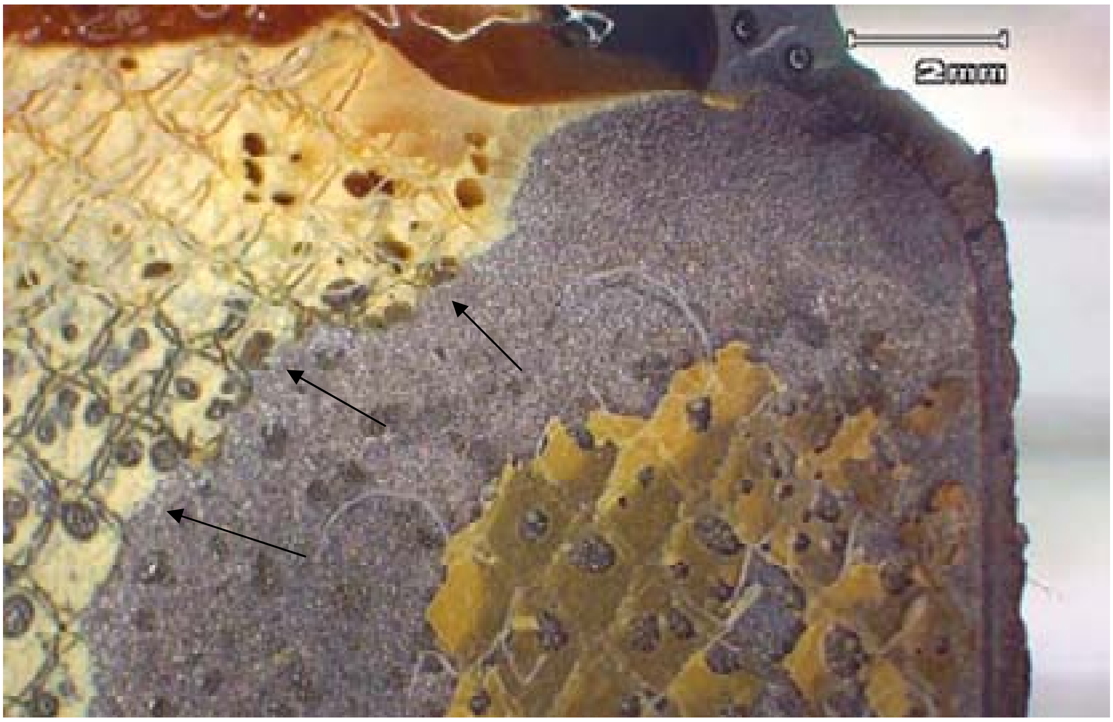
a) VH-OHA, s/n 6249A, fractured blade, showing a sharp corner between the spar end and the bonded surface, and, in addition, minimal surface roughening by grit blasting



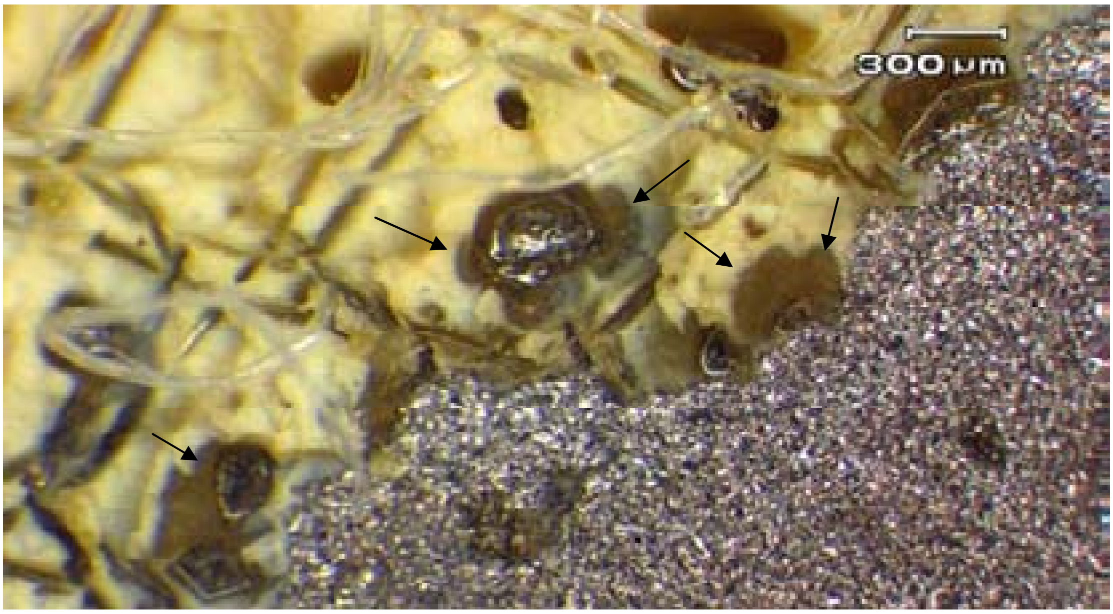
b) Blade s/n 8603B, showing a rounded corner between the spar end and bonded surface, plus surface roughening by grit blasting

In addition to the examples of fatigue crack growth from the end of the spar, an example of crack growth from voids within the adhesive joint, at the disbond boundary, was observed in one of the blades from the Australian sample (VH-UXF, s/n12616B).

Figure 39: Photomacrographs of the disbond boundary, blade s/n 12616B



a) The disbond boundary is arrowed.

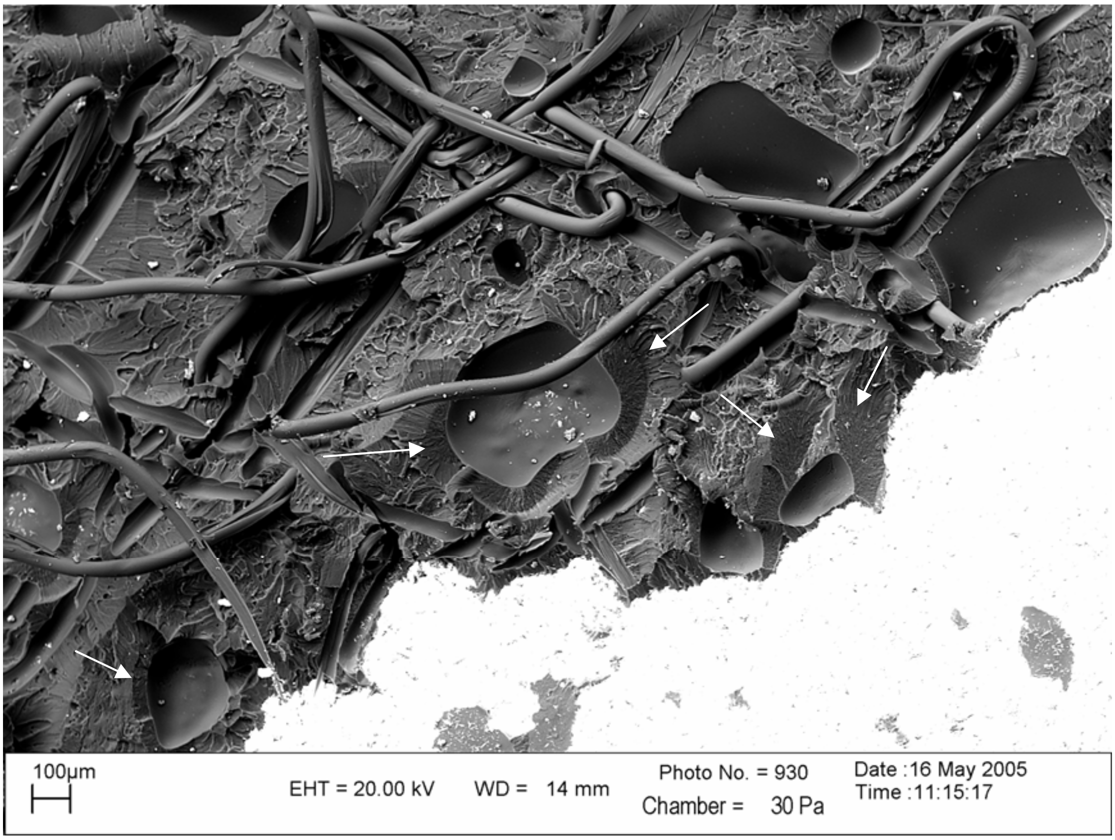


b) Regions of crack growth from voids near the disbond boundary are coloured darker than the region of fracture induced in the adhesive during strip down, examples are arrowed. The clear fibres are from the scrim fabric used in the manufacture of the adhesive film.

The region near the disbond boundary was examined further by scanning electron microscopy. The examination was conducted without coating the sample with conductive material. Variable chamber pressure was used to eliminate the effects of specimen charging.

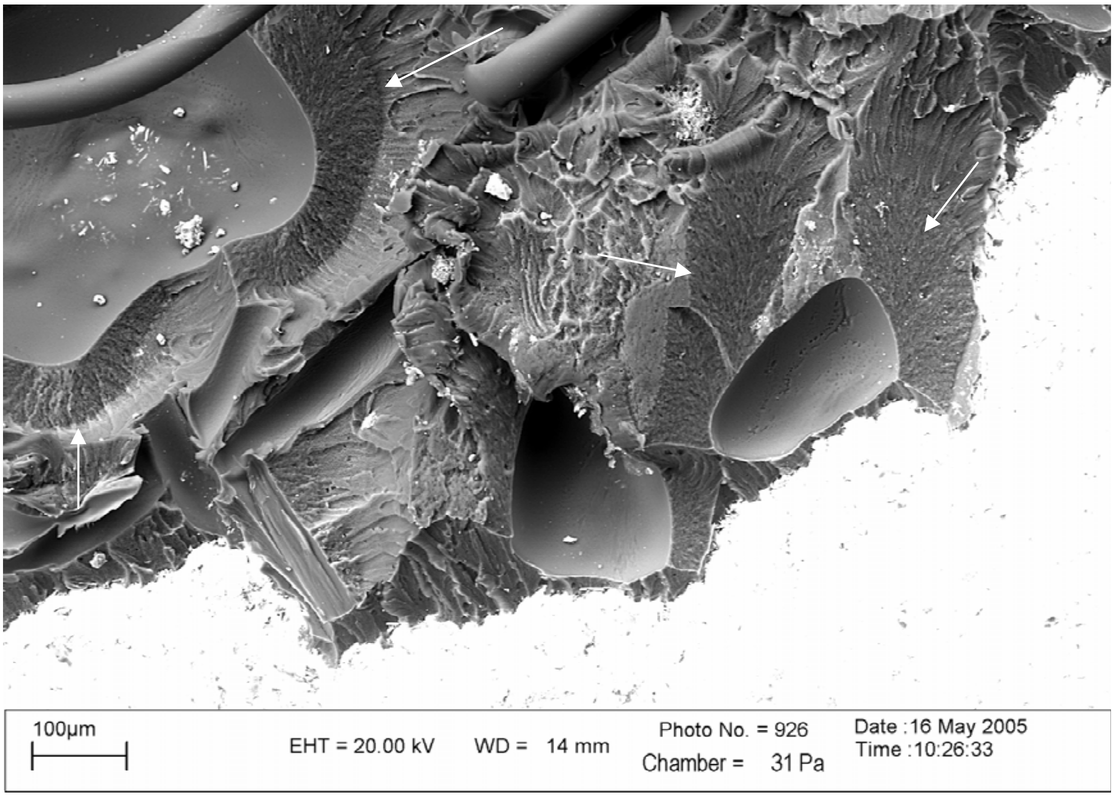
The regions of crack growth arrowed in figure 40 display surface features that are different from the surrounding surfaces that have been created by local tearing in the adhesive when the spar was peeled from the fitting. This difference in surface features supports the hypothesis that crack growth from voids near the disbond boundary occurs incrementally as a result of alternating stresses. A detailed view of the differences in surface features is shown in figure 40c.

Figure 40: Scanning electron micrographs of the disbond boundary, blade s/n 12616B

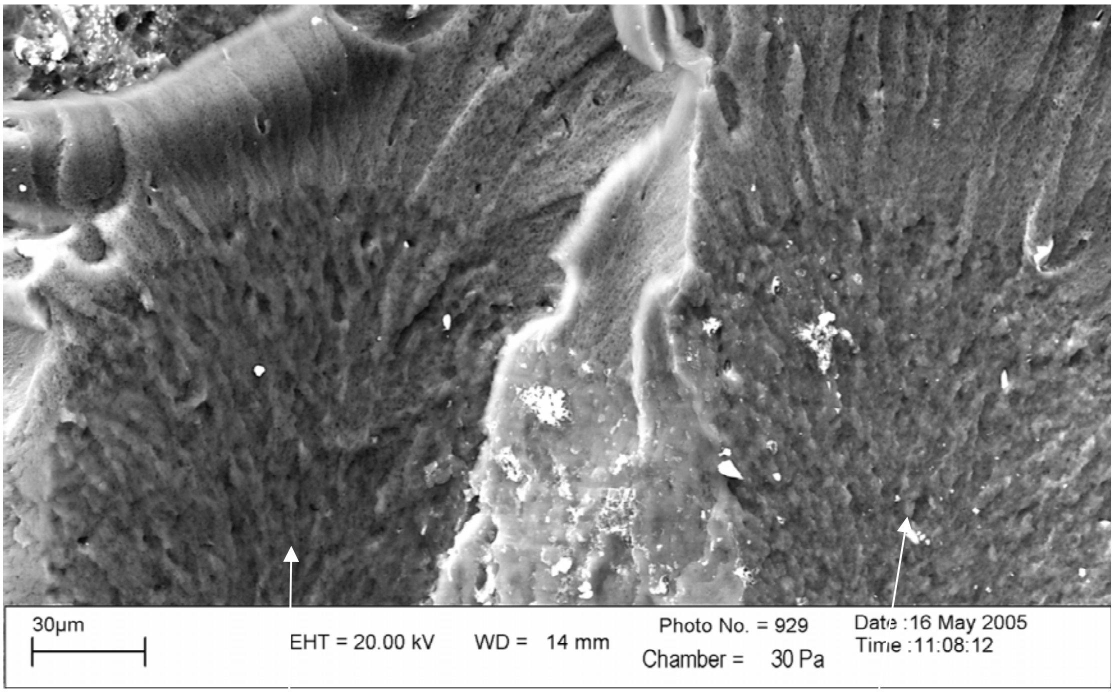


a) Regions of crack growth from voids near the disbond boundary are arrowed

Figure 40: continued



b) Region of Figure 40a at higher magnification, crack growth from voids is arrowed,



c) The boundary between a region of progressive crack growth (lower half of the micrograph) and induced tearing (upper half of the micrograph), the direction of crack growth is arrowed

5.4.3 Mitigating conditions

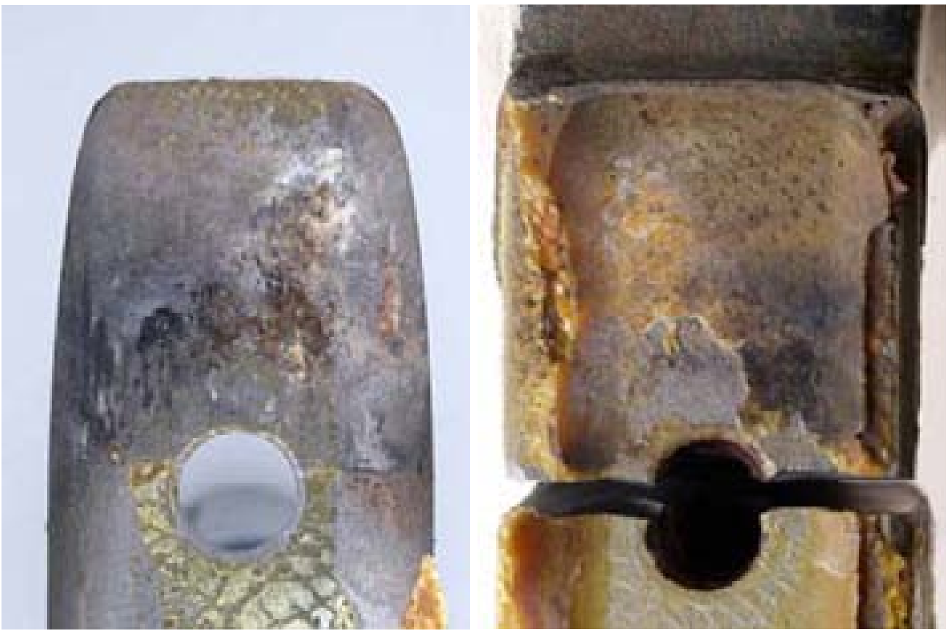
Hart-Smith[7] has reported that disbond growth in combined adhesively bonded and bolted joints may be arrested. The variable in this case is the ability of the inboard bolt to transfer load effectively as disbonding extends to the bolthole. It would be expected that a significant factor in the ability of the bolt to transfer load is the clamping force exerted by the bolt. Bolt clamping force is created by the tensile preload in the bolt established during the tightening process.

The evidence obtained from examinations of blade pairs, that is, the two blades fitted to a helicopter and both of which are subjected to the same operating loads and environment, shows that there is a marked difference in the extent of disbonding. In each case of fatigue cracking and blade fracture, disbonding has extended to the bolthole. While in the case of the blade paired with the fractured blade; disbonding has not extended to the edge of the inboard bolthole. This behaviour is shown in figures 41 and 42, the blade pairs from VH-OHA and VH-LDR. In contrast, blade s/n 8414B from the pair of blades fitted to VH-LOT demonstrates that bonding around the inboard hole can be protected by the installed bolt despite extensive disbonding from the end of the spar and between the two boltholes, see figure 43.

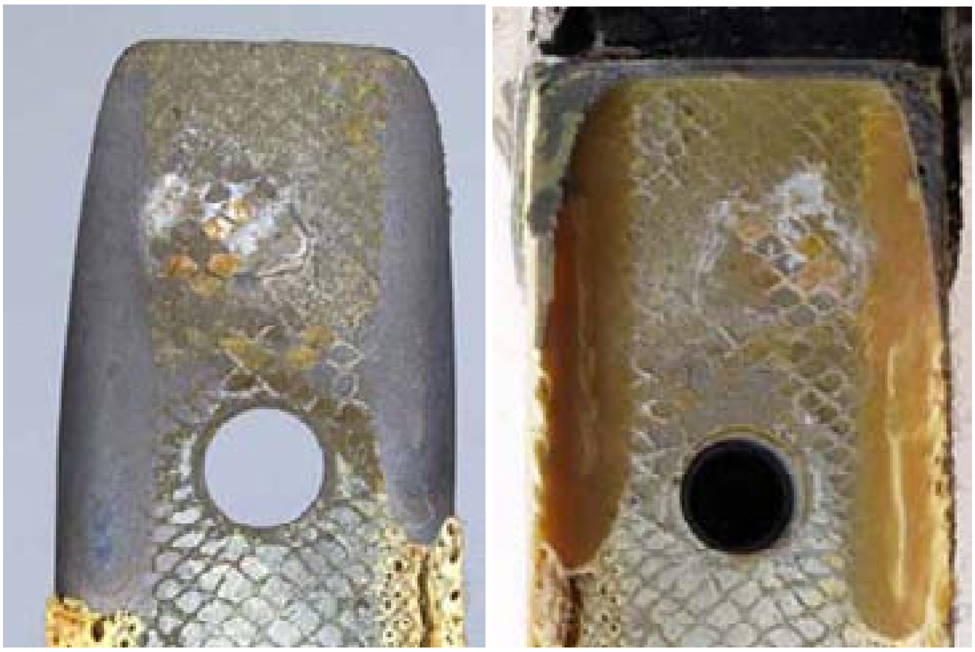
It is apparent that the key variable controlling disbond growth to the edge of the inboard bolthole is a variation in the structure of individual blades. It is not restricted to variations in operating conditions or environment.

[7] L J Hart-Smith, 'An Engineer's Viewpoint on Design and Analysis of Aircraft Structural Joints', International Conference on Aircraft Damage Assessment and Repair, Melbourne Australia August 26-28, 1991, Douglas Aircraft Company Paper MDC 91K0067

Figure 41: Extent of spar/fitting disbonding VH-OHA

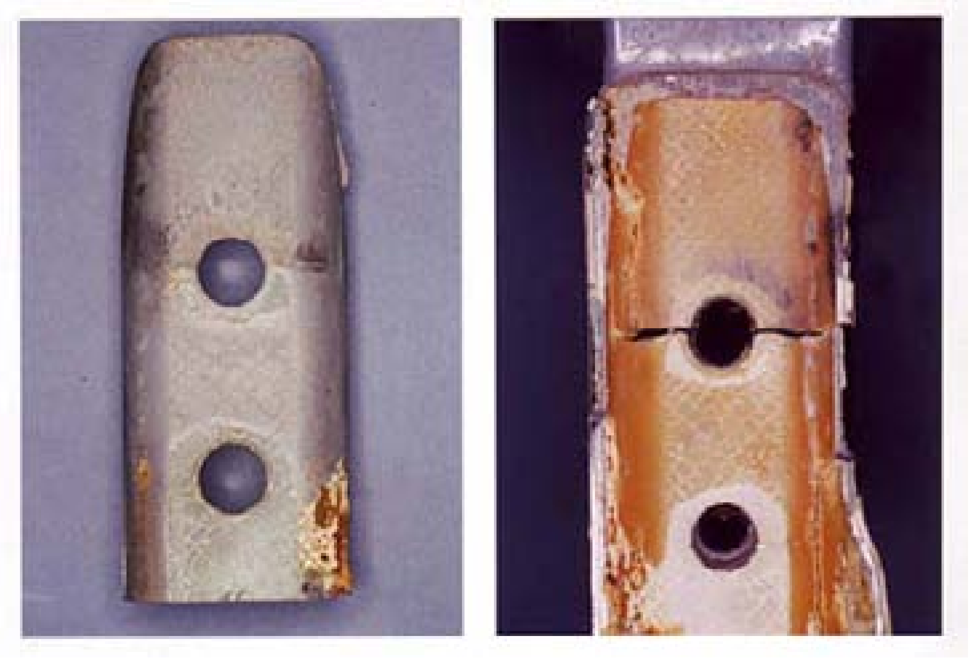


a) VH-OHA fractured blade

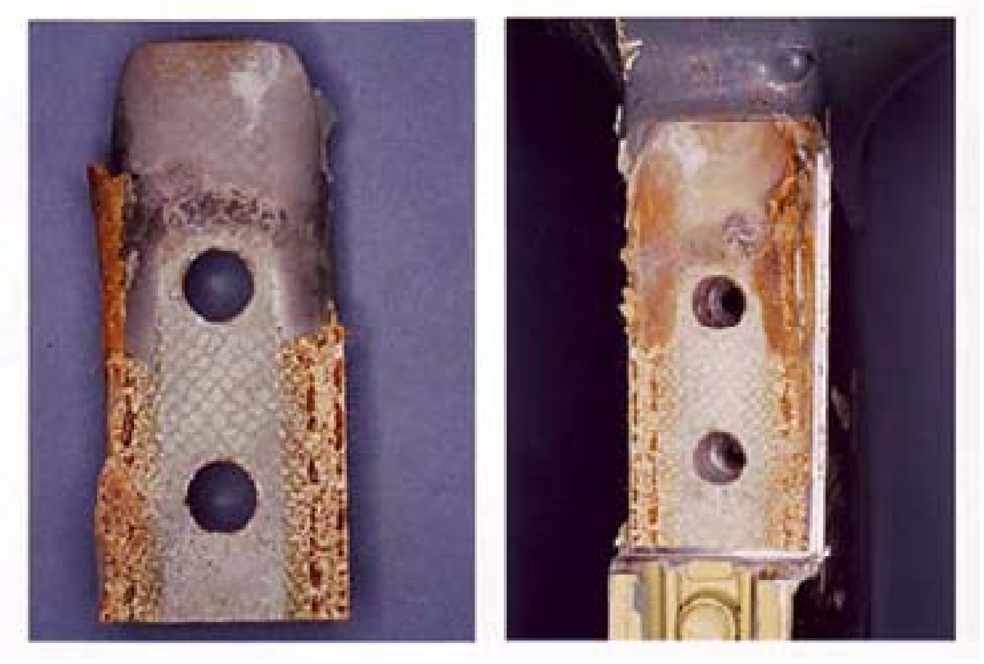


b) VH-OHA intact blade

Figure 42: Extent of spar/fitting disbonding, VH-LDR

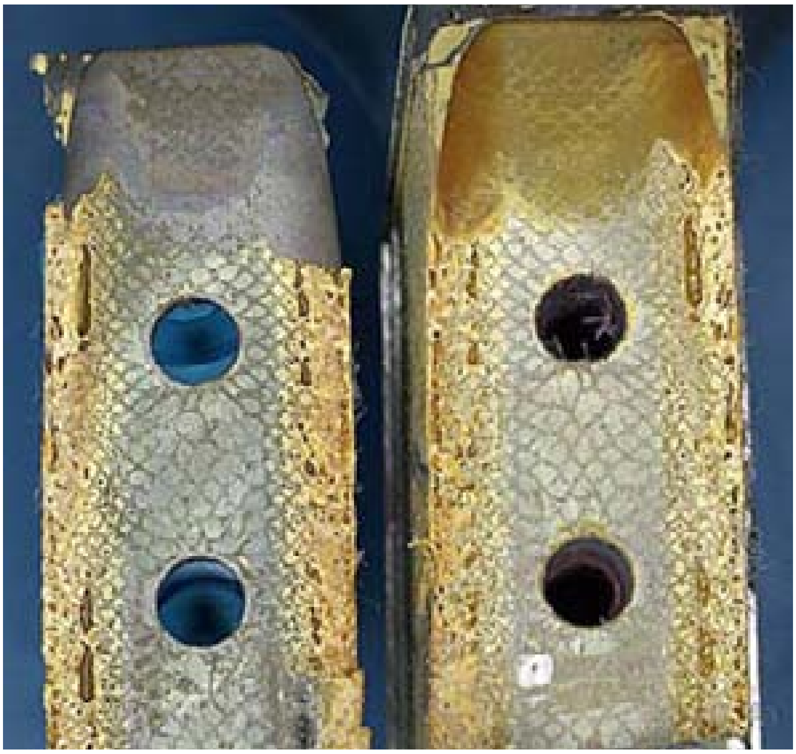


a) VH-LDR fractured blade



b) VH-LDR intact blade

Figure 43: Extent of spar/fitting disbonding, VH-LOT



a) VH-LOT, s/n 8411B

b) VH-LOT, s/n 8414B. Extensive disbonding has occurred, including disbonding between the two boltholes. However disbonding has not extended to the edge of the inboard bolthole.

5.5 Detection of Fatigue Cracking in the Blade Root Fitting

The safe life approach to fracture control does not require or rely on a series of scheduled non-destructive inspections unlike other fracture control plans, such as safety by inspection and damage tolerance which do rely on repeated non-destructive inspections directed at specific locations.

The prevention of R22 main rotor blade fracture is not dependent on repeated non-destructive inspections directed at specific locations. However, various signs and symptoms observed during operation, daily inspection, and maintenance may give an indication of a change in the condition of a main rotor blade. In the case of R22 main rotor blades, signs such as cracks in the blade skin and oil leaks, and symptoms such as main rotor vibration, have been identified as indicators of fatigue cracks in the blade structure prior to the accident involving VH-OHA. This information had been disseminated by the helicopter manufacturer through a number of airworthiness documents.

Airworthiness documents follow the general hierarchy of; directives – mandatory action to eliminate a specific hazard, alerts – notification of specific hazards, notices or letters – provision of information relating to safety issues.

Robinson Helicopter Company, R22 Service Letter SL-53, '*Visual Inspection of Main Rotor Blade Root Area*', issued 21 November. 2001, provides information relating to the potential development of main rotor blade fatigue cracks when the helicopter is operated under conditions where the loads on the main rotor exceed the design limits. It also provides information on potential indicators of blade fatigue cracking; main rotor vibration and the presence of skin cracks at the location of the inboard bolthole in the spar to root fitting joint.

R22 SERVICE LETTER SL-53

DATE: 21 Nov 01

TO ALL R22 Owners, Operators and Service Centers SUBJECT: Visual Inspection of Main Rotor Blade Root Area

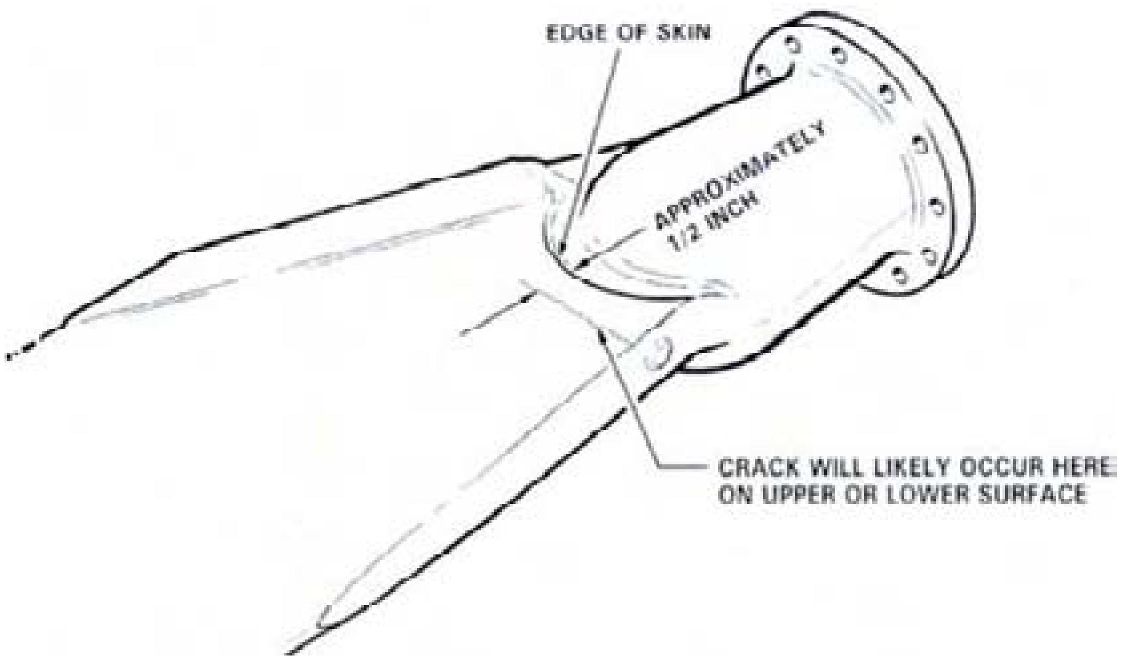
BACKGROUND: A main rotor blade fatigue failure could occur if the helicopter is repeatedly flown above its approved gross weight limit or operated above its approved manifold pressure limits. The first indication of a fatigue crack in progress may be a rotor that will not stay balanced after being adjusted. Another indication may be the appearance of a very fine hairline crack appearing in the areas shown in the Figure below.

COMPLIANCE PROCEDURE:

1. Visually examine both the upper and lower surface of each blade in the areas shown with a 10x magnifying glass.

2. If any indication of a crack is found, immediately ground the aircraft and return the suspect blade to the RHC factory for examination.

Figure 44: Diagram included in Service Letter SL-53



Robinson Helicopter Company, Safety Notice SN-37, '*Exceeding Approved Limitations Can Be Fatal'*, issued December 2001, provides a further warning of the effects of exceeding power and airspeed limitations on the development of fatigue cracking in main rotor blades. It contains the following warning.

> *WARNING*
>
> *1) Always operate the aircraft well below its approved Vne (never exceed speed), especially in turbulent wind conditions.*
>
> *2) Do not operate the engine above its placarded manifold pressure limits. 3) Do not load the aircraft above its approved gross weight limit.*
>
> *4) The most damaging conditions occur when flying or maneuvering at high airspeeds combined with high power setting.*
>
> *In response to the detection of a large fatigue crack in the root fitting of a R22 main rotor blade root fitting and an Air Accident Investigation Board (AAIB), United Kingdom investigation[8], Robinson helicopter Company issued a R22 Safety Alert on 25 June 2002. The wording of the safety alert was included in the AAIB report.*
>
> *UNUSUAL VIBRATION CAN INDICATE A MAIN ROTOR BLADE CRACK*

[8] AAIB Bulletin No: 9/2003, Ref: EW/C2002/05/04

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.

In this case, the fatigue crack initiated in the root fitting near the end of the blade spar and propagated into the blade spindle bearing housing. Crack growth did not occur in regions of the root fitting covered by adhesively bonded blade skin. The presence of the crack was detected through a visual inspection of the main rotor hub region following a flight during which main rotor vibrations increased in severity. Oil leaking from the general location of the main rotor hub drew the inspector's attention to the crack in the blade root fitting immediately.

Robinson Helicopter Company, Safety Notice SN-39, *'Unusual Vibration Can Indicate a Main Rotor Blade Crack'*, issued July 2003, provides advice on the linkage between main rotor vibrations and the presence of fatigue cracks without a direct association with excessive operational loads.

Safety Notice SN-39

Issued: Jul 2003

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 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. Have the rotor system thoroughly examined by a qualified mechanic before further flight.

If main rotor vibration rapidly increases or becomes severe during flight, make an immediate safe landing. Do not attempt to continue flight to a convenient destination.

In the period prior to the maintenance actions to address a rapid onset of main rotor vibrations in VH-OHA (April-June 2003) and the accident involving VH-OHA (20 June 2003), various airworthiness documents, published by Robinson Helicopter Company, provided information relating to the association between main rotor vibration, skin cracks and fatigue cracks in the blade root fitting. With the exception of the safety alert arising from the AAIB investigation into blade spindle bearing housing cracking, the information on fatigue cracking in the region of the inboard bolthole (blade spar to root fitting joint) is provided in the context of operations where gross weight, airspeed, and engine power (manifold pressure MAP) have been exceeded. It is important to note that increased gross weight, increased airspeed, increased engine power, for a constant rotor speed, will all result in increased drag forces on the main rotor blades.

In the case of VH-OHA, the main rotor blades were examined for the presence of skin cracking in the region indicated in service letter SL-53 during the maintenance actions undertaken to correct the main rotor vibrations. The examination extended to the removal of the paint applied to the under side of blade s/n 6249A (the fractured blade) in the designated region. Examination of the blade after fracture revealed that fatigue cracking in the root fitting had not transferred to either the upper or lower blade skins. Instead, disbonding between the root fitting and skins had occurred. No indication of cracking in the root fitting of blade s/n 6249A could have been obtained by an examination of the exterior surface of the blade upper and lower skins.

In situations where the plane of crack growth is perpendicular to the plane of the adhesive bond, the reliability of blade skin cracks as indicators of fatigue cracking in underlying blade structures is dependent on the strength of the adhesive bond. If the adhesive bond is strong then crack growth will extend from the root fitting through the adhesive and into the blade skin. If the adhesive bond is weak then crack growth in the fitting to the adhesive bond will result in disbonding at the fitting/adhesive interface. This dependency between adhesive bond strength and crack extension through an adhesive bond, roughly perpendicular to the plane of crack growth, has been reported widely and is the basis for the design and behaviour of fibre reinforced plastic composites[9]. The performance of fibre reinforced polymeric materials is optimised when the bond strength is controlled to favour disbond growth at the interface between fibres and the polymer matrix.

The detection of small areas of adhesive disbonding is difficult. Reliable methods are based on the through-transmission of ultrasound or standardised methods of vibrating the structure with sonic probes. Each method requires calibration against known good adhesive bonds and known poor adhesive bonds. Uncalibrated ‘tapping’ would not provide a reliable means of detecting disbonding between small areas at the inboard end of main rotor blade skins and the blade root fitting.

Preliminary experiments indicated that ultrasonic inspection, using an angle (45°) probe traversed over the blade skin parallel with the blade leading edge, over the leading edge of the root fitting, could interrogate the region around the inboard bolthole. The nature of the adhesive bonds between the blade skin and fitting was such that ultrasound could be transmitted without excessive attenuation.

This method of inspection would require development and validation of sensitivity of crack detection. Instances of excessive ultrasound attenuation may provide an indicator of loss of adhesion or other bonding defects.

[9] J E Gordon, 'The New Science of Strong Materials', Penquin Books Ltd, 1991, pp 112 - 124

6 CONCLUSIONS

Robinson R22 main rotor blade, p/n A016-2, s/n 6249A, fractured as the result of fatigue crack growth in the blade root fitting. Fatigue cracking initiated at the counterbore of the inboard bolthole in the bolted and adhesively bonded joint between the spar and rooting fitting. Fatigue cracking extended in a chordwise direction toward the blade trailing edge, on a plane perpendicular to the blade surface. Fatigue crack growth in the fitting did not extend into either the upper or lower blade skins.

The fracture of the blade occurred within the specified operational life limitation specified at the time of the accident, 2200 hours.

Fatigue fracture within the specified operational life of the blades is a failure of the fracture control plan developed to ensure reliable operation of a critical flight mechanism – the helicopter main rotor. The fracture control plan for R22 main rotor blades is based on retiring blades from service prior to the initiation of fatigue cracking in the blade structure – commonly known as the safe life approach.

Fatigue cracking in the blade root fitting was accompanied by disbonding of the spar/fitting adhesive joint, from the spar end to the inboard bolthole. An effect of disbond growth to the inboard bolthole is to change the nature of load transfer and the nature of local stress distribution in the joint resulting in an increase in the magnitude of alternating stresses in the inboard bolthole counterbore region. Increases in the magnitude of alternating stresses will reduce the operational time to fatigue crack initiation and failure.

The evaluation of several accidents involving blade fracture highlights the effect of disbond growth to the bolthole and the consequent effect of increased stress magnitude on the operational time to fatigue crack initiation. While disbond growth between the spar and root fitting of blades paired with fractured blades occurred, the adhesive bond surrounding the bolthole remained intact and continued to distribute load around the bolthole preventing an increase in local stress magnitude. No evidence of fatigue cracking was present in the inboard bolthole counterbore region of these blades despite the blades being subjected to the same operating loads and the same operating environment as their fractured pair.

A survey of 10 blades from a variety of Australian R22 helicopter operations and a survey of 51 blades representative of a variety of times in service and operations from many parts of the world showed that disbonding in the spar/fitting adhesive joint is widespread. The extent of disbonding is variable and does not appear to be related simply to any one type of operation, flight profile or environmental factor.

Detailed observation of disbond surfaces, the surface of the spar and the corresponding surface on the fitting, indicated that initial disbond growth had occurred through progressive crack growth in the adhesive. This form of adhesive bond breakdown is a process of fatigue and is affected by the magnitude of local alternating stresses, the number of stress cycles, their frequency of application, and the effects of operating environments (high temperatures, moisture absorption). The presence of stress concentrating features and adhesive bond defects in the highly stressed region of the adhesive joint (end of the spar), the development of high stresses during operation, and operation in hot/wet environments will decrease the time to the initiation of cracking in the adhesive and increase the rate of disbond growth.

Disbond growth in an adhesively bonded joint can be arrested by the presence of fasteners (bolts) in the joint. The ability of the bolt, installed in the inboard bolthole, to arrest disbond growth will be a function of its ability to transfer load in the joint. This ability will be determined by the clamping force created during the tightening of the bolt and the degree to which this clamping force is retained during operation.

A further effect of disbonding, in those instances where disbonding had extended to the inboard bolthole, is corrosion pitting in the bolthole counterbore region. The breakdown of the adhesive bond around the bolthole allows moisture and, in particular moisture containing chloride salts, to react with the aluminium alloy root fitting. Corrosion pitting in a critical stress region of the fitting will further reduce the operational time to fatigue crack initiation through increasing the local stress concentration and reducing the material resistance to fatigue crack initiation. Fatigue crack initiation in the fractured blade from VH-OHA was associated with localised pitting corrosion which had occurred after the ingress of moisture and chloride salts into the counterbore region of the inboard bolthole.

The reliability of visual inspection of the outer surface of blades as a means to detect underlying cracks in the blade root fitting is dependent on the mechanism of crack transfer across an adhesively bonded joint. Crack transfer across an adhesive bond is dependent on the adhesive bond strength. Crack transfer is favoured by high bond strengths while disbonding is favoured when the bond strength is reduced. If disbonding occurs then there will be no skin cracking to serve as an indicator of cracking in the root fitting.

For the case of the fractured blade from VH-OHA, disbonding between the root fitting and blade skins eliminated visual inspection of the blade surface as a means of detecting the underlying crack in the root fitting.

APPENDIX A
ROBINSON HELICOPTER MAIN ROTOR BLADE SPAR TO ROOT FITTING JOINT TEARDOWN INSPECTION PROTOCOL

Introduction

The investigation of the separation of a main rotor blade from a Robinson R22 helicopter during flight has identified progressive adhesive bond failure in the spar to root fitting joint as a factor in the blade failure. In order to gain an understanding of the mechanism of bond failure and the variables that may affect bond integrity it is necessary to conduct teardown inspections of spar to root fitting joints from a number of Robinson Helicopter blades. The blades inspected should represent a variety of operating environments and a variety of flight-loads spectra. In order that the results of these teardown inspections can be correlated with an initial teardown survey conducted by the Australian Transport safety Bureau the following inspection protocol has been set out.

Inspection Protocol

1. Photograph the blade root end fitting paying particular attention to capturing the condition of the paint on the root fitting and the presence of any cracks in the filler at the end of the lower skin and end of the spar; refer to Figures 1 and 2.

2. Photograph the blade part number and serial number decal on the lower skin; refer to Figure 3.

3. It is evident that the progressive failure of the adhesive bonding in the spar to root fitting joint initiates at the inboard end of the blade spar (stainless steel ‘D’ section) and at the inboard end of the lower skin towards the leading edge of the blade. In order to assess the nature of bonding in these areas it is necessary to remove the spar from the root fitting and remove both the upper and lower skins from the root fitting without creating secondary damage to the adhesive joint in the critical areas. The following steps detail the procedure used in the Australian survey.

4. Removal of the leading edge corners of the upper and lower skins.

   - Remove the leading edge cover plate at the inboard end of the spar.

   - Mark out a chordwise line 2.25 inches from the spar end (should be close to the end of the nut plate). Mark out a spanwise line extending from the inboard end of the blade skin, 0.75 to 1 inch from the end of the spar (the mark should just clear the blade serial number decal); refer to Figure 4.

- Cut through the blade skins (two layers) into the root fitting. An electrically powered oscillating saw (Fein saw) was found to be most effective. Any other fine rotary saw would be appropriate; refer to Figure 4.

- The leading edge corners of the upper and lower skins are removed by applying a peel force to the outboard end of section to be removed. The peeling of the skin is achieved by driving a small wedge between the leading edge of the skin and the spar. In practice, the most effective, easily obtainable, tool for this purpose is an old screwdriver. Once the adhesive bond has been broken at the outboard end of the section to be removed a peel force is applied by applying a controlled leverage on the wedge tool. The adhesive should fail progressively from the outboard end to the critical inboard end. Take care not to drive the wedge tool into the critical areas of the adhesive joint. Note; adhesive failure from the disassembly peel forces is typically cohesive with a light yellow colouration and the exposure of the adhesive film scrim fabric. Areas of adhesive bond failure, which had occurred during the blade's service life, are of a darker colouration and smoother in nature. Typically the spar, skin or fitting surface will be exposed; refer to Figure 4.

- The inboard end of the blade spar is removed by first removing the two bolts at this end of the spar. Cut through the spar just inboard of the third bolt. The section of spar is released by driving a fine wedge between the spar and root fitting adjacent to the cut. The peel stress created will fracture the adhesive from the cut to the inboard end of the spar. Once again the peel forces created during spar removal will result in a cohesive failure in the adhesive. Any areas of pre-existing bond breakdown can be seen in contrast; refer to Figures 5 and 6.

- Photograph the exposed surface of the root fitting and spar. Note the presence of bond failure, the presence of voids in the adhesive especially elongated voids at the edge of the root fitting and surface deposits or surface discolouration that may be associated with moisture ingress. Examine the surface of the root fitting for evidence of corrosion using a stereo light microscope at magnifications up to 25X, paying particular attention to the area immediately surrounding the inboard bolthole (RS 10.35) and the counterbore of this bolthole.

It is desirable to document as much information as is available on the type of flight spectrum each blade has been subjected to and the environment that the blade had been operated in, eg maritime, coastal, hot/dry, hot/wet, cold/wet etc).

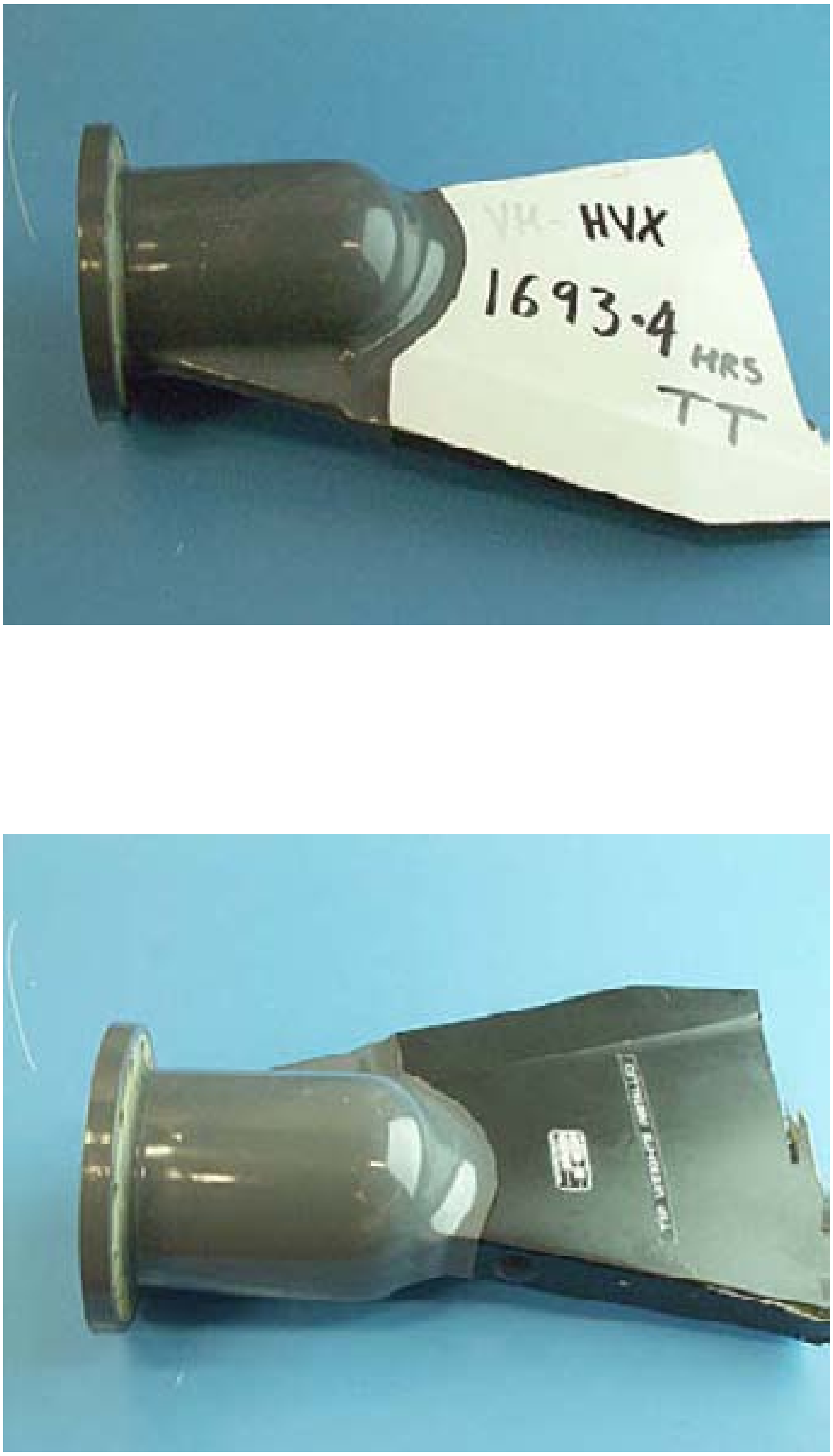


Figure 1. General condition of blade root end, upper and lower blade surfaces.

Figure 2. Serial number decal

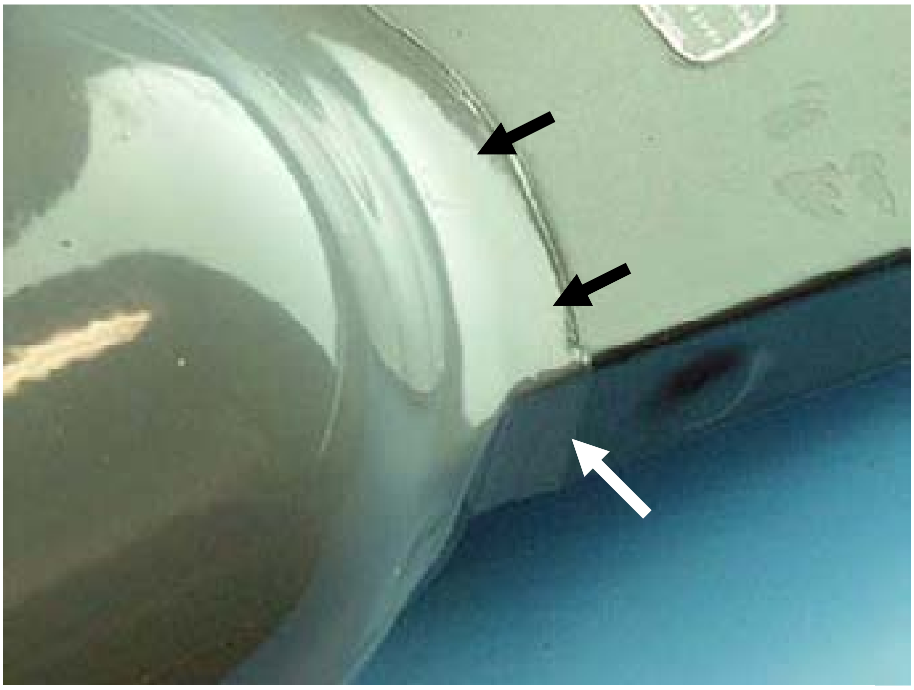


Figure 3. Cracks in filler at the end of the spar and lower skin

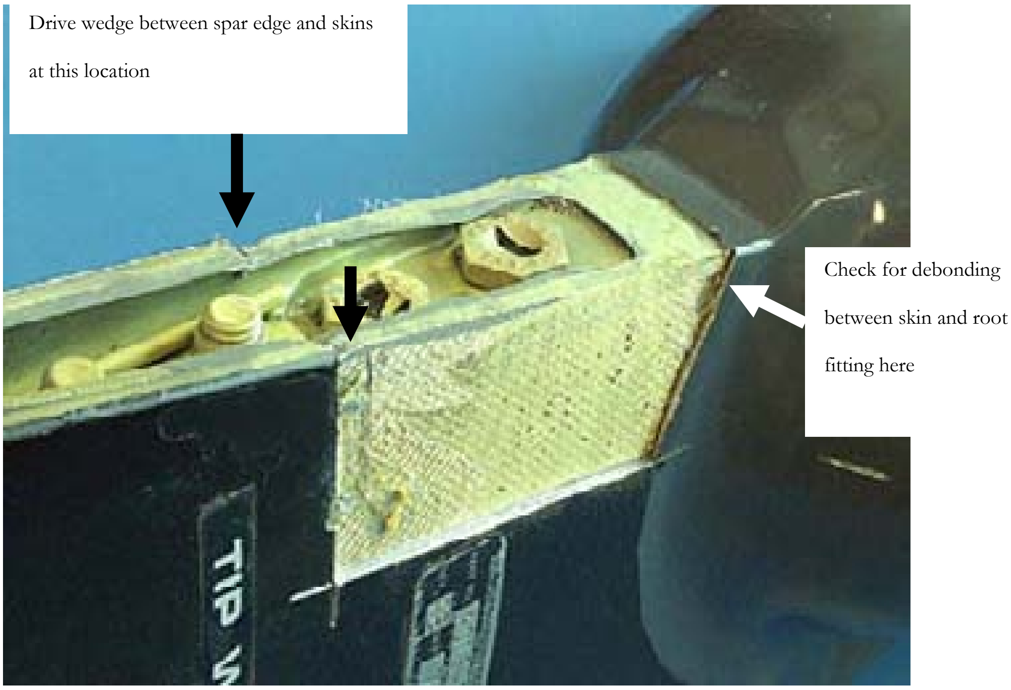


Figure 4. Blade skin removal

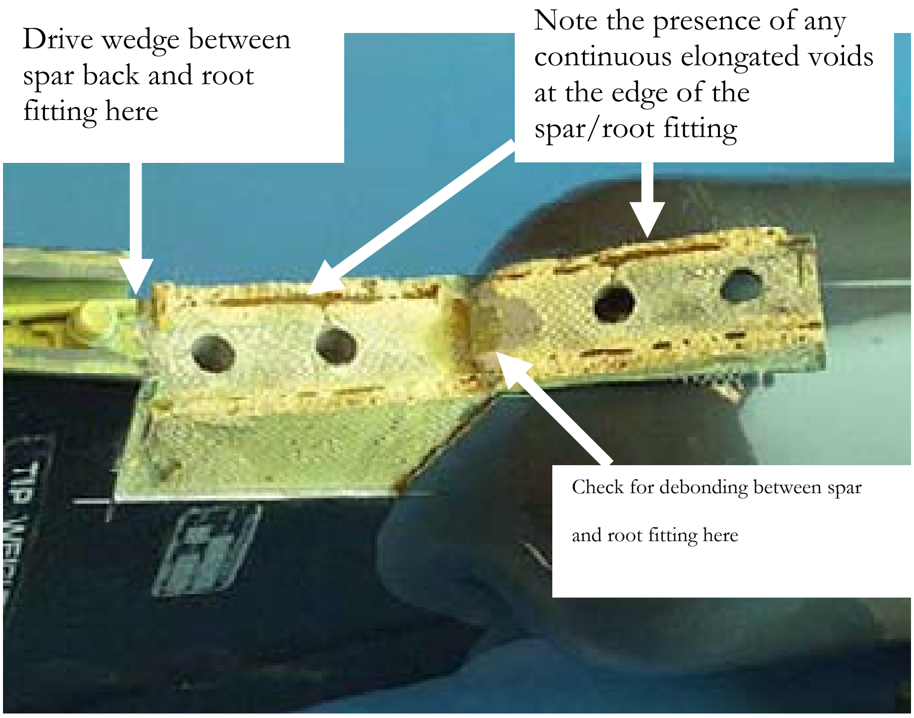
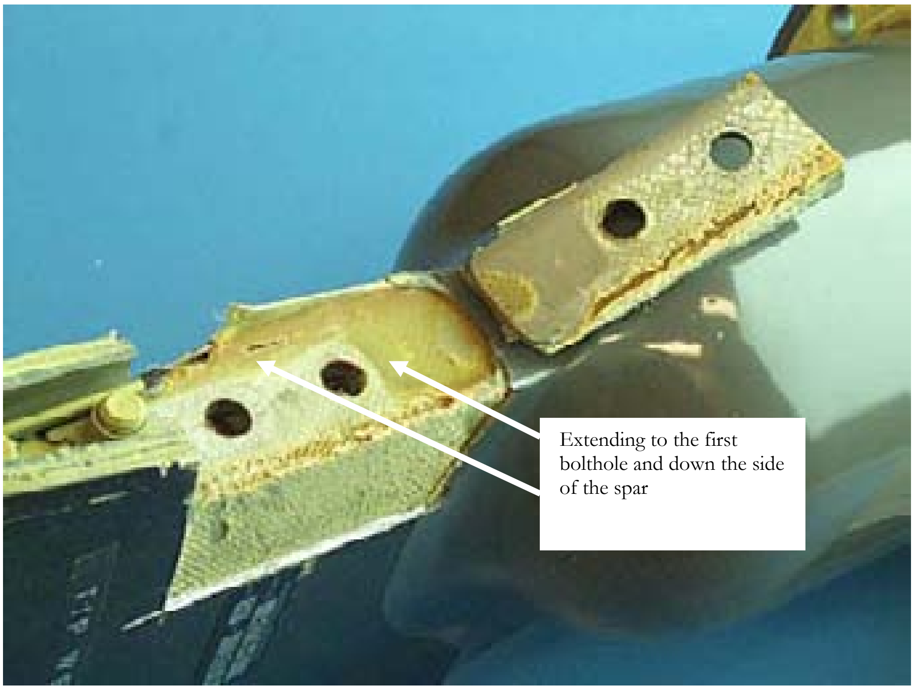


Figure 5. Blade spar removal



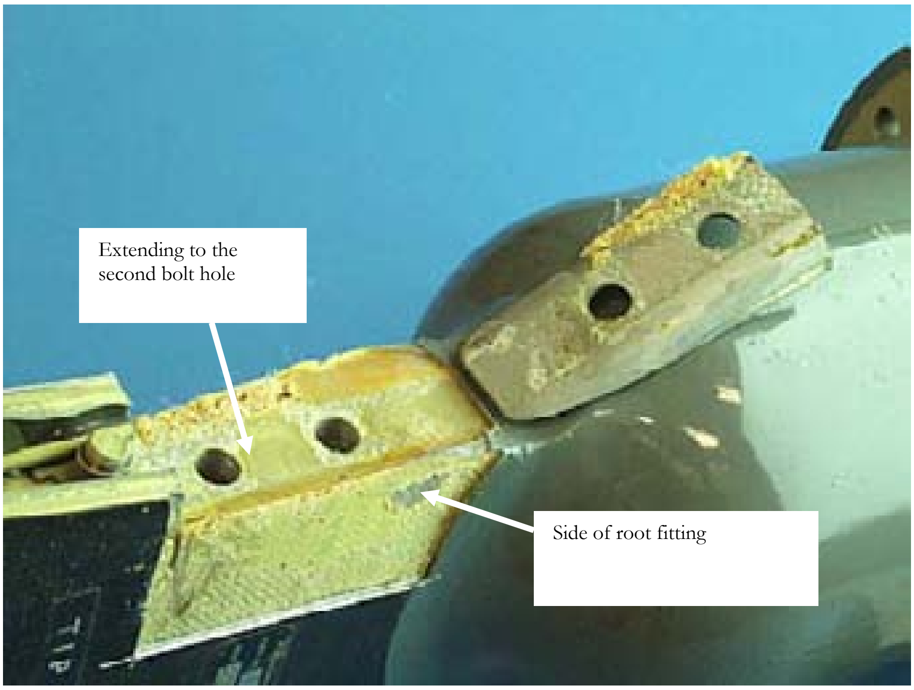


Figure 6. Examples of varying extents of adhesion failure (debonding)

APPENDIX B
DETAILS OF BLADES IN THE WORLDWIDE SAMPLE

Sample No.	Serial number	Revision	Total Time in Service	Date of Manufacture	Helicopter serial number	Aircraft Registration	Location
1	5816 B	AD	2265	08/02/91	1732	G-BTHI	LEEDS, ENGLAND
2	5834 B	AD	2265	14/02/91	1732	G-BTHI	LEEDS, ENGLAND
3	6052 C	AD	2062.5	18/04/91	1851	GFKNZ	QUEBEC, CANADA
4	6060 C	AD	2062.5	22/04/91	1851	GFKNZ	QUEBEC, CANADA
5	6588 B	AE	1976.09	17/10/91	1813	F-GLSF	FRONTENAS, FRANCE
6	6608 B	AE	1976.09	21/10/91	1813	F-GLSF	FRONTENAS, FRANCE
7	6841 A	AE	697.1	27/01/92	1747	N4072T	VAN NUYS, CA
8	6849 A	AE	697.1	03/02/92	1747	N4072T	VAN NUYS, CA
9	7004 C	AE	2202.2	06/04/92	2141	ZK-HXU	CHRISTCHURCH, N Z
10	7005 C	AE	2202.2	06/04/92	2141	ZK-HXU	CHRISTCHURCH, N Z
11	7406 B	AE	2221.6	12/11/92	2262	N2356M	SALINAS, CA
12	7416 B	AE	2221.6	18/11/92	2262	N2356M	SALINAS, CA
13	7441 A	AE	2192.2	07/12/92	2280	XK-HFL	CHRISTCHURCH, N Z
14	7448 A	AE	2192.2	11/12/92	2280	XK-HFL	CHRISTCHURCH, N Z
15	7529 C	AE	UNK	12/02/93	2409	N8118L	WEST PALM BEACH, FL
16	7550 C	AE	UNK	18/02/93	2409	N8118L	WEST PALM BEACH, FL
17	8238 B	AG	2200	22/03/94	2439	F-GPAR	FRONTENAS, FRANCE
18	8255 B	AG	2200	30/03/94	2439	F-GPAR	FRONTENAS, FRANCE
19	9784 B	AG	2144.9	21/02/97	1980	N980SM	TUCSON, AZ
20	9786 B	AG	2144.9	21/02/97	1980	N980SM	TUCSON, AZ
21	9968 C	AG	623.6	28/05/97	2715	N835SN	LONG BEACH, MS
22	10009C	AG	623.6	11/06/97	2715	N835SN	LONG BEACH, MS
23	10123C	AG	UNK	29/07/97	1773	ZS-RAR	EMPANGENI, SOUTH I AFRICA
24	10150A	AG	1679.6	12/08/97	1666	N4041W	SEBRING, FL

25	10158A	AG	1679.6	18/08/97	1666	N4041W	SEBRING, FL
26	10187C	AG	UNK	28/08/97	1773	ZS-RAR	EMPANGENI, SOUTH I AFRICA
27	10222A	AG	1962.7	10/09/97	1041	G-OMMG	NORTHAMPTON, U K
28	10226A	AG	1962.7	15/09/97	1041	G-OMMG	NORTHAMPTON, U K
29	10564B	AG	2101.8	06/03/98	2815 M	B-7013	TSUEN WAN, N.T. HONG KONG
30	10569B	AG	2101.8	06/03/98	2815M	B-7013	TSUEN WAN, N.T. HONG KONG
31	11218 B	AH	2200.	15/01/99	2450	N789RW	ELLINGTON, CT
32	11228C	AH	902.1	21/01/99	1831	F-GHUE	VIVIERS DU LAC, FRANCE
33	11280C	AH	902.1	17/02/99	1831	F-GHUE	VIVIERS DU LAC, FRANCE
34	11317B	AH	2200	01/03/99	2450	N789RW	ELLINGTON, CT
35	11568B	AI	2156.1	21/05/99	1028	VH KSC	KUNUNURRA, W.A. AUSTRALIA
36	11586B	AI	2156.1	02/06/99	1028	VH KSC	KUNUNURRA, W.A. AUSTRALIA
37	11764C	Al	2200	19/08/99	1499	ZK-HCG	CHRISTCHURCH, N Z
38	11770C	AI	2200	20/08/99	1499	ZK-HCG	CHRISTCHURCH, N Z
39	12386C	Al	2199.8	07/07/00	1149	N145RJ	BELLFLOWER, CA
40	12389C	Al	2199.8	07/07/00	1149	N145RJ	BELLFLOWER, CA
41	12491B	AT	2200	07/09/00	3153	N501HE	LONG BEACH, CA
42	12500B	Al	2200	07/09/00	3153	N501HE	LONG BEACH, CA
43	12921B	Al	2200	30/05/01	1780	N1118N	CHANDLER, AZ
44	12957B	Al	2200	19/06/01	1780	N1118N	CHANDLER, AZ
45	13601A	Al	518.1	19/06/02	3358	ZK-HCP	CHRISTCHURCH, N Z
46	13603A	AI	518.1	19/06/02	3358	ZK-HCP	CHRISTCHURCH, N Z
47	13938B	Al	400	14/12/02	2971	N7176S	HAYWARD, CA
48	14008B	AI	0 hr.	27/01/03	2331	CC-PPY	OSORNO, CHILE
49	14011B	Al	0 h.	27/01/03	2331	CGPPY	OSORNO, CHILE
50	14091A	Al	56.8	14/03/03	3449	N131MH	WATER, MI
51	14145A	Al	56.8	17/04/03	3449	N131MH	WATER, MI