APPENDIX A: MAIN ROTOR BLADE FAILURE ANALYSIS REPORT

ENGINEERED SYSTEM FAILURE ANALYSIS REPORT

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

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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

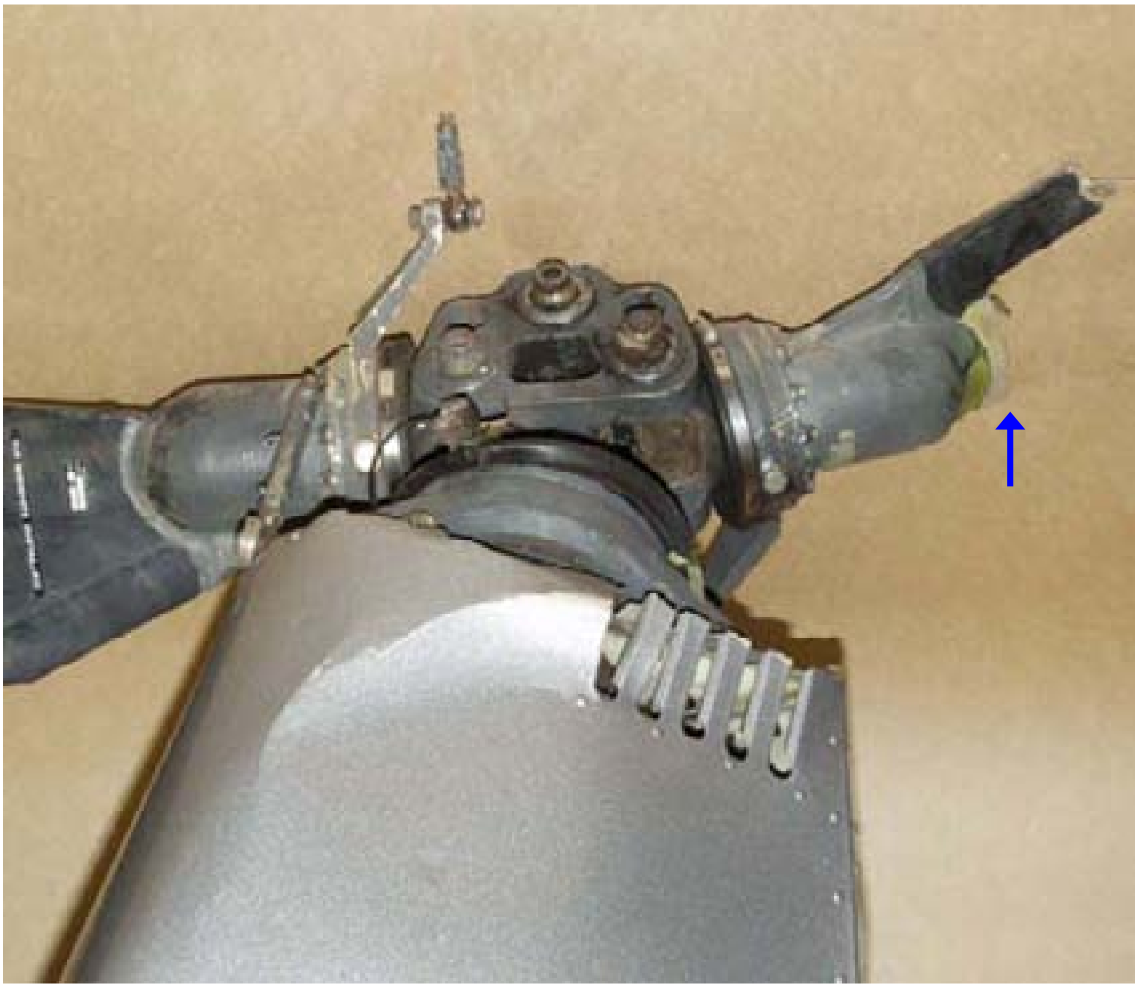
A Robinson R22 Mariner helicopter, registration VH-OHA, was involved in a fatal accident, 13km of NW Camden Airport, while being used for flying instruction.

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

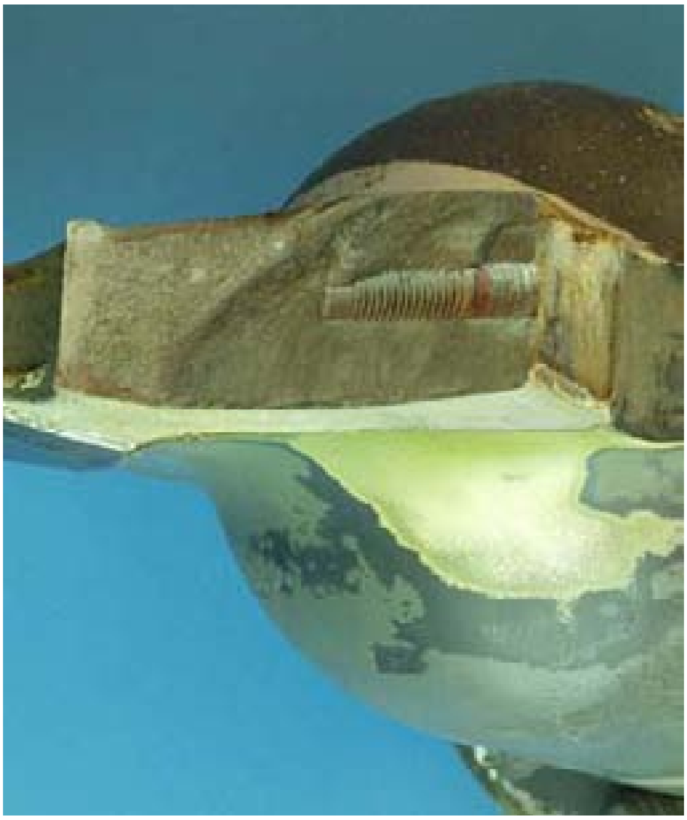
a) fractured blade

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.

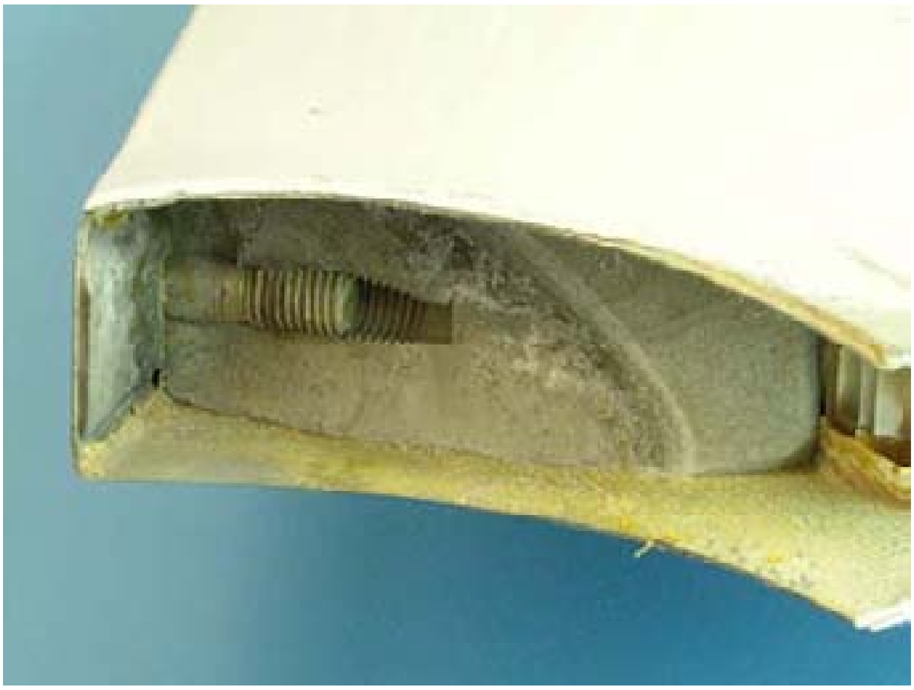
Figure 4: Fracture features, blade s/n 6249A – as recovered



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

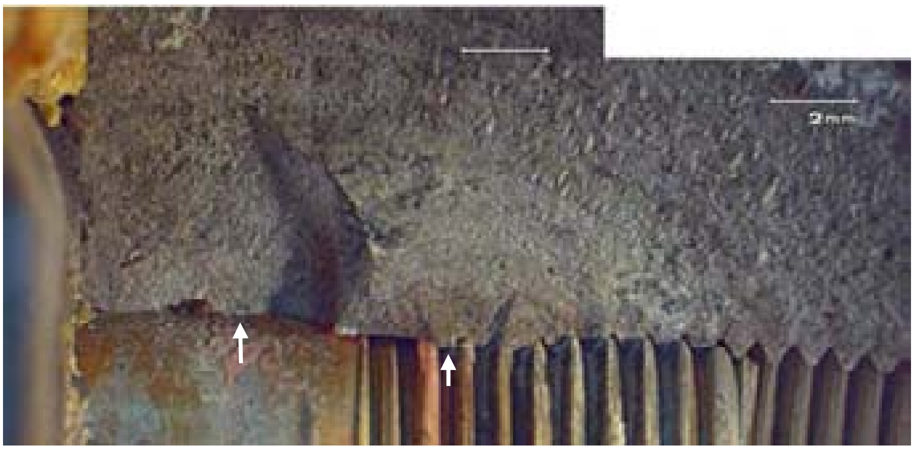
b) Looking up from the lower surface of the blade

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



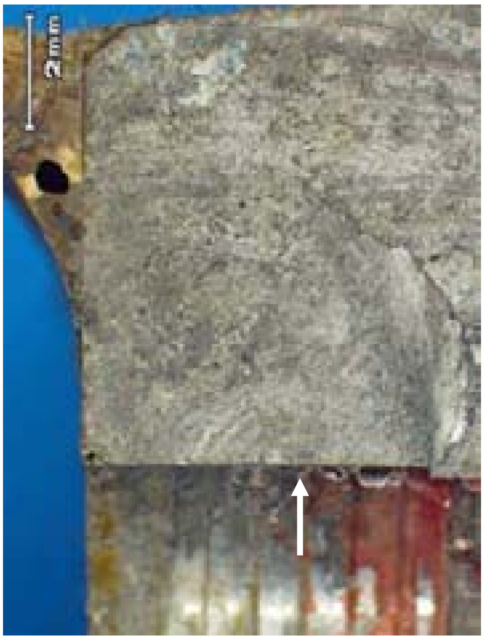
a) Upper side of the bolthole with the bolt in situ. The sites of fatigue crack initiation are arrowed

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.

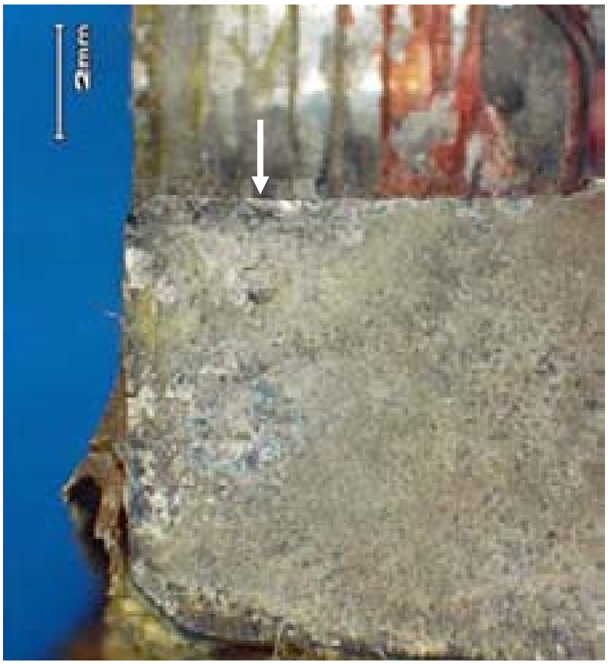
The nature of crack progression markings indicate that fatigue crack initiation occurred first at the lower side of the bolthole, secondly at the thread on the upper side of the bolthole and thirdly at the counterbore on the upper side of the bolthole.

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



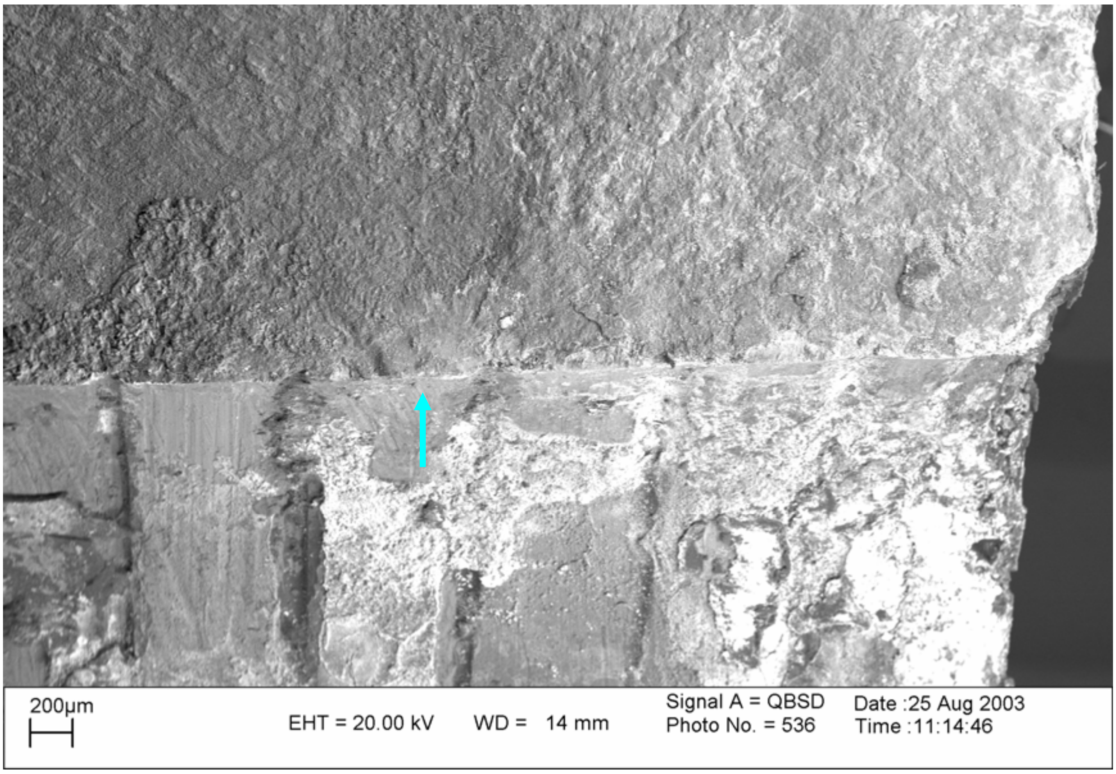
a) Third initiation site, crack growth from this site was influenced by the presence of the crack growing from second initiation site

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

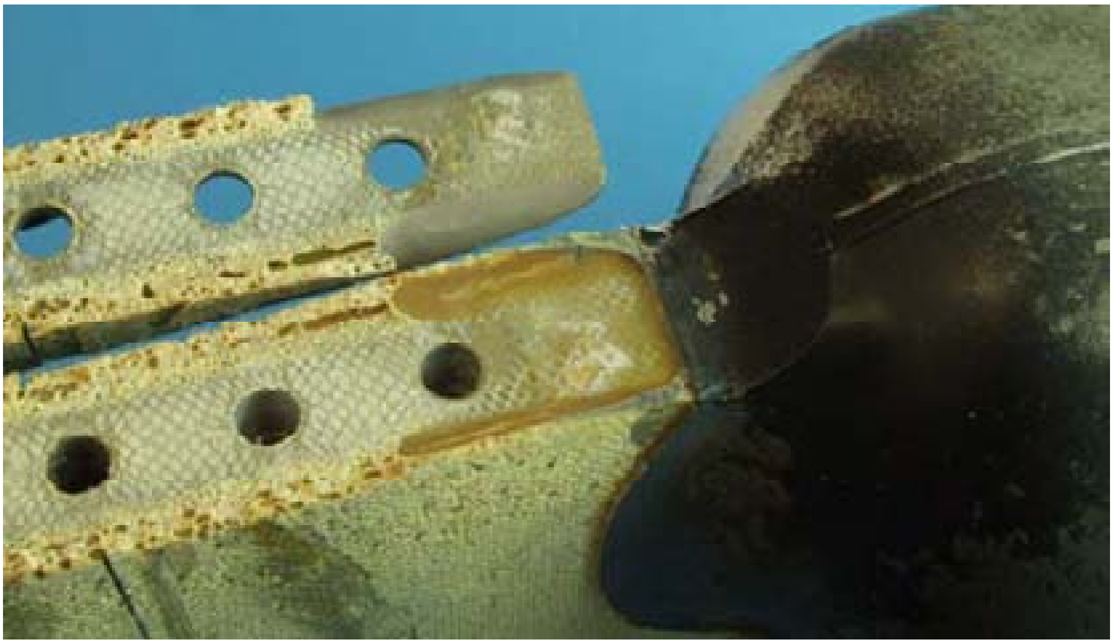
a) Disbonding extends from the inboard edge of the skin to the fracture. The paint film on the root fitting was badly weathered

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)

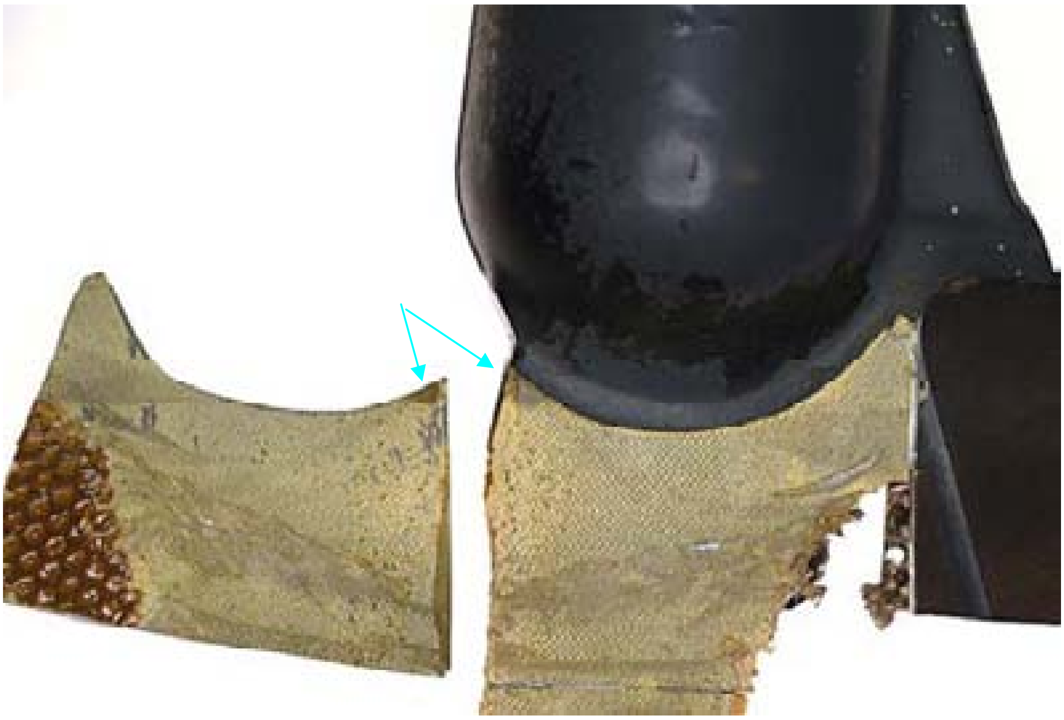


a) Disbonding has extended from the end of the spar to the edge of the inboard bolthole

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,

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[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

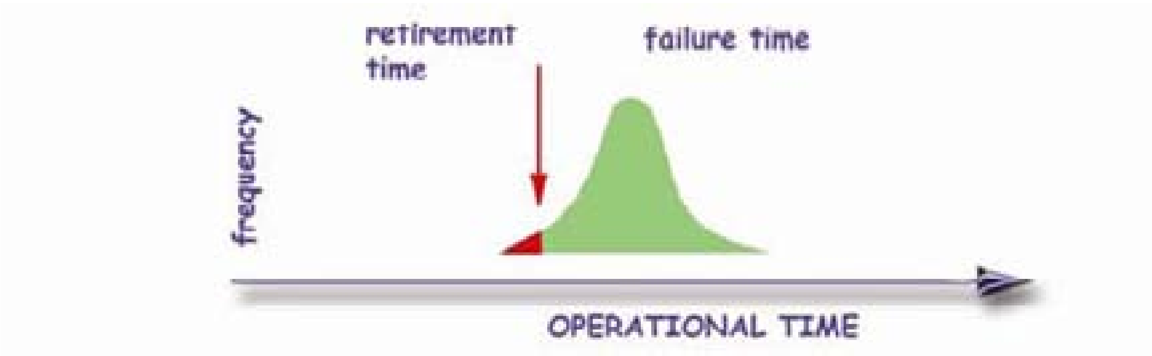
The essence of maintaining structural integrity through the use of a safe life approach is ensuring that components are retired from service before their strength is affected by the development of cracks or other deterioration. The safe life is defined in terms of an operational time.

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.

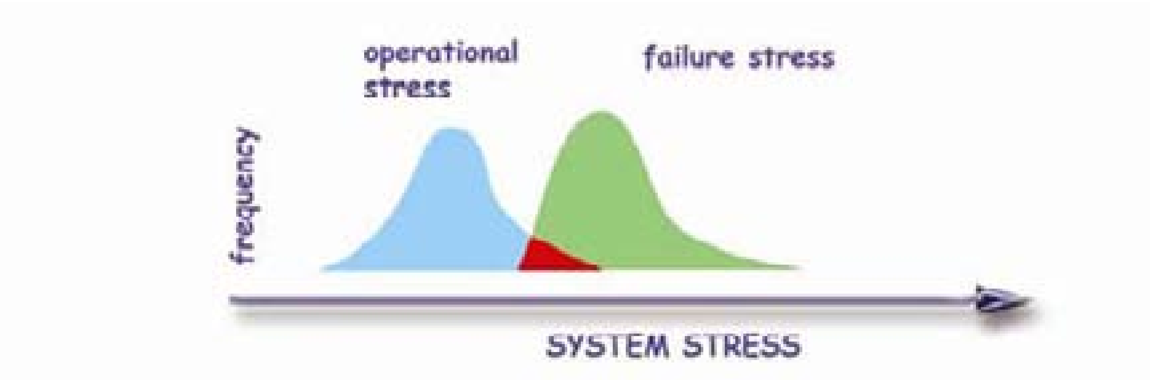
Figure 18: Schematic illustration of the probability of failure under conditions of variable operational stress and failure stress – no failure condition

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

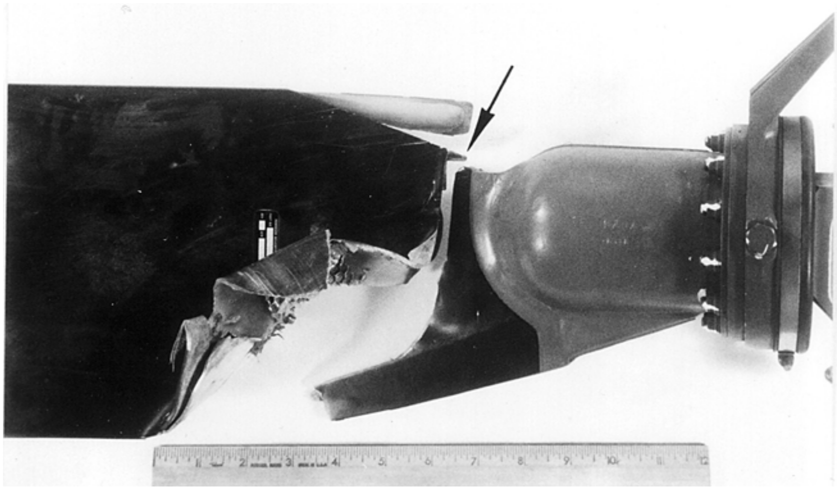
4.1 R22, N9065D, 1 Sep 1981

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

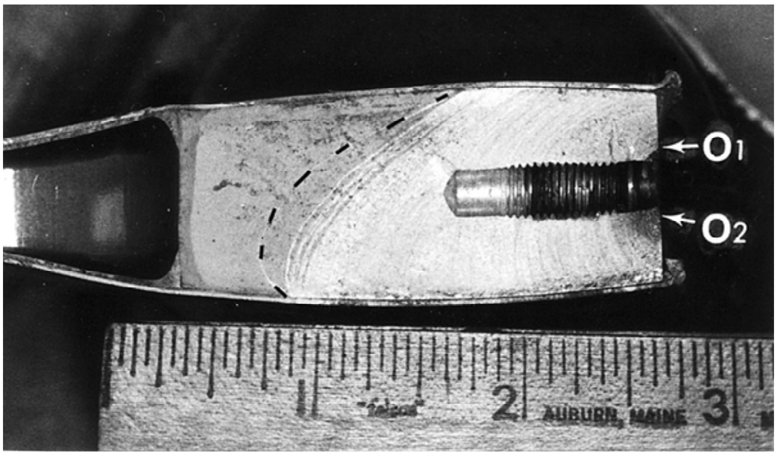
Blade part number: A016-1, revision V   Blade serial number: 370

Time since new: 690 hours

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

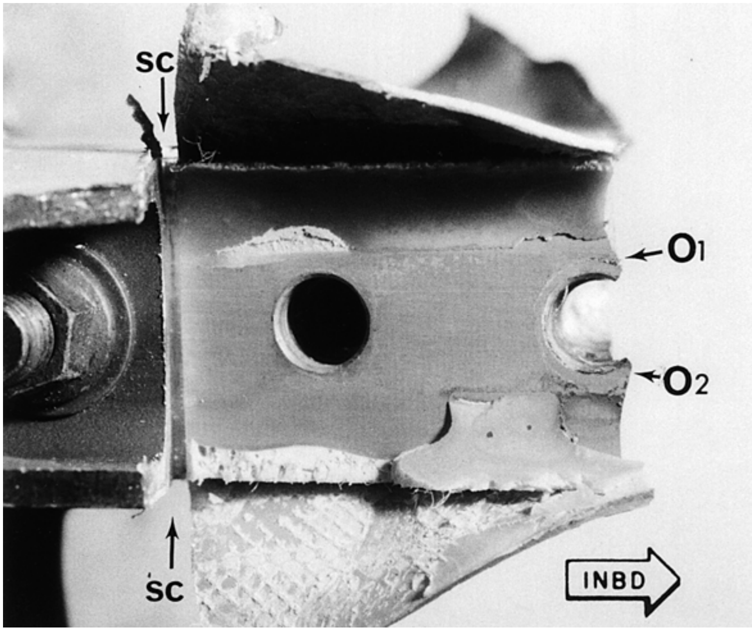


a) Fracture location



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

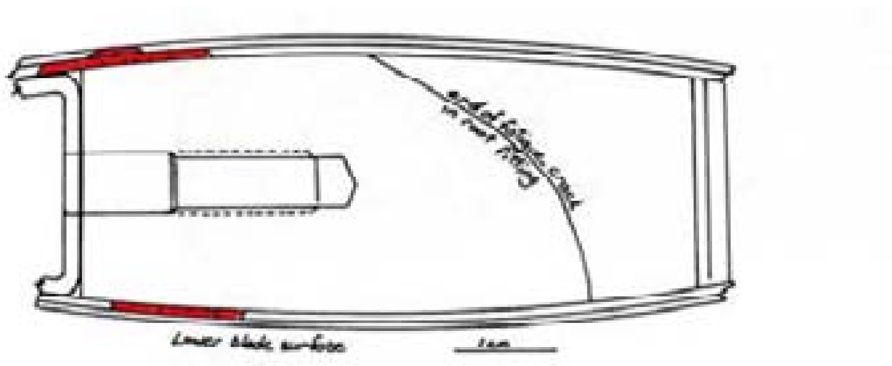
Blade part number: A016-2, revision AB   Blade serial number: 2961

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

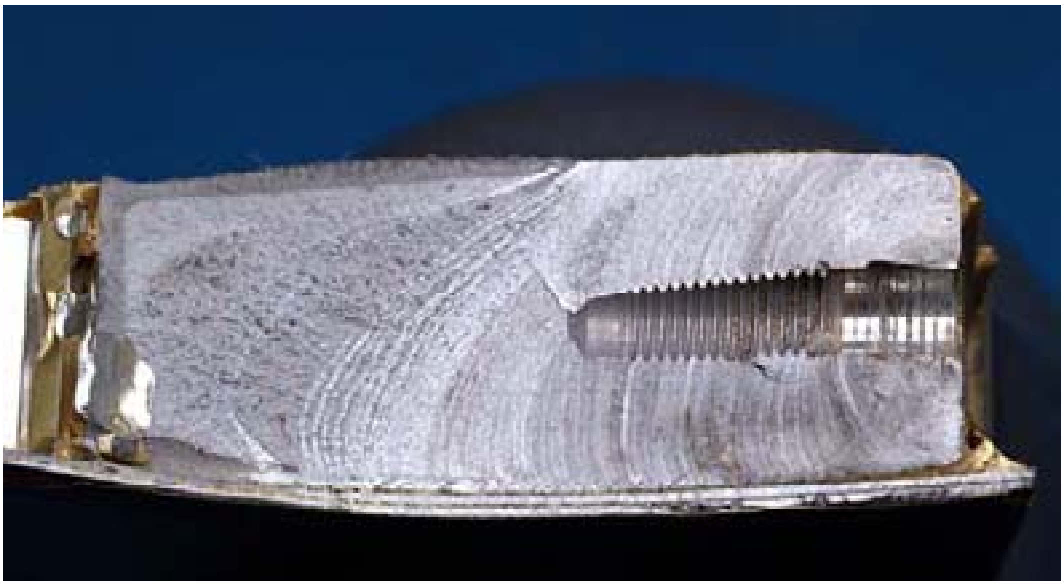
4.3 R22, VH-LDR, 29 July 2000

Australian investigation occurrence number 200003267

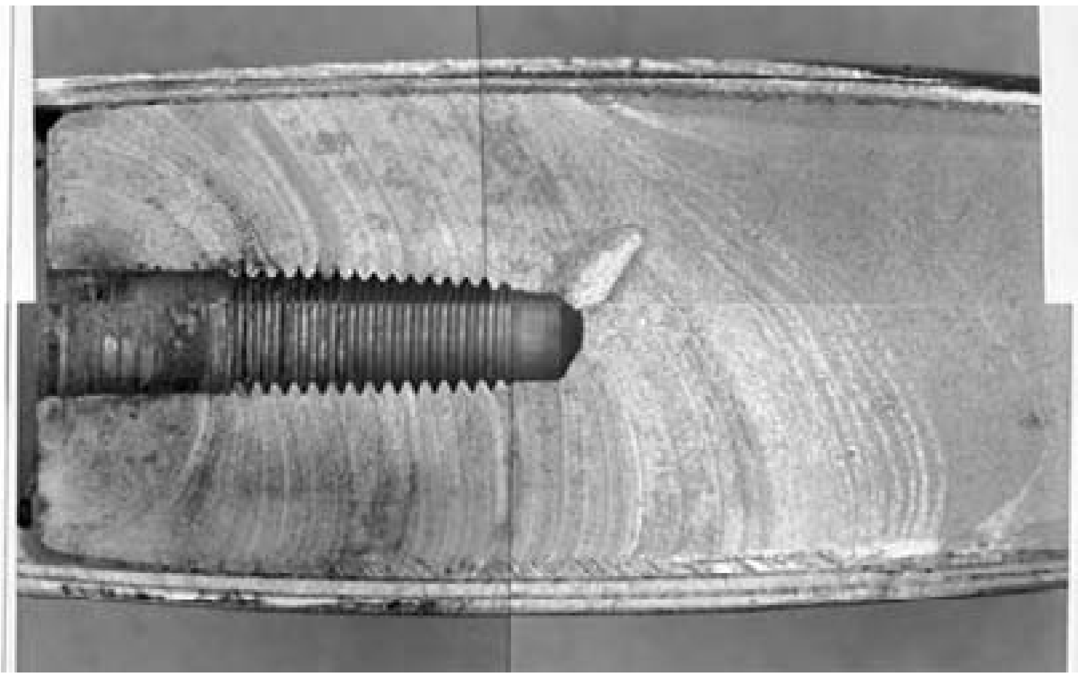
Blade part number: A016-2, revision AG, Blade serial number: 9278B

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

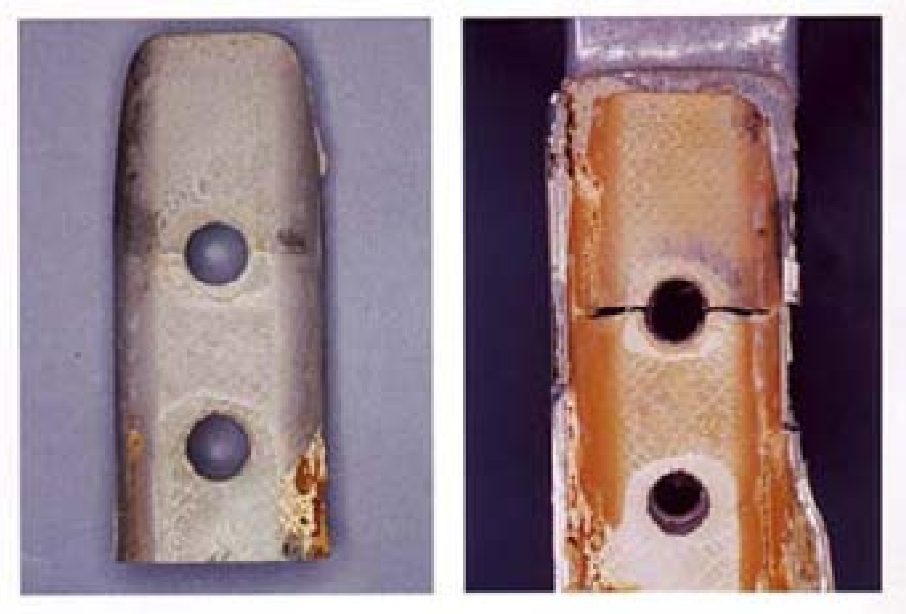


a) The root fitting side of the fracture

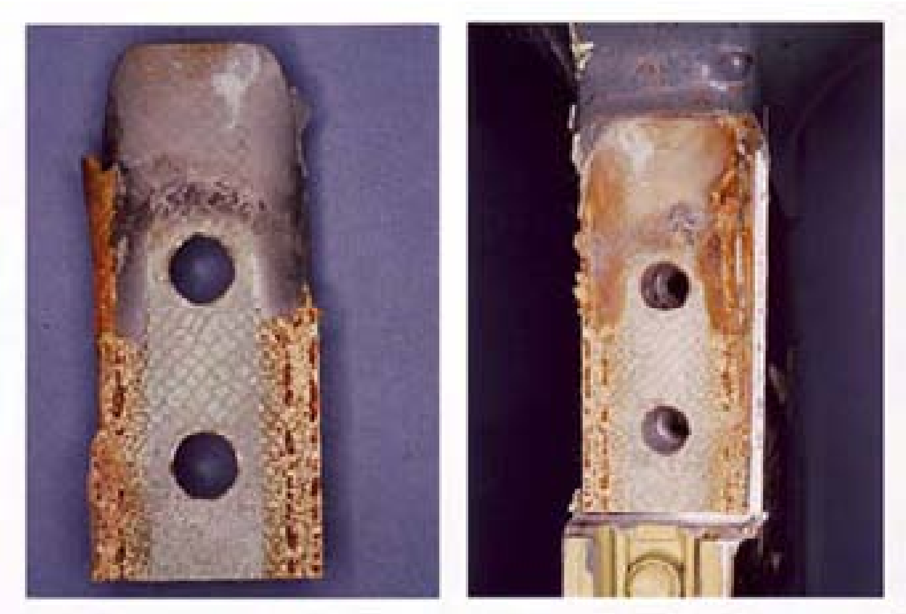


b) The blade side of the fracture

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

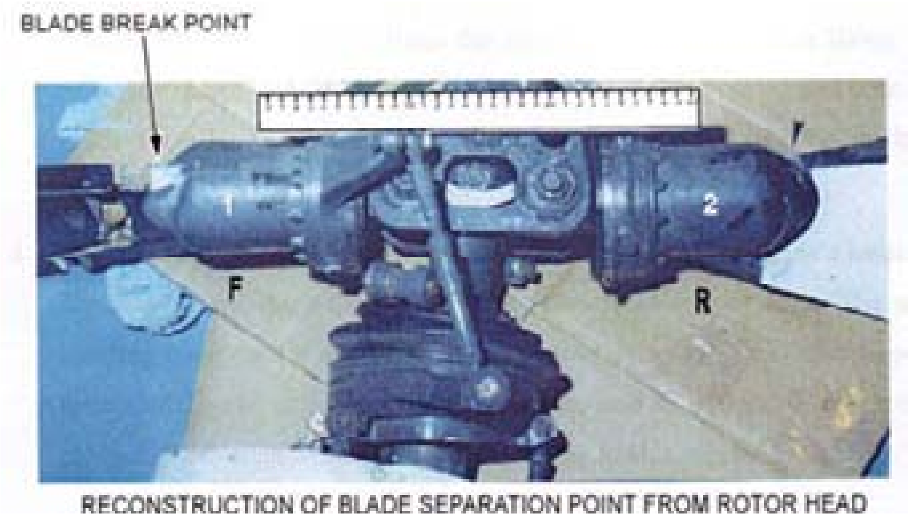
The State of Israel Ministry of Transportation investigation, Acc No 6-04

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

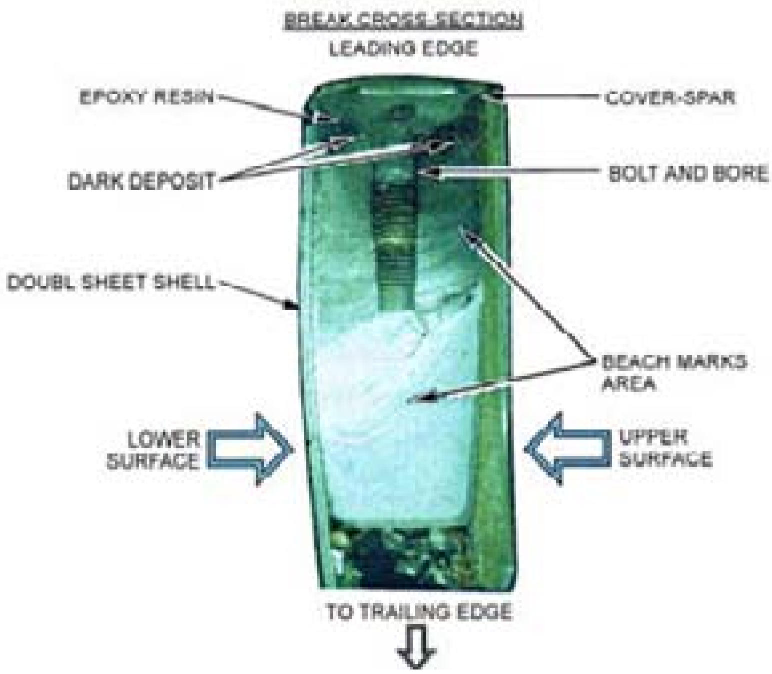
Time since new: approximately 1490 hours, 11.8 years

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

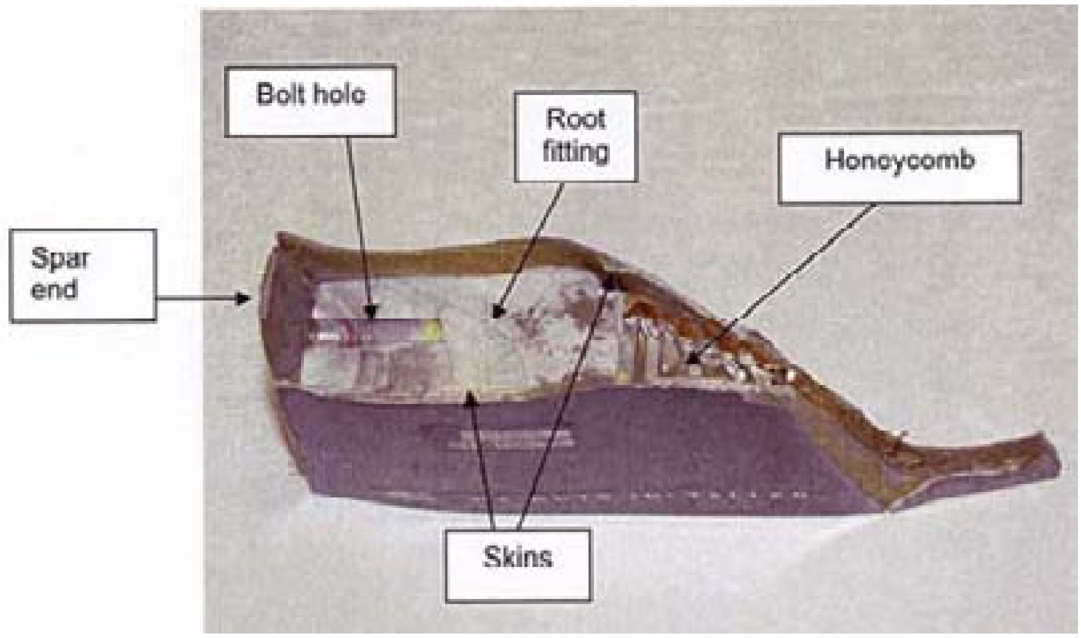
New Zealand Civil Aviation Authority investigation

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"

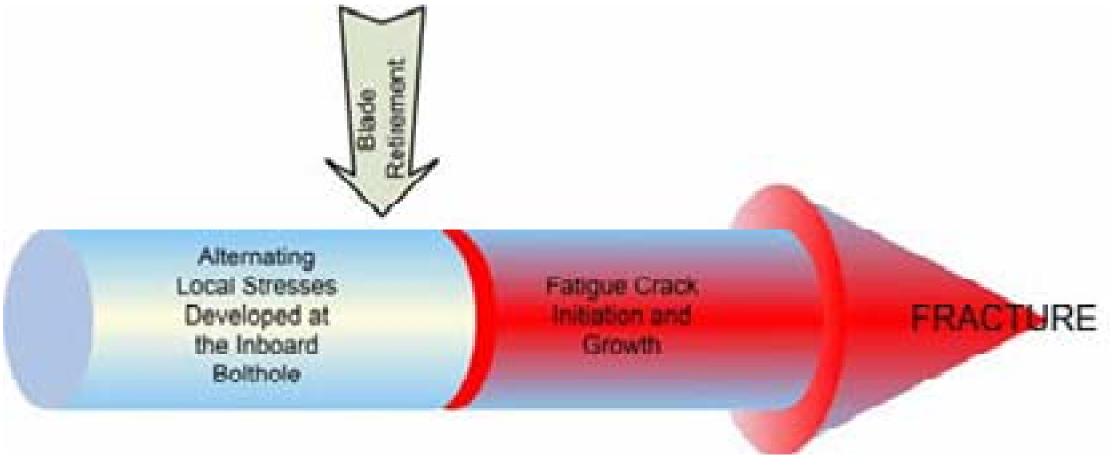
5 ANALYSIS

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.

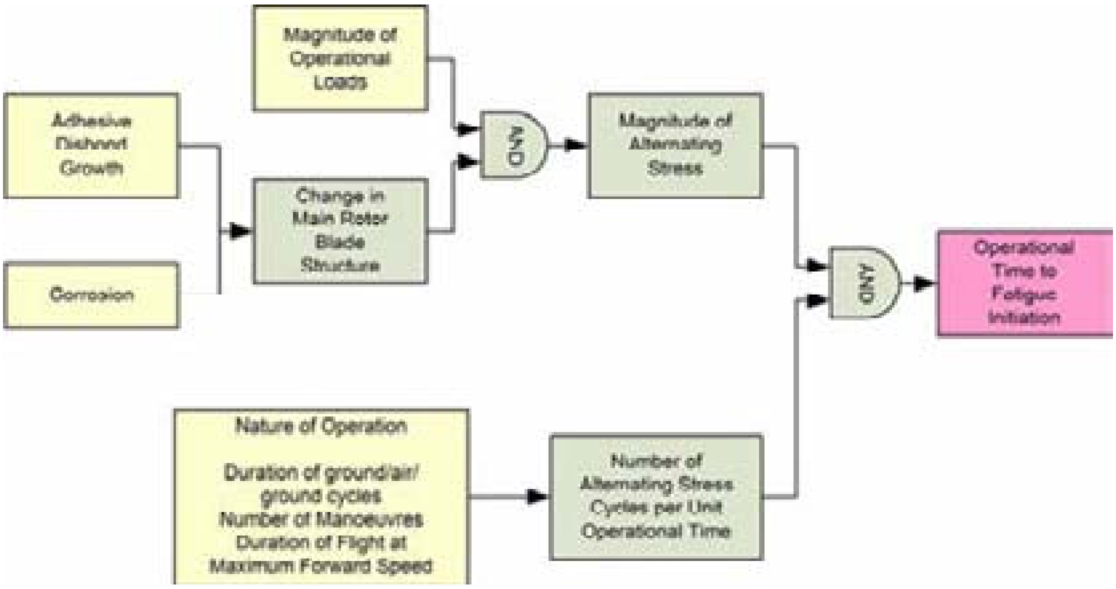
However, two instances of blade fracture have occurred at times within the specified retirement time (Israel, 4X-BCM; New Zealand, ZK-HWP).

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.

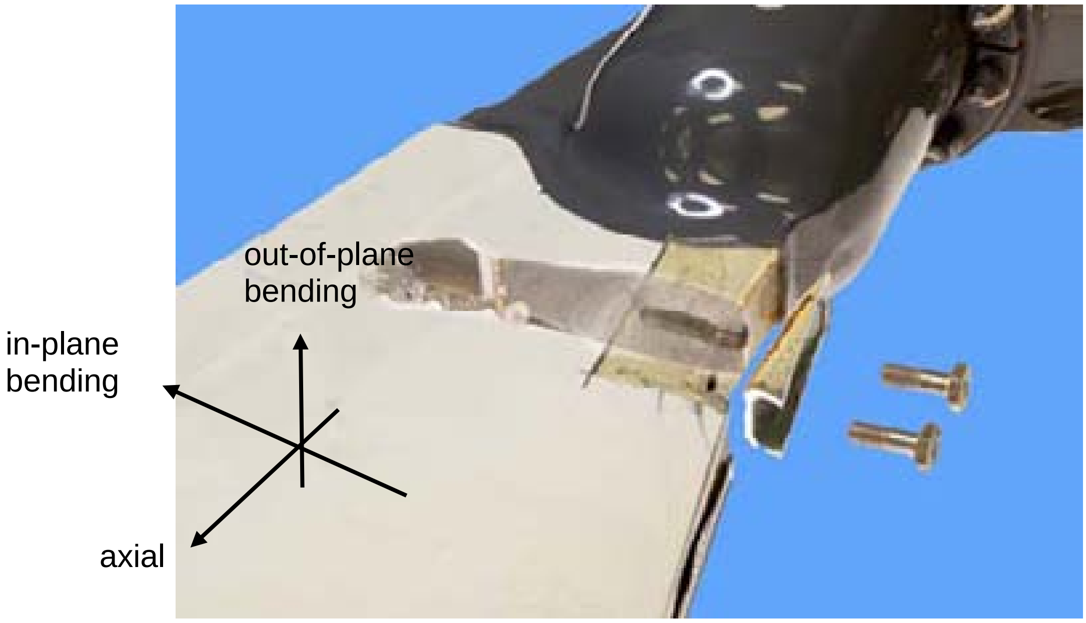
In-plane bending loads on a main rotor are created by the drag forces generated by main rotor rotation and helicopter flight. The magnitude of drag loads will vary with blade angle of attack and relative airspeed over the blade. Stress cycles may occur several times in each revolution of the main rotor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

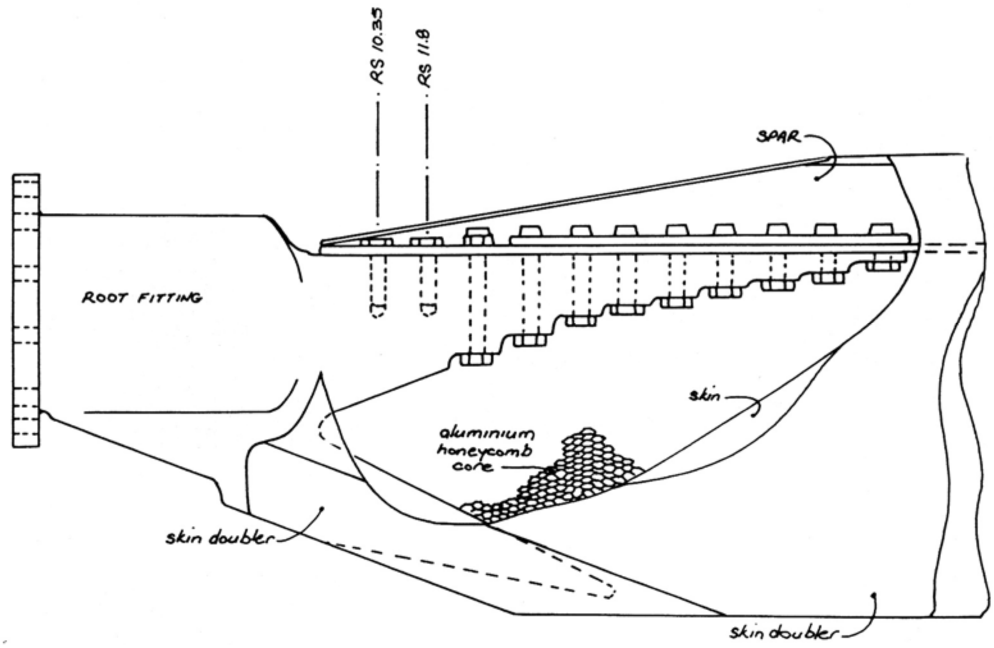


[2] R W Prouty, 'Practical Helicopter Aerodynamics', reprints of 'Aerodynamics' columns that have appeared in Rotor & Wing International magazine from 1979 to 1982

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

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[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.

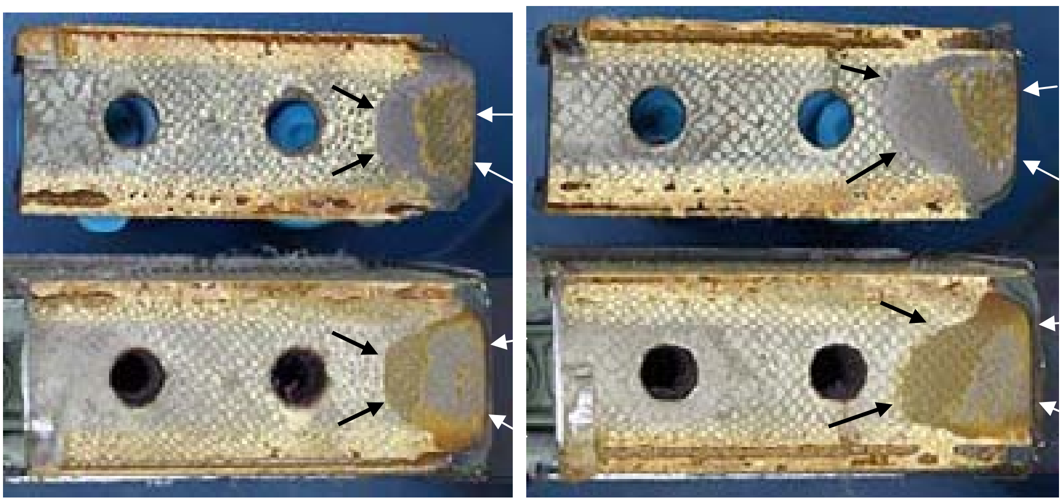
5.3 Adhesive Disbonding Survey

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.

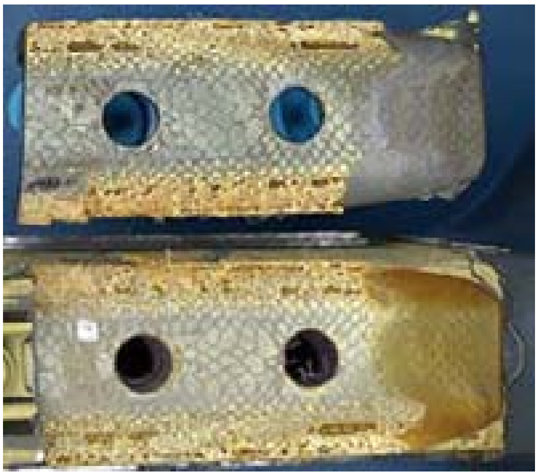
Figure 31: The extent of disbonding between the root fitting and inboard end of the blade spar – helicopters used in cattle mustering operations



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

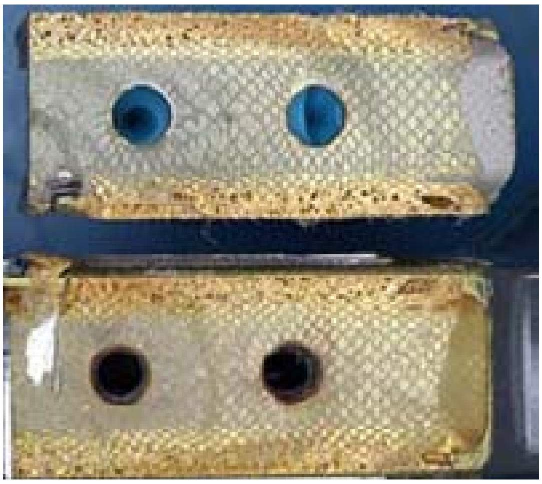
Regions of disbond are characterised by the darker, honey brown colour and regions were the metallic surface of the spar is exposed. Regions were the intact bond was fractured during the strip down process are characterised by a light cream colour and a rougher appearance. The cross-hatched appearance 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

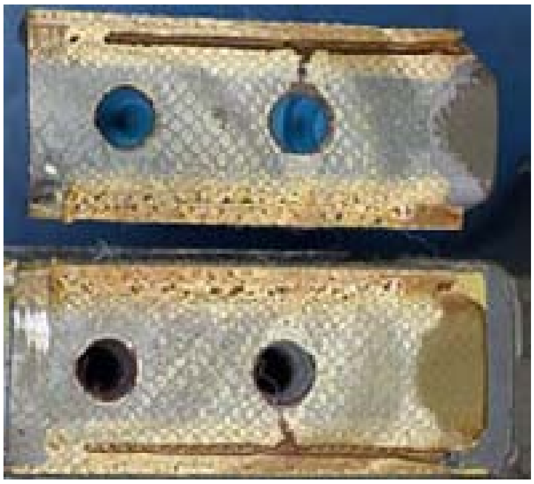


b) VH-LOT s/n 8411B

VH-LOT s/n 8414B (operational time approximately 1700 hours)

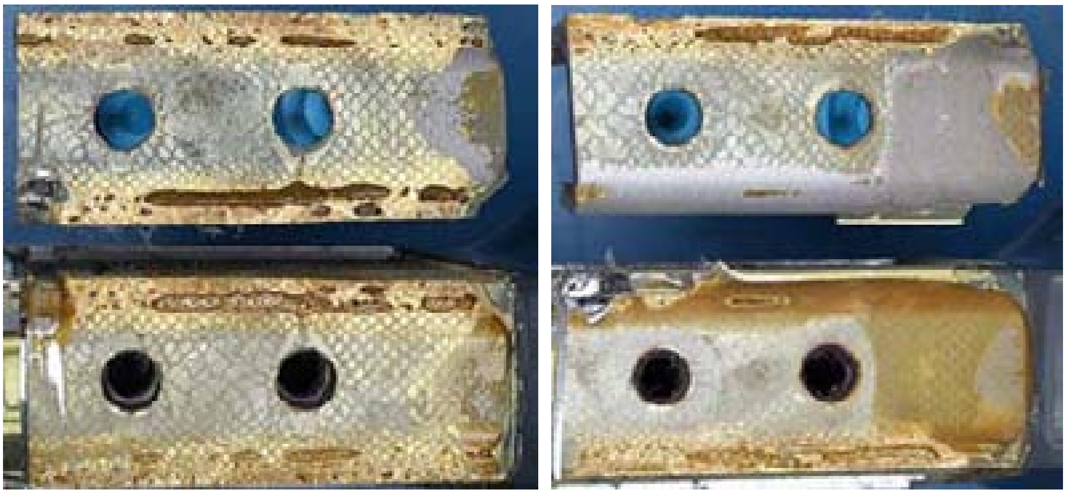


c) VH-HCF s/n11303A



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

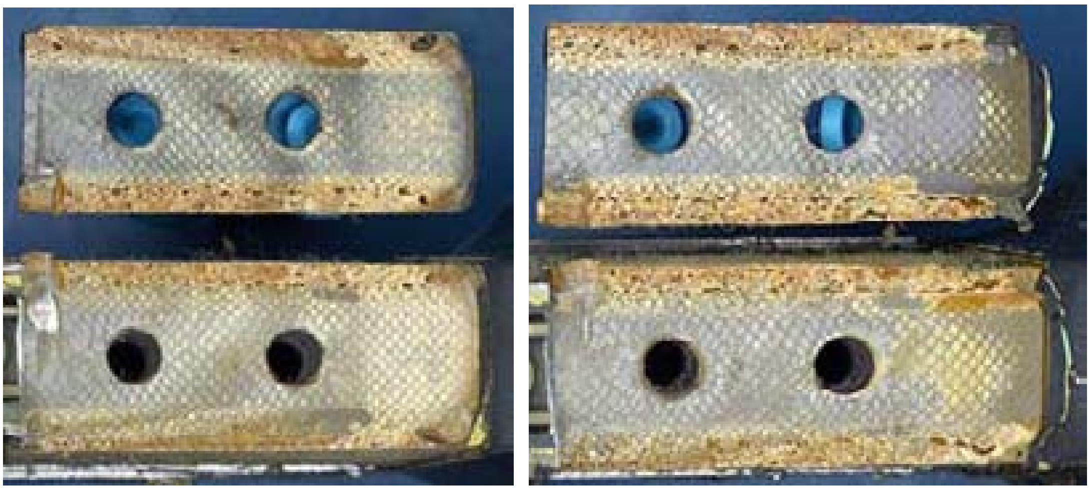
Figure 32: The extent of disbonding between the root fitting and inboard end of the blade spar – helicopters not used in cattle mustering operations



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.

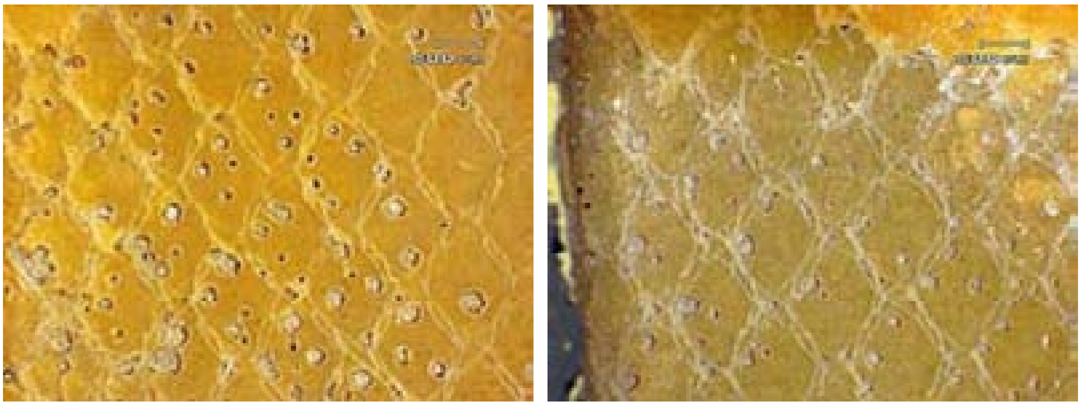
The factors that determine whether disbonding will occur during operation and those that determine the rate of disbond growth to the inboard bolthole can be grouped into three categories; initial conditions, operating conditions and mitigating conditions.

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.

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[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

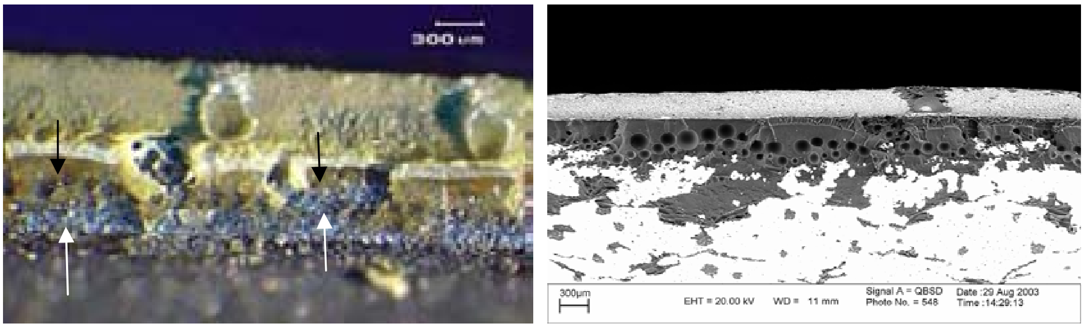
Figure 36: The distribution of adhesive material bonded to the spar, VH-LDR

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

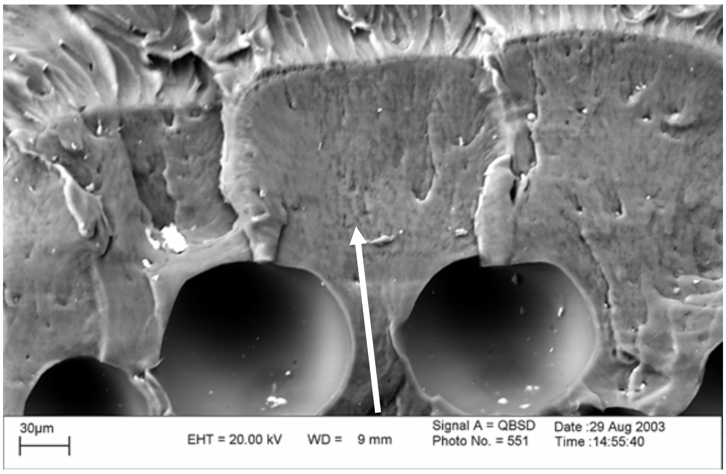
b) VH-LDR intact blade, adhesive bonded to the spar, at the spar end is arrowed

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.

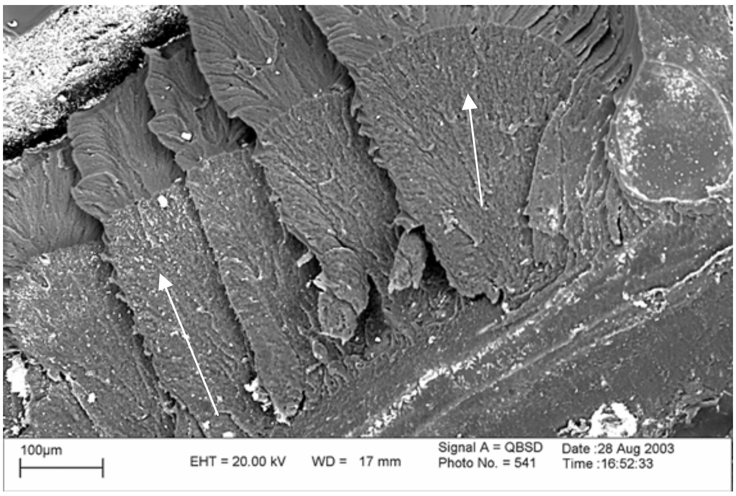
Figure 37: Examples of fatigue cracking in the adhesive fillet at the end of the adhesive joint



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.

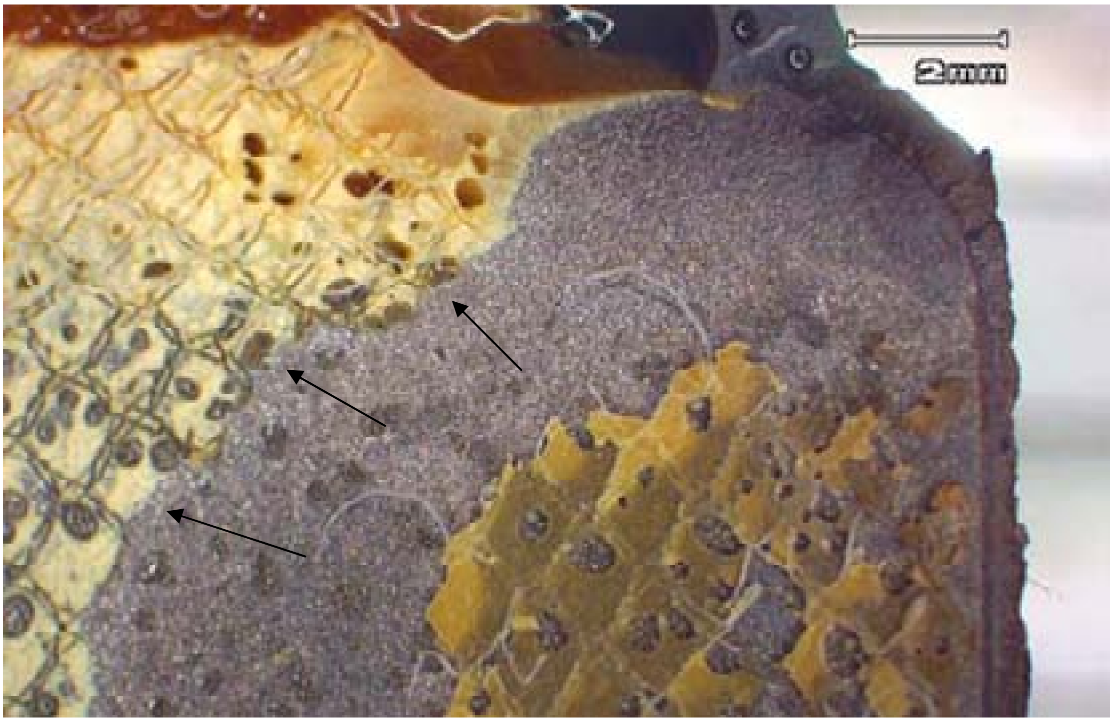
Figure 38: Examples of the geometric and surface finish variations at the spar end

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

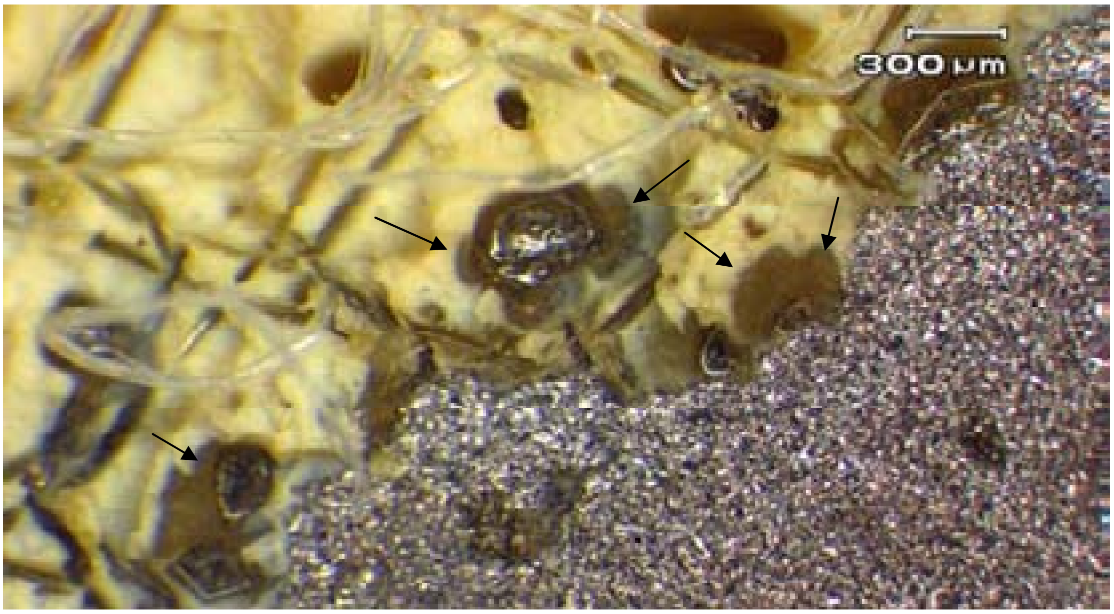
b) Blade s/n 8603B, showing a rounded corner between the spar end and bonded surface, plus surface roughening by grit blasting

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.

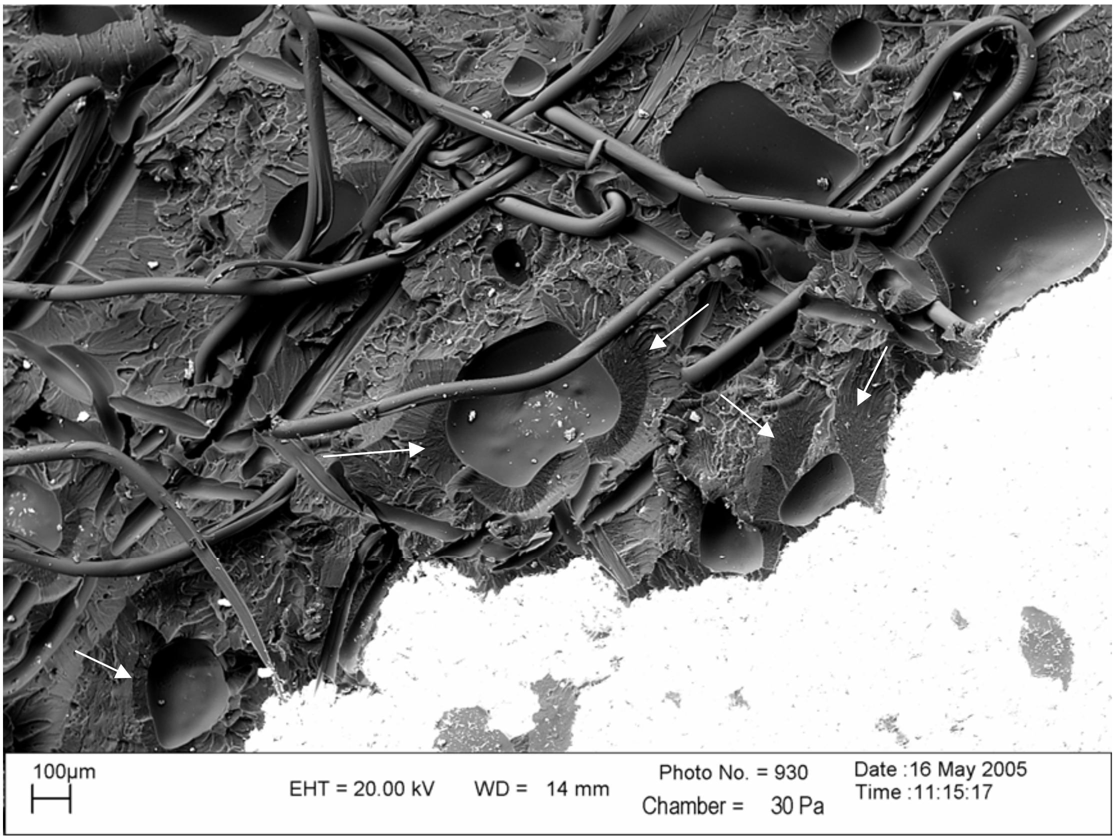


b) Regions of crack growth from voids near the disbond boundary are coloured darker than the region of fracture induced in the adhesive during strip down, examples are arrowed. The clear fibres are from the scrim fabric used in the manufacture of the adhesive film.

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.

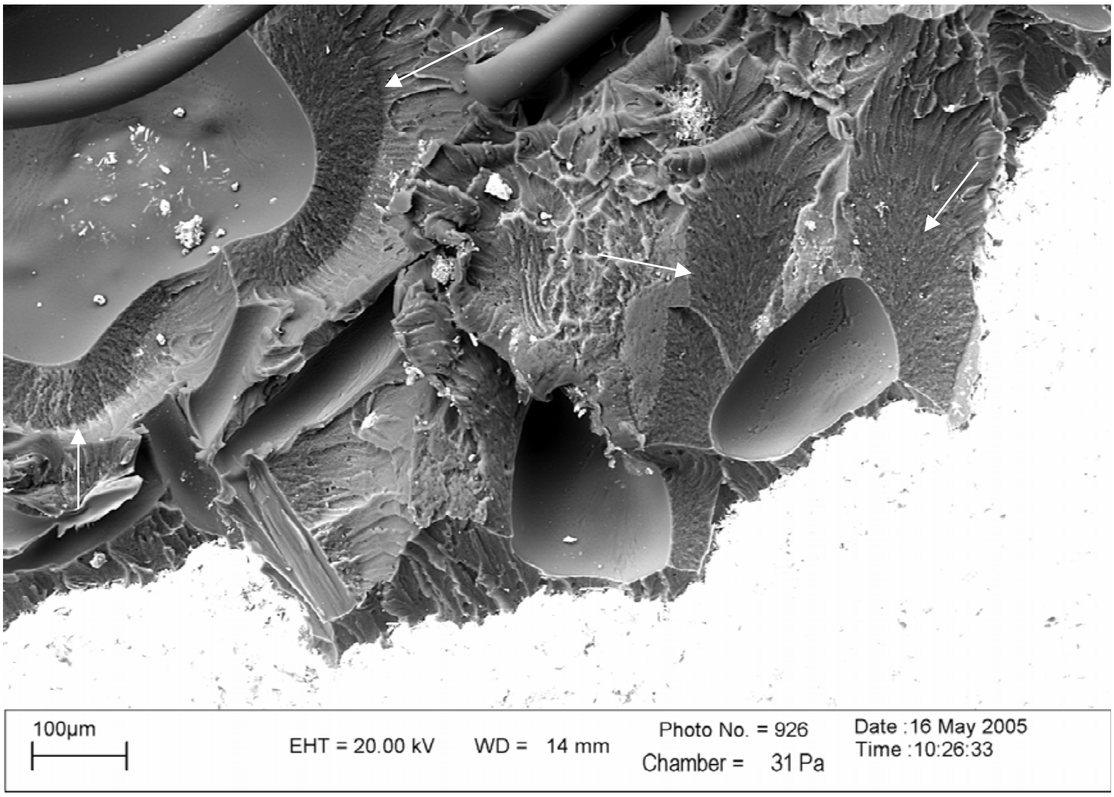
The regions of crack growth arrowed in figure 40 display surface features that are different from the surrounding surfaces that have been created by local tearing in the adhesive when the spar was peeled from the fitting. This difference in surface features supports the hypothesis that crack growth from 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

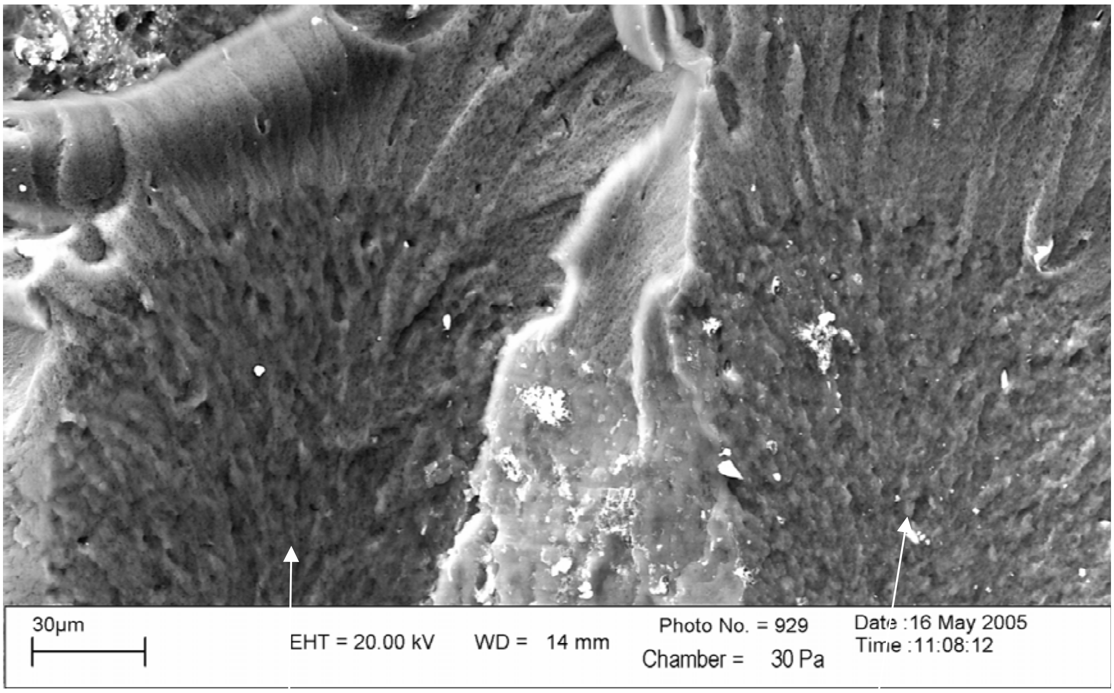


a) Regions of crack growth from voids near the disbond boundary are arrowed

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.

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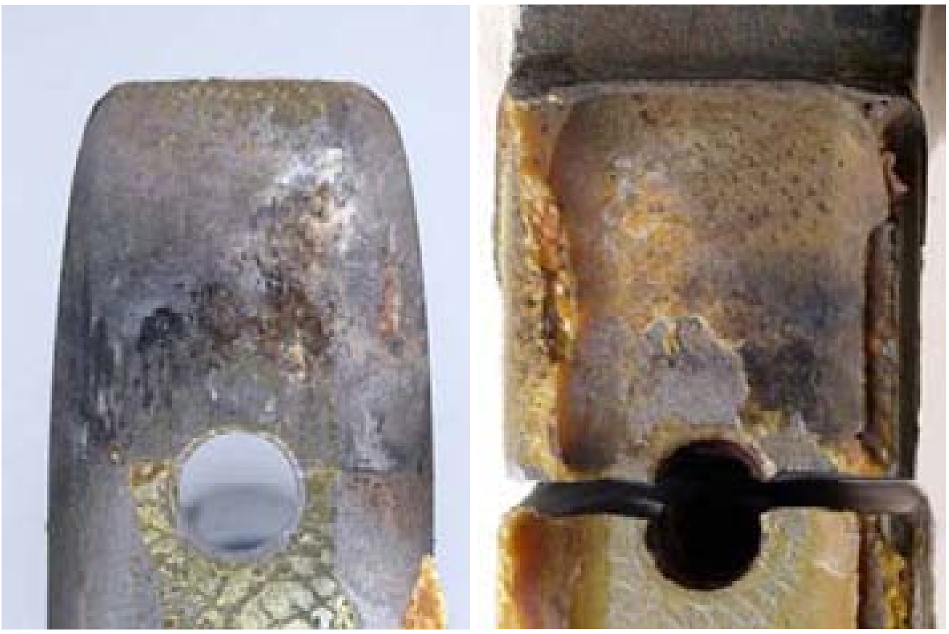
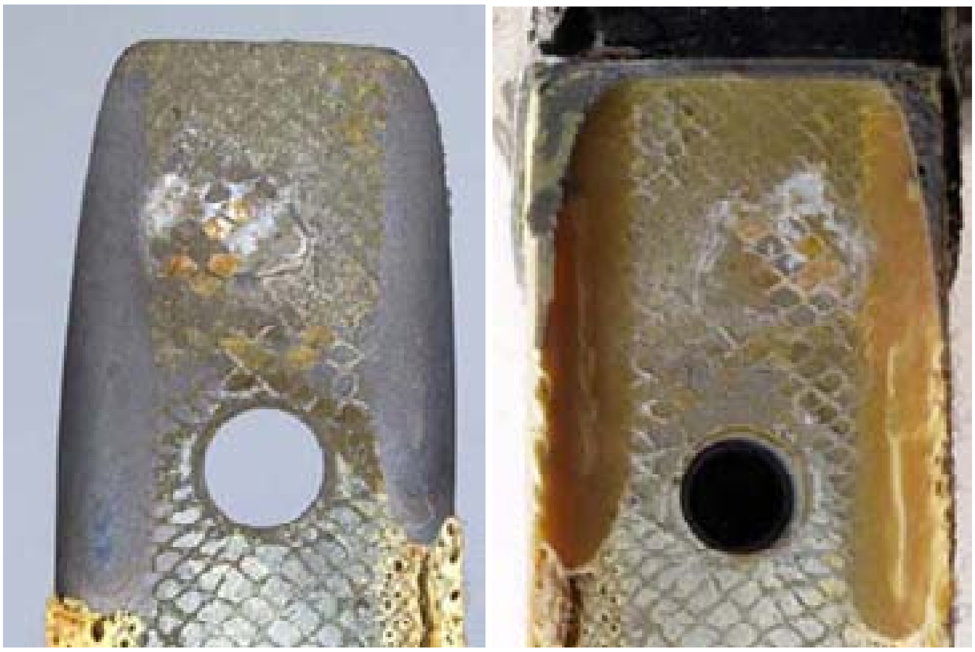
[7] L J Hart-Smith, 'An Engineer's Viewpoint on Design and Analysis of Aircraft Structural Joints', International Conference on Aircraft Damage Assessment and Repair, Melbourne Australia August 26-28, 1991, Douglas Aircraft Company Paper MDC 91K0067

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

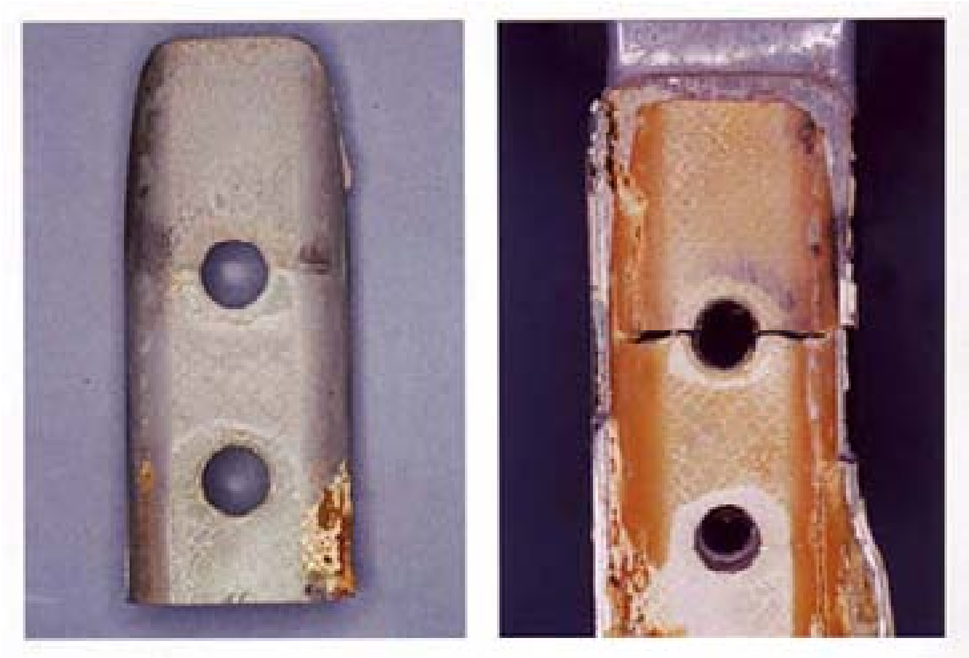


a) VH-OHA fractured blade

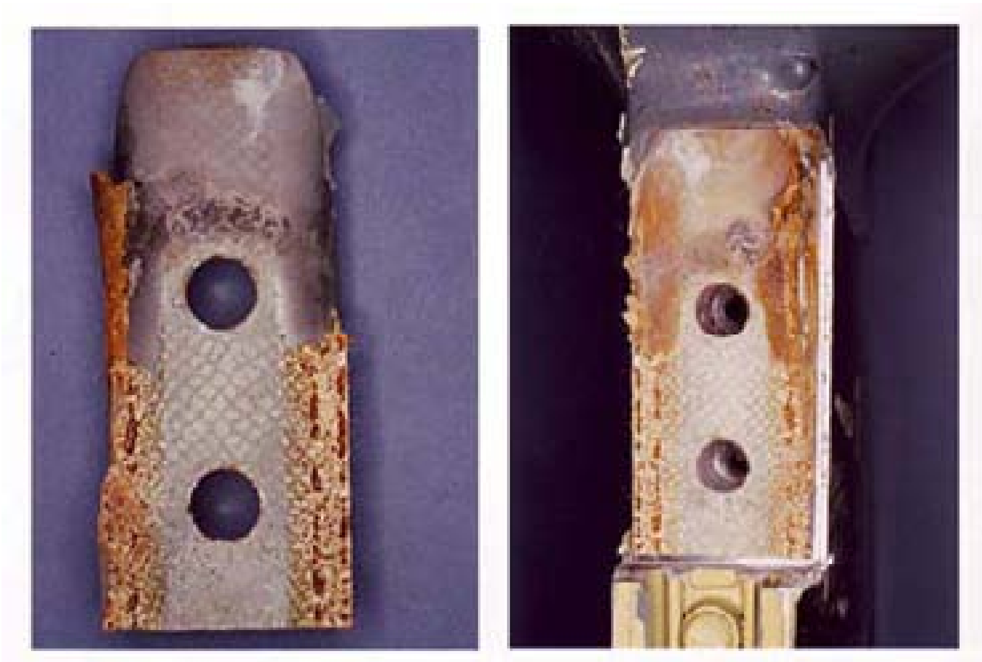


b) VH-OHA intact blade

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

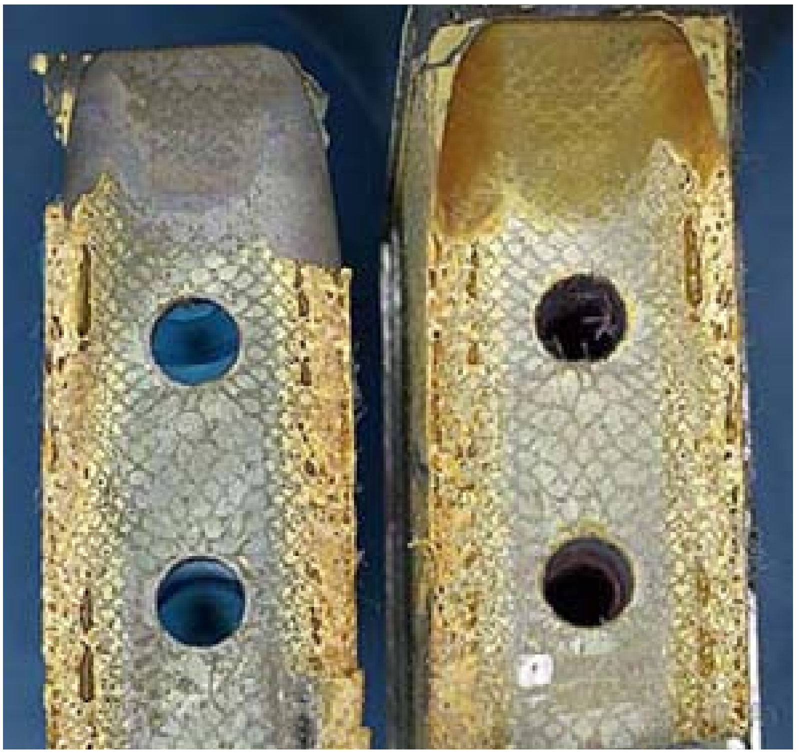


a) VH-LDR fractured blade



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

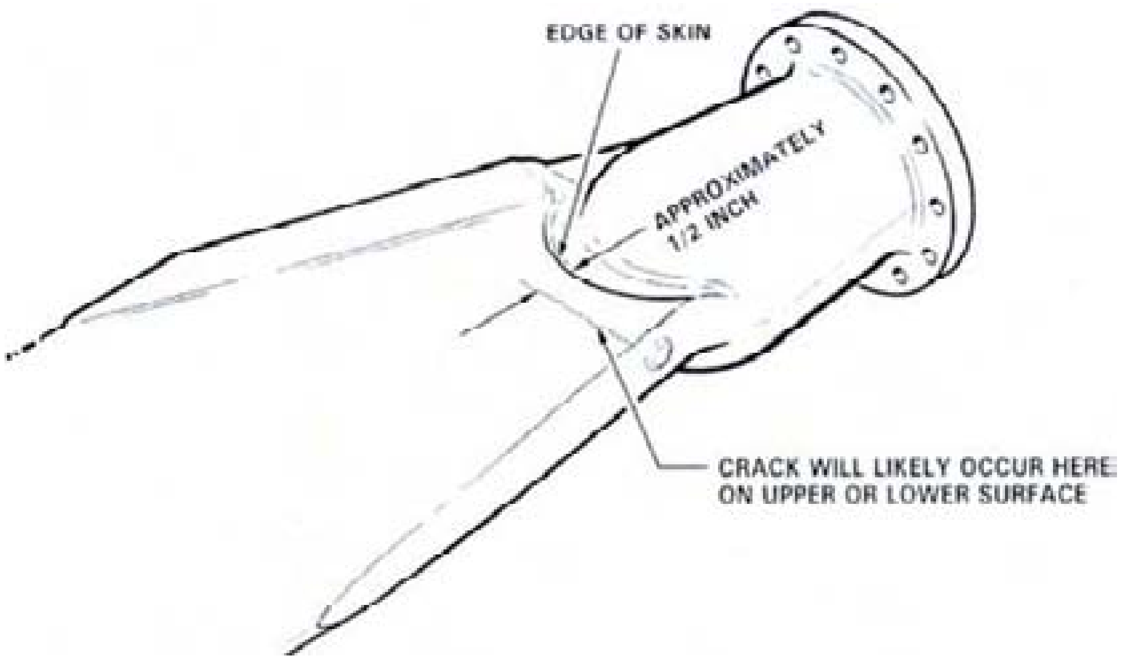
BACKGROUND: A main rotor blade fatigue failure could occur if the helicopter is repeatedly flown above its approved gross weight limit or operated above its approved manifold pressure limits. The first indication of a fatigue crack in progress may be a rotor that will not stay balanced after being adjusted.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

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[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).

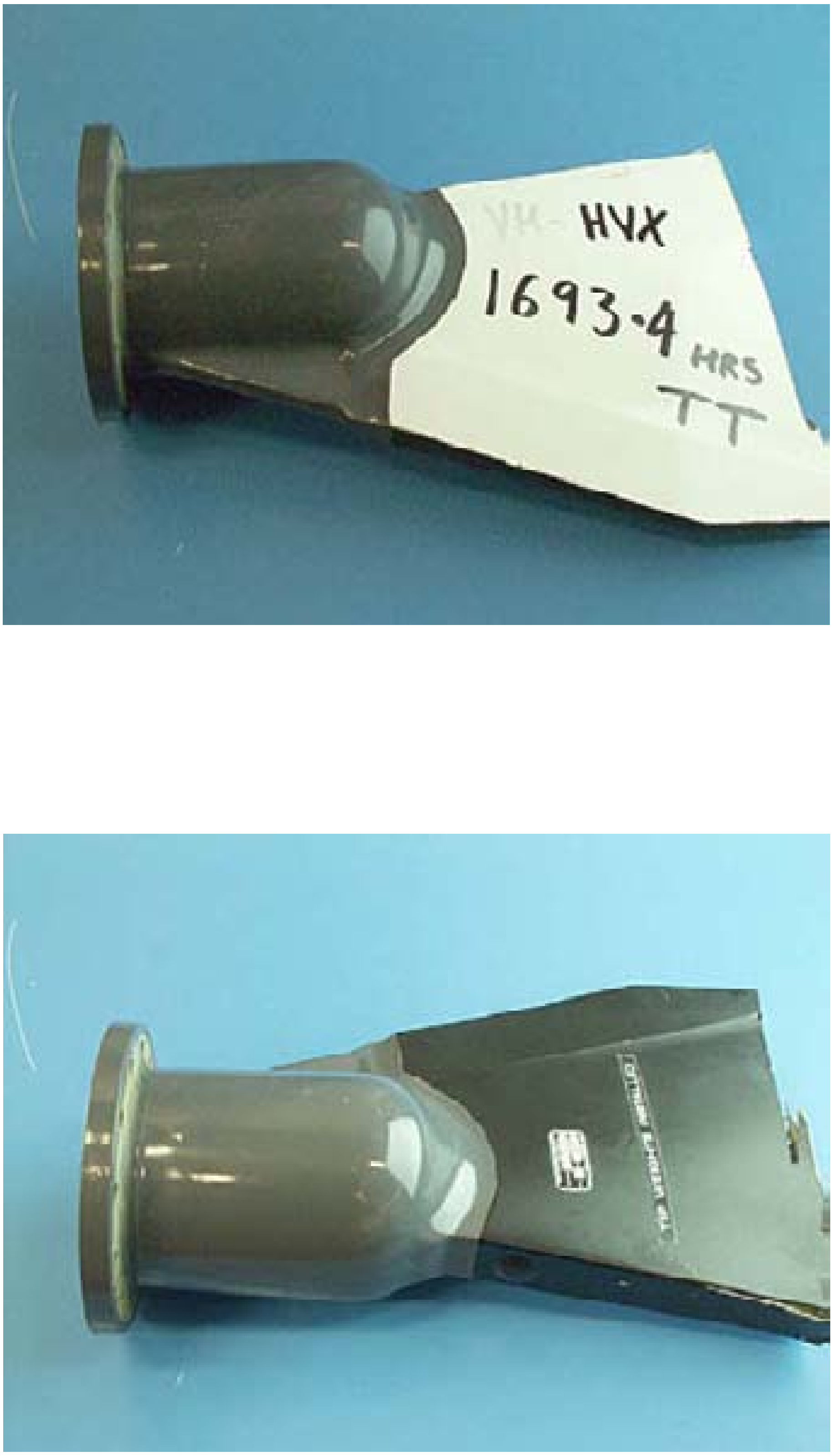


Figure 1. General condition of blade root end, upper and lower blade surfaces.

Figure 2. Serial number decal

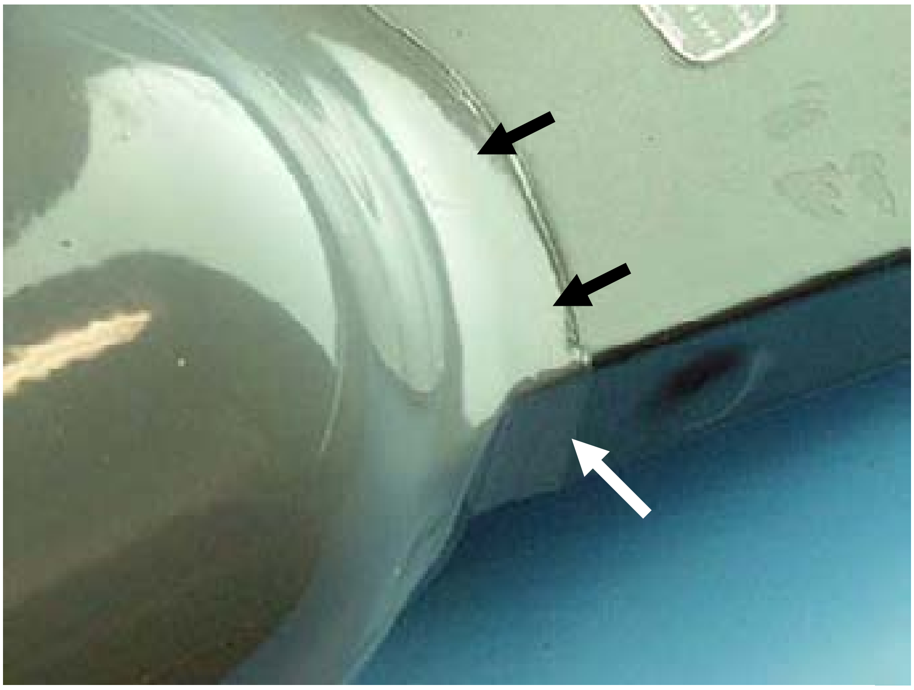


Figure 3. Cracks in filler at the end of the spar and lower skin

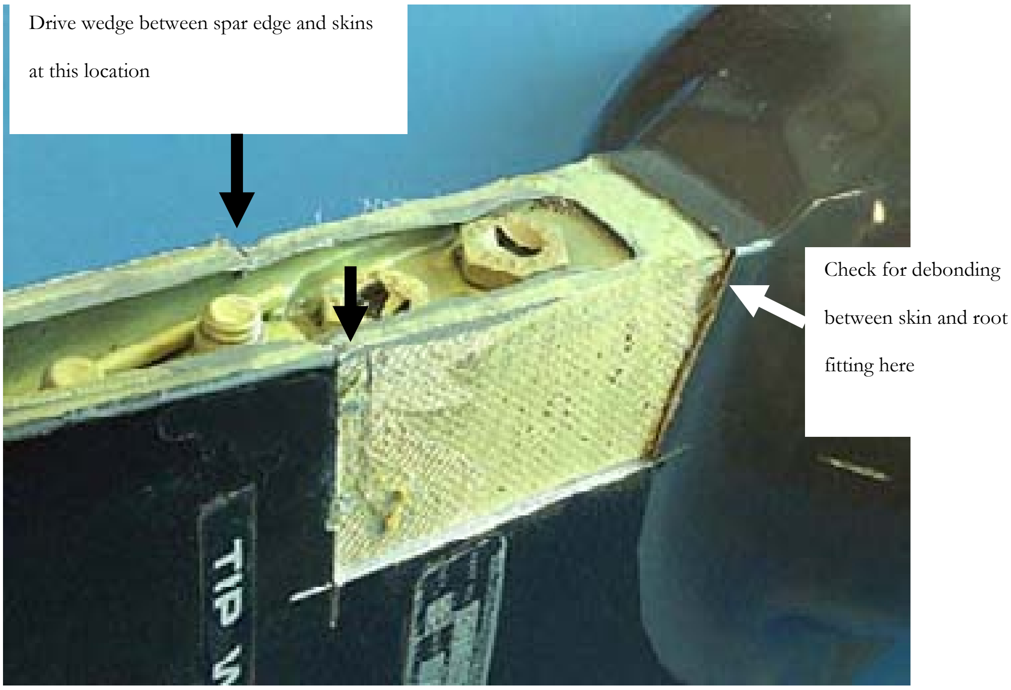


Figure 4. Blade skin removal

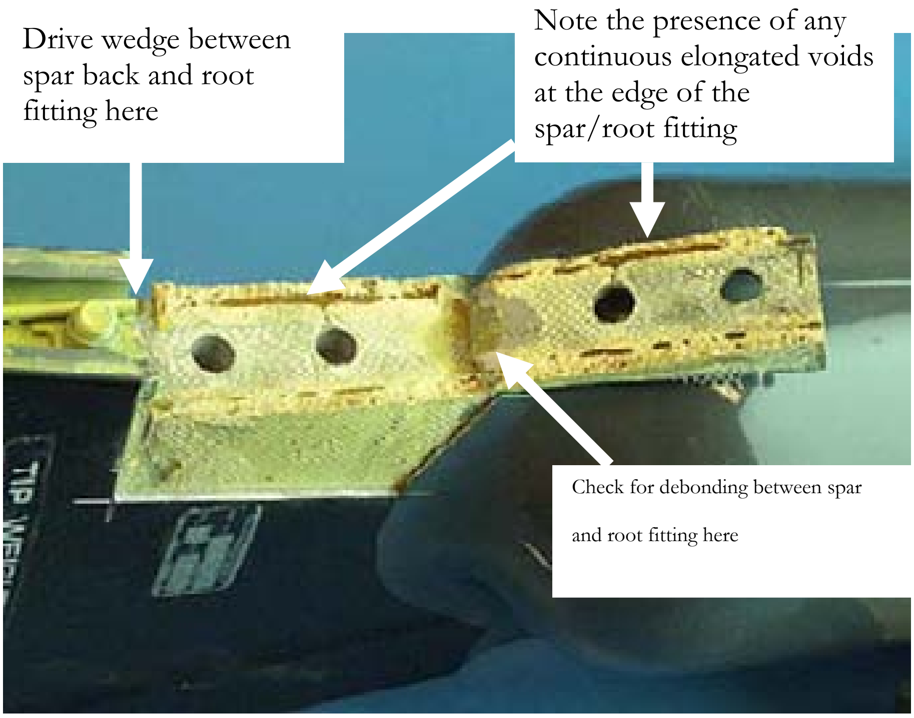
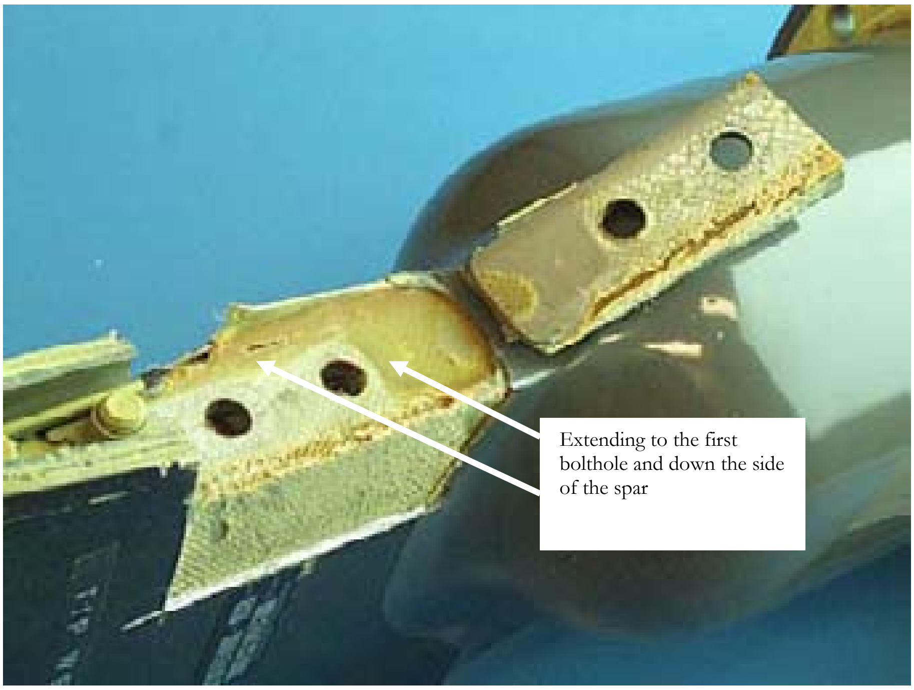


Figure 5. Blade spar removal



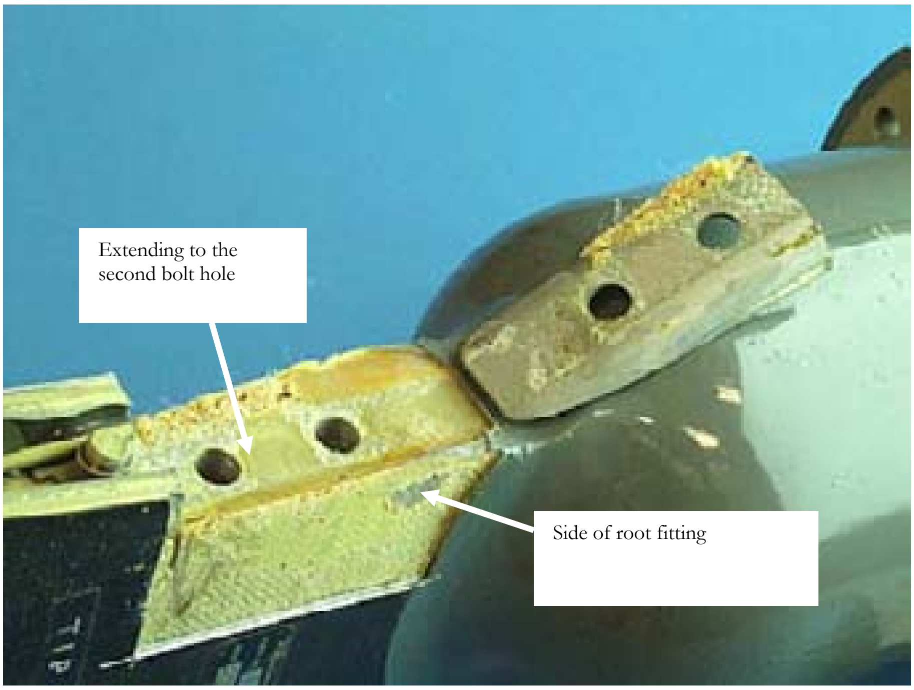


Figure 6. Examples of varying extents of adhesion failure (debonding)

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