Final Report AO-2015-003: Robinson R44, Main rotor blade failure,
Waikaia, Southland, 23 January 2015

Final Report

Aviation inquiry AO-2015-003
Robinson R44
Main rotor blade failure
Waikaia, Southland
23 January 2015

Approved for publication: May 2019

Transport Accident Investigation Commission

About the Transport Accident Investigation Commission

The Transport Accident Investigation Commission (the Commission) is a standing commission of inquiry and an independent Crown entity responsible for inquiring into maritime, aviation and rail accidents and incidents for New Zealand, and co-ordinating and co-operating with other accident investigation organisations overseas. The principal purpose of its inquiries is to determine the circumstances and causes of the occurrences with a view to avoiding similar occurrences in the future. Its purpose is not to ascribe blame to any person or agency or to pursue (or to assist an agency to pursue) criminal, civil or regulatory action against a person or agency. The Commission carries out its purpose by informing members of the transport sector and the public, both domestically and internationally, of the lessons that can be learnt from transport accidents and incidents.

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Important notes

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Citations and referencing

Information derived from interviews during the Commission's inquiry into the occurrence is not cited in this final report. Documents that would normally be accessible to industry participants only and not discoverable under the Official Information Act 1982 have been referenced as footnotes only. Other documents referred to during the Commission's inquiry that are publicly available are cited.

Photographs, diagrams, pictures

Unless otherwise specified, photographs, diagrams and pictures included in this final report are provided by, and owned by, the Commission.

Verbal probability expressions

The expressions listed in the following table are used in this report to describe the degree of probability (or likelihood) that an event happened or a condition existed in support of a hypothesis.

Terminology (adopted from the Intergovernmental Panel on Climate Change)	Likelihood of the occurrence/outcome	Equivalent terms
Virtually certain	> 99% probability of occurrence	Almost certain
Very likely	> 90% probability	Highly likely, very probable
Likely	> 66% probability	Probable
About as likely as not	33% to 66% probability	More or less likely
Unlikely	< 33% probability	Improbable
Very unlikely	< 10% probability	Highly unlikely
Exceptionally unlikely	< 1% probability	

ZK-HPC R44 II

Source: mapsof.net

Location of incident

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Abbreviations

[x]″ Hg	[x] inches of mercury
°C	degrees Celsius
AD	airworthiness directive
CAA	Civil Aviation Authority of New Zealand
CAR	Civil Aviation Rules
Commission	Transport Accident Investigation Commission
FAA	Federal Aviation Administration (United States)
NPRM	Notice of Proposed Rulemaking
NTSB	National Transportation Safety Board (United States)
Quest	Quest Integrity NZL Limited
Robinson	Robinson Helicopter Company
SB	service bulletin

Glossary

chord length	a straight line joining the centres of the leading and trailing edges of an aerofoil section, such as a wing or rotor blade
design life	the period of time during which an item is expected by its designers to work within its specified parameters; in other words, the life expectancy of the item
doubler	an additional piece of material used to strengthen the area of surrounding structure in a component
fatigue life	the number of applications of a given stress to which a sample of metal can be subjected before failing
knots	nautical miles per hour, equivalent to 1.85 kilometres per hour
service life	the lifespan of an aircraft or aircraft component from manufacture to 'end of life' (e.g. scrapping), which may be based on operating hours, cycles, landings or calendar time, or combinations of these
spar	the main load-carrying structural component of a wing or rotor blade
strain gauge	a device used to measure the strain (or stress) on an object. It consists of a conductive metallic strip that changes its electrical resistance when stretched

Data summary

Aircraft particulars

Aircraft registration:	ZK-HPC
Type and serial number:	Robinson R44 Raven II, 10525
Number and type of engines:	one Lycoming IO-540-AE1A5, normally aspirated
Year of manufacture:	2004
Operator:	High Country Helicopters Limited
Type of flight:	agricultural spraying
Persons on board:	one
Pilot's licence:	commercial pilot licence (helicopter)
Pilot's age:	27
Pilot's total flying experience:	1,630 hours
Date and time	23 January 2015, 1200[1]
Location	Waikaia
	latitude: 45° 38´ 30" S
	longitude: 168° 55´ 17" E
Injuries	nil
Damage	moderate

[1] Times are New Zealand Daylight Time (Co-ordinated Universal Time +13 hours) and in the 24-hour format.

1. Executive summary

1.1. On 23 January 2015 a Robinson R44 helicopter was being used to spray gorse near Waikaia. The pilot was making repeated spray runs and turning the helicopter through 180 degrees at the end of each run to position the helicopter for the next spray run.

1.2. The helicopter had just completed a turn when the pilot felt an unusual and significant vibration. The pilot landed the helicopter immediately and discovered a large crack in one of the main rotor blades.

1.3. The Transport Accident Investigation Commission (Commission) found that the point of origin of the crack in the main rotor blade had features consistent with metal fatigue, and that the crack was not the result of any manufacturing defects in the materials or in the construction of the blade.

1.4. The Commission also found that the helicopter had been operated primarily for agricultural flying, with the engine power likely to have exceeded the allowable limit at times. The types of turn used by the operator during spraying operations, especially when the helicopter was close to the maximum permitted all-up weight, likely subjected the main rotor blades to additional stresses not envisaged by the manufacturer.

1.5. The helicopter was not designed specifically for agricultural flying. The manufacturer had therefore not been required to consider the increased loads and cycles of agricultural flying when calculating the service life of the rotor blades.

1.6. A safety issue identified during the inquiry was that aircraft design organisations did not have to consider whether proposed major modifications to an aircraft would significantly alter the use of the aircraft or adversely affect the service life of any component.

1.7. The Commission recommended that the Director of Civil Aviation:

- consult the original equipment manufacturer when considering a modification or supplemental type certificate that, if approved, could result in any aircraft being used in a way that was significantly different from that which the manufacturer originally modelled and used as the basis for determining component fatigue life and the aircraft maintenance programme.

1.8. The key lessons identified during the inquiry into this occurrence were:

- metal fatigue occurs continuously in dynamic components. A fatigue crack can lengthen rapidly and the component lose its structural strength. If an unusual or severe vibration develops in flight, the pilot should land immediately and have the helicopter inspected before further flight
- the key to minimising stress in dynamic components is to fly conservatively, especially when operating close to the published weight, speed and power limits. The Commission has noted in previous inquiry reports that operating an aircraft outside the published limitations significantly erodes the safety margins factored into the service lives of components and can quickly lead to an early catastrophic failure
- operators and maintainers of aircraft that are subjected to cycles or flight profiles that are significantly different from those envisaged by the manufacturers when the aircraft were certificated should consider implementing shorter intervals for component inspections and earlier component replacement times.

2. Conduct of the inquiry

2.1. The Transport Accident Investigation Commission (Commission) became aware of this main rotor blade cracking incident when advised by the Civil Aviation Authority (CAA) during the course of the Commission's inquiry into an accident involving another Robinson R44 helicopter the following month (February 2015).

2.2. An inquiry was subsequently opened on 25 February 2015 under section 13 of the Transport Accident Investigation Commission Act 1990. By this stage the CAA had already sent both of the main rotor blades to the United States for inspection by the manufacturer, Robinson Helicopter Company (Robinson).

2.3. The United States, as the state of manufacture of the helicopter and in accordance with Annex 13 to the Convention on International Civil Aviation, appointed an investigator from the National Transportation Safety Board (NTSB) as its Accredited Representative to the inquiry. The Commission requested the Accredited Representative to oversee the inspections and examinations of the main rotor blades in the United States.

2.4. Contact was established through the Accredited Representative with the aircraft manufacturer, and with the local airworthiness authority, the Federal Aviation Administration (FAA). A visual inspection of the cracked blade was completed at the Robinson factory, with representatives of the NTSB and the FAA present. The examination notes were reviewed by the Commission's investigator in charge.

2.5. In early March 2015 the Commission's investigator in charge interviewed the pilot of the helicopter and the chief pilot for the operator.

2.6. The Commission received a set of time-expired main rotor blades from another R44 that had been frequently used for agricultural operations. These main rotor blades were inspected by staff of the New Zealand Defence Force, using non-destructive methods, to check for any fatigue cracks in the same area where the crack had occurred on the incident blade.

2.7. At the request of the Commission, Robinson undertook a series of flight tests in an R44 fitted with strain gauges on the main rotor blades, to measure the stresses in the rotor blades in normal flight as well as during simulated agricultural spraying operations.

2.8. The NTSB used a scanning electron microscope to examine the cracked blade and looked for indications of cracking on the other blade in the set. The Commission engaged the Defence Technology Agency of New Zealand to review the report produced by the NTSB.

2.9. Statistical data relating to the operation of agricultural helicopters in New Zealand was obtained from the CAA and forwarded to Robinson for the purposes of blade stress analysis.

2.10. On 24 August 2017 the Commission approved a draft report for circulation to interested persons for comment. The Commission received five submissions, and changes as a result of these were included in the final report.

2.11. On 13 December 2017 the Commission approved the publication of the final report.

2.12. On 18 July 2018, as a result of a further submission from the operator, the Commission resumed the inquiry and appointed a new investigator in charge. An expert metallurgist in New Zealand examined both main rotor blades from the incident helicopter and the investigators made further enquiries. A senior investigator of the Australian Transport Safety Bureau independently reviewed the evidential basis for aspects of the original report and the evidence obtained from the further enquiries.

2.13. The Commission considered the information from these further enquiries and changes were reflected in an amended draft report.

2.14. On 20 February 2019 the Commission approved a draft amended report to be circulated to interested persons for comment. The Commission received three submissions, and changes as a result of these have been included in the final amended report.

2.15. This report is the final report of the Commission on this occurrence and replaces its earlier report published on 18 January 2018.

3. Factual information

3.1. Narrative

3.1.1. On 23 January 2015 the pilot was to spray gorse on a block of land near the operator's base at Waikaia. The pilot conducted a pre-flight inspection of the helicopter, including the rotor blades, before the start of the day's flying. No irregularities were noted during this inspection. The pilot completed about three hours of flying that morning without incident.

3.1.2. At about midday, after a downhill spraying run with a light load, the pilot made a turn at about 30 knots[2] airspeed. When pulling out of the turn at an airspeed of 10-15 knots, the pilot felt an unusual and severe vibration, but had no difficulty in maintaining control.

3.1.3. The vibration began when the helicopter was about 200 metres from the landing site, so the pilot elected to land there. After landing, ground staff noticed an abnormal movement of one of the main rotor blades, so the pilot shut down the helicopter immediately. The helicopter was inspected and a large crack noticed on the lower surface of one of the main rotor blades, extending from the trailing edge to just behind the 'D spar[3]' leading edge (see Figures 1 and 2).

3.1.4. The helicopter was taken out of service and both main rotor blades were removed. The blades were shipped to the Robinson factory in the United States for detailed inspection. Here it was noted that the crack appeared to have started in a radius in the trailing edge where the chord length[4] of the blade increased, known as the 'chord transition[5]' (see Figures 2 and 3).

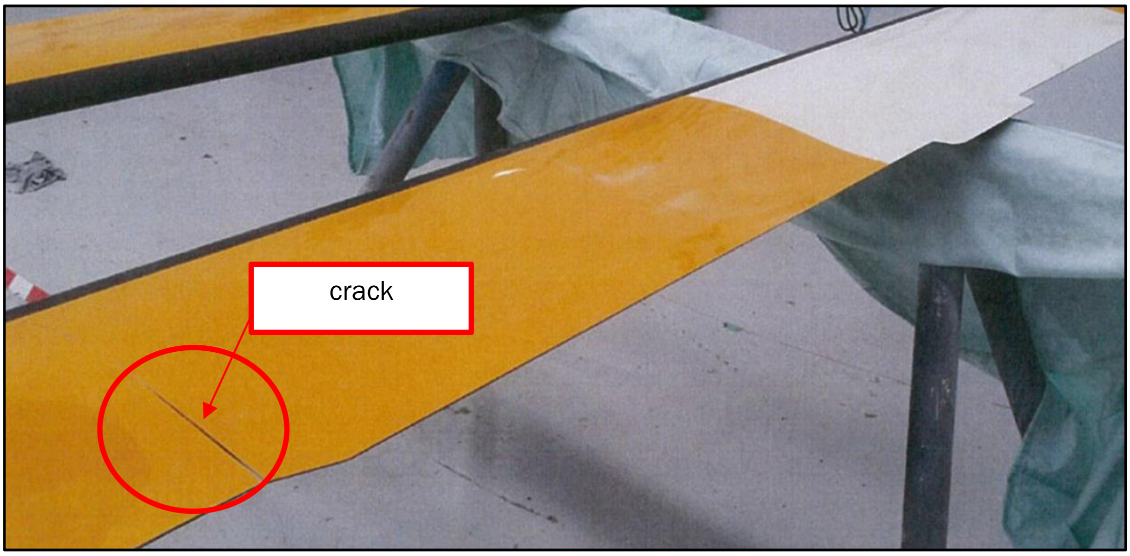

Figure 1
Cracked blade

Source: NTSB

[2] Nautical miles per hour, equivalent to 1.85 kilometres per hour.
[3] The main load-carrying structural component of a wing or rotor blade.
[4] The straight line joining the centres of the leading and trailing edges of a blade.
[5] The chord length increases, from 10 to 10.7 inches (25.4 to 27 centimetres), outboard of a point at approximately 120 inches(304 centimetres) from the centreline of the main rotor driveshaft.

Figure 2
Blade crack at chord transition

Source: NTSB

3.1.5. A section of the blade (part number C016-7, serial number 2031) that contained the crack was removed and sent to the NTSB laboratory, along with a similar section from the opposite blade. A detailed laboratory examination was carried out and a report was provided to the Commission (see Appendix 1).

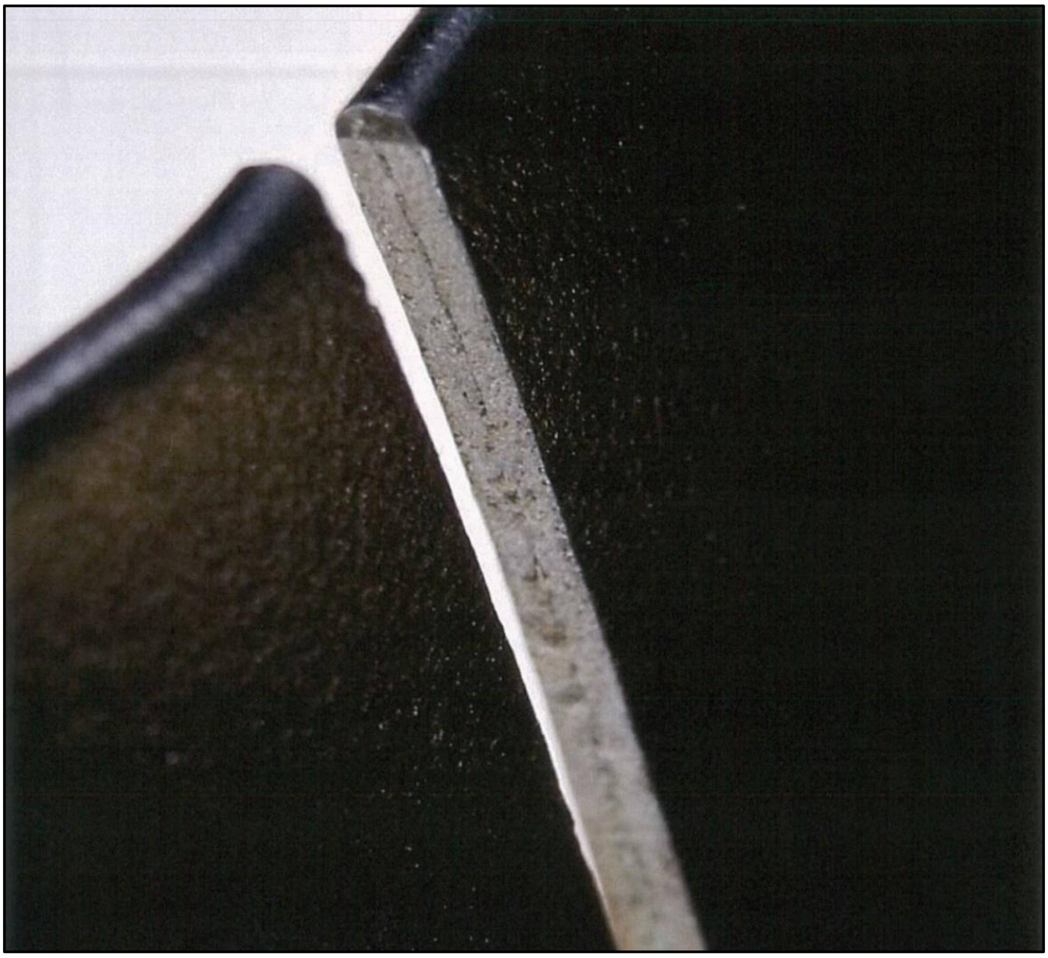

Figure 3
Closer view of crack from underside of trailing edge

Source: CAA

3.1.6. No material defects were found in the area of the fracture initiation in the aluminium skin, nor were there any relevant gaps or failures in the associated epoxy adhesive. The chemical compositions and characteristics of the cracked blade materials matched the design specifications.

3.1.7. The section from the opposite blade was visually examined. No indications of cracking or damage were found. The chord transition inboard radius was measured and it conformed with design specifications.

3.1.8. For comparison, the Defence Technology Agency of the New Zealand Defence Force[6] checked two C016-7 main rotor blades that had been used exclusively on agricultural operations and had reached their service lives of 2,200 hours. The blades were subjected to eddy current[7] and X-ray inspections in the chord transition area. No cracks were found, although indications of minor corrosion were discovered at the trailing edge of one blade.

3.2. Personnel information

3.2.1. The pilot of the incident helicopter had been employed initially as ground crew for the operator, and had been flying for the operator since September 2012. The pilot had been issued with a commercial pilot's licence (helicopter) in August 2010, and had logged a total of 1,630 flight hours, of which 1,173 hours had been on the R44. The pilot's most recent line check had been conducted in December 2013, and the most recent flight crew competency check had been made in February 2014. The pilot held a current Class 1 medical certificate.

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[6] The Royal New Zealand Air Force had expertise in the non-destructive inspection of helicopter main rotor blades.
[7] The use of electromagnetic induction to detect and characterise surface and sub-surface flaws in conductive materials.

3.3. Aircraft information

3.3.1. ZK-HPC was a Robinson R44 Raven II four-seat helicopter, with a maximum all-up weight[8] of 1,134 kilograms. It had been issued with an airworthiness certificate in the Standard Category, but had to be operated in the Restricted Category when the agricultural spray system[9] was installed. In effect, the category change required operations to be conducted under Civil Aviation Rules (CAR) Part 137, Agricultural Aircraft Operations.

3.3.2. The helicopter was primarily used in an agricultural role for spraying liquids. The spray system comprised: two spray booms; two tanks (one either side of the fuselage); and a pump powered by a small petrol engine. The total capacity of the helicopter spray tanks was approximately 500 litres,[10] but helicopter performance and weight limitations meant that less than half of this capacity could be carried. Each spray tank had a contents sight gauge marked in litres.

3.3.3. The main rotor blades were part number C016-7 'Revision AD' blades with aluminium top and bottom skins and an internal aluminium 'honeycomb' bonded to a load-bearing stainless-steel 'D spar'. Each blade's trailing edge was formed by bonding the upper and lower skins with structural epoxy adhesive. A stainless-steel doubler[11] was bonded between the upper and lower skins at the trailing edge (see Figures 4 and 5). The doubler terminated at about two-thirds of the length of the trailing edge from the blade root. A fillet of flexible epoxy sealant along the rear edge of the bond line provided protection from corrosion.

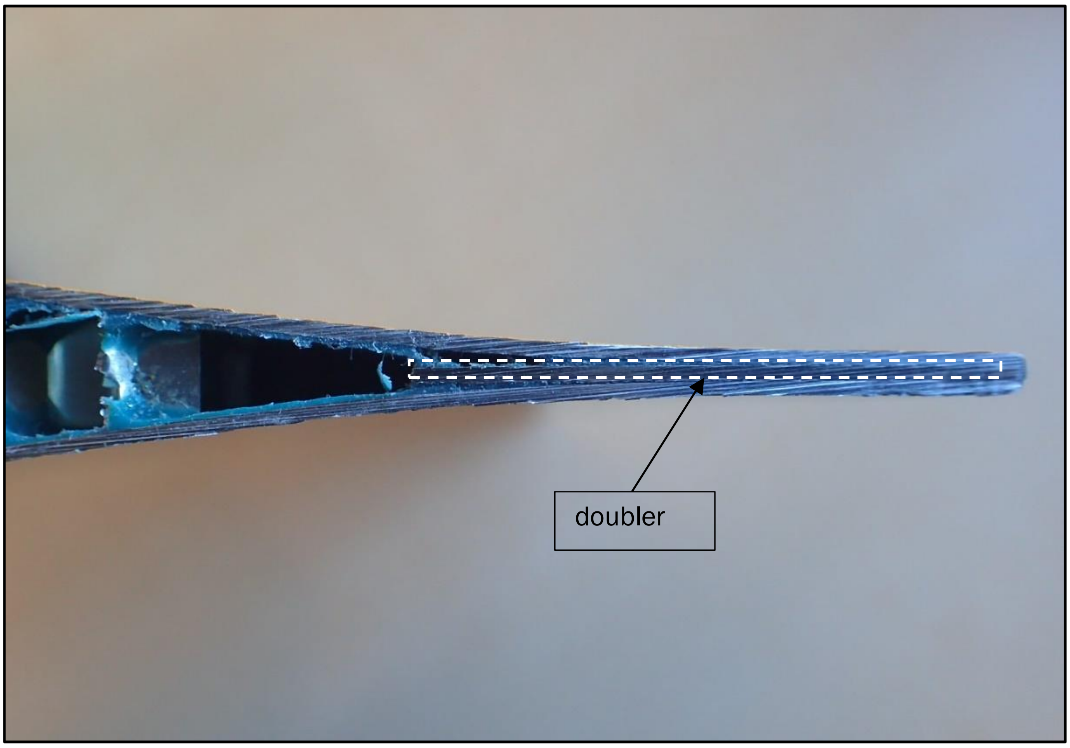

Figure 4
Cross-section of blade trailing edge, with doubler highlighted

[8] The limiting weight specified by the manufacturer and not to be exceeded.
[9] The spray system had been designed by an approved New Zealand aircraft design organisation.
[10] The tank was designed to be much larger than the actual volume of liquid that could legally be carried, to allow for 'foaming' of the liquid to be contained inside the tank.
[11] An additional piece of material used to strengthen the area of surrounding structure in a component.

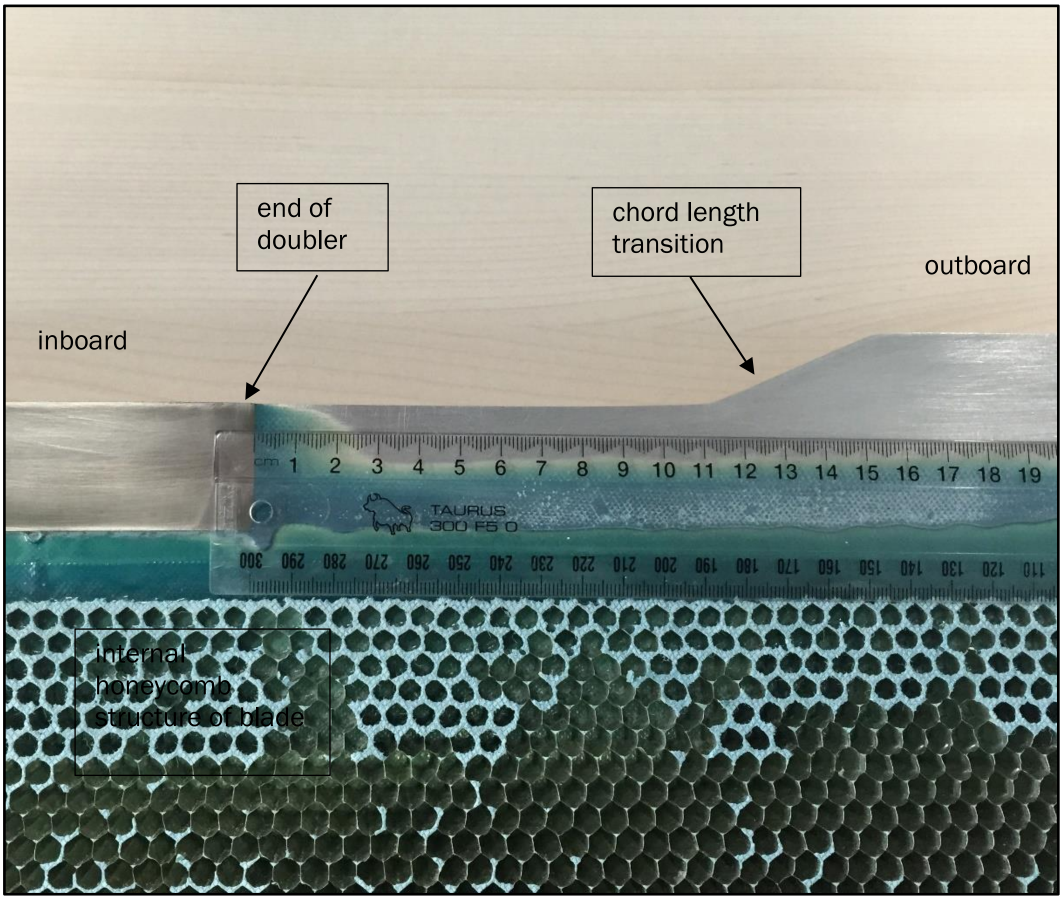

Figure 5
View of C016-7 blade with one skin removed, showing internal doubler

3.3.4. The main rotor blades had been imported into New Zealand in July 2012 and installed on the helicopter in March 2013 after having accrued 15.5 flight hours on another helicopter. They had accrued a further 831.5 flight hours in the 22 months in which they were fitted to the incident helicopter. The operator had experienced no prior defects or damage with this blade set.

R44 main rotor blade design history

3.3.5. The R44 was originally produced with part number C016-2 main rotor blades with stainless-steel skins. The blade planform[12] for the later model R44 Raven II was changed to increase the chord length of the outer section of the blades. The increased blade area produced more lift and allowed operations at heavier gross weights. All R44 blades incorporated stainless-steel trailing-edge doublers that terminated inboard of the chord transition (see Figure 5).

3.3.6. On 3 January 2008, prompted by reports of de-bonding[13] on blades with stainless-steel skins, the FAA issued airworthiness directive[14] (AD) AD 2007-26-12. This AD required a one-time visual inspection for skin separation on the lower surface of each blade.

3.3.7. On 2 June 2011 the AD was superseded by AD 2011-12-10, which required pilots to check the blade skin-to-spar joint area for any bare metal before the first flight of each day. Additionally, within 10 hours' time in service, and thereafter at 100-hour intervals or at each annual inspection, or if any bare metal were found during the pilot check, blades were to be inspected for corrosion, separation, gaps or dents by following certain procedures in Robinson's service bulletins (SBs).[15]

[12] The shape of an object as seen from above.
[13] An undesirable condition where the structural adhesive under the blade skin breaks down or fails as a result of surface erosion, corrosion or overstressing.
[14] A written directive issued by a national airworthiness authority to correct an unsafe condition. Compliance with these airworthiness instructions is mandatory.
[15] A written service instruction issued by the manufacturer. Compliance is mandatory in New Zealand if the issuing authority requires it, or if the service bulletin is included in operator certification requirements.

C016-7 aluminium-skinned blades

3.3.8. In 2007 Robinson developed a new C016-7 main rotor blade to resolve the skin separation and erosion issues with the R44 main rotor blades. The primary design change was the replacement of the stainless-steel skins with aluminium skins.

3.3.9. On 9 January 2015 FAA AD 2014-23-16 mandated the replacement within five years of all stainless-steel blades in service with the new aluminium blades. The embodiment of this AD removed the inspection requirements of AD 2011-12-10. Robinson exchanged part-life stainless-steel blades for discounted new C016-7 blades.

Actions after the incident

3.3.10. On 31 January 2015 the CAA issued Continuing Airworthiness Notice[16] 62-003, which contained details of this incident and photographs of the failed blade. The notice alerted operators to any unusual vibration, and recommended that blades be checked for defects at every pre-flight inspection.

3.3.11. On 23 February 2015 Robinson issued a safety alert[17] (see Appendix 3), which detailed the location of the incident blade's crack and recommended close visual inspections of the trailing edges of blades during daily pre-flight inspections. In addition, the FAA issued Special Airworthiness Information Bulletin SW-15-08, which reiterated the content of the safety alert.

3.3.12. The CAA subsequently issued AD DCA/R44/32, which mandated the inspection requirements of the Robinson safety alert and required an additional visual inspection before each flight.

3.3.13. On 30 March 2015 the safety alert was superseded by Robinson SB-89, which required C016-7 blades to be modified by increasing the radius of the inboard chord length transition (see Appendix 4). On 31 March 2015 the FAA issued 'Revision 1' of Special Airworthiness Information Bulletin SW-15-08, which included the content of SB-89.

3.3.14. In New Zealand, the CAA issued DCA/R44/32C, effective on 24 September 2015. This revision prescribed the modification actions outlined in SB-89, and specified that the modification was to be completed by 19 March 2016. The visual inspection requirements were also clarified in this revision to the AD.

3.3.15. Robinson changed the C016-7 production design to 'Revision AF' (25 March 2015), which included the re-profiled trailing edge at the chord length transition, similar to that achieved by SB-89. 'Revision AG' (31 March 2015) shifted the chord transition inboard by approximately six inches (") (15.2 centimetres), to overlap the trailing edge doubler (see Figure 6). The reasons for and effects of these changes are discussed in section 4.5.

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[16] A written communication published by the CAA, containing important airworthiness-related information.

[17] A written communication of important safety information issued by the manufacturer.

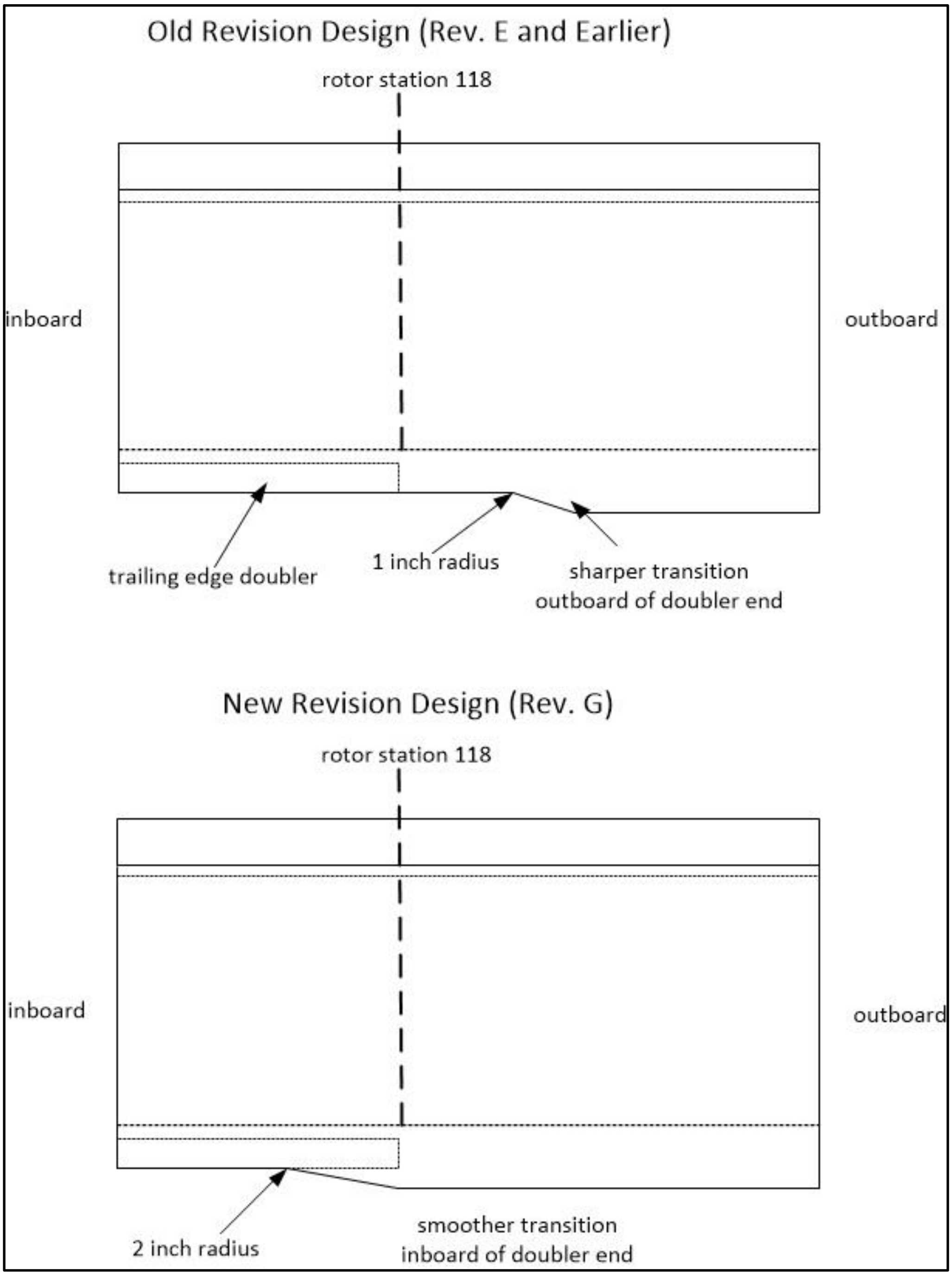

Figure 6
Design changes between C016-7 'Revision AE' and 'Revision AG' blades

Source: Robinson Helicopter Company

3.3.16. On 27 May 2016 the FAA issued a Notice of Proposed Rulemaking (NPRM-81 FR 33609, May-27-2016) for a new AD that included a one-time inspection of Robinson R44 (and R66) main rotor blades. The NPRM stated, in part, that:

> This proposed AD would require a one-time visual inspection of the main rotor blade (MRB) and either removing or altering it. This proposed AD is prompted by a report that a fatigue crack was found at an MRB's trailing edge and a determination that some MRBs may have reduced blade thickness due to blending out corrosion. The proposed actions are intended to prevent an MRB fatigue crack, which could lead to MRB failure and subsequent loss of helicopter control.

3.3.17. According to the FAA, it did not initially consider the New Zealand incident an airworthiness concern that warranted an AD. Following reports of corrosion that had remained undetected between scheduled inspections, and further reports that blending had been carried out in the area of the chord transition radius to remove the corrosion, the FAA re-evaluated the situation. It determined that the presence of corrosion or blending of the trailing edge could reduce the safety margins below an acceptable level.

3.3.18. The NPRM proposed that the modification required by SB-89 would correct this unsafe condition and help to prevent fatigue cracks. The proposed AD would be carried out within 100 hours' time in service or at the next annual inspection, whichever came first. The NPRM was subsequently accepted, and AD 2016-26-04 became effective on 8 February 2017.

3.4. Tests and research

3.4.1. Three independent metallurgical inspections and examinations were carried out in support of this inquiry. They are described in section 4.2 with further detail in Appendices 1 and 2.

3.4.2. After the incident Robinson fitted an R44 helicopter with test equipment to measure the stresses on the chord transition radius of the main rotor blades in normal flight, and during simulated agricultural spraying operations. Turns of low to moderate severity were performed, as well as 'cyclic procedure turns'.[18] These turns were conducted at high all-up weights but without spray equipment installed. The results of these flights are discussed in section 4.3.

3.4.3. The CAA reviewed statistical data pertaining to R44 agricultural helicopter operations in New Zealand for the years 2014 and 2015 to determine the average loads and cycles per hour for R44 helicopters used for spraying. The data was sourced from statistical information required to be reported by all aircraft operators in New Zealand, and from the quarterly operational statistics[19] submitted by agricultural operators. The data included total weights and volumes of products applied, load cycles and breakdowns of flight hours.

3.5. Organisational and management information

3.5.1. The company had operated Robinson helicopters since 2006 and had owned the incident helicopter for three years prior to the incident. The helicopter had been operated for between 600 and 700 flight hours a year, with most of that time on spraying operations. The helicopter had also been used for training, stock mustering and solid fertiliser application. The pilot and the chief pilot had flown the helicopter on a regular basis.

3.5.2. The two pilots stated that for agricultural operations the helicopter was usually configured for the maximum all-up weight at take-off, and carried sufficient fuel for 40 minutes of flight. The maximum payload[20] that could be carried, depending on the weight of the pilot and the specific gravity[21] of the product, was about 250 litres. The volume of liquid (water mixed with the product) put into the spray tanks was controlled through a flowmeter on the filler hose. At each turnaround the loader would check that the tanks were empty before refilling them with the maximum payload permissible, based on the operating weight of the helicopter[22] and the specific gravity of the liquid.

3.5.3. Both pilots described different methods of turning the helicopter by 180 degrees at the end of each spray run, depending on the weight of the helicopter and the helicopter speed during the spray run. The weight of the helicopter progressively reduced as the agricultural product was dispersed on each spray run. The speed of the helicopter could vary depending on the concentration of the product being applied.

3.5.4. Wider turns were made when the helicopter was heavy and/or it was flying at high speeds. When the helicopter was lighter the pilots referred to sometimes using a technique to reverse the direction of the helicopter more quickly in preparation for the next spray run. They referred to these as 'advanced turns', where the helicopter was pulled up until reaching near to or zero

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[18] A cyclic procedure turn is where the helicopter is turned around 180 degrees in a relatively wide, flat arc using the cyclic flight control only. These turns impose minimal added stresses on the dynamic components of the helicopter.

[19] Quarterly operational statistics are required to be submitted by CAR Part 137 Agricultural Aircraft Operations operators, and the data includes the amount of products applied and the area covered. It is not normally checked for accuracy.

[20] The difference between the operating weight and the maximum all-up weight.

[21] The relative density of a substance, usually compared to water. For example, a substance with a specific gravity of 1 has the same density as water, and 1.5 would be one and a half times the density of water.

[22] The empty weight of the helicopter with the spray system installed, plus the weight of the pilot and the fuel on board.

airspeed, then the torque from the main rotors and/or tail rotor thrust was used to pivot the helicopter around the main rotor mast until it was aligned with the next spray run.

3.5.5. The operator estimated that on average the helicopter accrued 10-12 take-offs[23] per hour, and sometimes up to 25 per hour, during spraying operations. The pilots stated that they usually used 'maximum allowable' engine power during take-offs and between 23 inches of mercury (Hg)[24] and 24″ Hg for the spray runs.

3.6. Additional information

Safety information

3.6.1. The manufacturer and national airworthiness authorities had issued a number of safety notices, gazette articles and airworthiness bulletins to highlight the dangers of overloading and overstressing helicopter dynamic components, particularly during agricultural flying.

3.6.2. In December 2001 Robinson had issued Safety Notice SN-37 – Exceeding Approved Limitations Can Be Fatal (see Appendix 5). The safety notice explained, in general terms, fatigue failures in helicopter components and the likely causes. A copy of this safety notice is included in the pilot's operating handbook carried in each helicopter.

3.6.3. The March/April 2005 issue of Vector magazine[25] included an article entitled 'Ag Work and the R22'. The article discussed the implications of using the Robinson R22 for agricultural operations in respect of loading limits and main rotor blade fatigue. The similarities between the R22 and the larger R44, including the main rotor designs, meant that this article was relevant to operators of R44 helicopters in the agricultural role.

3.6.4. In May 2006 the Civil Aviation Safety Authority of Australia had issued Airworthiness Bulletin AWB 02-015 Helicopter – Effects of Fatigue on Life Limited Components. The purpose of the bulletin was to "advise operators and maintenance organisations that the fatigue life of life-limited components may be adversely affected, or the safety margin reduced, depending on the operation and type of loading history that the components experience during its service life" (see Appendix 6).

3.6.5. One of the recommendations in the bulletin was that operators of aircraft engaged in agricultural flying, cattle mustering, or operations with more than four rotor full stops per flight hour consult the manufacturers and give full details of all operations for possible component life-limit revisions.

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[23] Normally a flight cycle includes a shut-down of the engine, but during spraying the engine is left running between loads.

[24] The pressure in the engine intake manifold, measured in inches of mercury (Hg), is a measure of the power setting of the engine. The pressure is regulated by the throttle setting. The absolute maximum take-off power is marked on the manifold pressure gauge by a single red, radial line.

[25] *Vector* magazine is published by the CAA every two months and contains safety notices, articles and news for the aviation industry.

4. Analysis

4.1. Introduction

4.1.1. Main rotor blade failures are rare, but can be catastrophic. In this case the failure occurred at low level and near a suitable landing site, which enabled the pilot to land safely.

4.1.2. Causes of rotor blade failure have included improper maintenance or material or manufacturing defects, although overloading and high stress and vibrations have also led to fatigue cracking. This failure was a type that had not occurred before and the failed blade had logged only 847 hours in service, well below the 2,200-hour service life.

4.1.3. This analysis describes the metallurgical tests performed on the failed blade, and examines the nature of agricultural flying. The cause of the failure was not determined conclusively, but the incident underlined the potential for intensive agricultural operations to contribute to component fatigue.

4.1.4. The fitting of a spray system to the R44 was a major modification that changed the usage of the helicopter and subjected it to flight stresses that had not been envisaged by the helicopter manufacturer. The analysis discusses this safety issue. Aircraft design organisations did not have to consider whether proposed major modifications to an aircraft would significantly alter the use of the aircraft or adversely affect the service life of any component.

4.1.5. The analysis also discusses operators and pilots not always knowing and adhering to the published limitations for their aircraft, an issue that has been raised in a previous Commission report.

4.2. Metallurgical inspections and examinations

4.2.1. Three independent metallurgical inspections and examinations were undertaken for this inquiry:

- The cracked main rotor blade and its paired blade were inspected at the Robinson factory with representatives of the NTSB and FAA present. Both blades were later examined in the NTSB materials laboratory. This is discussed further in paragraph 4.2.2.
- The section of cracked blade and the matching portion of its paired blade were examined by an expert metallurgist at Quest Integrity NZL Limited (Quest).[26] This is discussed further in paragraph 4.2.6.
- New Zealand Defence Force experts used X-ray and other non-destructive inspection methods to look for evidence of fatigue on a set of time-expired main rotor blades taken from another R44 that had been used almost exclusively on agricultural operations. Within the limits of the inspection techniques employed, no defects were identified.

NTSB examination

4.2.2. The NTSB laboratory examination report is included in Appendix 1. The examination determined that the cracks contained features consistent with fatigue failure, and that their origin was likely to have been at the trailing edge where the upper and lower skins were bonded (see Figure 8). The portion of the fracture surface consistent with fatigue cracking was 3.65″ (9.3 centimetres) in length and exhibited features typical of progressive crack growth. The remainder of the fracture exhibited mixed-mode[27] cracking, then a pure overstress failure (see Figure 9).

[26] The Quest examination was undertaken three years after that by the NTSB.
[27] A combination of overstress and fatigue cracking.

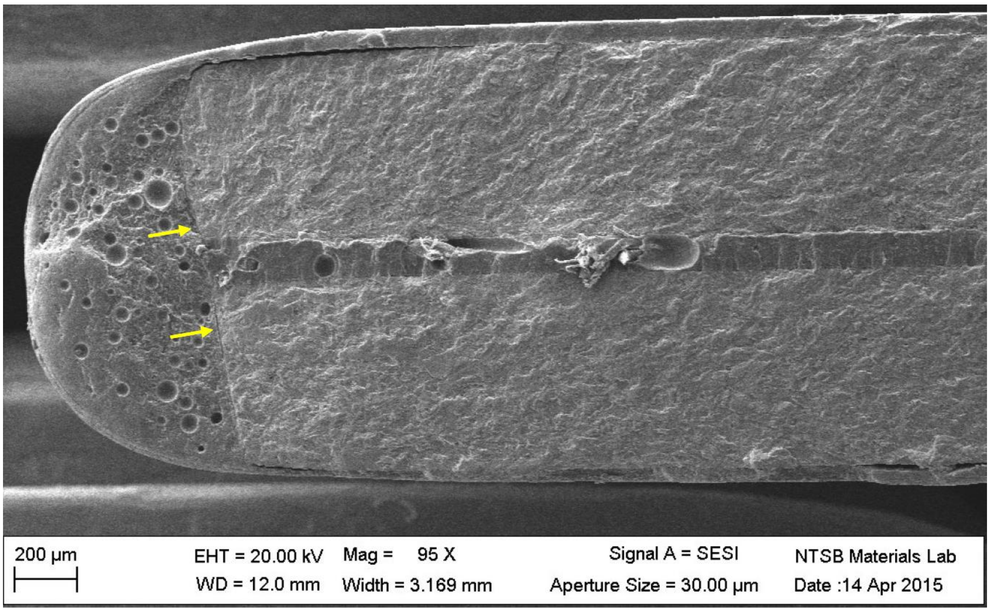

Figure 7
Crack origin

Source: NTSB Materials Laboratory Factual report 15-049

4.2.3. The NTSB examination report concluded that there were no manufacturing defects in the materials or in the construction of the blade, with all components of the blade meeting the design specifications. The materials experts from the NTSB also stated that the crack may have started in the epoxy sealant that coated the trailing edge, and that it could have spread into the aluminium skin from there. The epoxy sealant was not intended to be a structural or load-bearing component of the blade and was designed to flex with it. However, if the blade flexed significantly in service under cyclic loads, it is possible that the epoxy cracked first, creating a further stress concentration at that location.

4.2.4. The NTSB also visually inspected the paired blade in the vicinity of the chord transition and found "no indication of cracks, dents, or other damage. No anomalies were seen in the underlying sealant or aluminium skins".

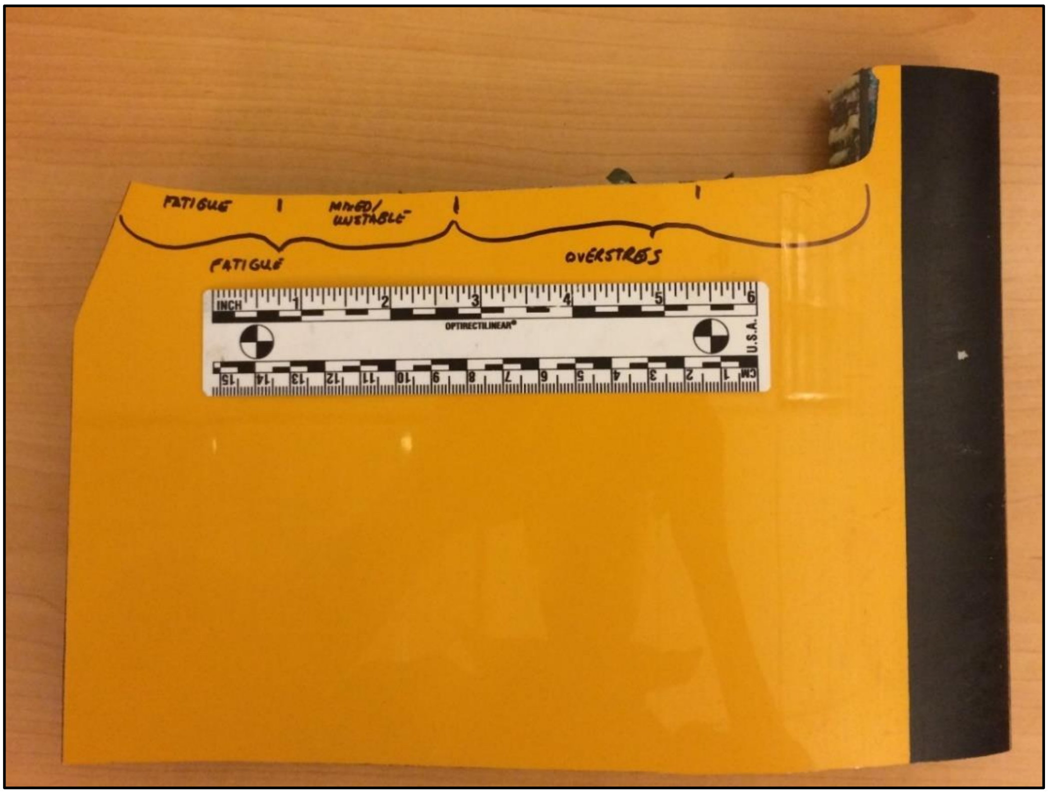

Figure 8
Upper surface of blade with annotations of crack failure mode regions

Source: NTSB Materials Laboratory Factual report 15-049

Independent New Zealand examination

4.2.5. The Quest examination report is included in Appendix 2. The fracture section (which had been prepared for the earlier NTSB examination) was examined with a scanning electron microscope. The report commented:

> The location of the origin, at the inner corner at the trailing edge of the aluminium layers does strongly suggest that part of the loading is due to bending or flexing around the minor axis of the blade section. It is understood that the nominal loading at this point is a combination of centrifugal loading from rotation, and bending/flexing around the major axis of the blade cross section due to aerodynamic loading. The additional bending load required to initiate a crack at the apparent location may be due to a particular vibrational mode in the blade, excited by other loads applied to the blade.
>
> The absence of any visible defects at the crack origins strongly suggests that the fatigue crack was caused by an applied load in excess of the capacity of the blade to meet the designed/designated fatigue life. The applied load would be a combination of varying centrifugal, aerodynamic and possibly vibrational stresses, resulting in the initiation and propagation of a fatigue crack at a critical location on the blade trailing edge (the stress concentration formed by the trailing edge transition).

4.2.6. Quest also inspected the paired blade from the helicopter for any sign of cracking or overstress in the area of the chord transition. As sectioning of the area was not feasible, Quest could only remove all coatings from the chord transition area and remove shallow surface defects before making a visual examination. Quest reported that "no defects or discontinuities were seen in the aluminium layers at the transition radius where the crack in the other blade had initiated." Quest further commented:

> Although both blades would have seen similar aerodynamic and mechanical loads in service, the absence of a fatigue crack in one blade does suggest a quantifiable difference between the two blades. This may be [due to] one or more of the following:
>
> - marginally stronger material, adhesive or fabrication resulted in a greater margin against fatigue crack initiation

- slight variations in fabrication and/or installation in the blade assembly resulted in a marginally different vibration response from the same loading, compared to the fractured blade, resulting in a slightly lower cyclic load, and hence no crack initiation or propagation at the time of removal from service.

Findings

1. There were no manufacturing defects in the materials or in the construction of the blade, and the examined components of the blade met the design specifications.
2. The point of origin of the crack in the main rotor blade trailing edge had features consistent with metal fatigue.

4.3. Operational factors considered

Overloading

4.3.1. Rotor blade failures that are caused by flying overweight, or excessive main rotor revolutions per minute (RPM),[28] typically involve cracks located closer to the blade root[29] where bending and centrifugal forces are at their highest. However, this failure occurred at a different location, closer to the tip of the blade where the chord length transitions and twisting forces are greater.

4.3.2. A comparison of the operational statistical returns[30] for the operator with those of other operators that used R44s for spraying did not indicate that the maximum all-up weight had been exceeded in service. The average load for the incident helicopter while on agricultural operations was consistent with the industry average in New Zealand.

Power settings

4.3.3. The chief pilot stated that their pilots used the maximum allowable power for take-off. The maximum allowable take-off power was determined by adding a fixed increment of 2.8″ Hg to the maximum continuous power permitted for the existing conditions of pressure altitude and outside air temperature. The pilot’s operating handbook contained a table, which was also displayed in the cockpit, that showed the maximum continuous power for different conditions (see Figure 10).

[28] A high rotor RPM increases the centrifugal force on the blade, which is also a function of the weight of the blade, so the stress will be higher towards the blade root where it is attached to the hub.
[29] Where the main rotor blade is attached to the main rotor head at the top of the mast.
[30] Agricultural operators are required to submit three-monthly returns to the CAA that contain the number of hours flown, the number of loads carried, the amount of product applied in tonnes or litres and the area covered in hectares.

LIMIT MANIFOLD PRESSURE - IN. HG								
MAXIMUM CONTINUOUS POWER								
PRESS	OAT - °C							
ALT-FT	-30	-20	-10	0	10	20	30	40
SL	21.5	21.8	22.1	22.4	22.6	22.9	23.1	23.3
2000	20.9	21.2	21.5	21.8	22.1	22.3	22.5	22.8
4000	20.4	20.7	21.0	21.3	21.5	21.8	22.0	22.2
6000	19.9	20.2	20.5	20.8	21.0	21.3	21.5	21.7
8000	19.5	19.8	20.1	20.3	20.6	20.8	21.0	21.3
10000	19.1	19.4	19.6	19.9	FULL THROTTLE			
12000								
FOR MAX TAKEOFF POWER (5 MIN), ADD 2.8 IN.								

Figure 9
Engine power limits

Source: Robinson Helicopter Company

4.3.4. The absolute maximum ('red line') power of 26.1″ Hg applied only when the outside air temperature was 40 degrees Celsius (°C) (at sea level (from Figure 10: 23.3″ Hg plus 2.8″ Hg). The operator was based in the lower South Island where temperatures were relatively low, and would almost certainly never have flown under those conditions. They would have been more likely to fly in temperatures between 0°C and 20°C at altitudes between sea level and 2,000 feet. Under those conditions, the maximum take-off power would have been between 24.6″ Hg and 25.7″ Hg. At interview, neither pilot could state the red line value. The pilots' statements about using maximum allowable power without having a correct awareness of the red line value or allowing for the ambient conditions suggested that they would likely have inadvertently exceeded at times the pilot's operating handbook limit for maximum take-off power.

4.3.5. The chief pilot said they operated the aircraft up to the limit, but not all the time, and not over the limit. The chief pilot said during the first interview that the power required during take-off was 23-24″ Hg, and up to 26″ Hg on occasions. The power was then reduced to about 20″ Hg for the spray runs. The maximum continuous power setting was 23.3″ Hg. Therefore it is very likely that the engine was inadvertently run at high power settings at times while the helicopter was spraying.

4.3.6. Although Robinson had set limits on the engine manifold pressure and the engine had been 'derated'[31] from its maximum power output, it was possible for pilots to demand more engine power than the helicopter's dynamic components were designed to absorb. It was therefore possible to overstress the helicopter by exceeding the power limit for the ambient conditions.

4.3.7. The integrity of components will be compromised if they are subjected to excessive stress. A single instance of over-stress may be sufficient to cause damage. The accumulation of fatigue can be rapid and reach the point of failure before early detection is possible. Robinson Safety Notice SN-37 (see Appendix 5), which is included in the pilot's operating handbook of every Robinson helicopter, highlighted the detrimental effects of excessive loading and engine power settings on the service life of dynamic components.

Turns

4.3.8. The in-flight strain survey carried out by Robinson included manoeuvres commonly used during agricultural flying. Turns typically used between spray runs, with cyclic and collective pull-ups of low to moderate severity, were performed during the flight testing. A direct comparison with some of the turn techniques used by the operator could not be achieved, because Robinson did not exactly replicate the same turn technique during its testing and the test aircraft did not have spray equipment fitted. From the in-flight strain survey, Robinson found that:

[31] The Lycoming IO-540 was originally used in fixed-wing aircraft and produced 300 horsepower at take-off, whereas in the R44 the power was limited to about 245 horsepower.

> The stresses were just below the level where they would contribute to fatigue damage. It is likely therefore that if the helicopter exceeded the maximum gross weight and/or the turns were performed more aggressively the loads would be damaging.

4.3.9. According to industry feedback, some of the turn techniques used by the operator were not widely used during spraying operations, and were not taught during basic agricultural flight training. A characteristic of these types of turn is that the helicopter reverses the direction in which it is heading while it has zero airspeed. If the turn is initiated earlier while the helicopter is still climbing, the heading of the helicopter during the latter part of the climb is not aligned with the flight path.

4.3.10. During this type of turn the helicopter changes from flying forwards to flying sideways, and possibly to flying backwards if the turn is initiated early enough. The helicopter must then translate to flying forward again. Consequently the turn can take place for the most part in 'transitional flight'.[32] According to a technical report that studied helicopter loads, fatigue and design, published by the Australian Department of Defence's Defence Science and Technology Organisation's Aeronautical Research Laboratory in 1993:

> 'Transitional flight' generates the highest vibratory stresses. When comparing to a 'transport mission', 'crop spraying' gives approximately the same rotor blade fatigue life, unless prolonged 'transitional flight' is included in the spectrum, in which case the fatigue life becomes less than half the life of the transport mission. The main rotor blades respond to transient loading by flapping, leading, lagging and twisting.

4.3.11. For a considerable period during such a turn there is reduced translational lift[33] as the helicopter airspeed reduces and the direction of the flight path is reversed. This type of turn differs from a conventional 'teardrop turn' where airspeed and therefore translational lift are conserved.

4.3.12. In regard to the consequences of this type of manoeuvring, Robinson stated:

> Since the relationship between loads and fatigue is exponential it is possible that operation consistently over maximum gross weight or excessively aggressive turns could cause an order of magnitude reduction[34] in fatigue life.

4.3.13. Robinson advised that the dominant variable in main rotor blade fatigue damage is blade angle of attack, which results in large variations in chord-wise bending loads. At the maximum all-up weight, pulling out of an aggressive turn requires significant increases in blade pitch angles (angles of attack) and engine power, along with an increase in gravity loading. The effect of increased aerodynamic forces and the loading of the blades during aggressive agricultural manoeuvres is the likely mechanism for the metal fatigue accumulation resulting in a significant fatigue penalty.

Exceeding aircraft limits

4.3.14. The operator's pilots had inadvertently used engine power settings that were likely, at times, to have been above the maximum allowable for the ambient conditions.

4.3.15. The Commission noted in a previous inquiry report[35] that New Zealand's helicopter accident rate was higher than that of other aviation sectors, and that there had been public criticism of how helicopters were operated in New Zealand, including a culture of operating outside the manufacturers' published and placarded 'never exceed' limitations. It noted that there was a possibility that such a culture had become normalised. The core safety issue would therefore lie within the wider helicopter sector, with flow-on effects to individual operators' safety systems.

4.3.16. In the previous inquiry report, the Commission did not make a recommendation because it was aware that the CAA was reviewing the 'sector risk profile' of commercial helicopter and small

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[32] The transient state as the helicopter transitions from one established flight phase to another, such as from a hover to forward flight or vice versa.

[33] The additional lift generated by the rotor system once the helicopter's airspeed increases beyond a certain point (normally about 15 knots).

[34] A reduction by a factor of 10.

[35] Report AO-2014-005, Eurocopter AS350-B2, ZK-HYO, collision with terrain, Mount Alta, 16 August 2014.

aeroplane operations, and had already recommended that the Director of Civil Aviation include the safety issue of helicopter operational culture in that review.

Findings

3. The helicopter had been flown mostly on spraying operations at high power settings, which were likely to have exceeded the allowable limit at times. The high power settings subjected the helicopter dynamic components to higher stress than they were designed for.
4. The types of turn used during spraying, particularly when the helicopter was at heavy weights, likely subjected the main rotor blades to stresses above those for which the blades were designed and contributed to the onset of fatigue at the chord transition radius.

4.4. Fatigue and cycle calculations during certification

4.4.1. Aircraft components are subject to a wide variety of mechanical and environmental stresses, and are particularly susceptible to accumulated fatigue. The continued safe operation of aircraft requires manufacturers to determine the expected lives of critical components. With that knowledge, appropriate inspection and replacement schedules can be implemented to ensure that components are replaced well before they are likely to fail in service.

4.4.2. Robinson specified that one engine and rotor 'stop-start'[36] was equivalent to one 'fatigue cycle' on the blade. This definition was used in its calculations to forecast component reliability, determine fatigue life and publish service life limits. Robinson advised that during the assessment of fatigue during main rotor blade design and certification, a rate of six stop-start cycles per flight hour was assumed. That cycle rate was combined with the in-flight fatigue spectrum[37] and the stress-cycle curve[38] for the blade to produce the blade's fatigue life[39] of 4,000 hours (24,000 cycles). After applying a safety margin, 2,200 hours (13,200 cycles) was specified for the main rotor blade service life.[40]

4.4.3. Figures provided by the operator showed that the helicopter flew an average of 12.5 product loads per hour during spraying, and about 15 turns between spray runs were conducted during the application of each load. Robinson submitted that the fatigue damage depended on the magnitude of the stress cycle, and that the damage associated with an aggressive turn was significantly less than that associated with a ground-air-ground cycle. However, stress cycles are accumulated more quickly during agricultural flying than they are during a 'standard' flight profile.

4.4.4. The type of flight operation performed affects the fatigue lives of helicopter components. A Civil Aviation Safety Authority of Australia bulletin[41] (see Appendix 6) described how component fatigue life may be affected by:

- operation of helicopters in agricultural operations where high loads may be encountered more frequently than envisaged by the manufacturer
- operations where there are a high number of landings and take-offs per hour
- frequent operations at or near maximum all up weight.

[36] One cycle is taken from the rotors starting to rotate until they stop rotating after engine shut-down.
[37] The fatigue spectrum is developed from in-flight stresses recorded during a 'standard' flight profile, which involves the normal manoeuvres expected in air transport, private and training flights, but not in agricultural flying.
[38] The stress life of a component is derived from testing the component with a specific load for a number of cycles until it fails. Tests are carried out at increasing loads that result in fewer cycles until failure. The results are plotted on a chart to produce an 'S-N' curve. This determines the fatigue limit, the point (applied load) at which any number (or for certain materials a very high number, typically 10^7) of cycles will not result in component failure.
[39] The minimum number of stop-start cycles and flight hours that stress (fatigue spectrum loads) can be applied to a component before a failure could occur.
[40] Finite or safe life is a usage limit specified for a component, in hours or calendar time, after which time it must be discarded. It is significantly lower than the fatigue life to ensure that the component will not fail in service.
[41] airworthiness bulletin (AWB 02-015)

4.4.5. The bulletin also stated that fatigue damage is accelerated when the cyclic loads on a component are of greater magnitude and frequency than envisaged during the design of the component. The standard flight profiles used by Robinson to determine the fatigue spectrum for the R44 blades did not include the types of manoeuvre or number of cycles associated with agricultural flying. Therefore Robinson's fatigue-life calculations and safe-life limits for components used in standard flight operations were unlikely to be valid for helicopters routinely used for agricultural operations.

4.4.6. After the incident Robinson recalculated the fatigue life of R44 main rotor blades used in agricultural operations, using an increased stop-start cycle rate of 10 per hour. The fatigue life reduced from 4,000 hours to 3,400 hours. However, Robinson determined that the safe life of 2,200 hours would not need to be reduced because an adequate safety margin still remained.

4.4.7. In contrast, the engine manufacturer had previously recognised that agricultural flying results in greater stresses in components, and had reduced the time between overhaul for engines used in aircraft performing that role. The standard engine fitted to an R44 normally has an overhaul period of 2,000 hours. However, if the helicopter is used for agricultural flying this is reduced to 1,500 hours. This limitation was introduced by the engine manufacturer because of the increased stresses during high power settings at heavier weights, and the associated reduction in the expected service lives of the engine's components.

4.4.8. The Quest examination of the fractured blade found evidence of high cycle fatigue. Small differences in material properties result in a variation in fatigue life between samples/objects that experience the same loading conditions. That is likely the reason for one blade failing before there was any indication of fatigue damage in the other blade.

Finding

5. The helicopter was not designed specifically for agricultural flying. The manufacturer had therefore not been required to consider the increased loads and cycles of agricultural flying when calculating the service life of the rotor blades.

4.5. Design factors

4.5.1. During the course of the inquiry, it was suggested to the Commission that the main rotor blade crack was the result of a failure in main rotor blade design. It was submitted that the presence of a design flaw was supported by Robinson subsequently modifying the main rotor blade to reduce the stress concentration around the chord transition, where this failure occurred.

4.5.2. This incident is the only recorded case of fatigue cracking in the chord length transition radius of a C016-7 main rotor blade. The fracture of the blade at the chord transition indicated that in-service stresses at the chord transition were higher than Robinson had anticipated. The critical stress location on the main rotor blade had shifted from being near the blade root, where it had been on earlier versions of the blade. However, Robinson advised that under design operating conditions the shift would not have been an issue, because even allowing for the additional stress of intensive agricultural operations, Robinson had calculated that there was still an adequate margin above the 2,200 hours' stated service life.

4.5.3. The 'Revision AG' blade incorporated a move of the chord length transition inboard of the stainless steel doubler termination. Robinson advised that this change was to provide a strength margin in case of corrosion or damage at or near the transition radius (see Figure 6).

4.5.4. The design change Robinson made as a result has improved the blade's safety margin for fatigue resistance. FAA AD 2016-26-04 and CAA AD DCA/R44/32C have mandated the field service modification specified in Safety Bulletin SB-89, which is an interim measure until the blades are replaced. Although the full benefits of the design improvements can only be achieved by replacement with 'Revision AG' blades, Robinson advised that SB-89 restores the calculated fatigue life for early revision blades to that of the new 'Revision AG' blade.

4.5.5. Therefore, the Commission does not consider that the fatigue cracking was the result of a design 'flaw'. If it had been, there would have been more recorded cases of fatigue cracking in the many other blades in service, many of which had significantly more accumulated hours in service than those fitted to the incident helicopter.

4.5.6. It is considered that the higher-than-normal stresses imparted on the blades during agricultural operations are likely to have culminated in a fatigue crack developing in the area of the blade most susceptible to those types of stressor. The modification of the blade design by Robinson was to increase the design margin in case trailing-edge damage occurred to the blade, or the particular helicopter operations exceeded the number of cycles and the nature of stresses envisaged when designing the life cycles of components.

Finding

6. 'Revision AE' and earlier C016-7 main rotor blades had a small radius in the trailing edge chord length transition. This created a higher stress concentration in this area and reduced the safety margin before fatigue cracking would occur.

4.6. Major aircraft modifications and role changes

Safety issue: Aircraft design organisations did not have to consider whether proposed major modifications to an aircraft would significantly alter the use of the aircraft or adversely affect the service life of any component.

4.6.1. The installation of spray equipment on the helicopter was a major modification that had been approved by a design organisation certificated under CAR Part 146 Aircraft Design Organisations – Certification. An aircraft design organisation holds a delegation from the CAA to approve design changes and aircraft modifications. CAR Part 21 Certification of Products and Parts requires all designs and design changes to be approved by the Director of Civil Aviation, but the Civil Aviation Act 1990 allows this to be delegated to other acceptable persons. CAR Part 146 provides details of the requirements for acceptable persons and the certification and operating procedures for design organisations.

4.6.2. Under CAR Part 146, an aircraft design organisation shall ensure that its designs and design changes have no "unsafe features", and shall establish procedures to:

- test and inspect specimens to [of] the type design to show compliance with airworthiness design standards: and
- produce, check, and control reports showing compliance with airworthiness design standards, including stress analysis and flight test reports.

4.6.3. Although the requirements for stress analysis and flight testing applied to the components of the modification itself, there was no requirement under CAR Part 146 for a stress analysis or fatigue assessment of the aircraft with the modification installed. Such an assessment would have helped to determine whether the installation of spray equipment on a helicopter, and the subsequent use of the helicopter for spraying, affected the fatigue life of the components of a helicopter that had not been originally certified with agricultural flying in mind.

4.6.4. An aircraft with a modification installed would normally be flight tested to determine whether there is any change in the aircraft's flight characteristics or performance, but in-flight stresses on aircraft components are not normally measured. For a full stress analysis of an aircraft with an installed spray modification, the aircraft would have to be fitted with strain gauges[42] and flown as it would be in service.

4.6.5. Robinson had not designed the R44 for agricultural flying and therefore the helicopter and its components had not been tested or certified with that use in mind. Had the increased stresses and cycles involved with agricultural flying been allowed for during certification, a shorter service

[42] A device used to measure the strain (or stress) on an object. It consists of a conductive metallic strip that changes its electrical resistance when stretched.

life may have been applied to critical dynamic components such as the main rotor blades, if used in that role.

4.6.6. Had the aircraft design organisation assessed the loads and cycles to which the modified helicopter would be subjected in the agricultural role, and been able to compare these with the original data from the manufacturer, the increased stresses would likely have been identified. However, Robinson submitted that manufacturers would be reluctant to release proprietary data like that to other parties, primarily for legal considerations. In any event, in this case Robinson calculated that the increased stresses did not warrant a shortening of the main rotor blade life.

4.6.7. Robinson had not been consulted during the development of any spray equipment for the R44, and had had no input to the design process or testing and certification stages of such equipment. There was no requirement for a design organisation to consult the aircraft manufacturer when developing a major modification for use in a specialist or unique role for which the aircraft was not originally designed.

4.6.8. National aviation authorities, like the CAA, that issue type certificates or type acceptance certificates appear to be best placed to obtain the co-operation of aircraft manufacturers in assessing the effects of a proposed modification on continuing airworthiness. Therefore the Commission recommended to the Director of Civil Aviation that they consult the original equipment manufacturer when considering a modification or supplemental type certificate that, if approved, could result in an aircraft being used in a way that was significantly different from that which the manufacturer originally modelled and used as the basis for determining component fatigue life and the aircraft maintenance programme.

Finding

7. The design organisation's stress analysis during the spray system's design and testing did not take into account, and was not required to take into account, the increased loads and cycles imposed on the dynamic components of a helicopter used for agricultural flying.

5. Findings

5.1. There were no manufacturing defects in the materials or in the construction of the blade, and the examined components of the blade met the design specifications.

5.2. The point of origin of the crack in the main rotor blade trailing edge had features consistent with metal fatigue.

5.3. The helicopter had been flown mostly on spraying operations at high power settings, which were likely to have exceeded the allowable take-off limit at times. The high power settings subjected the helicopter dynamic components to higher stress than they were designed for.

5.4. The types of turn used during spraying, particularly when the helicopter was at heavy weights, likely subjected the main rotor blades to stresses above those for which the blades were designed and contributed to the onset of fatigue at the chord transition radius.

5.5. The helicopter was not designed specifically for agricultural flying. The manufacturer had therefore not been required to consider the increased loads and cycles of agricultural flying when calculating the service life of the rotor blades.

5.6. 'Revision AE' and earlier C016-7 main rotor blades had a small radius in the trailing edge chord length transition. This created a higher stress concentration in this area and reduced the safety margin before fatigue cracking would occur.

5.7. The design organisation's stress analysis during the spray system's design and testing did not take into account, and was not required to take into account, the increased loads and cycles imposed on the dynamic components of a helicopter used for agricultural flying.

6. Safety issue

6.1. Aircraft design organisations did not have to consider whether proposed major modifications to an aircraft would significantly alter the use of the aircraft or adversely affect the service life of any component.

7. Safety actions

General

7.1. The Commission classifies safety actions by two types:

(a) safety actions taken by the regulator or an operator to address safety issues identified by the Commission during an inquiry that would otherwise result in the Commission issuing a recommendation

(b) safety actions taken by the regulator or an operator to address other safety issues that would not normally result in the Commission issuing a recommendation.

Safety actions addressing safety issues identified during an inquiry

Main rotor blade design

7.2. In response to this incident, Robinson issued SB-89 to improve safety margins in earlier version C016-7 blades. This introduced a field modification that reshaped the trailing edge profile and increased the inboard chord transition radius.

7.3. The actions specified in SB-89 had been mandated in New Zealand by the CAA through the issue of AD DCA/R44/32C. The CAA also sent a letter to all R44 helicopters operators advising them of the AD and requesting reports of any incidents or defects.

7.4. Robinson implemented design changes to the trailing edge profile of 'Revision AG' C016-7 blades, to improve safety margins by reducing stress concentrations and increasing fatigue tolerance. In addition to the increased inboard radius, the chord transition was moved approximately six inches (5.2 centimetres) inboard, to take advantage of the added strength of the trailing edge doubler. Both of these changes are included in the new 'Revision AG' blades.

7.5. The FAA issued AD 2016-26-04, which required inspections of the chord transition on C016-7 blades, and either the removal from service or an alteration of the inboard chord transition radius. This mandated the field modification provided by Robinson SB-89 for older revision blades still in service in the United States.

Safety actions addressing other safety issues

7.6. None identified

8. Recommendations

General

8.1. The Commission may issue, or give notice of, recommendations to any person or organisation that it considers the most appropriate to address the identified safety issues, depending on whether these safety issues are applicable to a single operator only or to the wider transport sector. In this case, a recommendation has been issued to the Director of Civil Aviation.

8.2. In the interests of transport safety, it is important that this recommendation is implemented without delay to help prevent similar accidents or incidents occurring in the future.

Previous recommendations

8.3. New Zealand's helicopter accident rate is higher than that of other aviation sectors. There has been public criticism of how helicopters are operated in New Zealand, including a culture of operating outside the manufacturers' published and placarded 'never exceed' limitations. Should this situation exist, there is a possibility that such a culture has become normalised. The core safety issue would therefore lie within the wider helicopter sector, with flow-on effects to individual operators' safety systems.

8.4. The Commission is aware that the CAA is currently reviewing the 'sector risk profile' of commercial helicopter and small aeroplane operations, and that that work will take a structured approach to risk identification and mitigation.

8.5. On 25 October 2017 the Commission recommended that the Director of Civil Aviation include the safety issue of helicopter operational culture in its current 'sector risk profile' review.[43]

On 13 November 2017 the Director replied, in part:

> The Part 135 sector risk profile (SRP) published in 2015 identified culture as a risk. Over the next two weeks workshops will confirm the 2015 risks and allocate treatment owners. The CAA will monitor the implementation of the treatments, however it must be stressed that it will take some years to convert in the aviation sector.

8.6. In June 2018 the CAA published a revised sector risk profile for Part 135 Air Operations – Helicopters and Small Aeroplanes. The revised profile did not address specifically the issue of pilots not adhering to flight manual limitations, but its key risk themes included:

- training and pilot experience
- organisational environment and culture
- sector safety culture and collaboration.

New recommendation

8.7. In New Zealand there is a large agricultural flying industry, with the R44 being the most commonly used light helicopter for spraying work. Over half of the R44 fleet is utilised for agricultural flying and some of it is also used for carrying passengers commercially.

8.8. The fitting of spray equipment to a helicopter for agricultural operations is one example of a modification that can result in an aircraft being used in a way that was not contemplated by the aircraft manufacturer. Another example is the conversion of an aeroplane for parachuting operations when that role was not envisaged. The different uses can result in flight profiles and engine handling that could adversely affect the fatigue lives of aircraft components. Without recognition of and due allowance for such adverse effects, the continuing airworthiness of the aircraft might be compromised. This is a potential safety issue, particularly for modified aircraft that also carry passengers in air operations.

Although modifications may be designed by approved aircraft design organisations, it is very unlikely that aircraft manufacturers will release the original fatigue data to independent organisations to enable them to assess the effects of proposed modifications on the fatigue lives of aircraft and their components. The reluctance to share data is based primarily on legal

[43] Commission recommendation 032/17.

considerations. However, aircraft manufacturers are likely to agree to requests from national aviation authorities, such as the CAA, for relevant data.

8.9. **On 14 December 2017 the Commission recommended to the Director of Civil Aviation that they consult the original equipment manufacturer when considering a modification or supplemental type certificate that, if approved, could result in any aircraft being used in a way that is significantly different from that which the manufacturer originally modelled and used as the basis for determining component fatigue life and the aircraft maintenance programme. (036/17)**

8.10. On 10 January 2018, the CAA replied:

> In considering the draft recommendation, the CAA refers the Commission to the Director's response on 10 October 2017 where the CAA considers that original equipment manufacturers are unlikely to release proprietary and commercially sensitive information on the majority of occasions when they are asked to do so.
>
> However, the Director is prepared to accept the recommendation on the basis that the CAA will seek a manufacturer's advice on the utilisation of an aircraft if it is considered that a modification or supplemental type certificate may place the aircraft's operations outside of those originally intended.

8.11. The CAA Aircraft Certification Unit amended its procedures for the issue of supplemental type certificates and its Major Design Change Authorisation Process Sheet by including a requirement to consider whether modifications could result in changes in operational use. The document amendments were completed on 16 March 2018. The Commission closed recommendation 036/17 on 23 April 2018.

9. Key lessons

9.1. Metal fatigue occurs continuously in dynamic components. A fatigue crack can lengthen rapidly and the component lose its structural strength. If an unusual or severe vibration develops in flight, the pilot should land immediately and have the helicopter inspected before further flight.

9.2. The key to minimising stress in dynamic components is to fly conservatively, especially when operating close to the published weight, speed and power limits. The Commission has noted in previous inquiry reports that operating an aircraft outside the published limitations significantly erodes the safety margins factored into the service lives of components and can quickly lead to an early catastrophic failure.

9.3. Operators and maintainers of aircraft that are subjected to cycles or flight profiles that are significantly different from those envisaged by the manufacturers when the aircraft were certificated should consider implementing shorter intervals for component inspections and earlier component replacement times.

10. Citations

Civil Aviation Safety Authority, 2006, *Airworthiness Bulletin AWB 02-015 Helicopter Effects of Fatigue on Life Limited Components*, Sydney, Australia.

Civil Aviation Authority, 2005. Ag Work and the R22, *Vector* magazine, March/April 2005, pp 6-9.

Appendix 1: NTSB Materials Laboratory report

NATIONAL TRANSPORTATION SAFETY BOARD
Office of Research and Engineering
Materials Laboratory Division
Washington, D.C. 20594

May 28, 2015

MATERIALS LABORATORY FACTUAL REPORT — Report No. 15-049

1. ACCIDENT

Place	: Waikaia, New Zealand
Date	: January 23, 2015
Vehicle	: Robinson R44-II, ZK-HPC
NTSB No.	: ENG15RA012
Investigator	: Chihoon Shin, AS-40

2. COMPONENTS EXAMINED

Sections of main rotor blades

3. DETAILS OF THE EXAMINATION

On January 23, 2015, a Robinson R44 II (S/N 10525) experienced a severe in-flight vibration, after which the pilot successfully landed the helicopter. A subsequent inspection found a crack on one of the main rotor blades. This incident was being investigated by the New Zealand Transport Accident Investigation Commission (TAIC), and the NTSB participated as the United States Accredited Representative to the New Zealand TAIC as the State of Design and State of Manufacture of the helicopter. The Civil Aviation Authority of New Zealand issued an airworthiness notice concerning these main rotor blades after the incident.[1]

The cracked main rotor blade was sectioned around the crack and shipped to the NTSB Materials Laboratory. In addition, a section of an exemplar blade along with exemplar samples of the rotor blade constituents were sent to the Materials Laboratory for comparative examination. The cracked blade, P/N C016-7, had undergone 832.5 hours in service installed on the helicopter.

3.1. Cracked Rotor Blade

Figure 1 and Figure 2 illustrate the sectioned rotor blade as received. The blade segment had been marked "S/N 2031" with red marker on the upper face (painted yellow, Figure 1). The lower face of the blade was painted black and exhibited transverse witness marks, nicks, and scratches consistent with impact from ground debris during blade rotation.

[1] Continuing Airworthiness Notice 62-003, issued January 26, 2015, was an advisory recommending inspection of main rotor blades for similar cracks fitted on all Robinson R44 series helicopters before each flight.

The segment contained a transverse crack approximately 9.2 inches long. The crack had initiated at the trailing edge of the rotor blade, progressing forward toward the leading edge. The crack had changed direction inboard and progressed approximately 1 inch. The location of the change in crack orientation was located at the forward spar in the rotor blade. Figure 4 illustrates the trailing edge at the location of the crack. The crack exhibited a relatively flat surface morphology, oriented perpendicular to the blade surfaces. The crack had penetrated the aluminum skin panels as well as the adhesive and trailing edge sealant. There were no macroscopic defects visually evident at this location.

The crack was backcut from the forward end and intentionally overstressed in the laboratory to expose the crack faces (or fracture surfaces) of the aluminum skin and honeycomb structure. The outboard face of the opened fracture surface is illustrated in Figure 5. As elaborated later in this section, the portion of the aluminum skin fracture surfaces towards the trailing edge exhibited features consistent with fatigue crack propagation. These features included crack arrest, or beach, marks that are typical of progressive crack growth. The portion of the fracture surface toward the forward side of the rotor blade exhibited a rougher surface texture and general 45° slant, consistent with failure from overstress.

It should be noted that the laboratory opening of the crack produced artifacts observed in Figure 5. The wing skins aft of honeycomb core had been bent upward. In addition, some of the honeycomb structure had been crushed and torn during the opening process. This was consistent with some of the core structure being intact before the crack was intentionally opened. The quantity of internal core structure that was intact could not be conclusively determined before or after the laboratory opening.

The crack initiation site is illustrated in Figure 6 and Figure 7. The fracture surface was sputter coated using a gold target to make it electrically conductive for examination in a scanning electron microscope (SEM). This portion of the rotor blade consisted of a composite of two aluminum sheet skins joined by a thermoset adhesive, FM 94.[2] The leftmost tip was a sealant, Scotch-Weld 2216 B/A, which was cured and later rounded mechanically.[3] In general, the sealant, adhesive, and aluminum sheets appeared to have been well adhered to each other.

The aluminum skins exhibited river patterns and crack arrest marks, consistent with progressive fracture that originated from the trailing edge. Likewise, the adhesive between the skins exhibited hackles and arrest marks consistent with crack propagation from the trailing edge. As shown in Figure 7 and Figure 8, the crack initiation sites of the aluminum skins were on the thin flat faces facing the trailing edge.

Figure 9 illustrates the crack initiation site at the trailing edge angled to show the aft surface. The rounded sealant exhibited features generally consistent with crack initiation at the surface. Microscopically, some crazing was observed, consistent with cracking features

[2] The adhesive, FM 94, is an above room temperature, modified epoxy film adhesive designed for bonding metallic and composite structures, manufactured by Cytec Engineered Materials, Tempe, AZ.
[3] The sealant, Scotch-Weld 2216 B/A, is a flexible, two-part, room-temperature curing epoxy, manufactured by 3M Corporation, St. Paul, MN.

typically observed in polymeric materials. However, much of the sealant surface had been ground away and obliterated, consistent with physical contact with the mating fracture surface.

The fatigue crack initiation site on the upper aluminum skin is illustrated in Figure 10. The site shows faceted features on the skin consistent with Stage I fatigue cracking.[4] However, no material defects, such as pores or non-metallic inclusions were observed at this area. There were no indications of gaps or poor bonding with the sealant material at this location. Figure 11 shows a closer view of the trailing edge tip in the epoxy sealant. The surface exhibited some lack of paint, but the edge of the crack sealant had been rounded. Small secondary cracks were found on the sealant fracture surface—these sub-cracks were consistent with fatigue fracture inward. Small pores were observed on the fracture surface in the epoxy sealant. However, there were no features found adjacent to these pores consistent with cracks or fracture emanating from them.

Figure 12 illustrates a representative area of the aluminum skin fracture surface near the trailing edge. This area exhibited striations consistent with fatigue failure. As stated above, and illustrated in Figure 3 and Figure 5, this region exhibited primarily fatigue striations. As one moved forward, the fracture surface exhibited sporadic dimple rupture features, as illustrated in Figure 13. This was consistent with mixed mode failure or unstable fatigue crack growth. However, fatigue striations were still the dominant feature in this portion of the aluminum skin fracture surface (see Figure 14).

Towards the end of the mixed-mode fatigue portion of the fracture surface, the fracture morphology included more discernible dimple rupture features (see Figure 15). At the end of the areas annotated fatigue and mixed mode in Figures 3 and 5, an abrupt change from fatigue striations to dimple rupture was observed on the fracture surface (see Figure 16). All areas of the fracture surface inspected forward of this point exhibited only dimple rupture, consistent with overstress failure (see Figure 17).

The portion of the fracture surface exhibiting features consistent with fatigue cracking measured approximately 3.65 inches from the trailing edge. The first 1.68 inches exhibited features consistent with primarily fatigue failure. The forward most 1.97 inches of the progressive crack exhibited mixed mode features: striations consistent with fatigue and dimple rupture consistent with overstress. Qualitatively, the proportion of the fracture surface exhibiting overstress features relative to the fatigue features increased as one moved from the trailing edge side to the forward edge side. The remaining length of the fracture surface exhibited features consistent with overstress. The 45° orientation of the fracture surface reversed twice—approximately every two inches.

The chemical compositions of the aluminum skin sheets were inspected using energy dispersive x-ray spectroscopy (EDS) and x-ray fluorescence (XRF). The chemical composition of the skins was consistent with AA 2024 aluminum alloy. The material hardness of the skins was inspected per ASTM E384, and the electrical conductivity was

[4] Stage I fatigue crack growth is the initial stage of fatigue crack nucleation and growth, typically occurring on shear planes (P.J.E. Forsyth, "A Two Stage Process of Fatigue Crack Growth," *Proceedings of the Symposium on Crack Propagation*, Cranfield, UK, 1961, pp. 76-94).

inspected per ASTM E1004.[5,6] The hardness of the upper and lower skins averaged 76 HRB (141 HV_{500}) and 72 HRB (129 HV_{500}), respectively. The conductivity of the upper and lower skins averaged 31.8 %IACS and 30.4 %IACS, respectively. These data were consistent with a T3 or T4 temper per AMS 2658.[7]

3.2. Intact Rotor Blade

Figure 18 and Figure 19 illustrate the other rotor blade section, as received. The blade segment had been marked "S/N 2027" with red marker on the upper face (painted yellow, Figure 18). The lower face of the blade was painted black, and exhibited transverse witness marks, nicks, and scratches consistent with impact from ground debris during blade rotation. There was no indication of cracks, dents, or other damage on this rotor blade section.

Figure 20 illustrates a protrusion observed on the trailing edge of the exemplar section. This feature was located in a similar position as the fatigue crack initiation site of the cracked rotor blade. The paint in this area was removed using a chemical paint stripper. After removal of the paint, no anomalies were observed in the underlying sealant or aluminum skins. The paint stripper was also found to attack the trailing edge epoxy compound with prolonged exposure. No defects or indications of cracking were observed in the underlying aluminum skins or adhesive.

The dimensions near the location on the trailing edge of the cracked blade were measured on the intact blade using a Keyence VHX-1000 digital microscope. These dimensions are shown on the intact blade in Figure 21. The radius of the blade extension and the length of the runout angle were consistent with the measurements in the blade skin drawing. The composition of the upper aluminum skin, measured using XRF, was consistent with AA 2024 aluminum. The electrical conductivity of the skin averaged 31.5 %IACS, consistent with that of the cracked blade upper skin.

3.3. Exemplar Sample Pieces

Samples from the tip sealant and internal adhesive were examined using a Fourier Transform Infrared (FTIR) spectrometer with a diamond attenuated total reflectance (ATR) accessory in accordance to ASTM E1252-98 and ASTM E334-01.[8] The spectrometer was used to collect and process infrared wavelength absorbance spectra of each sample.

The spectrum for the tip sealant was a strong visual match to a spectrum from a known sample of 3M Scotch-weld 2216 B/A. Both spectra exhibited a strong triplet at

[5] ASTM E384 – Standard Test Method for Knoop and Vickers Hardness of Materials
[6] ASTM E1004 – Standard Test Method for Determining Electrical Conductivity Using the Electromagnetic (Eddy-Current) Method
[7] AMS 2658 – Hardness and Conductivity Inspection of Wrought Aluminum Alloy Parts
[8] ASTM E1252 – *Standard Practice for General Techniques for Obtaining Infrared Spectra for Qualitative Analysis and American Society for Testing Materials*. ASTM E334 - *Standard Practice for General Techniques of Infrared Microanalysis*

~3750-3600 cm^{-1}, a broad peak at ~3350-3100 cm^{-1}, and a doublet at ~3000-2800 cm^{-1}. The spectrum for the internal adhesive was a strong visual match to a spectrum from a known sample of FM 94. Both spectra exhibited a broad peak at ~3600-3200 cm^{-1}, a triplet peak at ~3000-2800 cm^{-1}, and two sets of doublet peaks at ~1700-1600 cm^{-1} and 1550-1450 cm^{-1}.

Erik Mueller
Materials Research Engineer

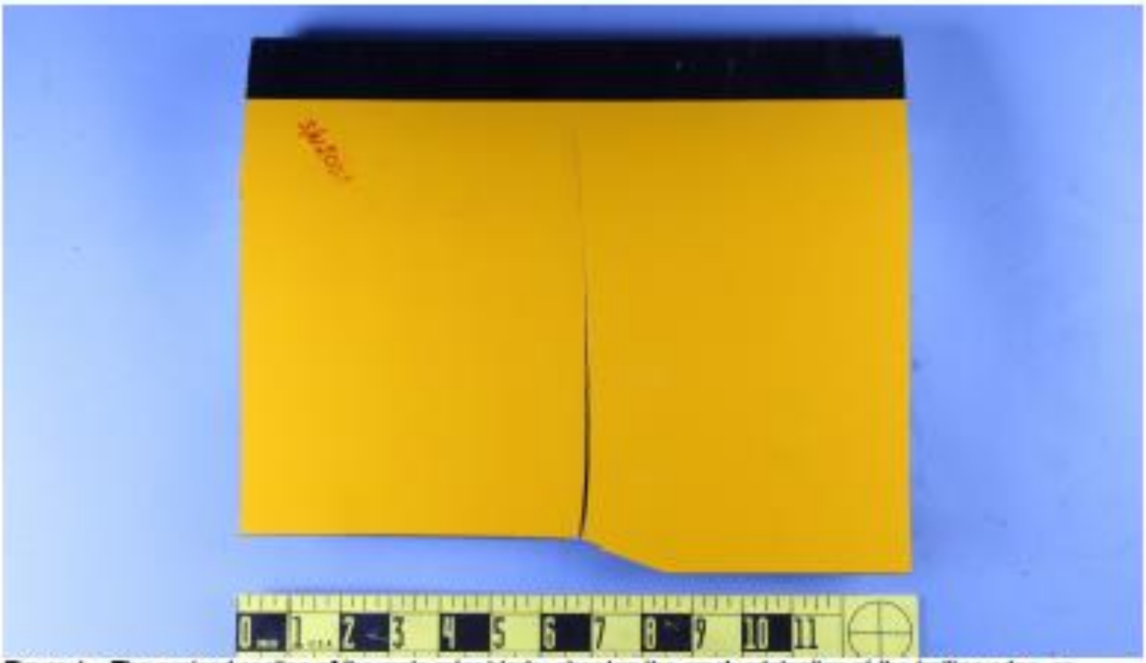


Figure 1 – The excised section of the main rotor blade, showing the crack originating at the trailing edge, as received.

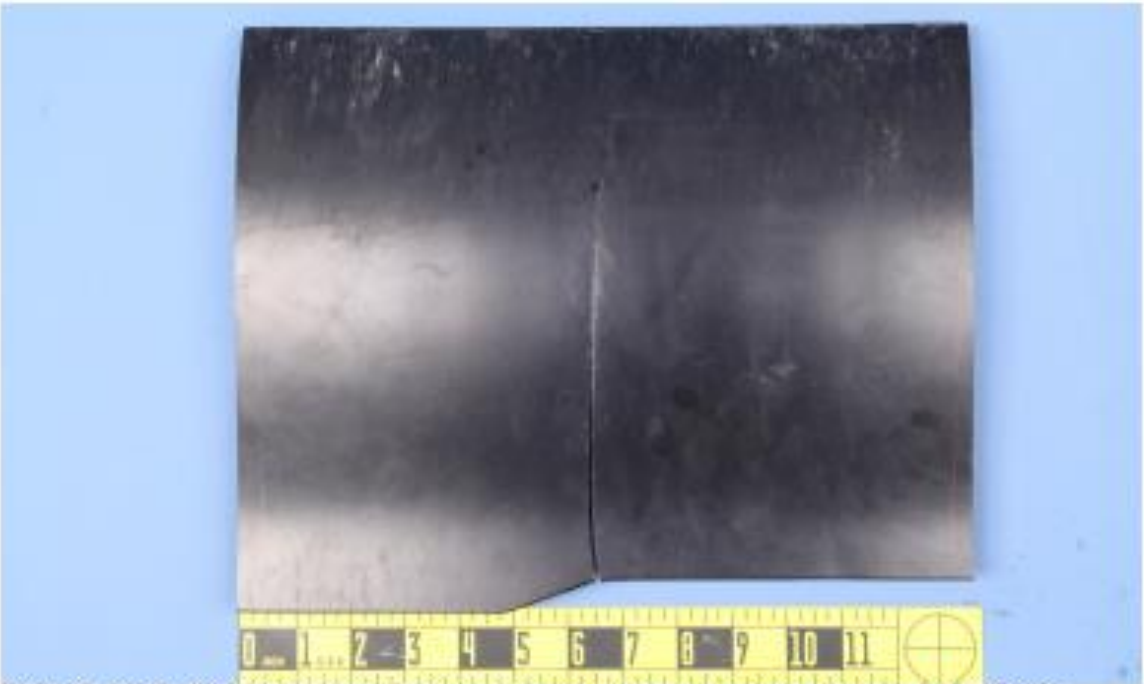


Figure 2 – The excised section of the main rotor blade, viewed from the reverse side shown in Figure 1, as received.

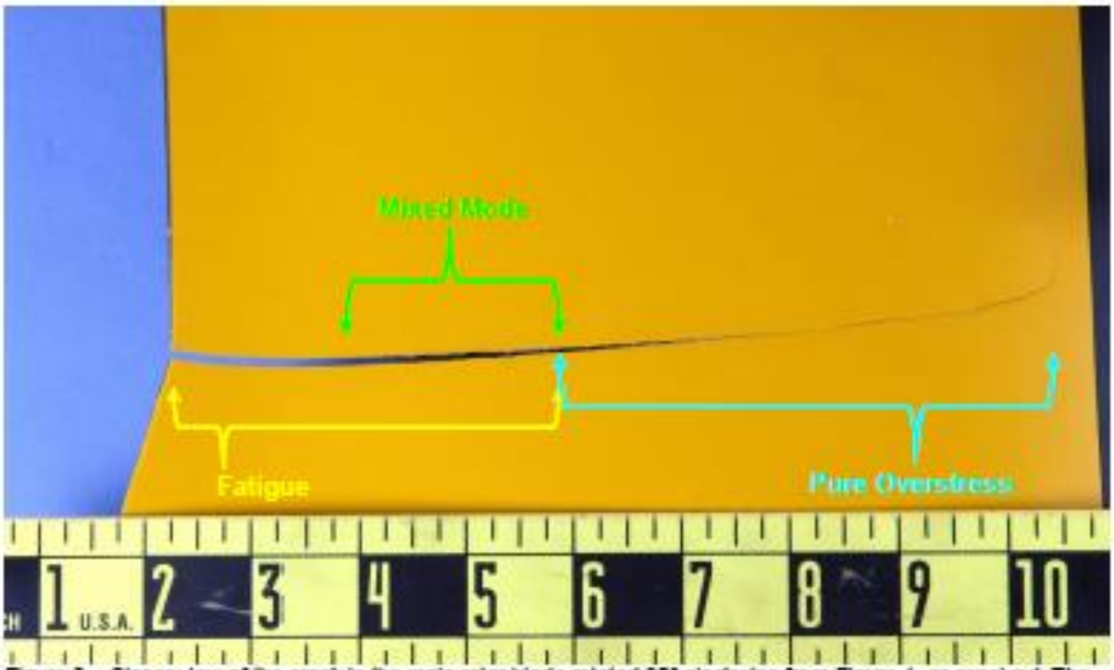


Figure 3 – Closer view of the crack in the main rotor blade rotated 90° clockwise from Figure 1, as received. The area later determined to be fatigue, mixed mode, and overstress are labeled on the figure.

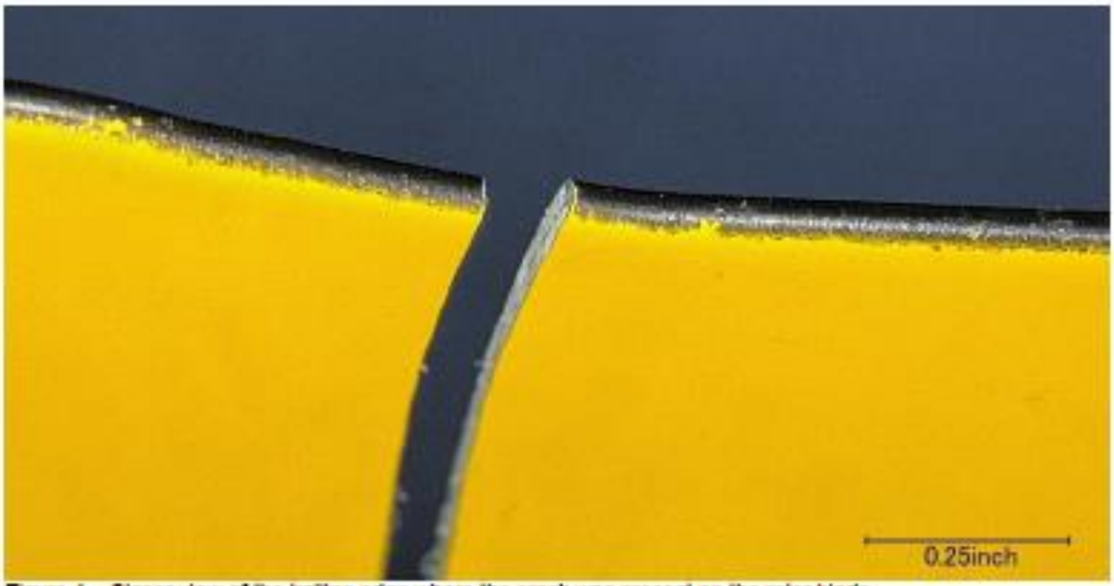


Figure 4 – Closer view of the trailing edge where the crack was present on the rotor blade.

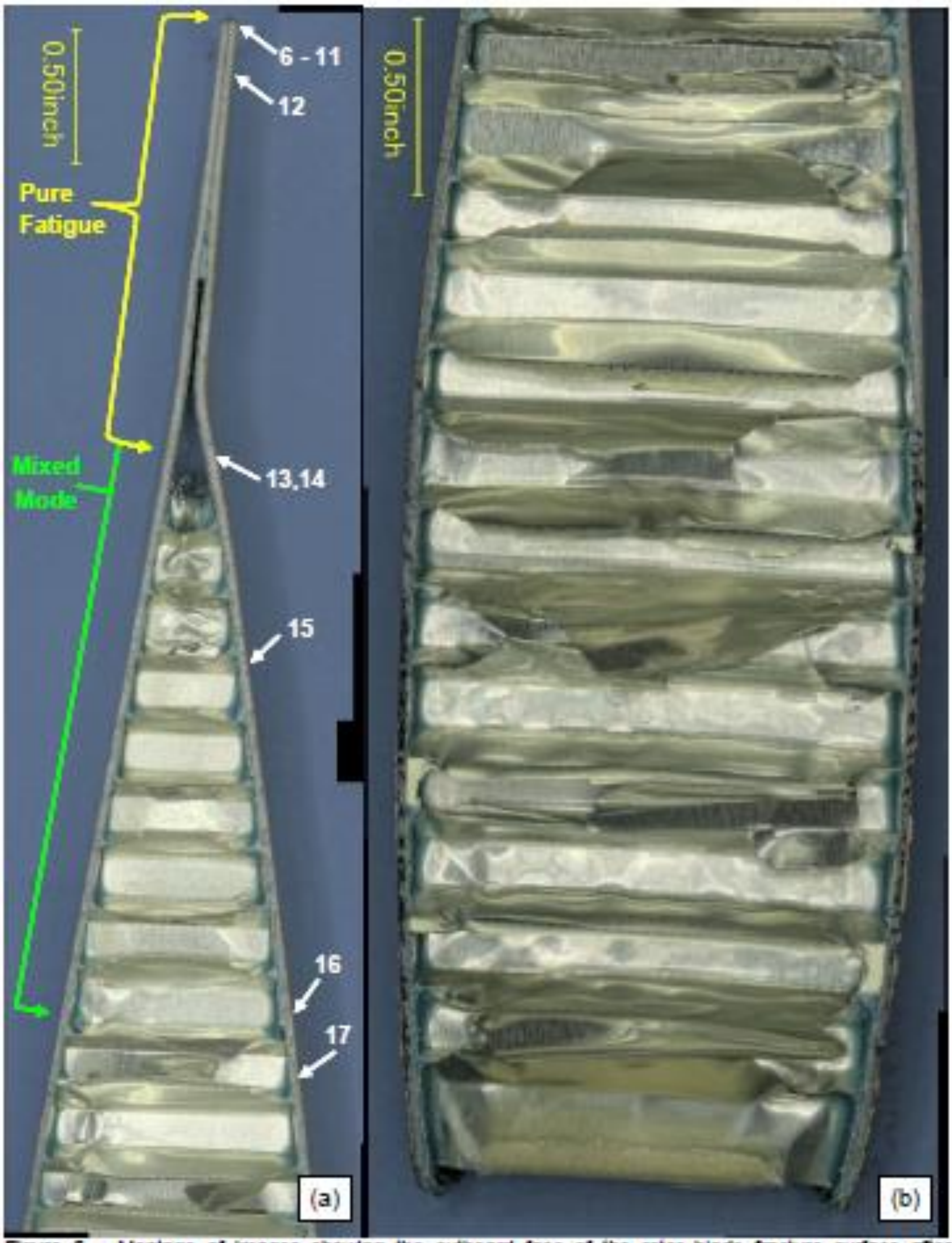


Figure 5 – Montage of images showing the outboard face of the rotor blade fracture surface after intentional crack opening in the laboratory from (a) the trailing edge side and (b) the leading edge side. The areas later determined to be fatigue, mixed mode, and overstress are annotated on the figures. Areas examined in the SEM later are labeled by figure number in (a).

Figure 6 – The initiation point of the fatigue crack at the trailing edge of the main rotor blade.

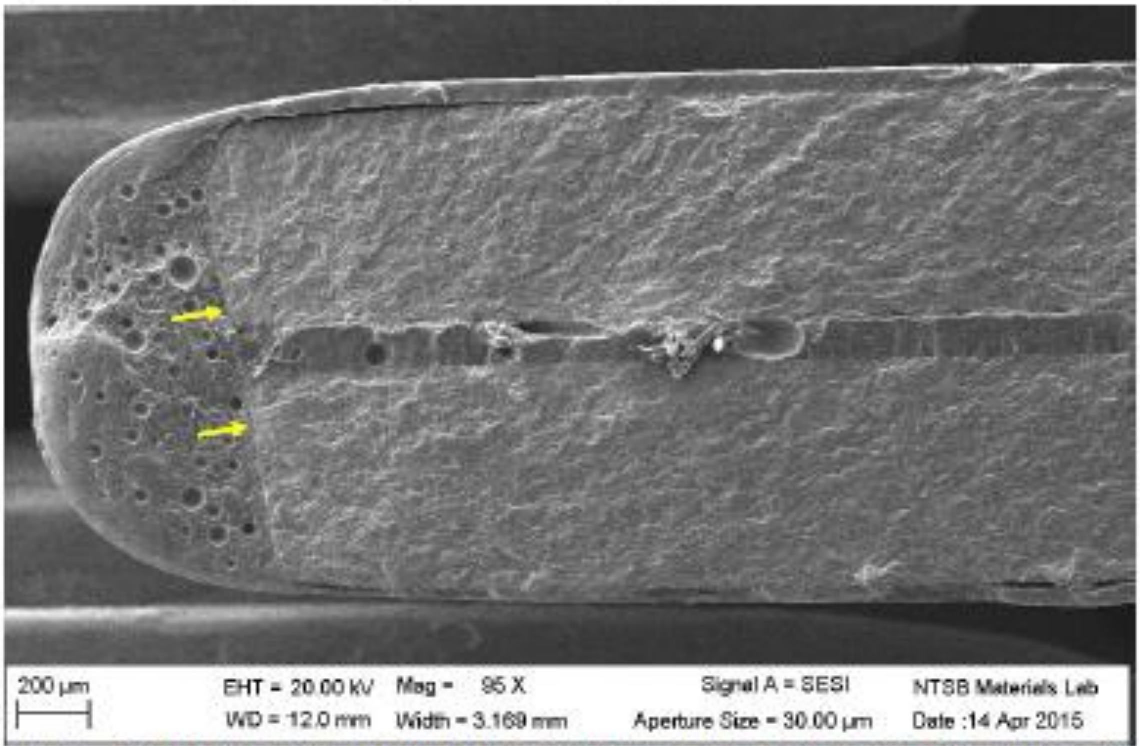


Figure 7 – Secondary electron (SE) micrograph of the fatigue crack initiation site in the composite rotor blade. The arrows denote the fatigue crack initiation sites in the aluminum sheets.

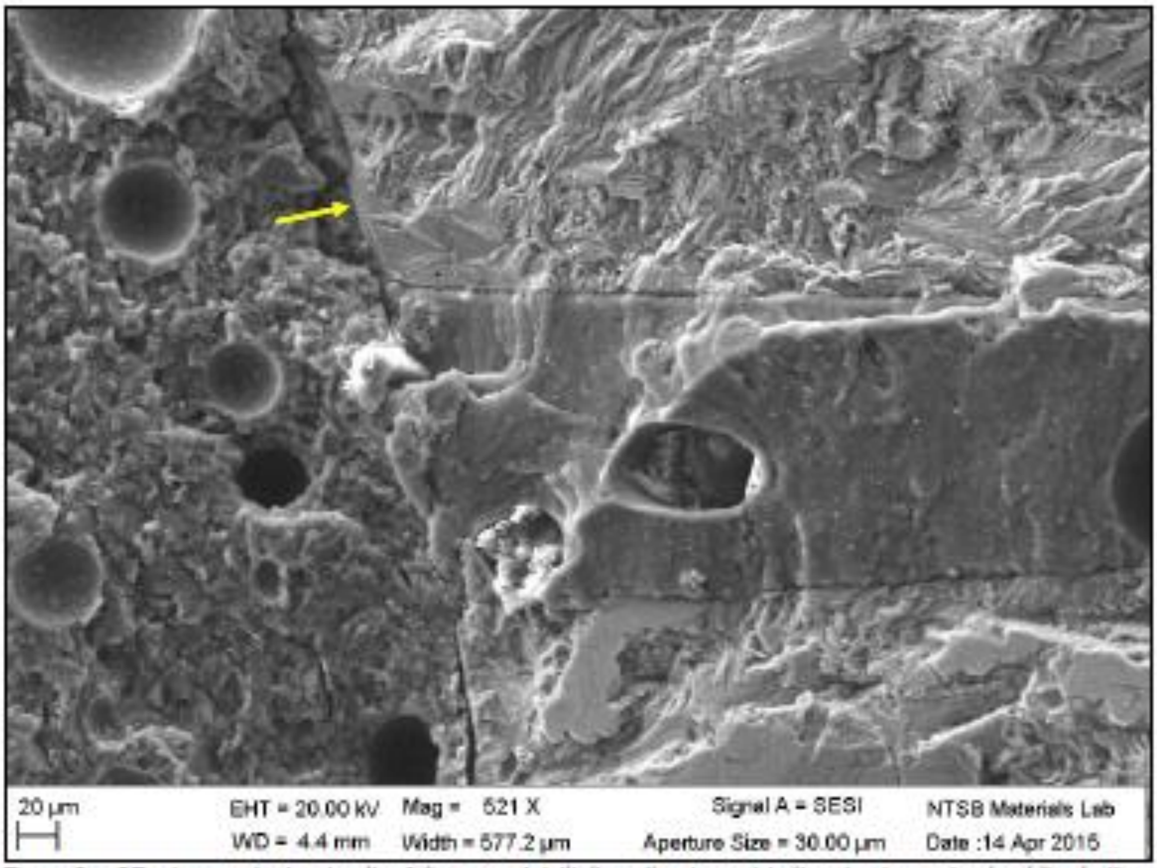


Figure 8 – SE micrograph showing the fatigue crack initiation sites (arrow) in the aluminum sheets at the trailing edge.

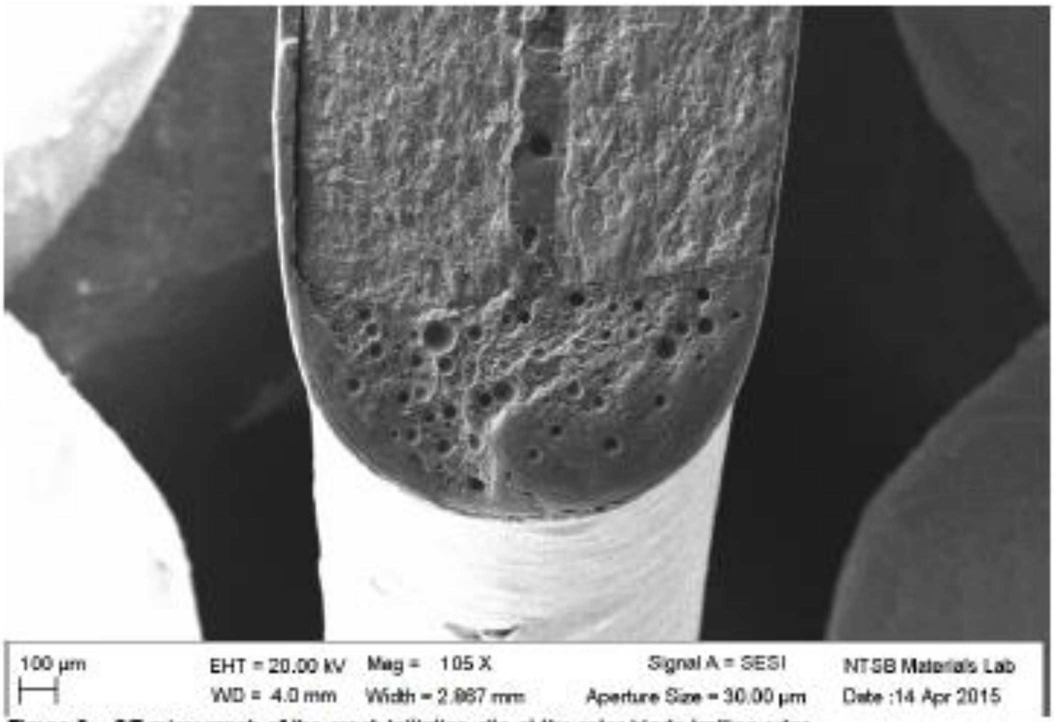


Figure 9 – SE micrograph of the crack initiation site at the rotor blade trailing edge.

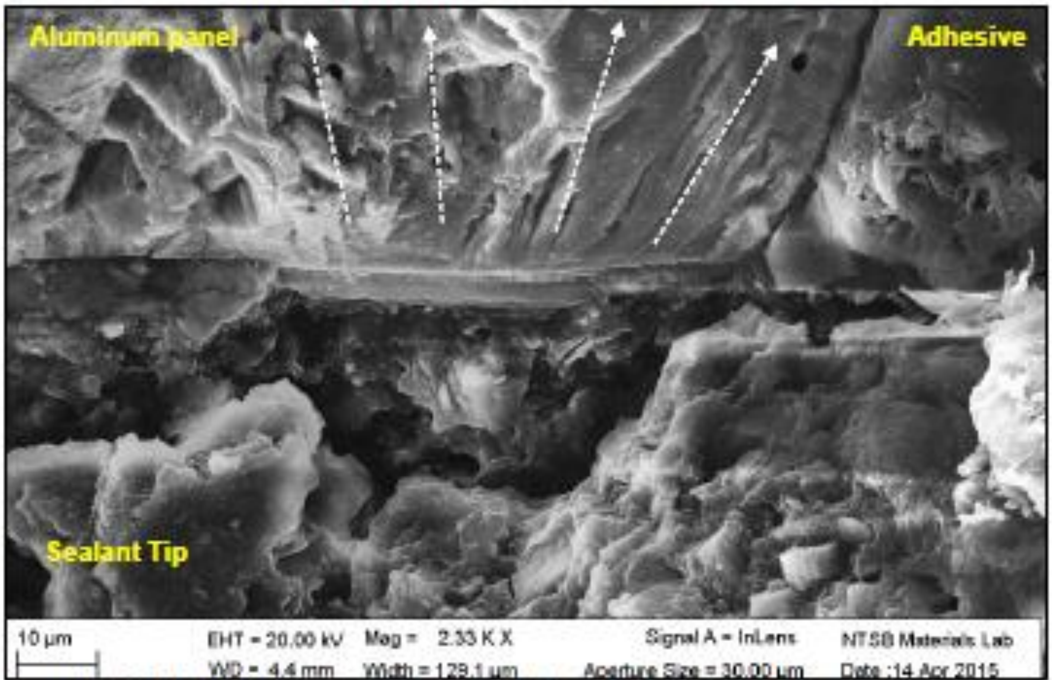


Figure 10 – SE micrograph of the upper aluminum sheet crack initiation site in Figure 7.

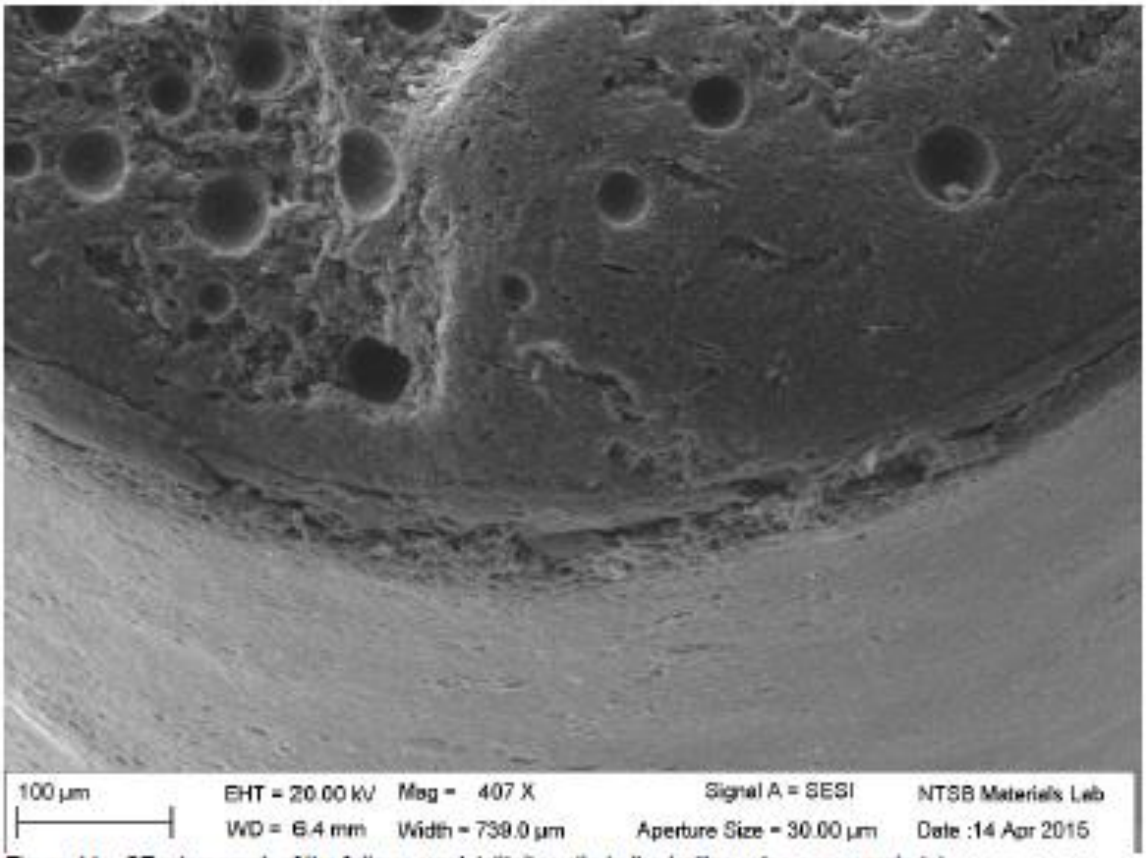


Figure 11 – SE micrograph of the fatigue crack initiation site in the trailing edge epoxy material.

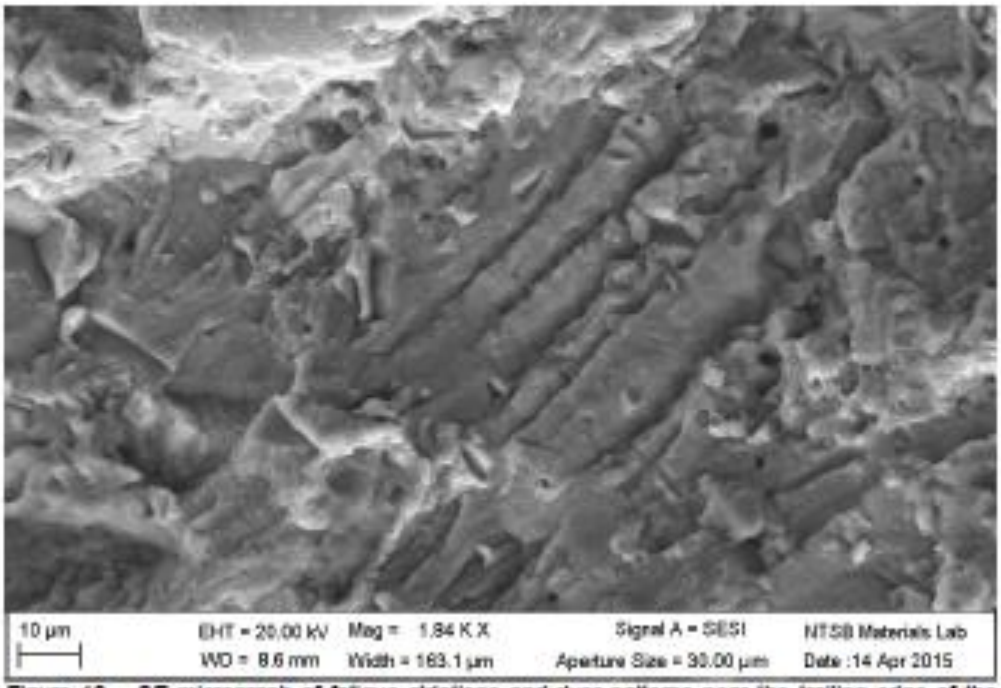


Figure 12 – SE micrograph of fatigue striations and river patterns near the trailing edge of the rotor blade in the upper aluminum sheet.

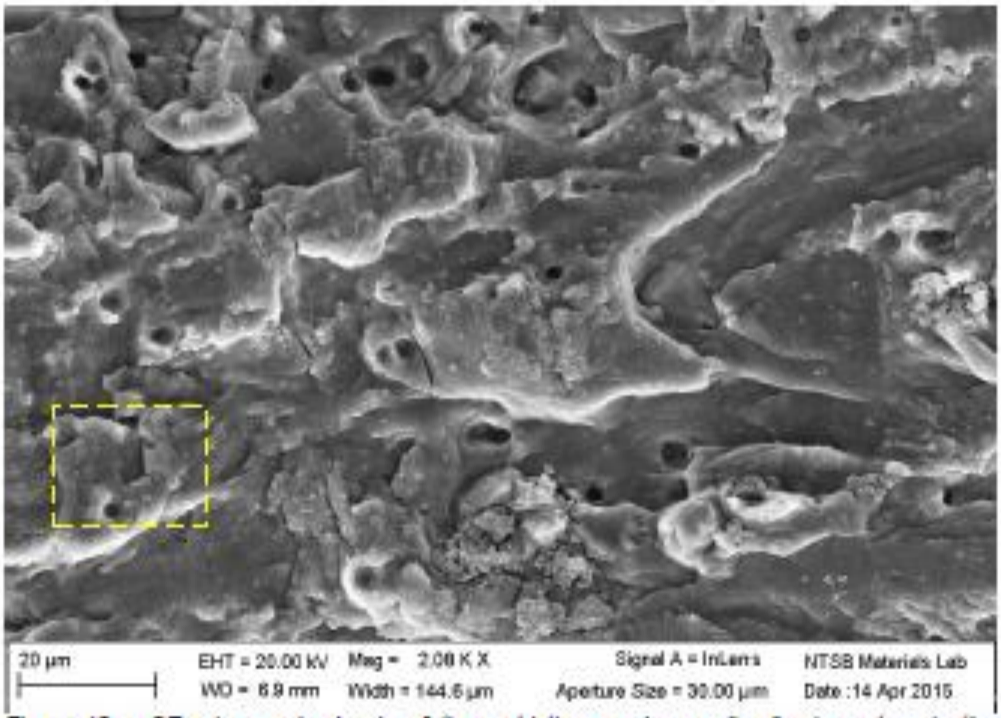


Figure 13 – SE micrograph showing fatigue striations and sporadic dimple rupture in the beginning of the mixed mode portion of the aluminum sheet fracture surface.

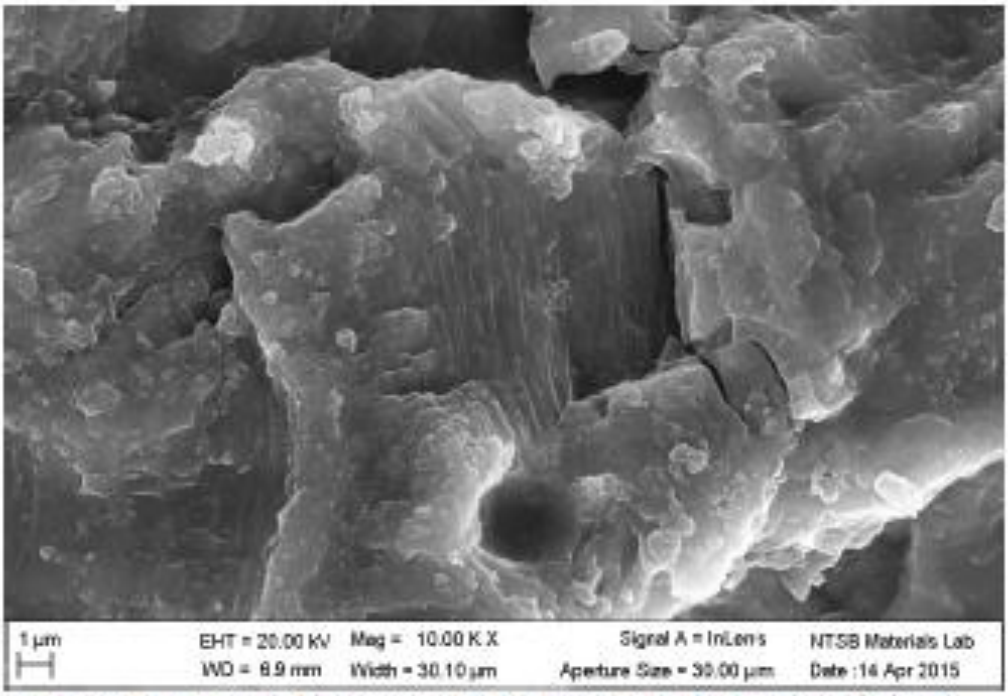


Figure 14 – SE micrograph of the yellow boxed area in Figure 13, showing fatigue striations.

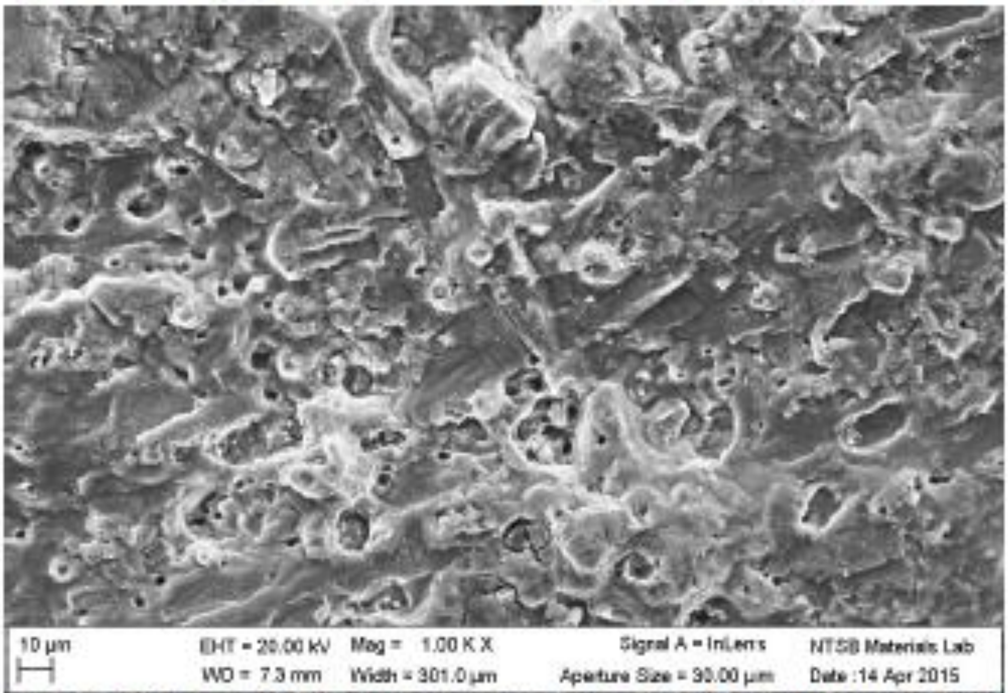


Figure 15 – SE micrograph of the aluminum fracture surface toward the forward side of the mixed mode region.

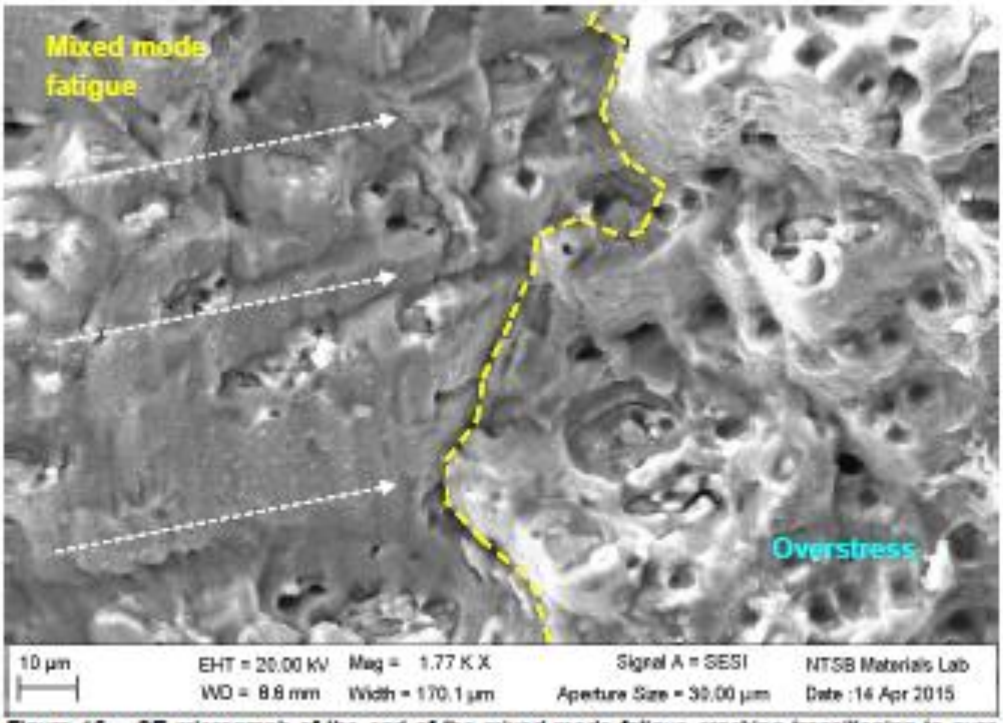


Figure 16 – SE micrograph of the end of the mixed mode fatigue cracking transitioning to pure overstress (dashed line).

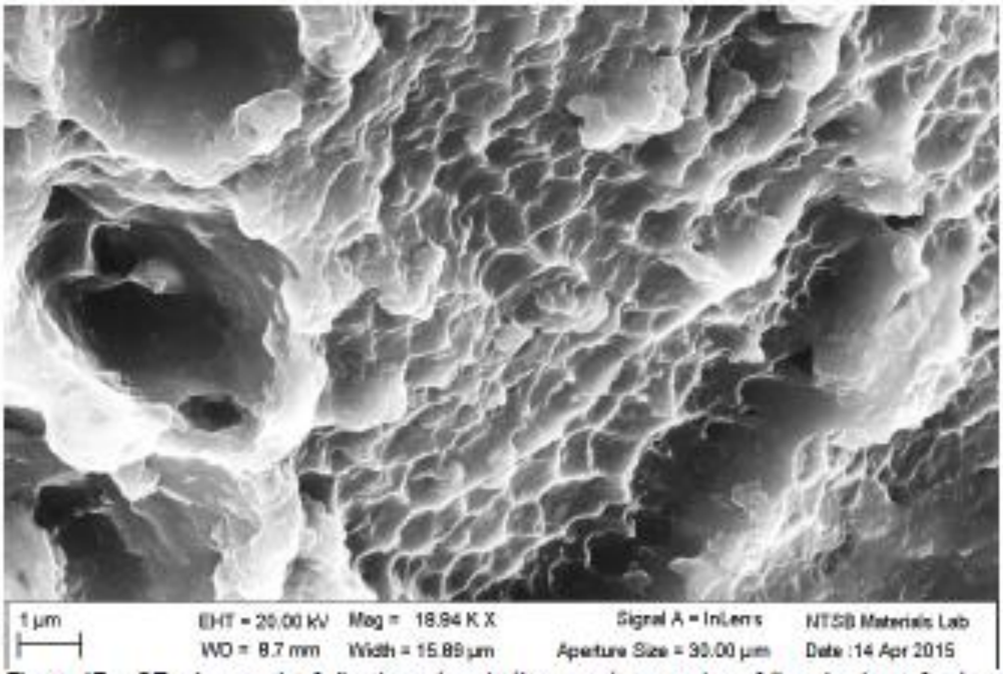


Figure 17 – SE micrograph of dimple rupture in the overstress region of the aluminum fracture surface.

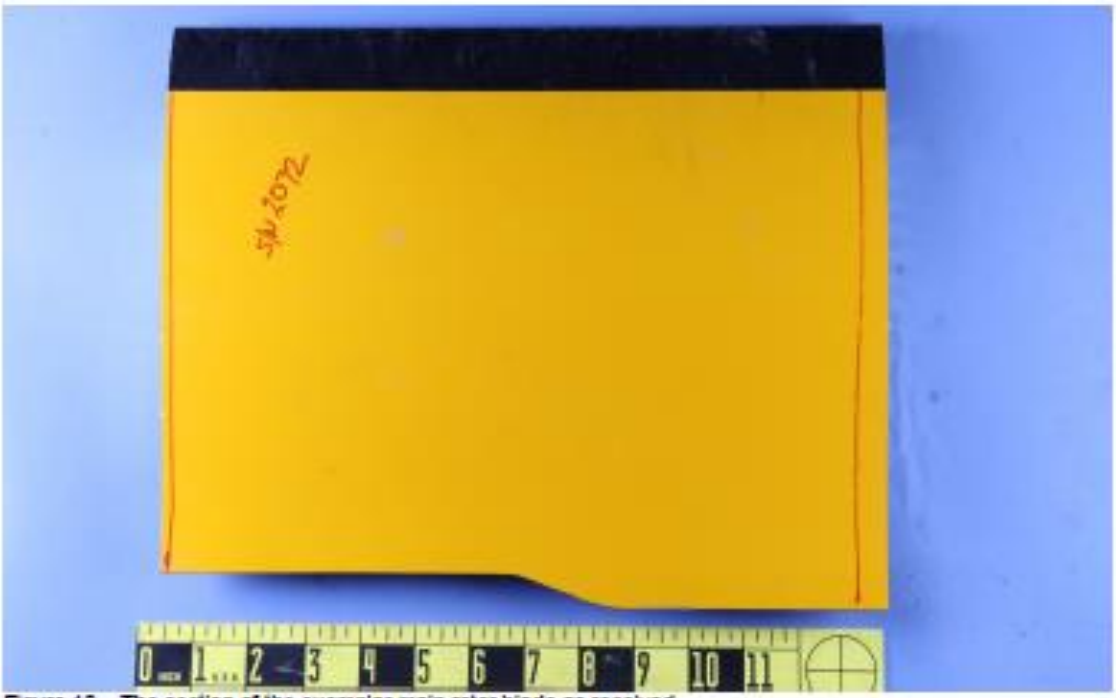


Figure 18 – The section of the exemplar main rotor blade as received.

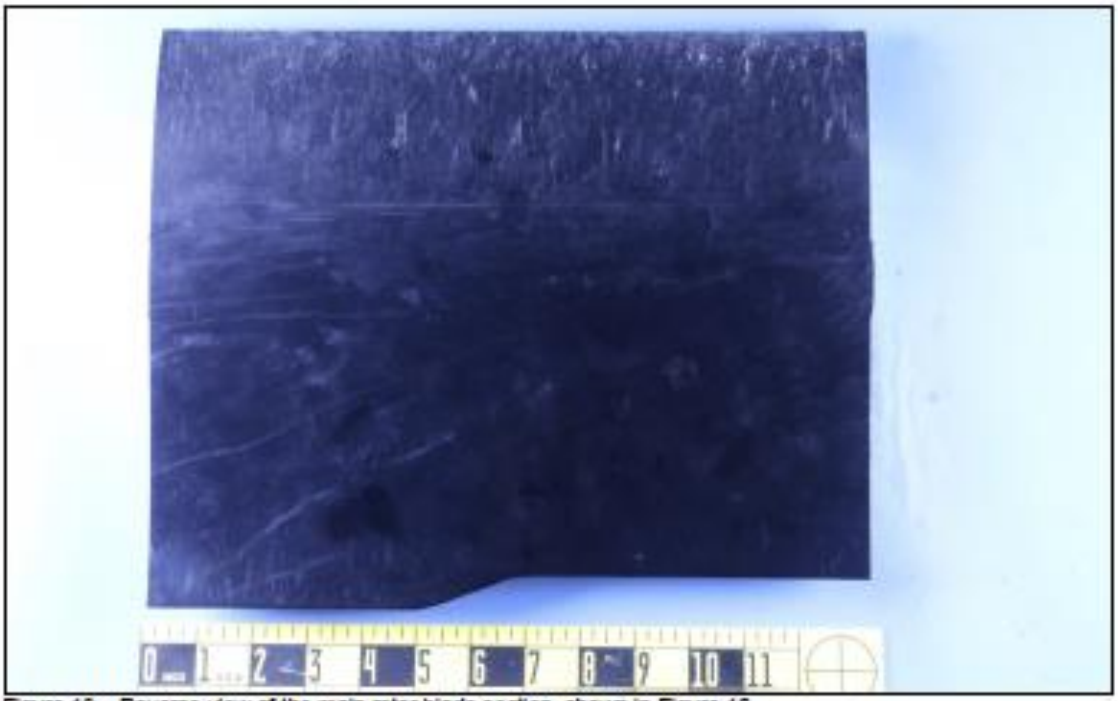


Figure 19 – Reverse view of the main rotor blade section, shown in Figure 18.

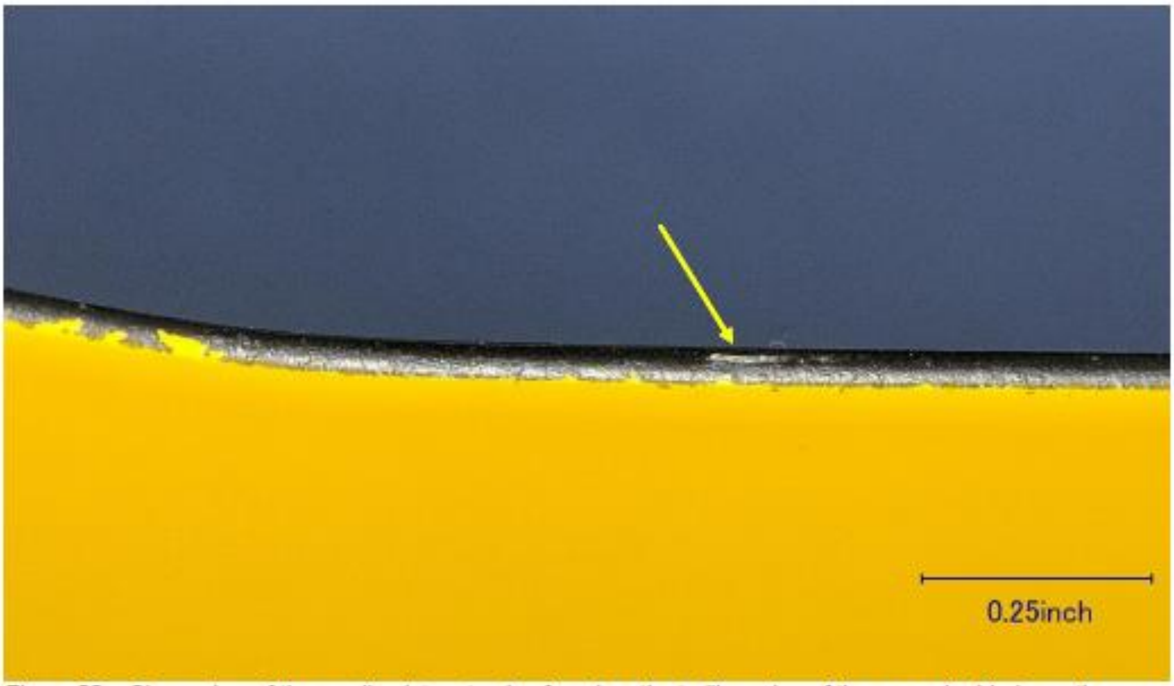


Figure 20 – Closer view of the small paint protrusion found on the trailing edge of the exemplar blade section (arrow), as received.

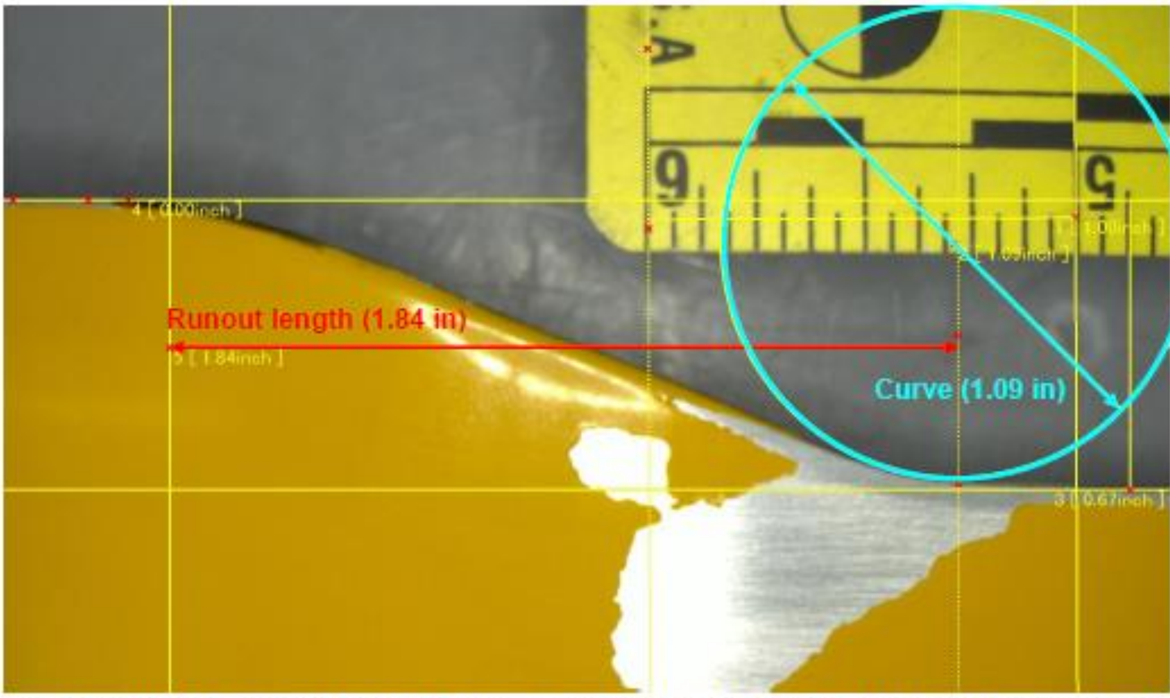


Figure 21 – The trailing edge of the intact blade section, showing several measurements from the Keyence VHX-1000 digital microscope.

Appendix 2: Quest Integrity NZL metallurgical examination report

Quest Integrity

Project Name:	R44 Blade Fracture Inspection
Attention:	Steve Walker
Client:	Transport Accident and Investigation Commission
Report Number:	241305.01
Confidential To:	Client
Email Address:	S.Walker@taic.org.nz
From:	Matt Smillie
Checked & Approved by:	Stephen Rowbotham
Date:	10 August 2018

www.QuestIntegrity.com
Quest Integrity NZL Limited, a TEAM Industrial Services Company

1. Introduction

The Transport Accident Investigation Commission requested a scope of work for the visual inspection of a fractured R44 rotor blade in order to review the failure mode, origin and whether there is any evidence of abnormal stress on the fracture surface. Quest Integrity performed the following scope of work:

- Visual and optical stereoscopic inspection of the supplied fracture surfaces.
- Scanning electron microscope (SEM) inspection of selected sections of the facture surface in order to determine fracture mode(s), e.g. ductile dimpling, cleavage, fatigue crack propagation.
- Short report summarising findings (this document).

2. Visual Inspection

Both fracture surfaces of the failed blade were supplied, along with the matching location on the other blade from the rotor (Figure 1). The following were observed in the as-received condition:

- The fracture initiated at or very close to a transition radius on the trailing edge of the blade.
- The initial fracture surface was perpendicular to the axis of the blade trailing edge transition, and macroscopically flat and smooth, suggestive of a tensile crack opening stress along the blade (Figure 2).
- Approximately 40 to 45 mm from the trailing edge, the fracture surface was visibly rougher and became less smooth.
- Between 85 to 90 mm from the trailing edge, the fracture surface transitioned to a 45 degree angle compared to the surface of the blade aerofoil, indicative of a ductile overload as the applied tensile stress resulted in a 45 degree shear lip.
- The shear lip on the upper surface fracture reversed to the opposed 45 degree shear lip at 160 mm. Both fractures turned parallel to the blade axis at the leading edge spar.

The overall morphology of the fracture propagation is consistent with a crack initiation at the trailing edge transition radius, propagating via a tensile cyclic stress, with an increasing mean stress, until the mean tensile stress increased to such a magnitude that a ductile overload occurred, resulting in a final fracture to the leading edge spar. This observation is also consistent with the annotations noted on the supplied sample, shown in Figure 1.

The matching location was examined under a stereoscopic microscope. At the same location as the fracture (the trailing edge transition), no indications of cracking was observed, although it is noted that the trailing edge was mostly covered by the coating/sealer present at the trailing edge, despite obvious removal of the surrounding coating by a chemical cleaner. It is not known if the removed coating contained any sign of distress, such as cracking or buckling that may indicate the presence of excessive strain at that location.

3. Scanning Electron Microscope Inspection

A supplied section of the fracture surface was examined in a FEI Quanta 450 scanning electron microscope (SEM). The supplied section included the crack origin and the fracture up to 80 mm from the trailing edge. The section had already been coated for SEM examination as supplied, so no cleaning or additional coating was performed before examination. The following were noted:

- The fracture surfaces at the apparent origin(s) were irregular, with extensive tear ridges visible aligned as river marks suggestive of the inner corner of the aluminium sheets as the fracture origin (Figure 3 to Figure 5). The upper sheet of aluminium had a faceted appearance at that point (Figure 4). The lower sheet had extensive mechanical damage (smearing) at that location (Figure 5).
- No obvious defects were visible in the aluminium. The adhesive layer between the two aluminium sheets was visibly adherent to the aluminium, to the degree where crack propagation was visibly matched in the aluminium and adhesive. This suggests the mechanical bond was strong enough to allow propagation of the crack between, or simultaneously along, the aluminium and adhesive layers (Figure 6).
- The coating/sealer present at the trailing edge had numerous pores, but the interface between the aluminium and the trailing edge suggests that there was no interaction between the coating/sealer.
- Striations were not visible close to the apparent fracture origins – the surface was obscured suggested a combination of oxidation, some corrosion product or excessive coating for SEM examination as represented by visible "mud cracking".
    - Energy dispersive spectroscopy (EDS) was performed to identify the presence of any corrosive species. The elements present on the surface were identified as aluminium and with minor traces of oxygen, magnesium, silicon, copper and gold. This was consistent with an age hardenable aluminium copper alloy (e.g. 2000 series alloy) with a gold sputter coating for SEM examination.
- Faint regular striations were visible a few hundred microns away from the apparent origin, with the propagation directions consistent with the origin being the inner corner of the aluminium layers (Figure 7 to Figure 9).
- Wider striations were visible at 15 mm from the trailing edge, along with some ductile dimpling, indicative of a mixed mode (fatigue plus overload) fracture (Figure 10).
- The striation spacing suggested that the propagation was marginally high cycle fatigue (on the order of 10,000 total striations (or 10,000 stress cycles) in the first 40 mm of fracture propagation).

4. Discussion of Findings

The visible assessment of the fracture surface leads to the following observations:

- The first 40 mm of the fracture propagated via a rapid high cycle fatigue mechanism
- The direction and appearance of the fracture is strongly suggestive of a tensile stress at a stress concentration formed by the transition of the leading edge.
- The location of the origin, at the inner corner at the trailing edge of the aluminium layers does strongly suggest that part of the loading is due to bending or flexing around the minor axis of the blade section. It is understood that the nominal loading at this point is a combination of centrifugal loading from rotation, and bending/flexing around the major axis of the blade cross section due to aerodynamic loading. The additional bending load required to initiate a crack at the apparent location may be due to a particular vibrational mode in the blade, excited by other loads applied to the blade.
- The absences of any visible defects at the crack origins strongly suggests that the fatigue crack was caused by an applied load in excess of the capacity of the blade to meet the designed/designated fatigue life. The applied load would be a combination of varying centrifugal, aerodynamic and possibly vibrational stresses, resulting in the initiation and propagation of a fatigue crack at a critical location on the blade trailing edge (the stress concentration formed by the trailing edge transition).

5. Conclusions and Recommendations

After visual and SEM inspection of the supplied samples from a fracture R44 rotor blade, the following is concluded:

- The fracture was due to fatigue initiating at the trailing edge transition radius.
- The absence of any material defect at the fracture origin suggests that the fatigue fraction was caused by a variable tensile load in excess of the capacity of the blade to meet the designed/designated fatigue life.

It is recommended that the following be considered

- Review of design data, or analysis of the blading, in order to assess possible vibrational modes that could have contributed to the loading on the blade.
- Strip the remainder of the coating from the matching blade sample and perform tests (visual, NDT, metallographic sectioning) in order to confirm the absence or presence of micro-cracks initiating at the matching location.

6. Figures

Figure 1: **Fractured blade sample shown at left, matching location on opposite blade shown at left. Trailing edge transition radius arrowed on both samples. Upper surface shown**

Figure 2: **Optical image of fracture surface at origin. Aluminium sheets joined by adhesive (blue) and covered with coating/sealer (dull grey at trailing edge, arrowed). Upper surface at top.**

Quest Integrity

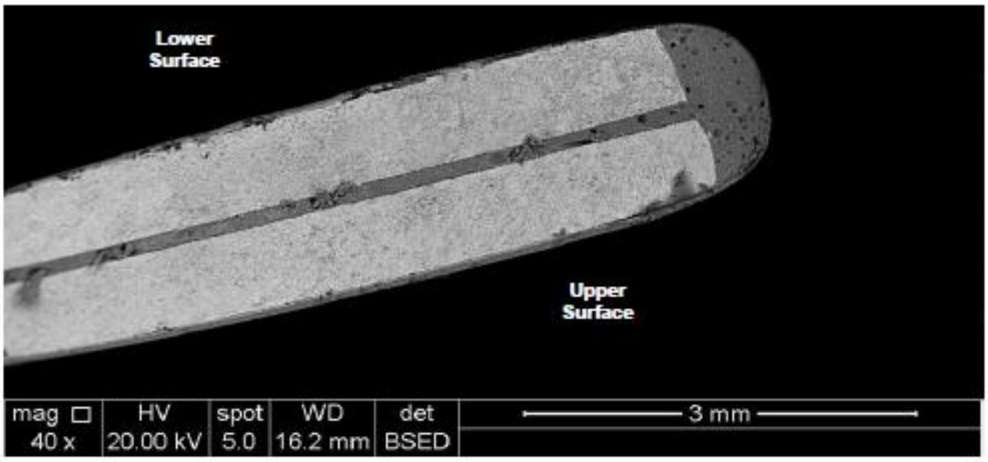


Figure 3: **Backscatter SEM image of trailing edge fracture. Upper surface at bottom. Adhesive and coating/sealer dark grey, aluminium light grey.**

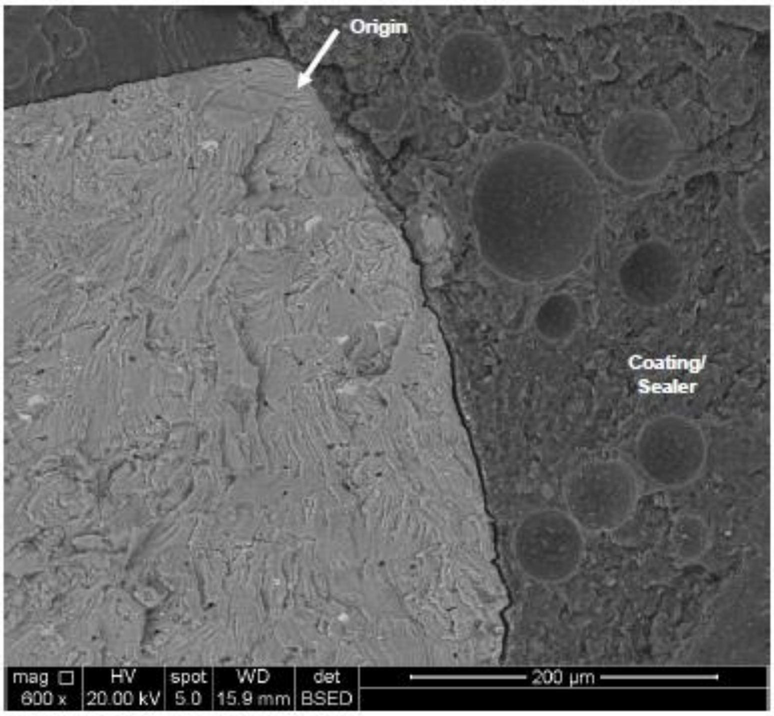


Figure 4: **Possible origin (arrowed) on upper aluminium layer as indicated by tear ridges, river marks and faceted appearance. Back scattered image. Note porosity and lack of adherence between aluminium and coating/sealer at trailing edge.**

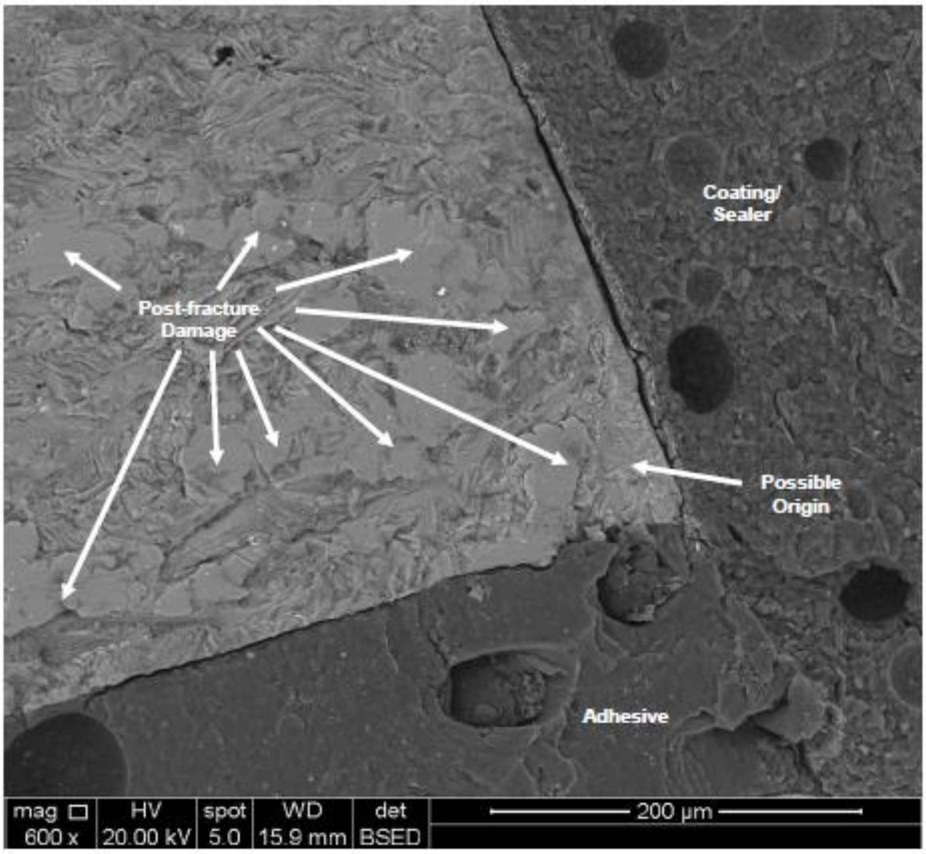


Figure 5: **Possible origin (arrowed) on lower aluminium layer. Associated fracture surface showing considerable post-fracture mechanical damage. Back scattered image. Note porosity and lack of adherence between aluminium and coating/sealer at trailing edge.**

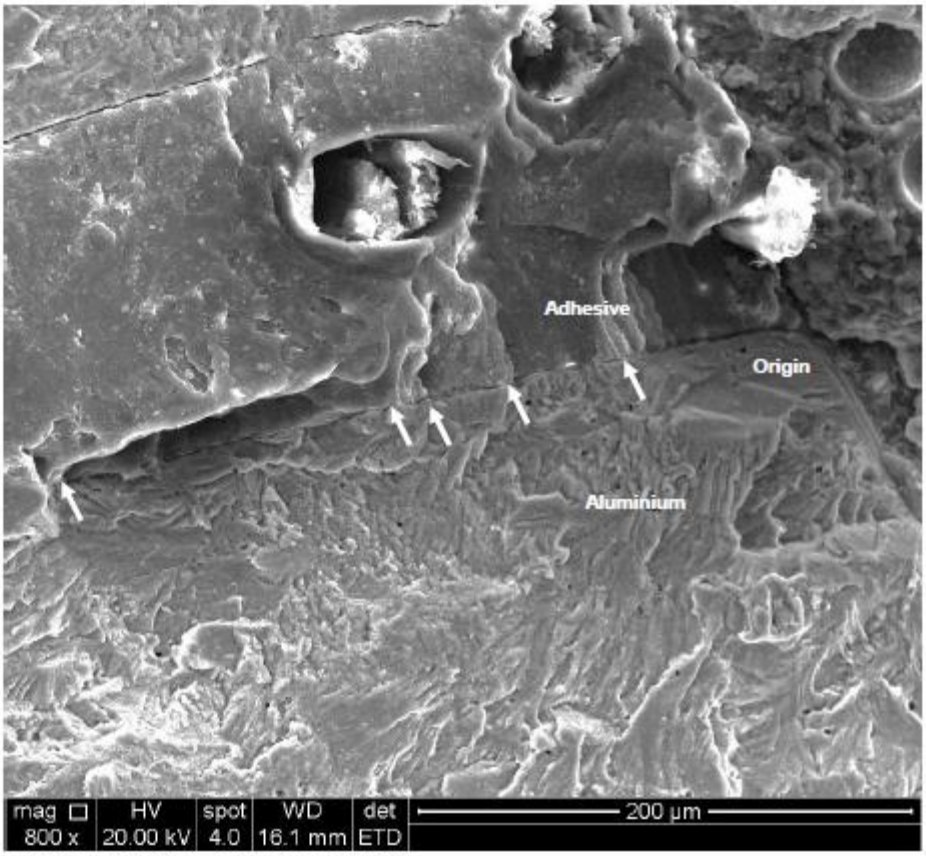


Figure 6: Detail near upper aluminium origin, showing apparent simultaneous propagation through adhesive and aluminium (matching fracture steps arrowed).

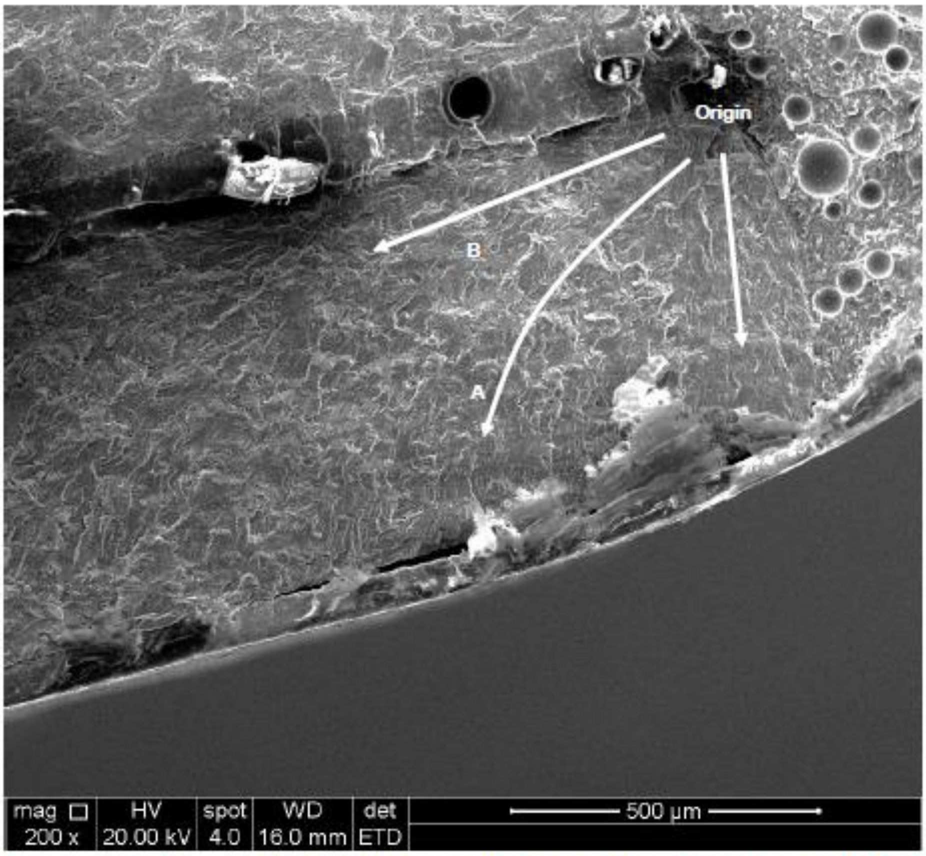


Figure 7: Secondary electron image of upper aluminium layer showing tear ridges radiating from origin (propagation direction arrowed). Detail regions labelled "A" and "B".

Quest Integrity

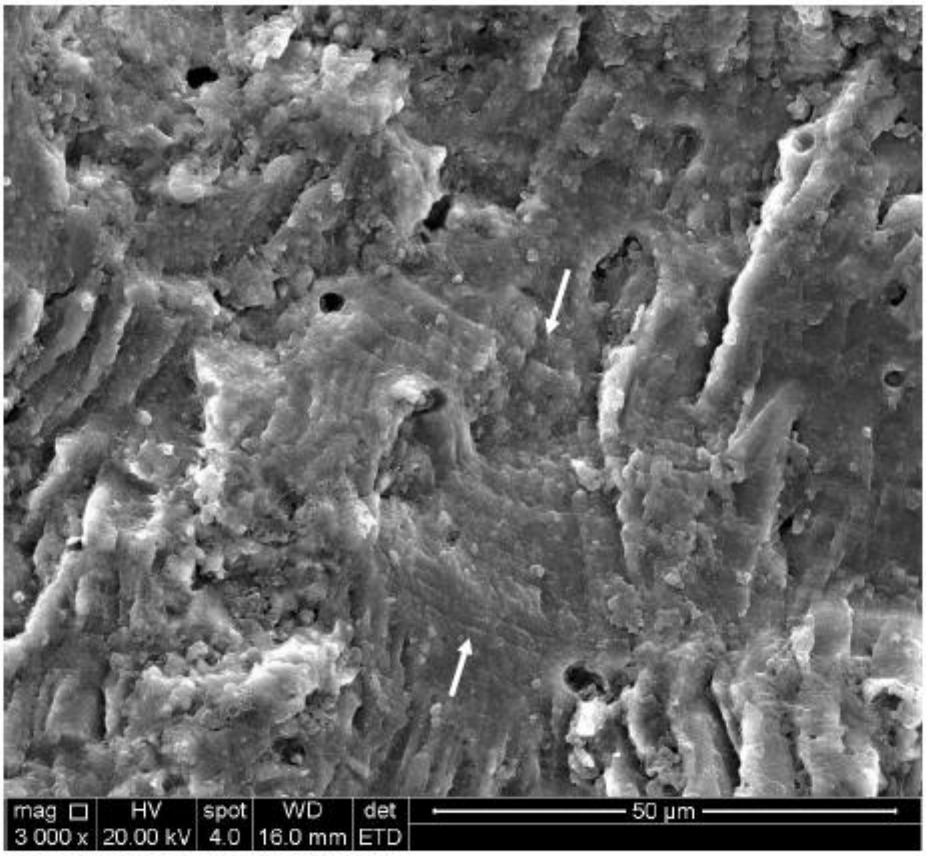


Figure 8: **Region "A" in Figure 7. Faint striations visible, perpendicular to tear ridges, at 2 µm to 3 µm separation, between arrows.**

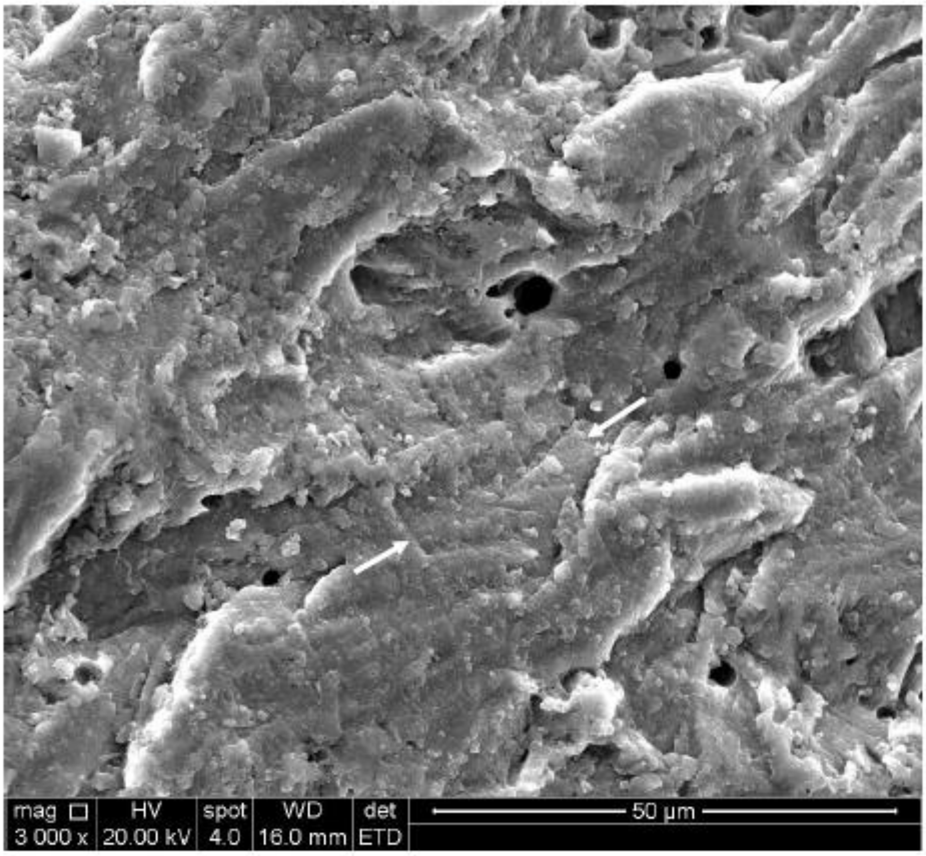


Figure 9: **Region "B" in Figure 7. Faint striations visible, perpendicular to tear ridges, at 2 µm to 3 µm separation, between arrows.**

Quest Integrity

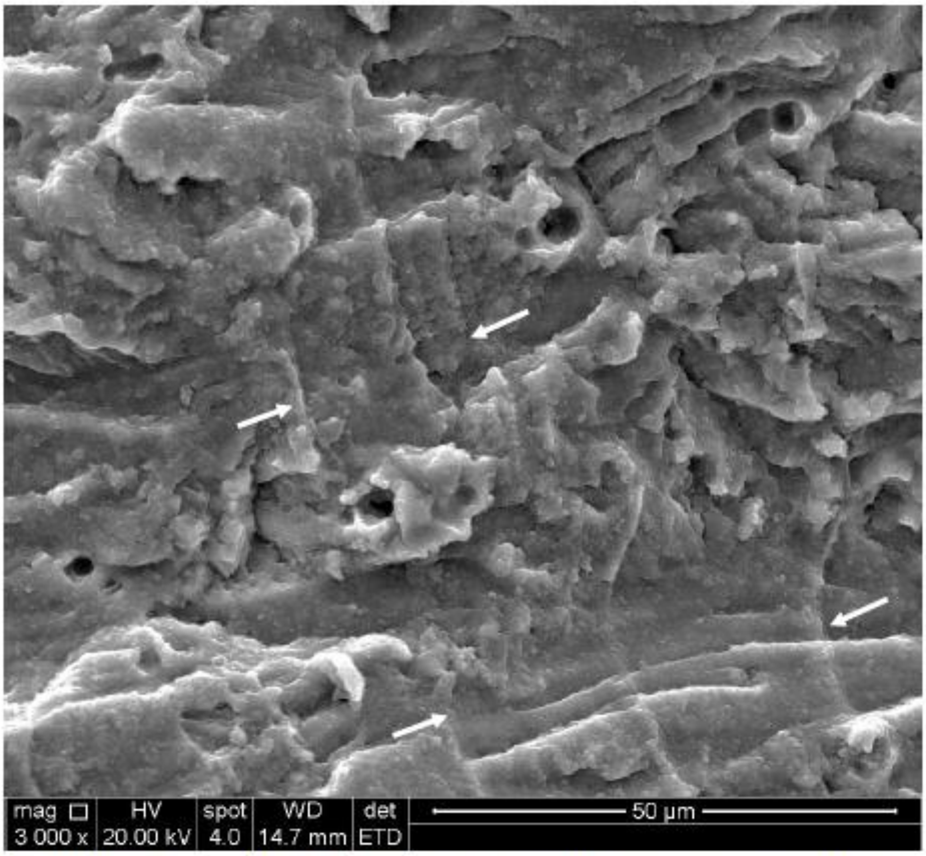


Figure 10: **Widely spaced (approximately 10 to 20 µm) striations approximately 15 mm from trailing edge in lower aluminium layer.**

ROBINSON
HELICOPTER COMPANY
2901 Airport Drive, Torrance, California 90505 Phone (310) 539-0508 Fax (310) 539-5198

R44 SAFETY ALERT

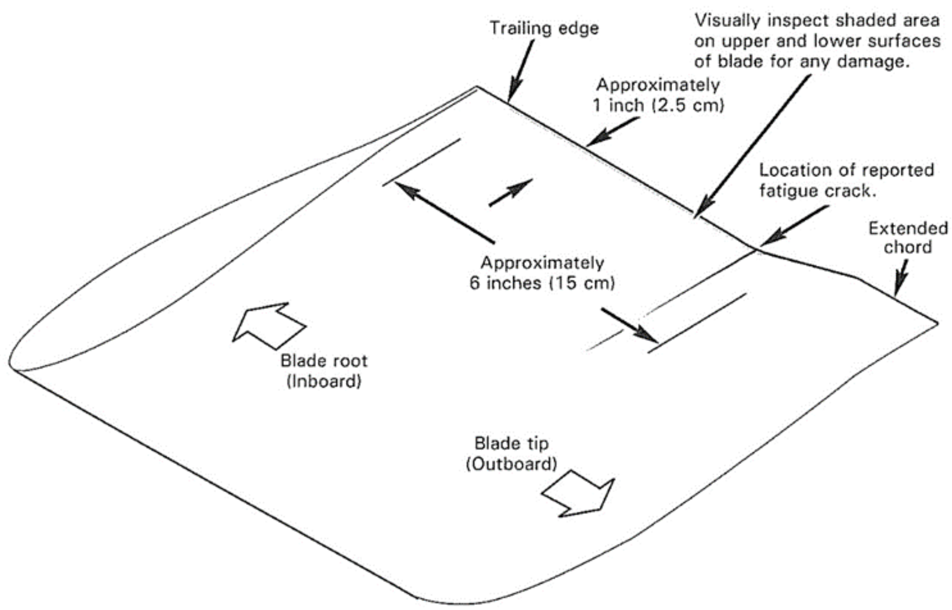
Issued: 23 February 2015

C016-7 MAIN ROTOR BLADE CRACK

RHC has received a report of an R44 C016-7 main rotor blade fatigue crack where the blade chord transitions to the extended chord (see figure). Careful visual inspection of this area during preflight will provide early detection and prevent a catastrophic failure.

During daily preflight inspection, visually inspect the trailing edge in the area shown on upper and lower surfaces of blade (a stepladder may be necessary) for any damage which may initiate a fatigue crack. If any damage is detected, the blade should be thoroughly examined by a qualified technician prior to further flight.

If unusual rotor system vibration is detected in flight, land immediately and have blades examined by a qualified technician.



C016-7 MAIN ROTOR BLADE SECTION
(View from the ground looking up)

Appendix 4: Robinson Service Bulletin SB-89

ROBINSON
HELICOPTER COMPANY
2901 Airport Drive, Torrance, California 90505 Phone (310) 539-0508 Fax (310) 539-5198

R44 SERVICE BULLETIN SB-89

(supersedes R44 Safety Alert dated 23 February 2015)

DATE: 30 March 2015

TO: R44 & R44 II Owners, Operators, and Maintenance Personnel

SUBJECT: Main Rotor Blade Modification

ROTORCRAFT AFFECTED: R44 & R44 II Helicopters equipped with C016-7 Revision AE or prior revision main rotor blades.

TIME OF COMPLIANCE: Within next 15 flight hours or by 31 May 2015, whichever occurs first.

BACKGROUND: RHC has received a report of a crack in a C016-7 main rotor blade. The crack initiated at the blade's trailing edge at the corner where the blade chord begins to increase. Although the cause of the crack has not yet been determined, this modification will improve safety margins.

Abrupt shape changes such as corners can concentrate stresses. This bulletin provides instructions to smooth the transition at the chord increase to reduce the stress concentration.

COMPLIANCE PROCEDURE:

NOTE: A short tutorial video for these instructions is available online at www.robinsonheli.com.

For each C016-7 Revision AE or prior main rotor blade:

1. Order one R7769 kit (reference page 6, one kit can perform R44 SB-89 or R66 SB-13 for 5 sets of 2 blades) from RHC Customer Service or acquire supplies locally.

NOTE
R7769-1 tube may be fabricated from smooth, rigid, 4.0 – 5.0 inch diameter tube approximately 3 inches long. R7769-6 slit tubes may be fabricated from A729-32 tubes slit lengthwise.

WARNING
Do not use power tools.

(OVER)

2. Rigidly support both main rotor blades and apply rotor brake, or remove main rotor blades per R44 Maintenance Manual (MM) § 9.111.

CAUTION
When performing modification with blades installed on helicopter, an appropriate work platform is required for the technician.

3. Refer to Figure 1. Clean blade in area shown in figure. Using 10x magnification and a bright light, visually inspect upper and lower blade surfaces and trailing edge in area indicated. Verify no cracks, corrosion, or damage such as nicks which may initiate a crack. If any of the preceding are found, remove blade from service and contact RHC Technical Support.

4. Using a fine-point marker, mark a line at transition to increased chord trailing edge as shown in Figure 1.

NOTE
Perform all filing and sanding in a spanwise direction.

5. Refer to Figure 2. Install R7769-6 slit tubes onto trailing edge to protect areas shown. Use two MS3367-6-0 ty-raps to secure tubes as shown. Remove material in marked, triangular-shaped area by filing. Hand-file spanwise towards blade tip from corner while holding file square to trailing edge. A sharp single-cut, flat file is recommended; clean file frequently to minimize burrs. Remove ty-raps and tubes after filing (tubes may be reused by cutting off area worn by filing).

6. Refer to Figure 3. Wrap R7769-1 tube with 220-grit wet-or-dry aluminum-oxide or silicon-carbide abrasive paper. Holding tube perpendicular to blade trailing edge, sand edge spanwise to create a 2.0 inch minimum inside radius.

7. Finish edge by sanding spanwise using tube wrapped with 320-grit wet-or-dry aluminum-oxide or silicon-carbide abrasive paper. Remove any remaining paint or primer by sanding to expose trailing edge bare metal for minimum 0.5 inch inboard of radius as shown. Also sand a 0.25 inch minimum outside radius at corner shown in Figure 3.

8. Using R7769-1 tube as a gage, verify radius is a minimum of 2.0 inches as shown in Figure 3 (a radius that is too small will leave a gap between tube and trailing edge). Verify minimum chord after rework per MM § 9.141 (9.90 inches minimum inboard of rework).

9. Hold a piece of 320-grit abrasive paper by edges so that tensioned paper wraps over trailing edge. Move tensioned paper spanwise along reworked area four passes to deburr skin edges.

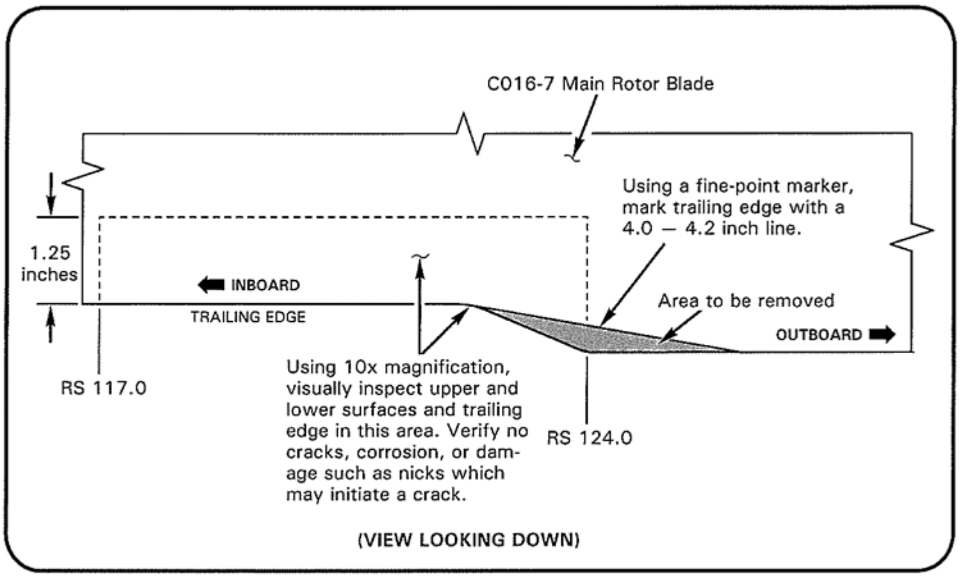


FIGURE 1

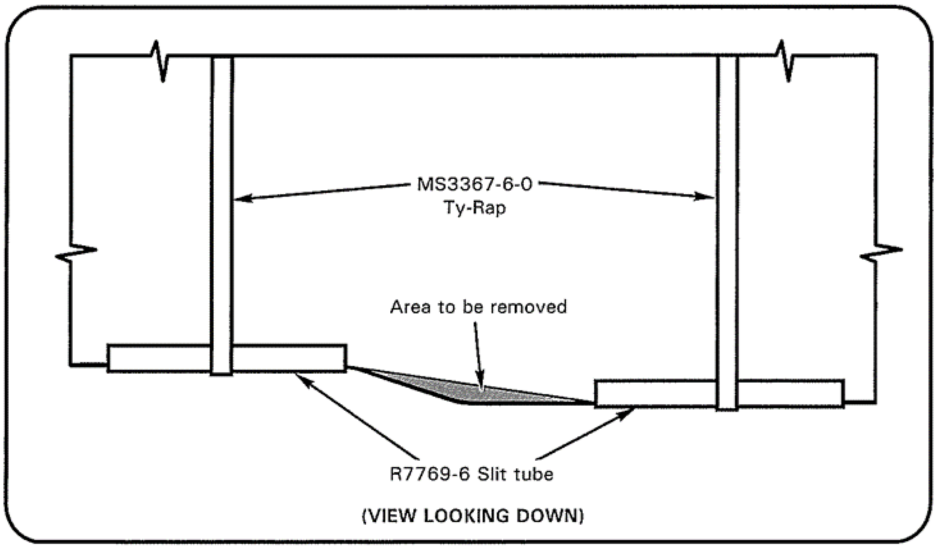


FIGURE 2

(OVER)

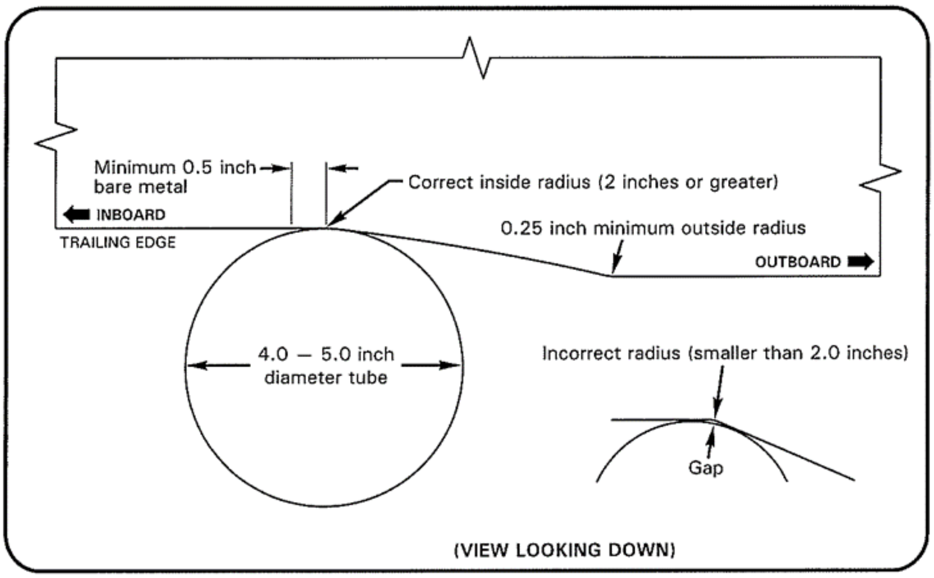


FIGURE 3

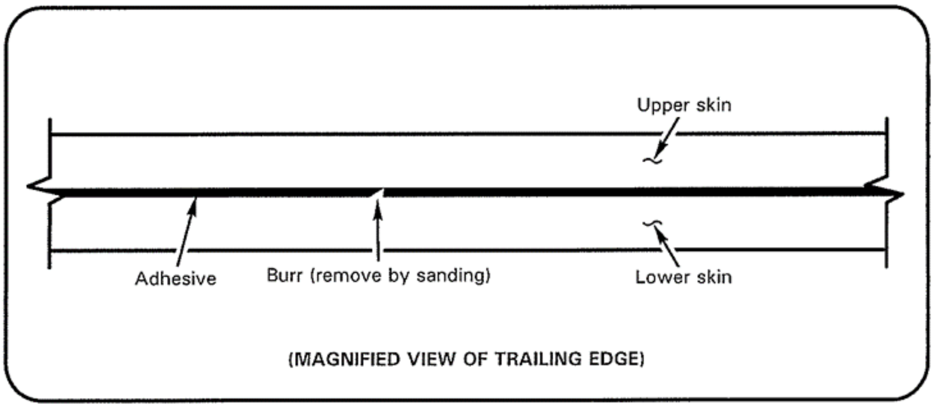


FIGURE 4

10. Refer to Figure 4. Clean bare metal with a lint-free cloth dampened with acetone. Using 10x magnification and a bright light, visually inspect bare metal area and 2.0 inch radius in particular for no sealant, nicks, or residual file marks. Verify no metal burrs protruding into adhesive layer from either upper or lower skin. Verify uniform bare metal finish with only fine, spanwise sanding marks remaining.

11. Tap test trailing edge in reworked area per MM § 9.134.

CAUTION
Review safety data for Alumiprep 33, Alodine 1201, primer, and paint. Wear appropriate protective equipment.

12. Mix Alumiprep 33 liquid and water per manufacturer's instructions and, using swab, apply to trailing edge bare metal for one to three minutes; do not allow Alumiprep mixture to dry. Wipe off Alumiprep mixture with lint-free cloths wet with clean water. Wipe dry.

13. Using swab, apply Alodine 1201 liquid to treated trailing edge bare metal for one to three minutes; do not allow Alodine to dry. Wipe off Alodine liquid with lint-free cloths wet with clean water. Wipe dry. Verify metal exhibits yellow or gold tint; repeat Alodine treatment as required.

CAUTION
Do not apply heat to shorten cure times in following steps. Blade can be damaged.

14. Mix sufficient primer to coat treated trailing edge. Apply thin coat of primer to dry, treated trailing edge and allow to cure for one hour.

15. Mix B270-9 sealant (ref. MM § 1.480) per manufacturer's instructions. Apply thin coat of sealant to dry, primed trailing edge and allow to cure for 12 hours.

16. Inspect sealant. Ensure sealant fully covers primed trailing edge. If required, smooth sealant with 320-grit abrasive paper followed by swabbing area with Alodine 1201 for one to three minutes. Wipe off Alodine liquid with lint-free cloths wet with clean water. Wipe dry.

17. Apply two additional coats of primer to trailing edge and allow to cure for two hours.

18. Paint trailing edge in area affected by work per MM § 9.142.

19. Using contrasting paint or primer, paint a 0.3 – 0.5 inch diameter dot adjacent to and outboard of data plate.

20. If removed, install main rotor blades per MM § 9.111.

21. Make appropriate maintenance entries.

(OVER)

[Note: Page 6 of SB89 contained only the estimated cost of parts and labour to embody the work.]

Safety Notice SN-37

Issued: Dec 01

EXCEEDING APPROVED LIMITATIONS CAN BE FATAL

Many pilots do not understand metal fatigue. Each time a metal component is loaded to a stress level above its fatigue limit, hidden damage occurs within the metal. There is no inspection method which can detect this invisible fatigue damage. The first indication will be a tiny microscopic crack in the metal, often hidden from view. The crack will grow with each repetition of the critical stress until the part suddenly breaks. Crack growth will occur quite rapidly in drive system parts from the high frequency torsional loads. It will also occur rapidly in rotor system components due to the high centrifugal force on the blades and hub. Damaging fatigue cycles occur with every revolution of an overload drive shaft or rotor blade.

If a pilot exceeds the power or airspeed limits on a few occasions without failure, he may be misled into believing he can safely operate at those high loads. Not true. Every second the limitations are exceeded, more stress cycles occur and additional fatigue damage can accumulate within the metal. Eventually, a fatigue crack will begin and grow until a sudden failure occurs. If the pilot is lucky, the part will have reached its approved service life and be replaced before failure. If not, there will likely be a serious or fatal accident.

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 settings.

Appendix 6: Civil Aviation Safety Authority AWB 02-015

Australian Government
Civil Aviation Safety Authority

AIRWORTHINESS BULLETIN

Helicopter - Effects of fatigue on life limited components

AWB 02-015 **Issue** : 1
Date : 19 May 2006

Applicability

All helicopter types and models.

Purpose

The purpose of this bulletin is to advise operators and maintenance organisations that the fatigue lives of life limited components may be adversely affected, or the safety margin reduced, depending on the type of operation and loading history that the component experiences during its service life.

Background

Fatigue becomes especially significant if a helicopter is operated outside what would be considered by the manufacturer (and specified in the Type Certificate) as the operations for which the helicopter was designed and certified. Examples of situations where fatigue lives may be affected are:

- Operation of helicopters in low level flying, agricultural, mustering or other operations where high loads may be encountered more frequently than envisaged by the designer/manufacturer.
- Operations where there are a high number of landings and takeoffs per operating hour.
- Where there is any history of design or flight manual limitations being exceeded, even if only for a short time and on an infrequent basis. Even a one time exceedance is enough to cause damage that may potentially lower the fatigue life of a component.
- Frequent operations at or near maximum all up weight, such as operations involving transport of underslung loads.

Fatigue damage may be caused to components and not be apparent during any routine inspections. The component may still be capable of taking limit loads, but may fail unexpectedly at a lower life than published. In many cases, there will be no warning of an impending fatigue failure, as progression from a small flaw to final fracture can be very fast.

Fatigue damage is accelerated when the cyclic loads in a component are of greater magnitude and frequency than that envisaged during the design of the component. These cyclic loads cause damage to the metal at the microstructural level. There is no easy way to tell the difference between two identical parts that have been subjected to differing fatigue environments, except by analysis under an electron microscope. This is what makes fatigue a hidden danger.

Australian Government
Civil Aviation Safety Authority

AIRWORTHINESS BULLETIN

Helicopter - Effects of fatigue on life limited components	**AWB** 02-015	**Issue** : 1
	Date : 19 May 2006	

Recommendation

It is recommended that all operators and maintainers keep a detailed record of the operational history, beyond the normal requirement of just recording hours & cycles. The types of things to record are:

- Times, dates and durations of any flight manual limit exceedances, no matter how transient – even if they occur only once. The manufacturer should be consulted for advice if a transient overstress or flight limit exceedance occurs. Generally, **follow the manufacturers flight and maintenance manuals for what to do if this occurs**.

- Times, dates and durations of unusual operations, for example - a helicopter goes from being a private helicopter used for personal transport to an agricultural or mustering machine. This change in operations needs to be managed and the risk properly assessed. The manufacturer should be contacted for advice on how the change in operations affects component fatigue lives.

- Times, dates and circumstances of changes in the helicopter's operational environment i.e. long period of storage in hot, humid, salty environments or storage in dry, dusty environments. Long periods of storage in an aggressive environment can sometimes lead to increased corrosion, thereby exacerbating the effects of fatigue cracking if it has already begun. In some cases, corrosion can initiate fatigue cracking in an otherwise serviceable part.

- Operators of aircraft engaged in operations such as agricultural, cattle mustering or operations where more than four rotor full stops per hour are achieved should consult the manufacturer giving full details, including previous history of all operations, for possible life revision.

- Operators should notify the manufacturer if they believe that their helicopter model is experiencing an operational environment that differs from the environment that the helicopter was designed & certified to operate in. The manufacturer may be able to provide advice on how to manage the risk.

Enquiries

Enquiries with regard to the content of this Airworthiness Bulletin should be made via the direct link e-mail address: AirworthinessBulletin@casa.gov.au

Or in writing, to:

Manufacturing, Certification and
New Technologies Office,
GPO Box 2005, Canberra, ACT, 2601

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Recent Aviation Occurrence Reports published by the Transport Accident Investigation Commission (most recent at top of list)

AO-2015-007	Airbus Helicopters AS350BA, ZK-HKU, Collision with terrain, Fox Glacier, 21 November 2015
AO-2017-007	Airbus A320 VH-VGY, Descent below clearance limit, Christchurch, 6 August 2017
AO-2016-007	Collision with terrain, Robinson R44, ZK-HTH, Glenbervie Forest, Northland, 31 October 2016
Interim Report AO-2018-009	MDHI (Hughes) 369D, registration ZK-HOJ, Wanaka, 18 October 2018
Interim Report AO-2018-006	Robinson R44, ZK-HTB, Stevensons Arm, Lake Wanaka, 21 July 2018
AO-2016-008	Robinson R66 helicopter, Partial power loss– forced landing, Hokonui Hills, Southland, 14 November 2016
AO-2015-009	Air traffic control incidents, Hamilton aerodrome,17 December 2015
AO-2017-001	Eurocopter AS350 BA, ZK-HKW, Collision with terrain, Port Hills, Christchurch, 14 February 2017
Interim Report AO-2017-004	Forced landing into Porirua Harbour (Pauatahanui Arm), MBB BK117A-3 Helicopter, ZK-IED, 2 May 2017
Interim AO-2017-009 and AO-2017-010	AO-2017-009: Boeing 787-9, registration ZK-NZE, Trent 1000-J2 engine failure near Auckland, 5 December 2017; and AO-2017-010: Boeing 787-9, registration ZK-NZF, Trent 1000-J2 engine failure, near Auckland, 6 December 2017
AO-2016-006	Eurocopter AS350-B2, ZK-HYY, Collision with terrain during scenic flight, Mount Sale, near Arrowtown, 12 September 2016
AO-2014-005	Eurocopter AS350-B2 (ZK-HYO), collision with terrain, during heli-skiing flight, Mount Alta, near Mount Aspiring National Park, 16 August 2014
AO-2015-005	Unplanned interruption to national air traffic control services, 23 June 2015
AO-2016-004	Guimbal Cabri G2, ZK-IIH, In-flight fire, near Rotorua Aerodrome, 15 April 2016
AO-2015-001	Pacific Aerospace Limited 750XL, ZK-SDT, Engine failure, Lake Taupō, 7 January 2015
AO-2013-010	Aérospatiale AS350B2 'Squirrel', ZK-IMJ, collision with parked helicopter, near Mount Tyndall, Otago, 28 October 2013

Price $26.00

ISSN 1179-9080 (Print)
ISSN 1179-9099 (Online)