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: December 2017

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

Nature of the final report

This final report has not been prepared for the purpose of supporting any criminal, civil or regulatory action against any person or agency. The Transport Accident Investigation Commission Act 1990 makes this final report inadmissible as evidence in any proceedings with the exception of a Coroner's inquest.

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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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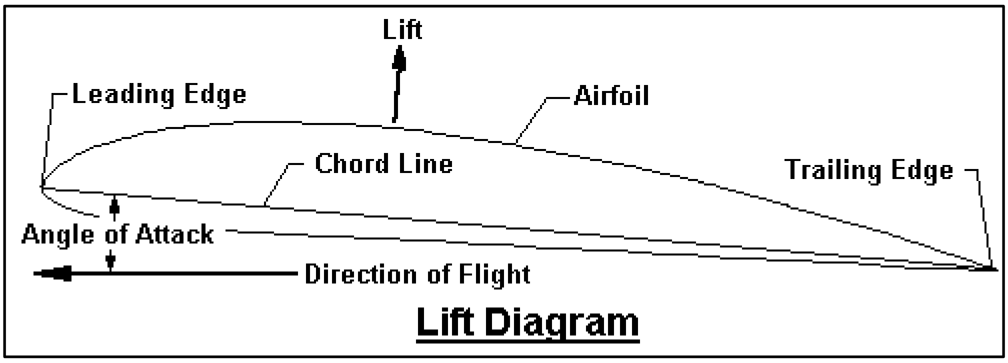
Abbreviations

AD	Airworthiness Directive
CAA	Civil Aviation Authority of New Zealand
Commission	Transport Accident Investigation Commission
FAA	Federal Aviation Administration (United States)
ft	feet
" Hg	inches of mercury
NTSB	National Transportation Safety Board (United States)
Robinson	Robinson Helicopter Company
SB	Service Bulletin
SL	Service Letter

Glossary

avoid curve	the boundary of height and airspeed combinations, inside of which a safe autorotation may not be possible in the event of an engine failure
chord or chord line	a straight line joining the centres of the leading and trailing edges of an aerofoil section, such as a wing or rotor blade (see Figure below)
design life	the period of time during which the 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 hours, equivalent to 1.85 kilometres per hour
safe-life design	safe-life products are designed to survive a specific design life with a chosen reserve
service life	in the case of Robinson safety-critical components, they are fatigue life-limited to a service life that is factored to be well below its design life
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 (consisting of a conductive metallic strip that changes its electrical resistance when stretched)
transitional flight	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
translational lift	additional lift generated by a main rotor system once the helicopter's airspeed increases beyond a certain point (normally about 15 knots)



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 Ltd
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:	1630 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. While spraying gorse near Waikaia on 23 January 2015, the pilot of a Robinson R44 helicopter felt an unusual and significant vibration. He landed immediately and discovered a large crack in the lower skin of one of the main rotor blades.

1.2. The crack's origin contained features characteristic of metal fatigue. The crack had started in a radius in the blade trailing edge, known as the 'chord length transition'. Flight testing by the manufacturer found that the stress in this area was higher than had been thought.

1.3. As a result of this incident, the manufacturer developed a modification for main rotor blades in service, and made design changes to new main rotor blades for R44 (and R66) helicopters.

1.4. The helicopter had been operated primarily for agricultural flying, usually at or over the maximum power settings. The use of a 'flick turn' while operating the helicopter close to the maximum all up weight very likely subjected the main rotor blades to additional high 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. Two safety issues were identified during the inquiry. The first is that aircraft design organisations do not currently have to consider whether proposed major modifications to aircraft will significantly alter the use of the aircraft or will adversely affect the service life of any component. The second is the routine exceedance by operators and pilots of published limitations for aircraft, an issue that has been previously raised by the Commission.

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

> consult the original equipment manufacturer when considering a modification or supplemental type certificate, which, 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 lives 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 very 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 flight manual weight, speed and power limits. The Commission has noted in previous inquiry reports that operating an aircraft outside the flight manual limitations significantly erodes the safety margins factored into the service lives of components and can quickly lead to an early catastrophic failure
- aggressive and unusual helicopter manoeuvres, such as flick turns during agricultural operations, can prolong the period spent inside the 'avoid curve', where a safe autorotation may not be possible in the event of an engine failure. Operators should require their pilots to fly in accordance with established industry guidelines and to continually demonstrate their commitment to a strong safety culture
- 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 reduced intervals for component inspections and earlier component replacement times.

2. Conduct of the inquiry

2.1. The 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 R44 accident 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, the 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 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 the NTSB and the FAA. 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. A set of time-expired main rotor blades were inspected by the Royal New Zealand Air Force, using radiography and eddy current non-destructive testing, 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 to look 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. Submissions were received from five persons and these were considered in the preparation of the final report.

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

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 airspeed. When he pulled out of the turn at an airspeed of 10-15 knots, the pilot felt an unusual and severe vibration, but he had no difficulty in maintaining control.

3.1.3. At the onset of the vibration the pilot was about 200 metres from the landing site so he 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. He inspected the helicopter and noticed a large crack on the lower surface of one of the main rotor blades, extending from the trailing edge to just behind the 'D spar' 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[2] of the blade increased, known as the 'chord transition[3]' (see Figures 2 and 3).

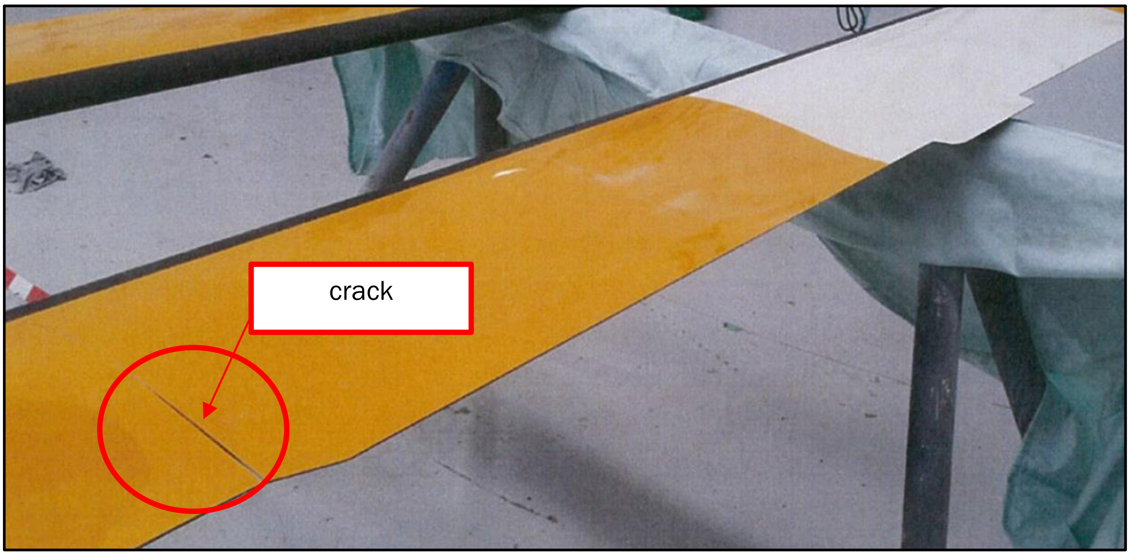


Figure 1
Cracked blade

[2] The distance between the trailing edge and leading edge of an aerofoil.

[3] The chord length increases, from 10 to 10.7 inches, outboard of a point at approximately 120 inches from the centreline of the main rotor driveshaft. This design feature takes advantage of the increased engine power of the R44 Raven II. The larger blade area generates more lift and enables the helicopter to operate at a higher gross weight.

Figure 2
Blade crack at chord transition

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 the similar section from the opposite blade. A detailed laboratory examination was carried out and a report was provided to the Commission (see Appendix 1).

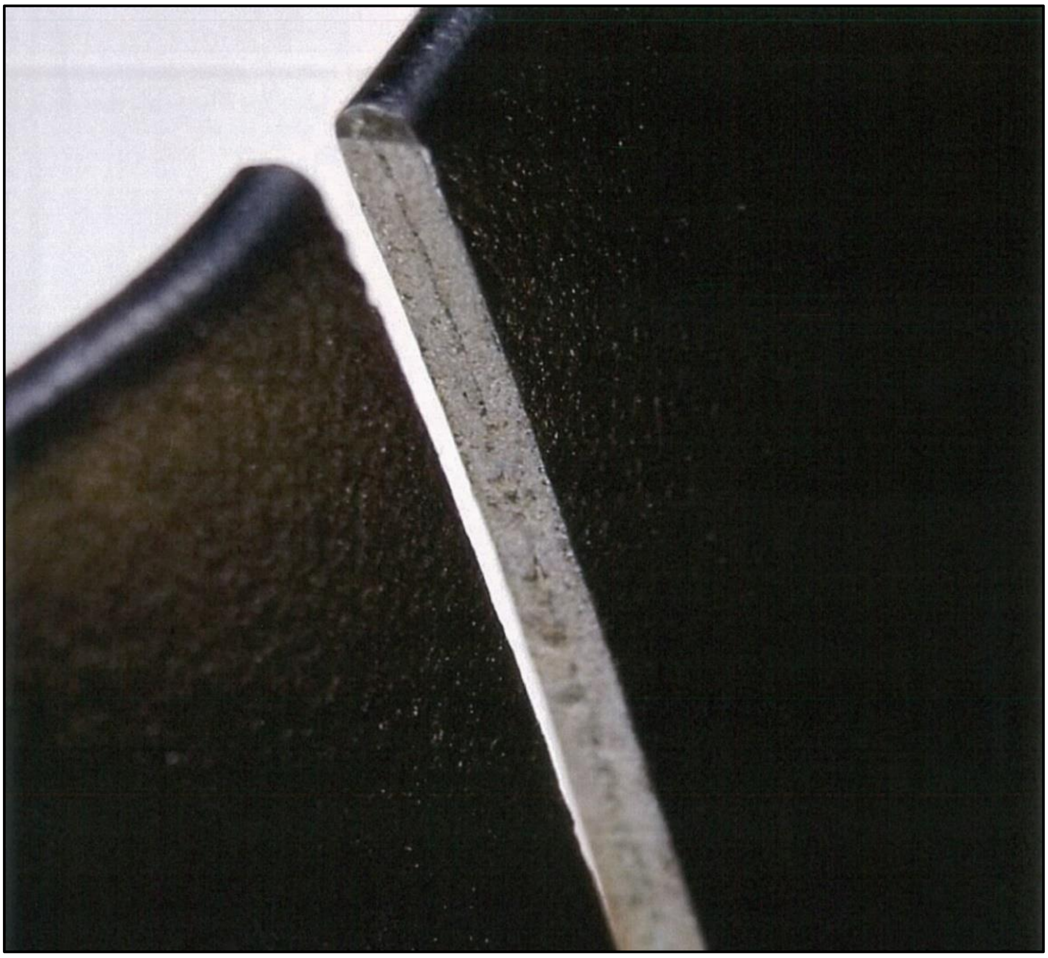


Figure 3
Closer view of crack from underside of trailing edge

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 to design specifications.

3.1.8. For comparison, the Royal New Zealand Air Force[4] checked two C016-7 main rotor blades that had been used exclusively on agricultural operations and had reached their service life of 2,200 hours. The blades were subjected to eddy current[5] 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 them since September 2012. He held a commercial pilot's licence (helicopter) first issued in August 2010, and had logged a total of 1,630 flight hours, of which 1,173 hours were on the R44. The pilot's last line check was conducted in December 2013, and his last flight crew competency check was in February 2014. He held a current Class 1 medical certificate.

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[4] The Air Force had expertise in the non-destructive inspection of helicopter main rotor blades.
[5] Eddy current inspection is 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[6] 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[7] 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 was comprised of two spray booms, two tanks 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,[8] but 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'. The blade's trailing edge was formed by bonding the upper and lower skins with structural epoxy adhesive. A stainless steel doubler was bonded between the upper and lower skins at the trailing edge (see Figures 4 and 5). The doubler terminated 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.

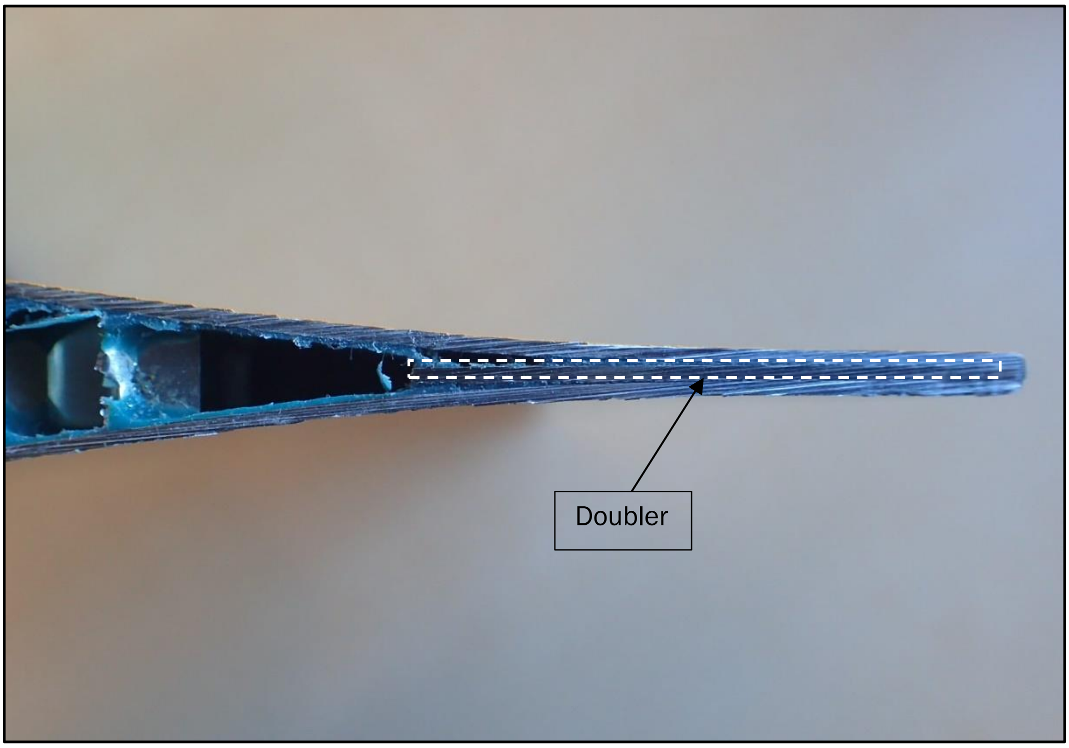

Figure 4
Cross-section of blade trailing edge, with doubler highlighted

[6] The limiting weight specified by the manufacturer and not to be exceeded.
[7] The spray system had been designed by an approved New Zealand aircraft design organisation.
[8] The tank was designed to be much larger than the actual volume of liquid that could legally be carried, to allow for the 'foaming' of the liquid to be contained inside the tank.

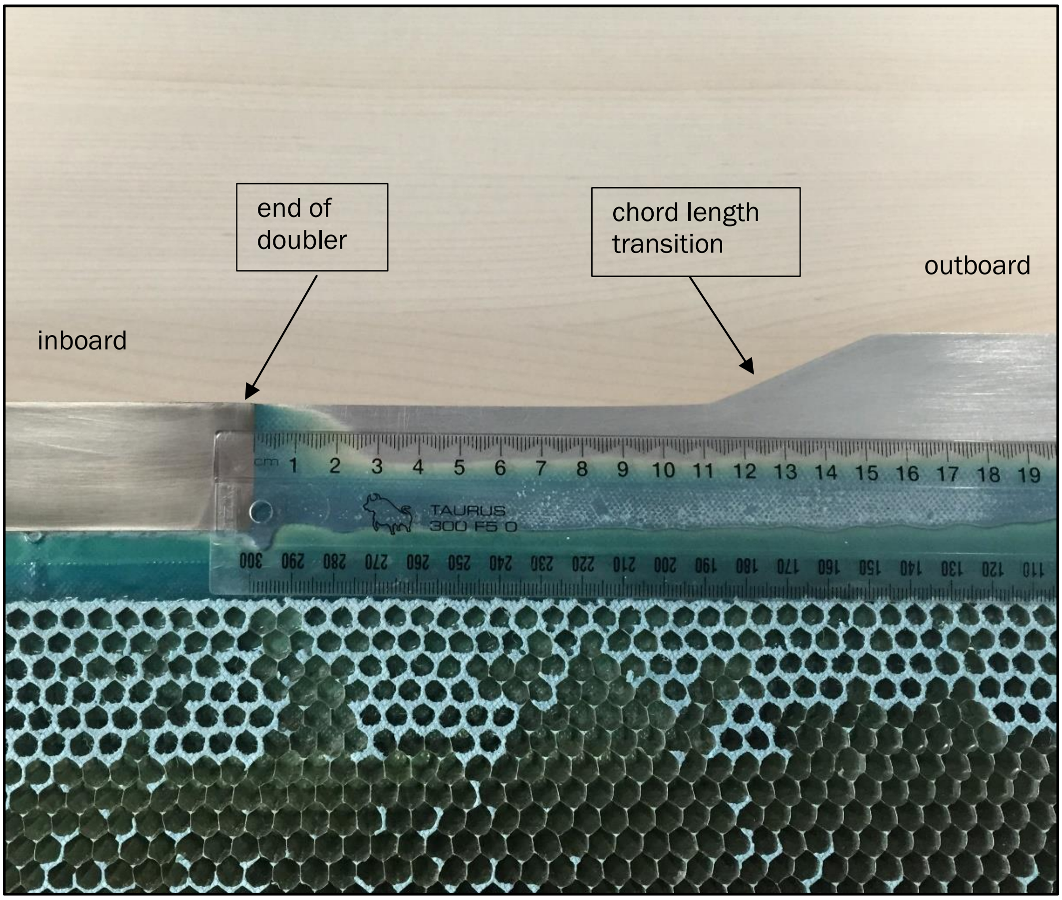

Figure 5
View of C016-7 blade with one skin removed, showing internal doubler

3.3.4. The main rotor blades were 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 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 C016-5 main rotor blades with stainless steel skins. The blade planform[9] for the later model R44 Raven II was changed to increase the chord length of the outer section of the blade. The increased blade area produced more lift and allowed operations at heavier gross weights. The point where the chord length increased was called the chord transition. All R44 blades incorporated a stainless steel trailing edge doubler that terminated inboard of the chord transition (see Figure 5).

3.3.6. On 3 January 2008, prompted by reports of de-bonding[10] on blades with stainless steel skins, the FAA issued Airworthiness Directive[11] (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 2007-26-12 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 was found during the pilot check, blades were

[9] Planform is the shape of an object as seen from above.
[10] An undesirable condition where the structural adhesive under the blade skin breaks down or fails, as a result of surface erosion, corrosion or overstressing.
[11] A written directive issued by a national airworthiness authority to correct an unsafe condition. Compliance with these airworthiness instructions is mandatory.

to be inspected for corrosion, separation, gaps or dents by following certain procedures in Robinson's Service Bulletins (SBs).[12]

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. Embodiment of this AD removed the inspection requirements of AD 2011-12-10. Robinson exchanged discounted new C016-7 blades for part-life stainless steel blades.

Actions after the incident

3.3.10. On 31 January 2015, the CAA issued Continuing Airworthiness Notice[13] 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 at every pre-flight inspection for defects.

3.3.11. On 23 February 2015, Robinson issued a Safety Alert[14] (see Appendix 2), 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 3). 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 by this revision to the AD.

3.3.15. Robinson changed to the C016-7 production design. 'Revision AF' (25 March 2015) 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, to overlap the trailing edge doubler (see Figure 6).

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[12] A written service instruction issued by the manufacturer. Compliance is mandatory in New Zealand if the issuing authority requires it, or if the AD is included in operator certification requirements.

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

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

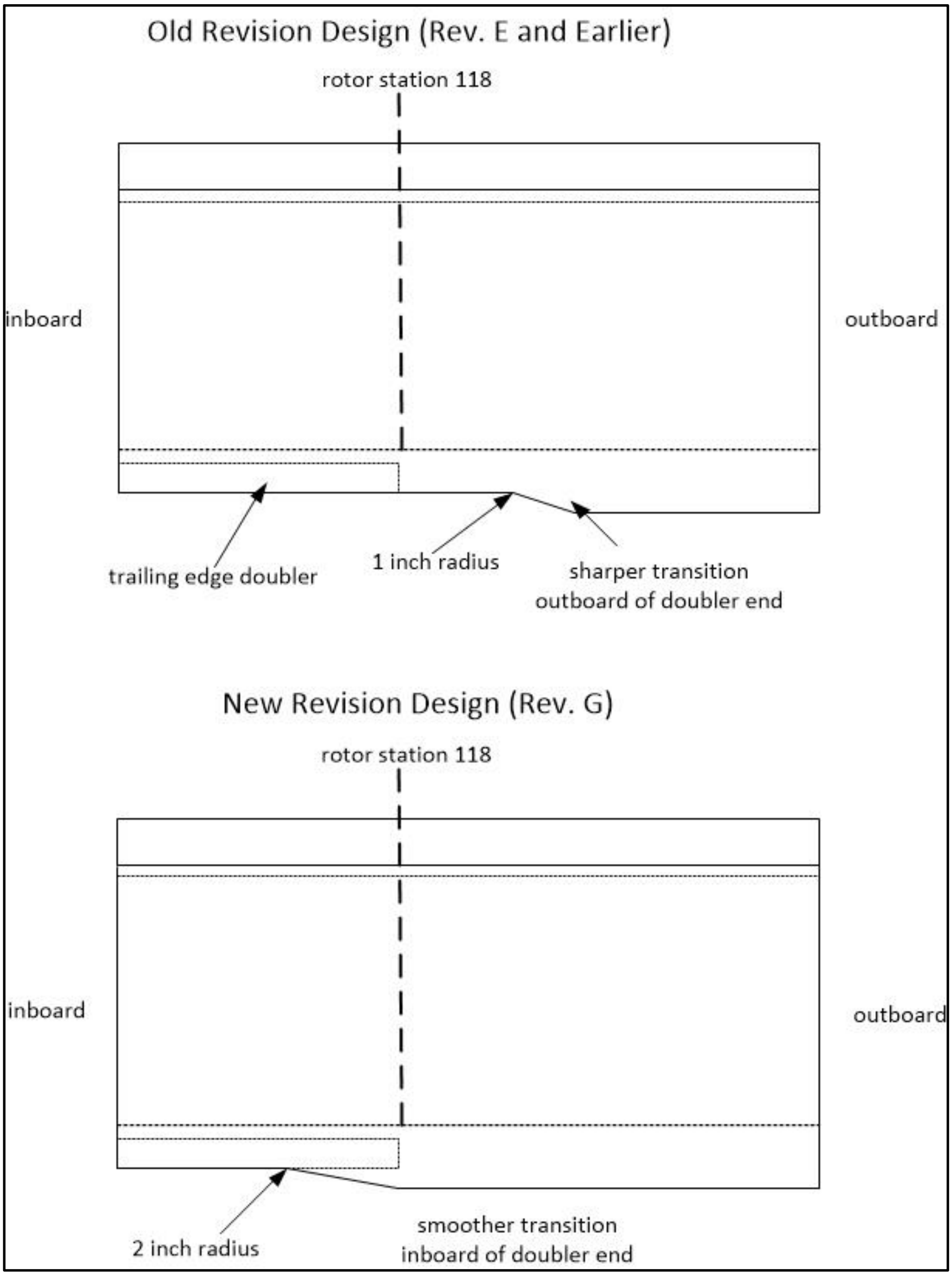

Figure 6
Design changes between C016-7 'Revision AE' and 'Revision AG' blades

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, they did not initially consider the New Zealand incident to have been 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 reevaluated the situation. It determined 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 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. 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, such as 'hammerhead turns[15]' and 'procedure turns,[16]' and these were conducted at high all up weights but without spray equipment installed. The results of these flights are discussed in paragraph 4.15.

3.4.2. A review of statistical data pertaining to R44 agricultural helicopter operations in New Zealand for the years 2014 and 2015 was conducted, to determine the average loads and cycles per hour for R44 helicopters used for spraying. The data was provided by the CAA from statistical information required to be reported by all aircraft operators in New Zealand, and from the quarterly operational statistics[17] submitted by agricultural operators. The data included total weights and volumes of products applied, load cycles and the breakdown of flight hours.

3.5. Organisational and management information

3.5.1. The company had operated Robinson helicopters since 2006 and 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 was one of two company pilots who 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 that could be carried, depending on the weight of the pilot and the specific gravity[18] 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 with the maximum payload permissible,[19] based on the operating weight of the helicopter[20] and the specific gravity of the liquid.

3.5.3. Both pilots described using a manoeuvre that they referred to as a 'flick turn' to turn the helicopter quickly at the end of each spray run. The manoeuvre was observed during the course of the investigation. The turn involved the tail being flicked around in a 180° pedal turn[21] at about 30 knots, with the helicopter continuing to climb sideways then backwards until the apex of the turn. Once the helicopter stopped climbing, it was dived towards the ground and then pulled out of the dive to recommence spraying. The operator estimated that

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[15] A hammerhead turn is where a helicopter is climbed at a steep angle, normally about 45° until the airspeed decreases to zero, and then the helicopter is made to spin around the main rotor driveshaft using the tail rotor pedals so it is facing back towards the ground. It is similar to a stall turn in a fixed-wing aircraft.
[16] A cyclic procedure turn is where the helicopter is turned around 180° in a relatively wide flat arc using the cyclic flight control only. These turns involve minimal added stresses on the dynamic components of the helicopter.
[17] Quarterly operational statistics are required to be submitted by Part 137 Agricultural operators, and the data includes the amount of products applied and the area covered, and is not normally checked for accuracy.
[18] Specific gravity is the weight of a substance relative to water, i.e. a specific gravity of 1 is the same as the weight of water, or 1.5 would be one-and-a-half times the weight of water.
[19] The maximum payload was the difference between the operating weight and the maximum all-up weight.
[20] Operating weight was the empty weight of the helicopter with spray system installed, plus the weight of the pilot and the fuel onboard.
[21] A pedal turn is where only the tail rotor pedals are used to 'spin' the helicopter around its vertical axis (its main rotor driveshaft).

there was about a '2G'[22] pull-up at the bottom of the dive. Generally, there would be 10- 20 of these turns per load, on average, and up to 30-40 for some loads.

3.5.4. The operator estimated that on average the helicopter accrued 10-12 take-offs[23] per hour, and sometimes up to 25 per hour. The pilots stated that during take-off full power or 'red line' (26.1" Hg[24]) was always demanded from the engine, with about 23-24 inches of manifold pressure used during the flick turns and about 20 inches for the spray runs.

3.6. Additional information

Subsequent clutch shaft failure on incident helicopter

3.6.1. On 25 November 2015, a different pilot flying the helicopter for the operator felt a sudden and high frequency vibration accompanied by the clutch[25] light flickering rapidly. He landed immediately and shut down the helicopter. A significant crack was found in the clutch shaft, which was removed and sent to a metallurgy specialist for closer examination.

3.6.2. The metallurgist determined the crack was a torsional fatigue fracture,[26] and only 13% of the shaft's circumference remained intact. The fracture had initiated aft of two bolts used to attach the clutch shaft to the main rotor system (see Figure 7).

Figure 7
Cracked clutch shaft

3.6.3. The metallurgist found two maintenance factors that had contributed to the shaft failure: one of two attachment bolts was below its nominal torque value, and an incorrect jointing compound had been used. The CAA subsequently issued Airworthiness Directive DCA/R44/33, which stated, in part, that the torque on the bolt probably reduced due to the

[22] A 2G pull-up is when an aircraft is subjected to twice the force of gravity while pulling out of a dive, and the effect is the same as weighing twice as much as it did before the pull-up manoeuvre.
[23] Normally a flight cycle includes a shut down of the engine, but during spraying the engine is left running between loads.
[24] Pressure in the engine induction manifold is measured in inches of mercury (Hg) and is regulated by the throttle setting. The manifold pressure indicates the power setting of the engine.
[25] The clutch connects the engine output driveshaft to the main rotor gearbox drive input, and like a car allows some slippage before fully engaging.
[26] A torsional fatigue fracture is a fatigue failure caused by torsional (twisting) forces.

breakdown of the incorrect jointing compound. The CAA also required all R44s in New Zealand to be inspected to ensure the correct jointing compound was used and that there were no cracks or under-torqued bolts.

3.6.4. The clutch shaft was installed on the helicopter on 22 November 2011 and had accrued 1,844.7 hours' time in service. The finite life of the shaft is 2,200 hours.

Safety information

3.6.5. 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.

3.6.6. In December 2001, Robinson issued Safety Notice SN-37 – Exceeding Approved Limitations Can Be Fatal (see Appendix 4). 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 aircraft flight manual carried in each helicopter.

3.6.7. The March/April 2005 issue of *Vector* magazine[27] 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 R44, including the main rotor designs, meant that this article was relevant to operators of R44 helicopters in the agricultural role.

3.6.8. In May 2006, the Civil Aviation Safety Authority of Australia 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 lives 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 receives in its life' (see Appendix 5).

3.6.9. 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 should consult the manufacturer, and give full details of all operations, for a possible component life-limit revision.

Other incident

3.6.10. In December 2016, a pilot of an R22 in Australia experienced unusual vibrations while conducting aerial stock mustering in Queensland. The pilot landed the helicopter immediately, and after shutting down the helicopter he discovered a crack in one of the main rotor blades. The crack was similar to the one found on ZK-HPC, except its location was further inboard from the chord transition area. The investigation by the Australian Transport Safety Bureau (ATSB) was ongoing at the time of this report's publication.

[27] *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 extremely rare, but are usually catastrophic for those on board the helicopter. In this case, the failure occurred at very low level and near a suitable landing site, which enabled the pilot to land safely.

4.1.2. The failed blade had accumulated only 847 hours in service, well below the 2,200 hours service life. The most common causes of rotor blade failure have been material or manufacturing defects, although overloading and high stress and vibrations have also led to fatigue cracking.

4.1.3. This analysis describes the metallurgical tests performed on the failed blade, and examines the nature of the agricultural flying performed by the operator. 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 were not envisaged by the helicopter manufacturer. The analysis discusses this safety issue. Aircraft design organisations do not currently have to consider whether proposed major modifications to aircraft will significantly alter the use of the aircraft or adversely affect the service lives of any of their components.

4.1.5. The analysis also discusses the safety issue of operators and pilots routinely exceeding published limitations for aircraft, an issue that has been raised in a previous Commission report.

4.2. NTSB Material Laboratory testing

4.2.1. The NTSB laboratory examination determined that the crack initiation site contained features consistent with fatigue failure, and that its point of origin was at the trailing edges of the upper and lower skins, where they were bonded together (see Figure 8). The portion of the fracture surface consistent with fatigue cracking was 3.65 inches in length and exhibited features typical of progressive crack growth. The remainder of the fracture exhibited mixed-mode[28] cracking, then a pure overstress failure (see Figure 9).

[28] A combination of overstress and fatigue cracking.

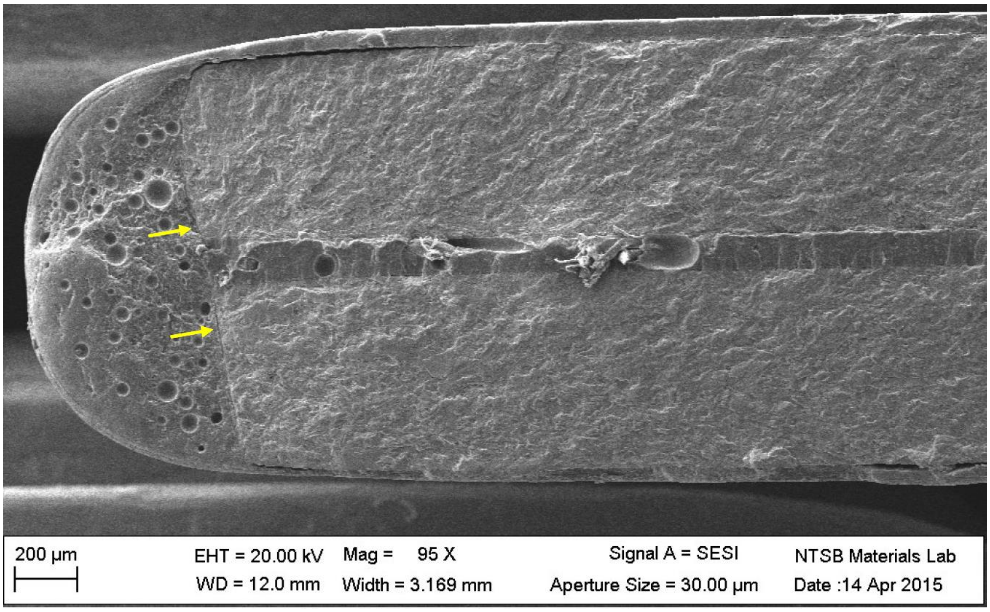

Figure 8
Crack origin

4.2.2. The 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 is designed to flex with it. However, if the blade was to flex or bend significantly in service under cyclic loads, it is possible the epoxy could have cracked first, creating a further stress concentration at that location.

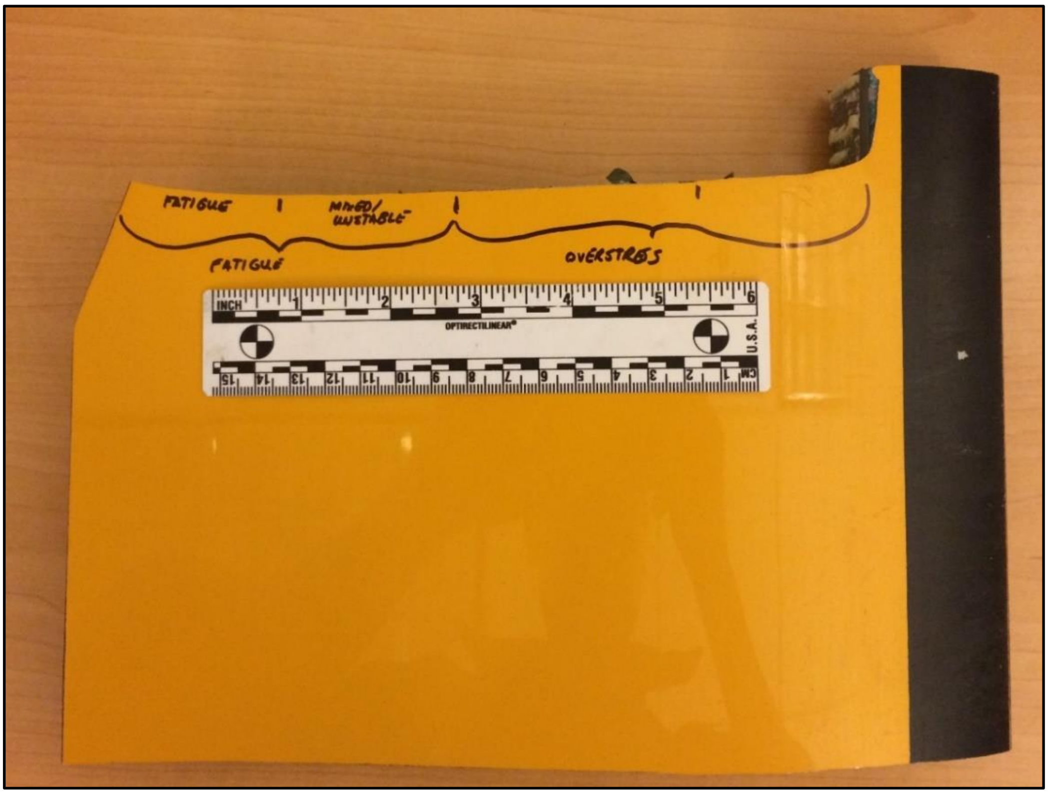

Figure 9
Upper surface of blade with annotations of crack failure mode regions

Finding:

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

4.3. Operational factors

Overloading

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

4.3.2. A comparison of the operational statistical returns[30] for the operator with 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.

4.3.3. However, there was also a torsional fatigue failure of the helicopter's clutch shaft 10 months after the main rotor blade failed. The premature failure of two critical dynamic components on the same helicopter cannot be disregarded. The failures supported the proposition that, through pilot handling and operating practices, the dynamic components[31] of the helicopter had been consistently subjected to excessive stresses which encroached into the safety margins, eventually resulting in the two fatigue failures. It was noted that maintenance practices likely contributed to the premature clutch shaft fatigue failure.[32]

Power settings

4.3.4. The operator stated that full power or 'red line' manifold pressure was set for every take-off, which equated to 26.1 inches of mercury ("Hg). However, the flight manual contained the following table for maximum continuous and maximum take-off power settings under different ambient conditions (see Figure 10):

[29] 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.
[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.
[31] Dynamic components in helicopters include all the rotating parts that transmit the engine's power to the main and tail rotors, i.e. the clutch shaft, reduction gearbox, drive shafts and rotor systems.
[32] See paragraph 3.6.3.

LIMIT MANIFOLD PRESSURE - IN. HG								
MAXIMUM CONTINUOUS POWER								
PRESS ALT-FT	OAT - °C							
	-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 10
Engine power limits

4.3.5. The operator was based in the lower South Island where temperatures are relatively low, and they also sprayed a number of high country farms near its base at Waikaia. The maximum ('red line') limit for take-off of 26.1"Hg is based on a temperature of 40°C at sea level (from Figure 10, 23.3"Hg plus 2.8"Hg for maximum take-off power). The operator would never be flying under these ambient conditions and would more likely be flying in conditions between 0°C and 20°C and between sea level (SL) and 2000 feet. These conditions would mean a range of 24.6" to 25.7"Hg, and a median figure of 25.1"Hg for maximum take-off power. The pilots' statements about using red line every time suggested they had not observed the flight manual limits for maximum take-off power for the engine.

4.3.6. The two pilots who had carried out flick turns during spraying stated that the power required during these turns was 23-24"Hg, which exceeded the maximum continuous power setting, and was near the maximum take-off power under certain ambient conditions. It is likely that the engine was run at high power settings for most of time the helicopter was spraying, and that very high power settings were used for both take-off and during the flick turns.

4.3.7. Robinson had set limits on the engine manifold pressure and the engine had been 'derated'[33] from its maximum power output, although 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 dynamic components if a manifold pressure (power setting) higher than the limit for ambient conditions was consistently used in service.

4.3.8. The integrity of components will be compromised early in their service lives if they are subjected to consistently excessive levels of stress, particularly when a stress concentration exists within the component. The accumulation of fatigue can be rapid and could reach the point of failure before early detection was possible by visual means. The premature failure of the clutch shaft and the main rotor blade was typical of the manner in which operational overstress can compromise the integrity of dynamic components. Robinson Safety Notice 37 (see Appendix 4), 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 lives of such components.

Flick turns

4.3.9. The in-flight strain survey carried out by Robinson included manoeuvres commonly used during agricultural flying. Spray turns, such as 'hammerhead turns', with cyclic and collective pull-ups of low-to-moderate severity, were performed during the flight testing. A direct comparison with the incident pilot's technique was limited, because Robinson did not perform

[33] 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 more aggressive flick turn manoeuvre during their testing. From the flight strain survey, Robinson found that:

> 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.

4.3.10. According to industry feedback, the flick turn is not widely used during spraying operations, and was not taught during basic agricultural flight training. A characteristic of this type of turn is that the helicopter reverses the direction in which it is heading while it is climbing. This means that the heading of the helicopter during the climb is not aligned with the flight path. The flick turn is used to point the nose toward the ground soon after the end of the spray run and enables the pilot to quickly start spraying again.

4.3.11. During the flick turn, the helicopter changes from flying forwards, to flying sideways, to flying backwards, and then translating to flying forward again. Therefore, the flick turn occurs for the most part in 'transitional flight'.[34] According to a technical report that studied helicopter loads, fatigue and design, published by the Australian Department of Defence Science and Technology Organisation 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.

4.3.12. For a considerable period of time during the flick turn there is reduced translational lift[35] as the helicopter airspeed reduces and the flight path reverses. The flick turn differs from a conventional 'teardrop turn' where airspeed and therefore translational lift is conserved, and from the 'hammerhead turn' where no sideways or backwards flight is involved.

4.3.13. If the helicopter is heavily loaded when the flick turn is performed, and where there is a period of reduced translational lift, the stresses on the main rotor blades during transitional flight could be significant. Additionally, there is an increased 'G' load (vertical acceleration) associated with the more abrupt pull-up from the dive at a low height above ground, resulting in greater stresses on the rotor.

4.3.14. In regard to the consequences of this abrupt manoeuvring, Robinson suggested:

> 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[36] in fatigue life.

4.3.15. 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 the flick turn requires significant increases in blade pitch angles (angles of attack) and engine power, along with an increase in G 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. Due to the additional stresses imparted by the flick turns, it is likely that each of these turns results in a significant fatigue penalty or stress cycle.

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[34] 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.

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

[36] A reduction by a factor of 10.

Exceeding aircraft limits

Safety Issue: There may be a culture in New Zealand of some pilots operating their aircraft outside the manufacturers' published and placarded 'never exceed' limitations. Should this situation exist, there is a possibility that such a culture has become normalised.

4.3.16. The operator's adoption of a flight manoeuvre that was not considered normal and their use of power settings that were likely at times to have been above the maximum permitted were signs that a culture existed of routinely exceeding an aircraft's published limitations.

4.3.17. The Commission has identified a safety issue in a previous inquiry report[37] that some New Zealand helicopter pilots may have a culture of operating their aircraft beyond the manufacturers' published and placarded limits, with the possibility that such a culture has become normalised.

4.3.18. In that report the Commission noted that New Zealand's helicopter accident rate is higher than that of other aviation sectors, and that 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.

4.3.19. The Commission was aware that the CAA was reviewing the 'sector risk profile' of commercial helicopter and small aeroplane operations, and has already recommended that the Director of Civil Aviation include the safety issue of helicopter operational culture in that review.

Finding:

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

3. The frequent use of the flick turn while operating the helicopter at heavy weights likely subjected the main rotor blades to stresses above those for which the blades were designed, and accelerated the 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 drawn up 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'[38] was equivalent to one 'fatigue cycle' on the blade. This definition was used in its calculations to forecast component reliability, determine a 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

[37] Report AO-2014-005, Eurocopter AS350-B2, ZK-HYO, collision with terrain, Mount Alta, 16 August 2014.
[38] One cycle is taken from the rotors starting to rotate until they stop rotating after engine shut-down.

fatigue spectrum[39] and the stress-cycle curve[40] for the blade to produce the blade's fatigue life[41] 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.[42]

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 flick turns were conducted during the application of each load. Robinson submitted that the fatigue damage depended on the magnitude of the stress cycle, and the damage associated with a flick turn was significantly less than a ground-air-ground cycle. However, stress cycles are accumulated more quickly during agricultural flying than during a 'standard' flight profile.

4.4.4. The type of flight operation performed affects the fatigue life of helicopter components. A Civil Aviation Safety Authority of Australia bulletin (see Appendix 5) describes how component fatigue lives 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.

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 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 the safe life of 2,200 hours would not need to be reduced because an adequate safety margin 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. A Lycoming 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 Lycoming because of the increased stresses during high power settings at heavier weights, and the associated reduction in the expected service life of the engine's components.

Finding:

4. 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.

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[39] 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.

[40] 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.

[41] Fatigue life is 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.

[42] 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 the component will not fail in service.

4.5. Design factors

4.5.1. Although this incident is the only recorded instance of fatigue cracking in the chord length transition radius of a C016-7 main rotor blade, the safety actions and design changes made as a result have 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 SB-89, which is an interim measure until the blades are replaced. The full benefits of the design improvements can only be achieved by replacement with 'Revision AG' blades. However, Robinson advised that SB-89 restores the calculated fatigue life for early revision blades to that of the new 'Revision AG' blade.

4.5.2. Blade 'Revision AG' also moved the chord length transition inboard of the 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).

> **Finding:**
>
> 5. 'Revision AE' and earlier C016-7 main rotor blades had a small radius in the trailing edge chord length transition. This created a critical stress concentration and reduced the safety margin for fatigue cracking in this area.

4.6. Major aircraft modifications and role changes

Safety issue: Aircraft design organisations do not currently have to consider whether proposed major modifications to aircraft will significantly alter the use of the aircraft or will 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. An aircraft design organisation holds a delegation from the CAA to approve design changes and aircraft modifications. CAR Part 21 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 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 help 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. Flight testing of an aircraft with a modification installed is normally carried out to determine whether there is any change to the aircraft's flight characteristics or performance, but in-flight stresses on the aircraft's 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[43] 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 lower service 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 that the modified helicopter would be subjected to 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 reduction in 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 no input into 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 that the aircraft was not originally designed for.

4.6.8. National aviaton authorities, like the CAA, which 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 he consult the original equipment manufacturer when considering a modification or supplemental type certificate which, if approved, could result in the 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 lives and the aircraft maintenance programme.

Finding:

6. 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.

[43] A strain gauge is 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.

5. Findings

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

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

5.3. The frequent use of the flick turn while operating the helicopter at heavy weights likely subjected the main rotor blades to stresses above those for which the blades were designed, and accelerated the fatigue at the chord transition radius.

5.4. 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.

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

5.6. 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 issues

6.1. There may be a culture in New Zealand of some pilots operating their aircraft outside the manufacturers' published and placarded 'never exceed' limitations. Should this situation exist, there is a possibility that such a culture has become normalised

6.2. Aircraft design organisations do not currently have to consider whether proposed major modifications to aircraft will significantly alter the use of the aircraft or adversely affect the service life of any components.

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; and

(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 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 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 removal from service or an alteration of the inboard chord transition radius. This has 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. Nil.

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 these recommendations are 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.[44]

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.

New recommendations

8.6. 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 those are also used for carrying passengers commercially.

8.7. 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 life of aircraft components. Without recognition of and due allowance for such adverse effects, the continuing airworthiness of the aircraft might be compromised. That is a potential safety issue, particularly for modified aircraft that also carry passengers in air operations.

8.8. 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 the aircraft and their components. The reluctance to share data is based primarily on legal considerations. However, aircraft manaufacturers are likely to agree to requests from national aviation authorities, such as the CAA, for relevant data.

[44] Commission recommendation 032/17.

8.9. **On 14 December 2017, the Commission recommended to the Director of Civil Aviation that he consult with the original equipment manufacturer when considering a modification or supplemental type certificate which, 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 lives and the aircraft maintenance programme (036/17).**

On 10th January 2018, the Civil Aviation Authority replied:

> In considering the draft recommendation, the CAA refers the Commission to the Directors 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 that originally intended.

9. Key lessons

9.1. Metal fatigue occurs continuously in dynamic components. A fatigue crack can lengthen very 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 flight manual weight, speed and power limits. The Commission has noted in previous inquiry reports that operating an aircraft outside the flight manual limitations significantly erodes the safety margins factored into the service lives of components and can quickly lead to an early catastrophic failure.

9.3. Aggressive and unusual helicopter manoeuvres, such as flick turns during agricultural operations, can prolong the period spent in the 'avoid curve', where a safe autorotation may not be possible in the event of an engine failure. Operators should require their pilots to fly in accordance with established industry guidelines and to continually demonstrate their commitment to a strong safety culture.

9.4. 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 reduced 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

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R44 SAFETY ALERT

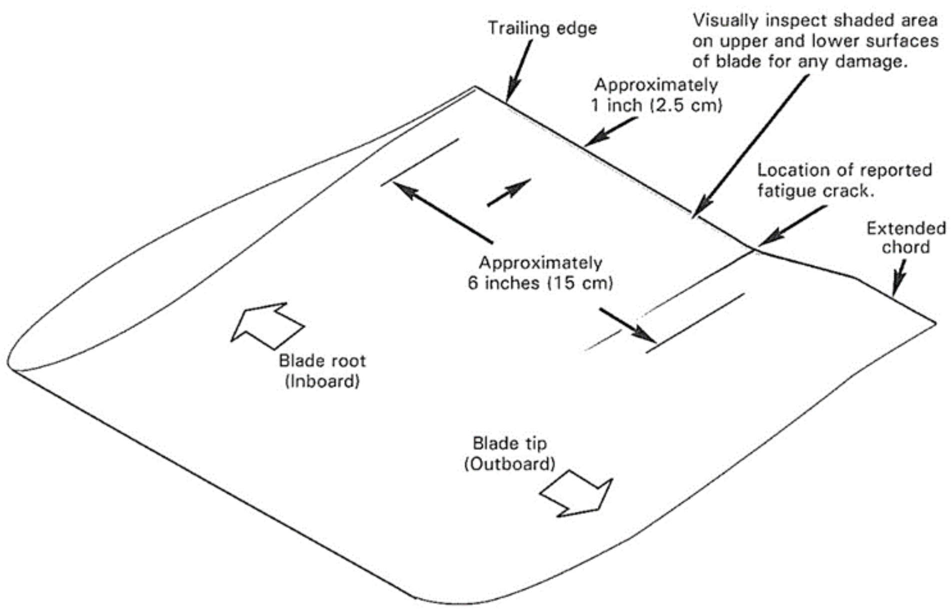
Issued: 23 February 2015

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 3: 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.

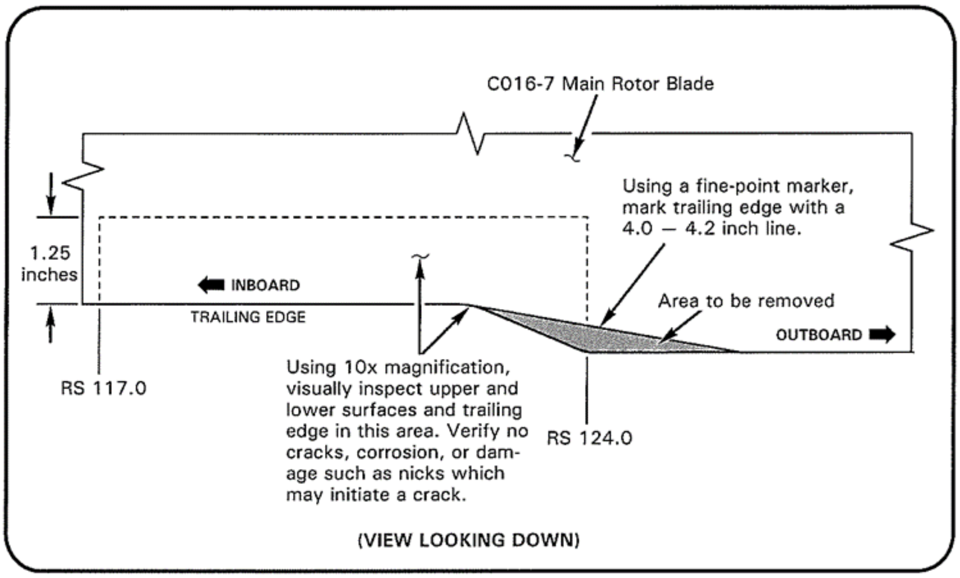


FIGURE 1

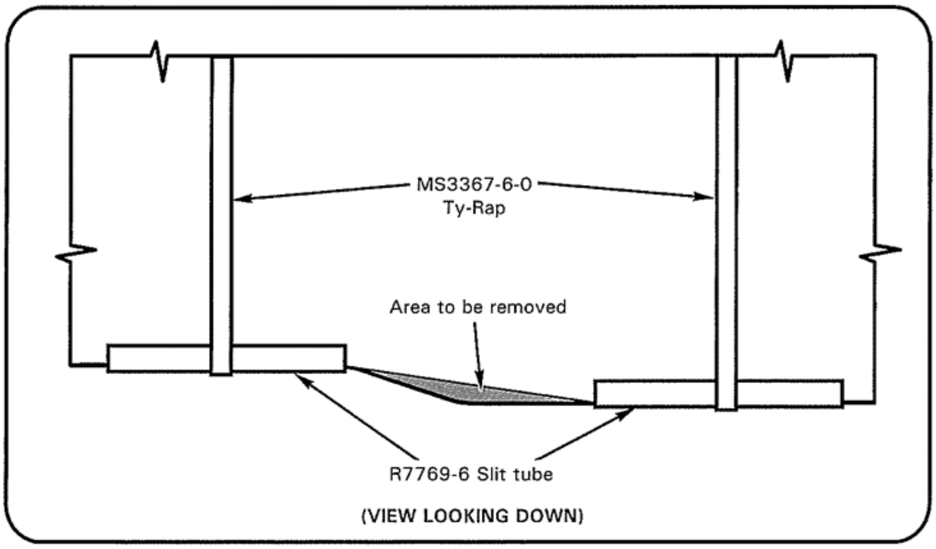


FIGURE 2

(OVER)

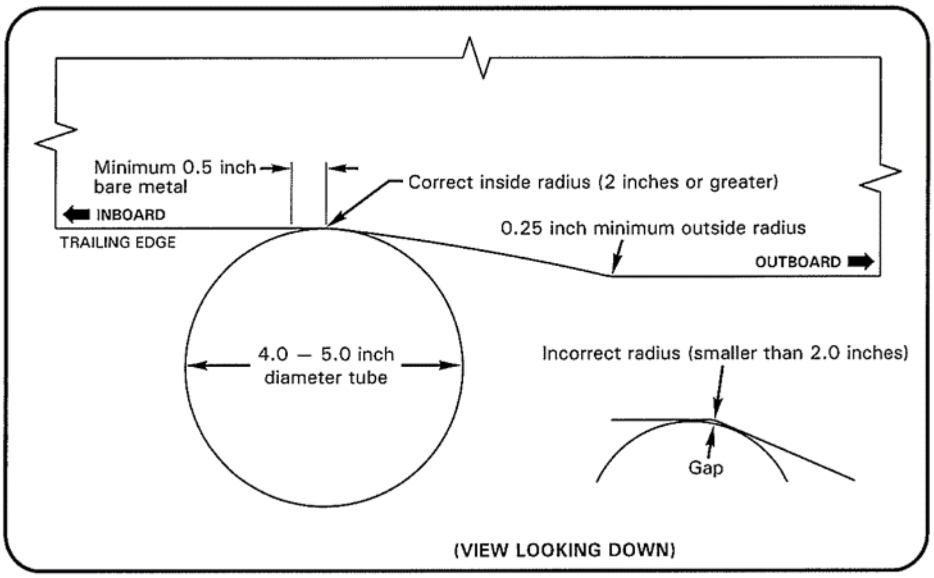


FIGURE 3

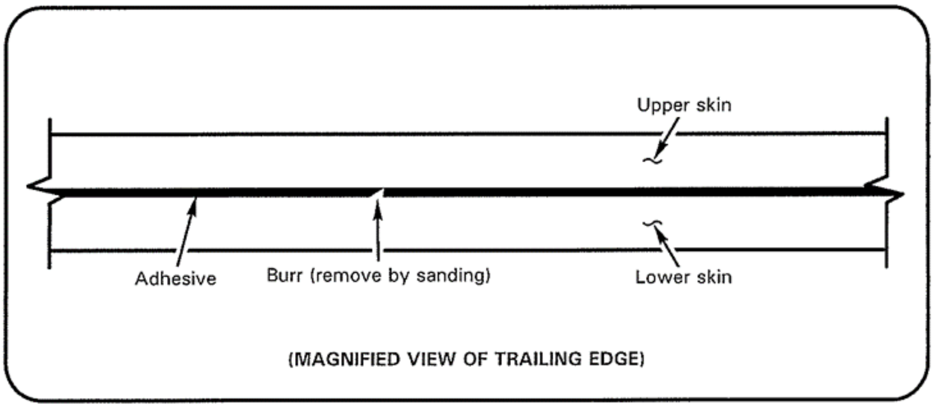


FIGURE 4

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)

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 5: 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.

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

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
Addendum to final report AO-2015-002	Mast bump and in-flight break-up, Robinson R44, ZK-IPY, Lochy River, near Queenstown, 19 February 2015
Interim Report AO-2017-001	Collision with terrain, Eurocopter AS350-BA, ZK-HKW, Port Hills, Christchurch, 14 February 2017
AO-2013-011	Runway excursion, British Aerospace Jetstream 32, ZK-VAH, Auckland Airport, 2 November 2013
AO-2014-006	Robinson R44 II, ZK-HBQ, mast-bump and in-flight break-up, Kahurangi National Park, 7 October 2014
Interim Report AO-2016-007	Collision with terrain, Robinson R44, ZK-HTH, Glenbervie Forest, Northland, 31 October 2016
AO-2014-004	Piper PA32-300, ZK-DOJ, Collision with terrain, Near Poolburn Reservoir, Central Otago, 5 August 2014
AO-2015-002	Mast bump and in-flight break-up, Robinson R44, ZK-IPY, Lochy River, near Queenstown, 19 February 2015
AO-2013-008	Boeing 737-300, ZK-NGI, Loss of cabin pressure, near Raglan, Waikato, 30 August 2013
AO-2013-003	Robinson R66, ZK-IHU, Mast bump and in-flight break-up, Kaweka Range, 9 March 2013
AO-2014-002	Kawasaki BK117 B-2, ZK-HJC, Double engine power loss, Near Springston, Canterbury, 5 May 2014

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