**AIRCRAFT ACCIDENT REPORT 1/2023**

Air Accidents Investigation Branch

Report on the accident to
**Leonardo AW169, G-VSKP**
at King Power Stadium, Leicester
on 27 October 2018

# APPENDICES

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## APPENDICES

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**Appendix A**

## SUMMARY OF THE ACCIDENT PILOT'S FLYING EXPERIENCE

Table A1 summarises the accident pilot's flying experience and instructor ratings.

| Aircraft Type | Total Hours | Hours PIC[1] | Instructor rating |
|---|---|---|---|
| **Helicopter flying** | | | |
| AS332 Super Puma | 1,804 | 1,630 | |
| AS350/355 Eurocopter Squirrel | 290 | 234 | TRI[2] |
| AgustaWestland AW109 | 1,252 | 1,245 | TRI |
| Leonardo AW169 | 177 | 174 | TRI |
| Bell 206 Jet Ranger | 312 | 287 | |
| Robinson R22/R44 | 506 | 403 | FI(H)[3] |
| Sikorsky S76 | 441 | 390 | |
| Eurocopter Gazelle SA341/342 | 69 | 69 | |
| Sub-totals | 4,784 | 4,360 | |
| **Fixed wing flying** | | | |
| Boeing 737 (B737) | 3,010 | 2,940 | TRI, TRE[4] (B737 300-900) |
| Boeing Business Jet | 150 | 150 | TRI, TRE |
| Cessna 150/152/172/182 | 118 | 110 | |
| Cessna Citation C525/550/560XLS | 1,670 | 1,603 | TRI |
| Hawker Siddeley/BAe HS125/800 XP | 30 | 6 | |
| Gulfstream G450/550/560/IV/V | 2,851 | 2,473 | TRI |
| Piper Aircraft PA28/31/34 | 222 | 145 | |
| Robin HR200 | 18 | 18 | |
| Socata GY-80 | 27 | 27 | |
| Sub-totals | 8,096 | 7,472 | |
| **Total flying hours:** | **12,947** | **11,904** | |

**Table A1**
The pilot previously held ratings as a simulator flying instructor and examiner for aeroplanes (SFI(A)/SFE(A)) and had been a class rating instructor for single and multi-engine aircraft.

1 Pilot in command.
2 Type Rating Instructor.
3 Flying Instructor (Helicopter).
4 Type Rating Examiner.

**Appendix B**

# AW169 FLIGHT SIMULATOR TRIALS

## 1. Synopsis

The AAIB conducted a series of trials on the AW169 flight simulators at the helicopter manufacturer's facilities in Italy. The trials consisted of automated replays of the accident flight as well as manually flown exercises to explore related emergency handling aspects.

The trials indicated that the failure prompted no system-generated alerts as to the nature of the emergency and pilots needed to rely on their own sensory cues to identify the problem. Only instantaneous recognition of the loss of tail rotor control and immediate lowering of the collective lever gave any opportunity for effective attitude control using the cyclic. Even with the collective fully-lowered within 1.5 seconds of the failure onset, the high residual rate of yaw meant that effective control over the helicopter's horizontal trajectory was not possible.

Lowering the collective was the required course of action to reduce the de-stabilising main rotor torque, but this placed the helicopter in a high rate of descent condition. Descending rapidly with limited cues made it difficult to judge when to raise the collective to cushion the subsequent touchdown. None of the trial profiles ended with a controlled landing below the simulator's crash detection threshold.

The rate of climb on the accident flight departure exceeded the RFM Cat A profile parameters but the trials indicated that, in the short time available before impact, the consequent elevated torque level did not significantly influence the post-failure controllability of the helicopter.

## 2. Aim of the trials

The investigation used the simulator trials to:

- Gain an insight into routine AW169 procedures before comparing them with the flight profile described in G-VSKP's FDR recordings.

- Understand what the accident sequence was like from inside the cockpit, including what warnings and cues might have been presented to the pilot at the time.

- Determine the level of post-failure controllability available to the pilot and to what extent applied main rotor torque might have affected it.

## 3 Simulator resources

The investigation's trials were conducted on two different simulators:

*Full flight simulator (FFS)*

The FFS was a Level D flight simulation training device (FSTD), certified in accordance with the EASA CS FTSD(H) standards[1]. It was accredited for zero-flight time conversion training. Accident flight reconstructions used the Olympic Stadium in Rome, rather than the King Power Stadium, as the takeoff location because the latter was not accurately modelled in the simulator terrain and obstacle database.

*Engineering simulator (ES)*

The ES was a fixed-base facility with a visual projection system and generic cockpit layout. It was normally used to develop and evaluate flight control software and it had the capability to record flight parameters for subsequent analysis. For the investigation's trials it was loaded with the AW169-specific flight model and representative flight control and avionics software for the accident flight. The King Power Stadium was modelled with enough fidelity for it to be used for the trial profiles.

### 3.1 Personnel

The primary pilot for profiles requiring manual flight was the manufacturer's AW169 development test pilot (TP). AAIB Inspectors of Air Accidents (Operations) and an Inspector of Air Accidents (Human Factors) observed automated flight profiles from both cockpit seats.

One of the AAIB Inspectors of Air Accidents (Operations) manually flew several profiles to help inform context regarding indicative pilot workload, cockpit ergonomics and failure symptoms.

### 3.2 Accident flight model

The manufacturer had used their AWARE[2] facility in the ES to develop a 'pilot in the loop' simulation model of the accident flight. The validity of the model was confirmed by comparison of FDR parameters from GVSKP and those generated during the simulation. This model assumed that the helicopter was stable and in trim in the hover after lift-off. From that point onward, the Cat A departure, as flown by the pilot, was recreated using FDR flightpath parameters. On passing 360 ft (280 ft radio altitude) the tail rotor pitch control actuator failure was injected.

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1 EASA Certification Specifications for Helicopter Flight Simulation Training Devices (CS-FSTD(H)) dated 26 June 2012. Available at https://www.easa.europa.eu/document-library/certification-specifications/cs-fstdh-initial-issue (accessed 28 July 2023).

2 AgustaWestland Advanced Rotorcraft Environment for simulation and design.

**Appendix B cont**

### 3.3 Failure event model

The modelled failure event consisted of the tail rotor pitch being driven to its fully negative angle of attack (-10°) over a period of 2.5 seconds. After failure injection, pilot control inputs as recorded by the FDR were used to drive the simulation to conclusion.

## 4 Conduct of trials

### 4.1 Normal procedures

The FFS was used to familiarise the investigation team with relevant normal procedures for the AW169 helicopter. These included full flight deck preparation and start sequence as well as the quick-start procedure which was used on the accident flight. The TP flew one complete Cat A departure from lift-off to an established climb at $V_Y$ with landing gear retracted.

### 4.2 Accident flight profiles (FFS)

Several automated replays were flown so that the accident sequence could be fully understood. Due to the lack of floodlights in the modelled Olympic Stadium and limited visual texture during simulated night conditions, the majority of profiles were flown in simulated daylight. Once sufficient experience of the full accident flight sequence had been gained, a number of manually flown profiles, were completed. Differing rates of climb and pilot response strategies were used to explore their effect on post-failure controllability of the helicopter.

### 4.3 Data gathering profiles (ES)

The ES was used to collect flight data to more accurately analyse the effects of the different rates of climb and pilot response times that had been observed in the FFS trial. The variable parameters used were main rotor torque (TQ), and thereby rate of climb, and response time from failure injection to lowering the collective lever. To facilitate consistent entry parameters for the different scenarios, the failure event was injected while the helicopter was in a steady rearward climb rather than during the dynamic transition to forward turning flight as seen with G-VSKP.

*Main rotor torque settings*

The simulated helicopter all-up-weight (AUW) used for the ES trials was lower than G-VSKP's AUW on the accident flight. The investigation considered the weight discrepancy to be acceptable because it would exaggerate the effect of additional torque, potentially making visual assessment of simulator response easier and it could be compensated for later during the subsequent data analysis.

**Appendix B cont**

The lower of the two torque settings used for the trials was approximately 86%, derived by establishing a 300 ft/min rate of climb during the simulated Cat A departures.

The second setting was 103% TQ which was representative of that used by G-VSKP's pilot. In the ES, the higher TQ setting resulted in rates of climb of approximately 1,500 ft/min, equivalent to 210% of the stabilised rate of climb on the accident flight. The investigation accounted for the un-representative simulated AUW when assessing the degree to which the elevated TQ levels affected post-failure controllability of the helicopter.

*Intervention time*

To establish what effect the timing of collective lever lowering had on the helicopter flight path the investigation chose three different target response times following failure injection: near-instantaneous to simulate the earliest possible intervention, then a two second interval and finally six seconds, similar to that on the accident flight. For each TQ setting the TP manually flew three Cat A departures with failure injection followed by lowering of the collective after the target interval. Manually lowering the collective introduced an element of variability between simulation runs. On the fifth simulated climb the target intervention time (2 seconds) was exceeded by approximately 1.7 seconds. Data from the six departure profiles was recorded and used for graphical analysis.

## 5 Trial results

FFS trials

*Automated accident flight profiles*

On simulated accident flights the helicopter climbed noticeably quicker than on the demonstration Cat A departure. It became apparent that after lowering the pitch attitude to begin the transition to forward flight, the pilot had rolled the helicopter right to establish a gently banked accelerating turn. The angle of bank briefly stabilised before the helicopter then experienced a marked yaw divergence to the right. With increasing yaw rate, the helicopter became visibly ever more unstable in both pitch and roll.

In the simulated environment, the AAIB did not observe any audio warnings or display cues that might have alerted the pilot to the tail rotor pitch control runaway. He would have relied on his visual and vestibular senses to identify the nature of the problem.

After the collective had been lowered the helicopter began descending rapidly and was still yawing at a high rate as it fell. On the automated replay flights, due to the modelled Rome Olympic Stadium being larger than the King Power Stadium, the helicopter had not travelled beyond the roof before descending. All automated trial profiles ended in collisions with the stadium roof rather than the ground.



 

**Appendix B cont**

*Manually flown profiles*

The TP manually completed several failure profiles under the investigation team's direction. Some of these were flown from an airfield location to avoid the possibility of collision with the stadium roof. Failures were injected during the climb with the rotor disc in a level, rather than banked, attitude.

Visually, there was no significant difference between profiles flown with an initial 300 ft/min rate of climb and those using a higher TQ setting to achieve a higher climb rate. In one profile where the collective was lowered immediately after failure injection the peak rate of yaw was observably lower than for a longer intervention time. In this instance the TP was able to make effective use of the cyclic to control helicopter attitude. While able to stabilise the helicopter's attitude, the residual rate of rotation meant that control over its horizontal trajectory was not possible.

With a two second wait the observed yaw rate was noticeably faster and it was no longer possible to achieve effective attitude control with the cyclic. On one occasion where a six-second intervention time was used, when the collective was lowered the helicopter rolled rapidly left in excess of 45° and became increasingly unstable in all three axes.

Judging rate of descent and when to raise the collective to cushion the touchdown was challenging due to the lack of visual texture in the simulated environment. All touchdowns exceeded the simulator's crash detection threshold.

*ES trials*

On all ES trial profiles the onset of yaw departure was rapid and uncontrollable. Peak yaw acceleration rates approached 50°/s$^2$. Even with near-instantaneous lowering of the collective lever, peak yaw rates exceeded 100°/s. Visually, the onset rates did not appear to differ significantly between those profiles using 86% TQ and those with 103% TQ applied. This was later corroborated by analysis of the recorded flight parameters (Figure B1). For comparable response times at the two different TQ settings, the resultant yaw, pitch and roll rate traces did not diverge markedly.

On all profiles the helicopter initially rotated in a relatively flat plane (about the normal axis). The greater the interval between failure injection and lowering the collective, the more unstable the helicopter became as the amplitude of pitch and roll oscillations increased (Figure B1).

**Appendix B cont**

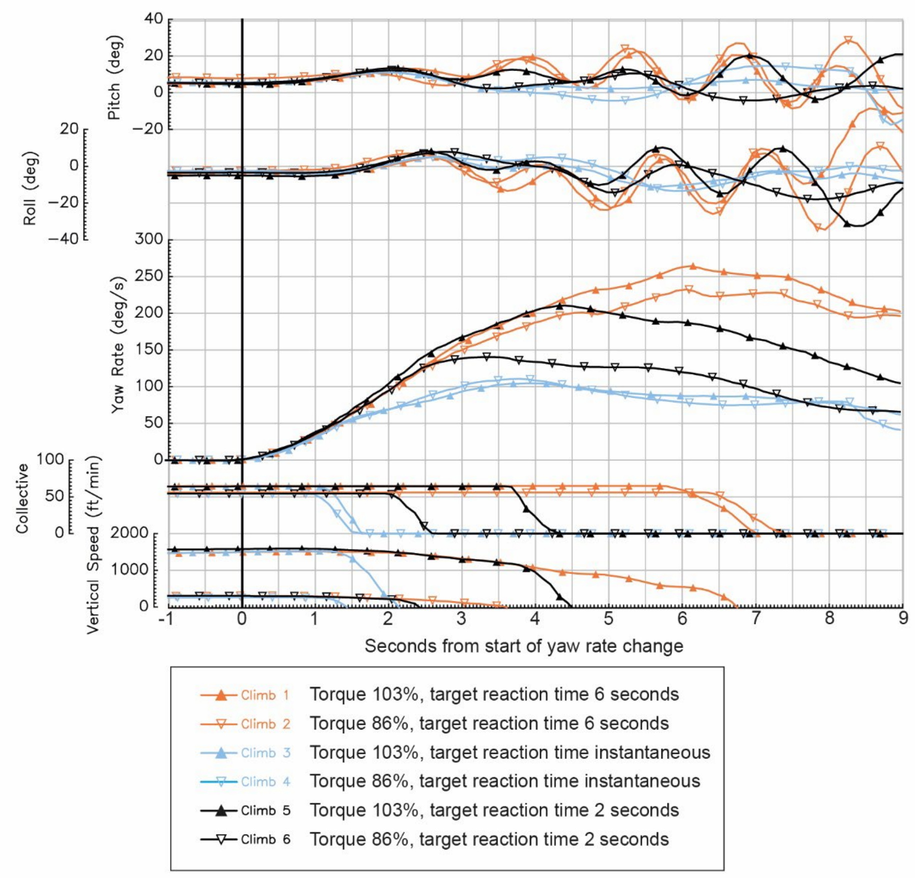


**Figure B1**

ES data output for TQ (rate of climb) and response time analysis

When analysing the potential effect of elevated TQ on the accident flight, the investigation compared the simulated helicopter's behaviour on the two climbs with a target intervention time of six seconds (Figure B2). While the post-failure collective lowering times were similar to that on G-VSKP, both climbs exceeded the 209°/s maximum yaw rate seen on the accident flight. Climb 1 reached 260°/s and Climb 2 peaked at 230°/s. It was noted that the instantaneous yaw rates started to diverge approximately 2.5 seconds after failure injection. They converged again approximately 3 seconds after the collective was lowered and the TQ reduction had taken effect. Using ES data for the two climbs and linear interpolation to compensate for the AUW discrepancy, the additional TQ on the accident departure would likely have increased the maximum yaw rate by approximately 7%. The observed amplitude and frequency of pitch and roll deviations did not differ markedly between Climb 1 and Climb 2 despite the different TQ levels.

Appendices

 

**Appendix B cont**

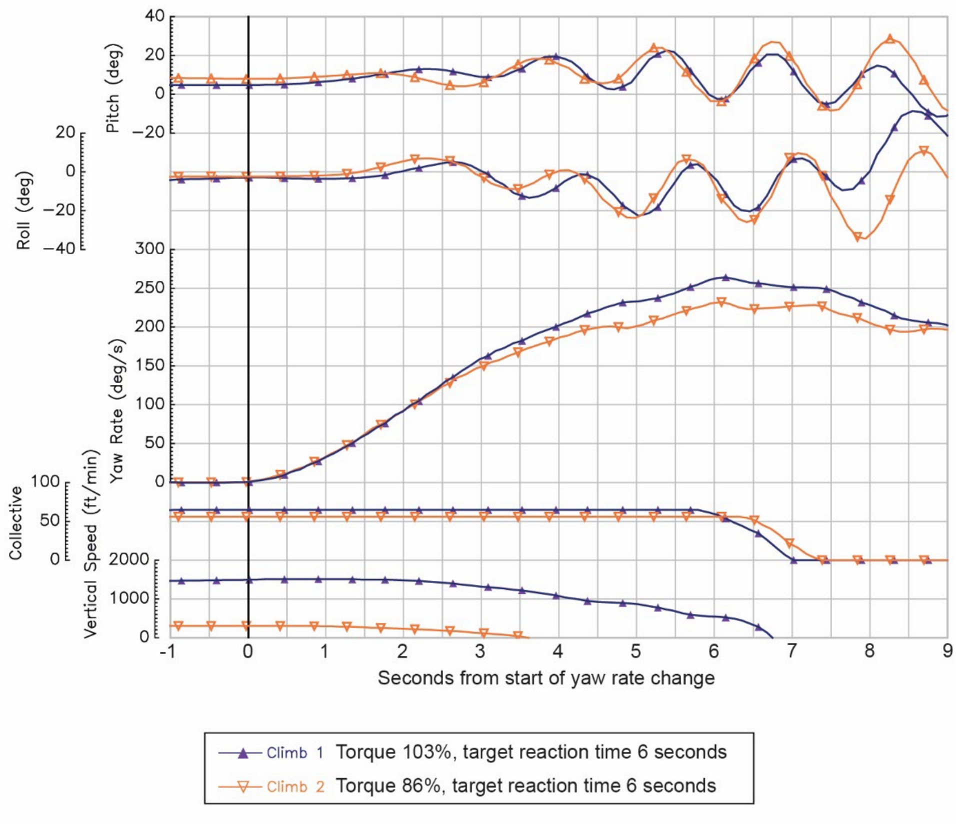


**Figure B2**
Comparison of Climbs 1 and 2

It was not possible, on any of the trial profiles, to reduce the yaw rate to a level where control of the helicopter's horizontal trajectory could be established. Every simulated accident flight terminated in an uncontrolled touchdown which exceeded the simulator's crash detection threshold.

## 6 Discussion

Nature of the failure

The trials confirmed that the pilot was presented with an unexpected failure for which there were no system-generated warnings or cues. He would have relied on visual and vestibular senses to identify the nature of the problem. The onset of yaw departure exceeded 50°/s$^2$ and could not be prevented by use of the yaw pedals. With the peak yaw rate exceeding 200°/s and significant pitch and roll divergences the emergency was likely to have been extremely disorientating[3].

3 The limited range of travel of the simulator's motion cueing systems meant that the full extent of physical sensations experienced by the pilot as a result of the longitudinal, lateral and normal accelerations could not be accurately assessed.



 

Post-failure controllability

Analysis of the ES data revealed no marked post-failure stability characteristic differences on Cat A departures flown with differing TQ levels. The most significant influence on post-failure controllability was the response time for lowering the collective lever. When using a near-instantaneous response the TP was able to exercise a degree of attitude control as the helicopter descended but not enough to direct the flight path. When the reduction in collective pitch was initiated beyond two seconds the magnitude of pitch and roll instability became increasingly exaggerated and positive attitude control was not achievable.

Lowering the collective to reduce the destabilising TQ had the expected secondary effect of generating a high rate of descent. The dynamic nature of the descent and limited cues made judging when to raise the collective to cushion the touchdown extremely challenging.

Even discounting the intervention of the simulated roof in the automated replays, the scenario played out extremely quickly, leaving little time for the pilot to consider shutting down the engines.

## 7 Conclusion

The trials found that the loss of tail rotor control was sudden and irrecoverable. With no system generated cues, the pilot had to rely on what he could see and feel, together with his past experience, to make sense of the situation and decide on an appropriate course of action. With no specific drill for a tail rotor pitch control runway, lowering the collective lever was the most appropriate action to take.

The greater the time between the injection of the failure and lowering the collective, the more unstable the helicopter became and the more difficult it was to control its attitude. Even with an instantaneous lowering of the collective lever, positive control of the helicopter's trajectory was impossible after the tail rotor pitch moved to full deflection. With limited time and height available the only reliable flight control was the collective lever. Lowering it reduced the destabilising torque but generated a high rate of descent. Judging when to cushion the touchdown relied on the pilot's instinctive assessment of the critical point at which to raise the collective lever. None of the simulated accident flight profiles ended in controlled touchdowns below the simulator's crash detection threshold.

The rate of climb on the accident flight departure exceeded the RFM Cat A profile parameters, but the ES trials indicated that the higher TQ level required to achieve it did not significantly influence the post-failure controllability of the helicopter.

**Appendix C**

## SUMMARY OF PILOT RESPONSE TIMES FROM PREVIOUS ACCIDENTS*

| Event | Situation | Observable symptoms of failure | Time between failure onset and pedal application | Time taken to apply full pedal | Time between full pedal and start of lowering the collective | Time between failure onset and start of lowering the collective | Time between failure onset and completely lowering the collective |
|---|---|---|---|---|---|---|---|
| G-WNSR 1 | Immediately after lift-off from offshore installation | Uncommanded right yaw | Immediate | 3 s | 2 s | 5 s | Not relevant due to immediate landing |
| G-WNSR 2 | Before landing 4 ft above offshore installation. Second instance of uncommanded yaw on the same day. | Uncommanded right yaw | Immediate | 2 s | Collective began to be lowered as max left pedal reached -0.5 s | 1.5 s | Not relevant due to immediate landing |
| B-MHJ | 2-pilot operation. Climbing over water 350 ft AMSL and 70 kt | Loud bang and vibrations. Loss of authority of yaw pedals | Investigation reported that the pilot 'immediately' put the helicopter into autorotation | | | | |
| PR-SEK | 2-pilot operation. Stabilised climb over water with the autopilot engaged at 130 kt | Abnormal noise, 'abrupt' right yaw and left roll | - Pedal applied immediately by autopilot | - | - | 1.3 s first lowering which was then reversed and possibly inadvertent. 4.5 - 5 s second lowering. | 5 - 5.5 s |
| G-VSKP | Single pilot. Low speed climbing transition from rearwards to forward flight above a structure. | Rapid uncommanded right yaw | Immediately < 1 s | 1 s | 5 s | 6 s | 8 s |
| Chapellow research | Routine simulator training. Level flight at 2000 ft | Left yaw of approximately 80°/s | Not reported | | | Range 0.58 - 3.21 s Mean 1.53 s | Range 3.26 - 8.26 s Mean 4.9 s |

* Times from previous events are approximate due to limitations of flight data or the data included in previous reports.

**Appendix D**

# FLIGHT RECORDER ISSUES

## 1 Overview

There were a number of issues with the flight recorder system that generated challenges to the investigation. They did not affect the findings given the nature of the accident but may affect future investigations. These relate to:

- Timing dither in the recorded data creating problems deriving rate of change parameters.
- Accelerometer saturating due to location, sensing range and high yaw rates.
- The quality of the audio from the Cockpit Area Microphone (CAM).
- Recorded Global Positioning System (GPS) anomalies.

## 2 Certification standards

The helicopter manufacturer stated that the flight recorder installation was certified in accordance with:

- CS 29 'Certification Specifications, Including Airworthiness Code and Acceptable Means of Compliance, for large rotorcraft', Amendment 2, dated 17 November 2008.
- EUROCAE ED-55[1] 'MINIMUM OPERATIONAL PERFORMANCE SPECIFICATION FOR FLIGHT DATA RECORDER SYSTEMS' and ED-56A 'MINIMUM OPERATIONAL PERFORMANCE SPECIFICATION FOR COCKPIT VOICE RECORDER SYSTEM', Amendment 1.
- United Kingdom Civil Aviation Authority specifications 10, 10A, 11 and 18.
- Federal Aviation Administration (FAA) TSO-C123a and TSO-C124a.

The EASA Air Operations Regulation also has flight recorder requirements. The pertinent specification in this case is in its Annex VI (Part-NCC) along with the associated Acceptable Means of Compliance (AMC) and Guidance Material (GM). AMC1 NCC.IDE.H.165

1 European Organisation for Civil Aviation Equipment (EUROCAE). ED is EUROCAE Document (ED).

'Flight data recorder' states that the "operational performance requirements for flight data recorders should be those in EUROCAE Document ED-112 (Minimum Operating Performance Specification for Crash Protected Airborne Recorder Systems) dated March 2003, including amendments n°1 and n°2, or any later equivalent standard produced by EUROCAE." It further refers to a list of FDR parameters required for 'All helicopters'.

ED-112 superseded the original ED-55 and ED-56A specifications associated with the installation and has been superseded by ED-112A.

The requirements with regards to the issues identified have not appreciably changed from ED-55 to ED-112A. EUROCAE has an ongoing working group updating this to ED-112B.

### 3 Timing dither in the recorded data

The AW169 DAFR installation uses the two AMMCs to supply the majority of the data to the flight recorder. The AMMCs supply the parameter updates using data buses. The update samples are provided at a faster rate than the recorder is required to capture. The recorder does not capture all the samples provided to it. It still records more samples per second overall than required but drops some of the supplied samples in an unpredictable manner. This adds significant short-term errors when calculating how fast the parameters are changing.

The recorder loads the AMMC data to parameter buffers as it receives it, for example every 160 ms for some parameters. It then reads the buffer values at its prescribed recording rate, every 250 ms for this example, putting the latest values from the buffer into the flight recorder memory module. In this case, sometimes the buffer is only updated once between the recording samples being taken and other times there has been two updates. This means the time between each recorded sample dithers between 1 and 2 whole original sample periods as illustrated below (Figure D1).

The supplied-interval/recorded-interval pairing 160 ms/250 ms of the example is not the only one with the same resultant problems. 80 ms/125 ms, 320 ms/500 ms and 640 ms/1000 ms pairings are also used.

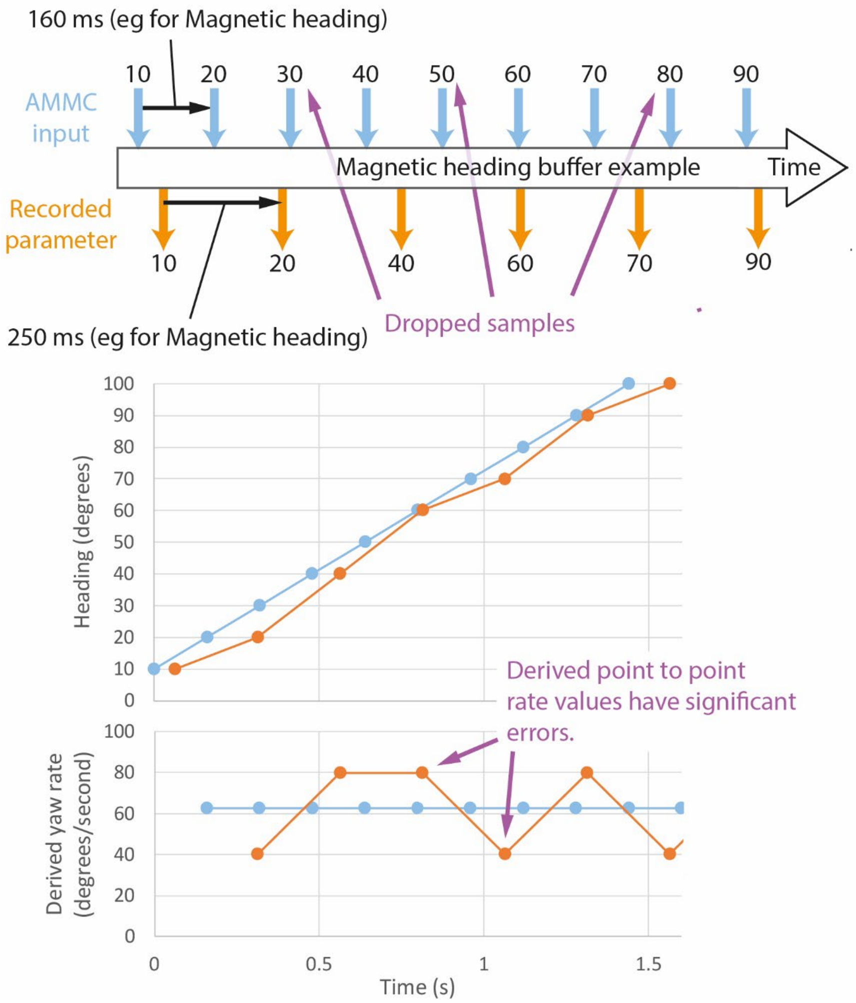


**Figure D1**
Illustration of dither in recorded data and derived rate parameters using Magnetic heading as an example

Using the recorded values to derive point-to-point rates of change gives large erratic errors.

Smoothing the data over time reduces the error during periods where the parameter changes are stable but does not reflect a parameter with dynamic rates of change.

This is the same for all the recorded parameters where the recorder is not sampling an analogue input itself but picking data from a data bus.

The problems with the derived yaw rate parameter added further error to calculations of g forces calculated based on the derived yaw rate. This problem also complicated the use of the recorded data to establish descent rates from radio height data, needed to assess impact forces.

 

**Appendix D cont**

### 3.1 Requirements

The relevant requirements are given in ED-55, Chapter 3, 'DATA RECORDING REQUIREMENTS', section 3.2.2, and ED-112A, paragraph II-2.1.15.2 'Data Sampling'. This states:

> '*Successive recorded values of each parameter shall be derived from new readings obtained from the input interface of the flight recorder system. The interval between these readings shall be that specified in the parameter tables within a tolerance of 1/64$^{th}$ of a second…*'

The input interface has been interpreted as where the recorder receives the data from the AMMCs. So the time between samples is taken as time between sampling the data from the bus, leading to this problem. This issue is likely not unique to this aircraft type or manufacturer.

ED-112A is in the process of being updated to ED-112B. Additional text has been proposed to address the issue.

### 4 Accelerometer limitations

A tri-axial accelerometer is fitted to the helicopter so the recorder can capture the fore/aft, left/right and up/down accelerations of the helicopter. Rotational motion will also induce an acceleration at locations not at the CG of the helicopter. The further from the CG, the larger the rotation induced acceleration will be. To limit the effect of this on the recorded accelerations, the accelerometer is required to be located within the CG limits of the helicopter.

However, these are within the passenger cabin space so the accelerometer is located in the rear avionics bay on this helicopter type, aft of the CG limits.

During the accident, the yaw rate induced a longitudinal acceleration at the sensor which exceeded its range. With sufficient recording of the rotational motion of the helicopter, it is possible to approximately remove the rotational effects from the measured accelerations. However, the accelerometer saturated, making such a task impossible.

The location of the accelerometer is addressed in CS 29.1459, ED-55 and ED-112A. CS 29.1459, '*Flight recorder*', requires that:

> '…
>
> *(2) The vertical acceleration sensor is rigidly attached, and located longitudinally within the approved centre of gravity limits of the rotorcraft;*
>
> …'

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ED-55 2.11.2, 'LOCATION OF EQUIPMENT', paragraph c states:

> '*Acceleration data shall be obtained from sensors which are rigidly attached and located longitudinally either (i) in helicopters, within the approved centre of gravity limits, or (ii) in aeroplanes, within a distance forward or aft of the centre of gravity limits that does not exceed 25% of the aeroplane mean aerodynamic chord…*'

It also references A2.5.2 of the same document, which states:

> '*The requirements for acceleration sensors are given in paragraph 2.11.2c.. The use of alternative sources of acceleration data, e.g. Inertial Reference Units, is not recommended particularly where the units are located outside the prescribed limits. Translation of the such [sic] parameters involves many variables in a complex calculation where the algorithm depends on the location of the inertial reference unit relative the particular aircraft centre of gravity. It would be difficult, if not impossible, to verify the algorithm for the accident environment. Furthermore, equipment interchangeability would be adversely affected*.'

ED-112A, Chapter II-6, 'EQUIPEMENT INSTALLATION AND INSTALLED PERFORMANCE', section II-6.2.2 uses very similar wording to ED-55 with regards to the accelerometer.

The Centre of Gravity (CG) limits on this helicopter are in the passenger cabin volume. The accelerometer is mounted in the rear avionics bay, 1.37 m to 1.81 m aft of the CG limits. A longitudinal accelerometer 1.81 m from the CG will sense approximately 2.5 g purely due to a yaw rate of the 209˚/s (the peak rotation rate during the accident sequence). The accelerometer has a longitudinal range of ±1 g.

Rapid pitching and a gravity component when not level would add to this. With the pitch angles recorded, the gravity component would have reached more than 0.5 g alone.

This yaw rate induced acceleration reduces the closer the accelerometer is to the CG. However, had the accelerometer been located within the CG limits as per the standards, there is still a potential problem with the limited accelerometer sensing range due to rotation induced accelerations. The 209˚/s yaw rate induced component of longitudinal acceleration, with the CG at one edge of the CG limits and an accelerometer at the opposite CG limit, would have been approximately 0.6 g. That along with the gravity component of longitudinal g during the more extreme pitch attitudes of the helicopter during the accident flight would have exceeded the 1 g range limit of the accelerometer.



**Appendix D cont**

The range of the recorded parameters are defined in the latest recorder requirements, ED-112A, in ANNEX II-A ‘DEFINITION OF PARAMETERS TO BE RECORDED’. This includes tables detailing the parameters to be recorded and additional text that highlights when this may be deviated from. ED-112A, Table II-A.2: ‘PARAMETERS TO BE RECORDED – HELICOPTERS’ includes ‘*Longitudinal Acceleration (body axis)*’ with a range of ±1 g. This has a maximum recording interval of 0.25 seconds, the same as the pitch, roll and heading parameters. ‘*Normal acceleration*’ has a maximum recording interval of 0.125 seconds. These are improved recording rates for the rotational parameters over the ED-55 requirements. Without the timing jitter associated with the parameters, the sample rates of the latest standard support back calculation of the actual forces at the CG. Even without the timing jitter, the rates are not sufficient for calculating the higher sample rate vertical accelerations at the CG.

Additionally, ED-112A, ANNEX II-A, II-A.1 paragraph b states:

> 'The choice of recorder class and the parameters to be recorded is the prerogative of the responsible regulatory agency.'

ED-112A is in the process of being updated to ED-112B. Additional text is being proposed that highlights the problem and provides additional requirements for installations where the accelerometers are not mounted within a prescribed location relative to the centre of gravity.

### 5 Cockpit Area Microphone channel audio quality

During the short accident flight, there was significant disruption to the CAM audio recording. Further inspection also found brief disruptions during the previous flights. These were often spaced 9 seconds apart though some were more sporadic.

Figure D2 shows the CAM waveform envelope from when the first engine was being started for the accident flight, to the impact. Parameters are added to provide some reference to the timeline. This also shows the results of an algorithm designed to detect periods when the waveform has stopped crossing the zero value for an extended number of audio samples. Periodic disruptions, occurring every nine seconds can be seen to initiate in the waveform envelope, detected by the algorithm. It also shows two other periods of more erratic problems. The first is during the rotor run-up on the ground and the second is after the bearing failure in the air.

**Appendix D cont**

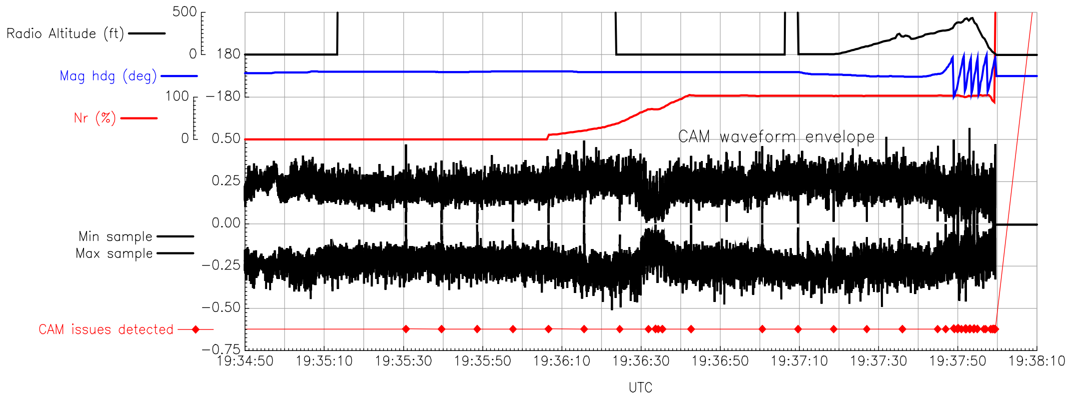
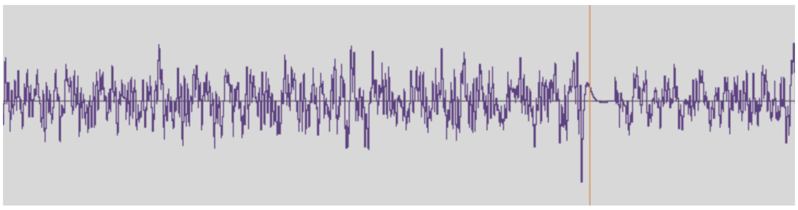



**Figure D2**
Envelope of the CAM audio, the detections of CAM audio issues and recorded parameters for the accident flight

Figure D3 shows audio waveforms from 0.3 seconds of CAM recordings.

The CAM channel should be showing continuous activity during these periods. Given the use of active gain control in the acquisition of the audio, the absolute values of the amplitudes are not relevant, the repeated loss of signal is the key issue.



a)

b)

**Figure D3**
Two extracts of CAM audio of 0.3 second duration, a) during one of the typical 9 second periodic disruptions, and b) during the accident flight.

Figure D3 shows the extent of the problem during the accident flight, after the loss of control and before impact. This degraded audio analysis techniques.

**Appendix D cont**

The disruptions to the recorded audio were not observed on the other recorded audio channels.

### 5.1 Component locations

In common with other installations on other aircraft, the weak CAM signals feed into a Cockpit Control Unit (CCU) pre-amplifier which then drives the input to the recorder. Ideally the CCU is close to the CAM to minimise the distance over which the unamplified signal travels.

In this installation design, the CCU is close to the DAFR in the rear avionics area, remote from the CAM in the cockpit roof.

The fire damage did not allow an assessment of the audio components, the wiring between the CAM and the preamplifier, or the shielding of the wiring.

### 5.2 Helicopter manufacturer review

The helicopter manufacturer was asked to review CAM recordings from other helicopters of the same type to establish whether this was unique to the accident helicopter. They found other isolated examples of brief disruptions. They associated this with the proximity of the top Traffic Alert and Collision Avoidance System (TCAS) antenna to the CAM, indicating Electromagnetic Interference (EMI) as the likely cause.

No links between TCAS and nine second periodic activity have been identified.

### 5.3 DAFR manufacturer review

The DAFR manufacturer was asked to review their designs for anything that could relate to regular disruptions every nine seconds, and how the recorder would react to different signal disruption scenarios.

The only aspect of the recorder system with regular timings are associated with the storing of recordings to memory. A bad memory chip or block of memory within a chip could conceivably create a periodic disruption in a downloaded recording. However, the timings do not correlate with a nine second repeating problem.

The DAFR manufacturer tested the CCU and DAFR CAM input responses to intermittent disruption of the CAM wiring. The results showed different recovery times for the different wires disconnected but none were consistent with many of the observed disruptions of the CAM recording. The testing explored some timing scenarios but not an exhaustive list. This does not rule out very short duration disruptions of some sort.

### 5.4 Audio quality conclusion

No definitive cause of the CAM audio anomalies was identified. The helicopter manufacturer linked the issue to proximity of the CAM in the cockpit roof to the TCAS antenna location. This is not a pure systematic cause as the problem is not prevalent on other installations. The system could not be tested due to fire damage so CAM audio issues specific to the accident helicopter could not be identified.

The helicopter manufacturer has changed the flight recorder system used for new helicopters of this model, and relocated the CAM away from the TCAS antenna. No anomalous CAM behaviour has been observed with the new installation. There are no retrofit plans.

## 6 Recorded Global Positioning System (GPS) anomalies

The recorded GPS path after the loss of control had significant errors, ending approximately 180 m from the wreckage location. Additionally, there were timing issues with the AMMCs, which update their internal clocks using GPS. The DAFR records the AMMC times, these changed their behaviour at approximately 19:37:52. The AMMC internal logs recorded a warning but failed to retrieve its own timestamp for the log. This occurred between 1937:52 hrs and 1937:55 hrs. The ADAHRS units also receive GPS time and record this against its logged faults. Despite one unit having faults span an elapsed time of 23 seconds, the same GPS time was recorded against them, less than a second before 1937:51.9 hrs. The time recorded against the faults on the other ADAHRS unit was 1937:51.7 hrs. This points to a problem occurring approximately 8 seconds before impact.

Two possible reasons for GPS disruption not ruled out were loss of satellite tracking and excessive antenna rotation.

There are many GPS satellites in a constellation, constantly moving across the sky. GPS receivers only need to track a few of them to function properly but they should be widely distributed around the sky for good accuracy. The GPS receiver antenna has to be in line-of-sight of the satellites it is receiving signals from.

The antenna is mounted on the top of the tail structure, a short distance from where it joins the main fuselage. This will give it a clear view of much of the sky, except for directly forward. The forward structure will obscure more of the forward view of the sky when the helicopter is pitched nose-up. The structure to the rear of it may also block satellite views when the helicopter is pitched nose-down. The dynamic attitude changes may have resulted in sufficient disruption to the line of sight to satellites to lose lock on some of their signals. It is not known how well the installed system can cope with this. Satellites with high elevations at the time will not have lost line-of-sight but need to be supplemented by lower elevation satellites for better accuracy.



**Appendix D cont**

The signals from the satellites have circular polarisation. This means that the electromagnetic wave that carries the information effectively corkscrews as it travels from the satellite to the receiver antenna. Spinning the antenna can effectively unwind or wind the carrier signal. The helicopter manufacturer stated that this phenomenon could account for the GPS problems during the accident flight.

Appendices

**Appendix E**

# AW169 FLIGHT TEST LOAD SURVEY AND CONTACT PRESSURE ANALYSIS

Table 1E shows the highest recorded axial ($F_Z$ or TH1) load for the AW169 and AW189 and the test conditions when they occurred. These were the maximum loads recorded under dynamic loading. As such, they may only have occurred for short durations during the manoeuvre. The AW169 highest load was recorded during flight manoeuvres, the AW189 recorded its highest load during ground taxi manoeuvres, so the highest load in flight has also been included.

| AW169 | | | | AW189 | | | | |
|---|---|---|---|---|---|---|---|---|
| Gross Weight (kg) | Centre of gravity (long) | Altitude (ft) | Axial load (N) | Flight phase | Gross Weight (kg) | Centre of gravity (long) | Altitude (ft) | Axial load (N) |
| 4,800 | 5,460 (aft) | 5,000 | 8,985 | flight | 8,600 | 5167 (fwd) | 0 | 13,652 |
| | | | | ground taxi | 8,600 | 5167 (fwd) | 0 | 14,400 |

**Table 1E**

Highest axial tail rotor loads recorded during AW169 and AW189 flight test

All the AW169 flight test load survey results for axial load and bending moment were collated and analysed by the investigation. They were then filtered to identify test points with an axial load ($F_Z$ or TH1) of 7,000 N or greater and a calculated PRL or NRL bearing bending moment (M) equal to or above the highest considered by the design spectrum of 42.2 Nm. The points which met this criterion are shown in Table E2. Note: The sign convention of positive or negative just denotes the direction of the load relative to a datum. The magnitude of the recorded load was used to filter the results.

Appendices

## Appendix E cont

| Flight test conditions | | | | | | Flight test point | Calculated and recorded loads | | |
|---|---|---|---|---|---|---|---|---|---|
| Flight No | GW (kg) | CG (Long STA) | CG (Lat STA) | Alt (ft) | underslung load (kg) | Description | Approx bearing moment NRL (daNm) | Approx bearing moment PRL (daNm) | Axial Load (daN) |
| 263 | 4800 | 5120 | 0 | 0 | 0 | Spot turn HIGE 30° / sec. Left | -5.46 | -4.23 | 787.396 |
| 265 | 4800 | 5120 | 0 | 0 | 0 | Azimuth 090° at 50 Kts G.S. entry | -4.79 | 4.51 | 762.736 |
| 286 | 4800 | 5464 | 0 | 0 | 0 | Spot turn HIGE 30° / sec. Left | -5.75 | -3.53 | 768.712 |
| 286 | 4800 | 5464 | 0 | 0 | 0 | Azimuth 090° at 50 Kts G.S. entry | -4.44 | -4.05 | 794.784 |
| 297 | 4200 | 5207 | 0 | 0 | 0 | Azimuth 270° at 50 Kts G.S. recovery | -5.94 | -6.15 | -766.393 |
| 349 | 4800 | 5120 | 60 | 0 | 0 | Azimuth 270° at 50 Kts G.S. entry | -7.19 | -5.59 | -784.607 |
| 349 | 4800 | 5120 | 60 | 0 | 0 | Azimuth 270° at 50 Kts G.S. steady | -6.22 | -3.61 | -784.607 |
| 349 | 4800 | 5120 | 60 | 0 | 0 | Taxiing RH turn 180° on concrete surface | -4.95 | -3.2 | -763.749 |
| 349 | 4800 | 5120 | 60 | 0 | 0 | Azimuth 270° at 50 Kts G.S. recovery | -6.15 | 5.52 | -703.781 |
| 349 | 4800 | 5120 | 60 | 0 | 0 | Azimuth 090° at 50 Kts G.S. entry | -5.46 | 4.21 | 779.761 |
| 356 | 4800 | 5464 | 0 | 15000 | 0 | VAUTO autorotation @ NRMAX 30° bank RH turn | -5.54 | -6.93 | -727.135 |
| 359 | 4200 | 5207 | 0 | 15000 | 0 | VAUTO autorotation @ NRMAX 30° bank RH turn | -5.22 | -6.17 | -769.879 |
| 359 | 4200 | 5207 | 0 | 15000 | 0 | VNE autorotation @ NRMAX 30° bank RH turn | -6.17 | -7.99 | -741.199 |
| 360 | 4800 | 5120 | 0 | 0 | 0 | VAUTO autorotation @ NRMAX 30° bank RH turn | -5.23 | -6.28 | -714.345 |
| 360 | 4800 | 5120 | 0 | 0 | 0 | VNE autorotation @ NRMAX 30° bank RH turn | -6.33 | -7.79 | -711.738 |
| 360 | 4800 | 5120 | 0 | 0 | 0 | Azimuth 060° at 50 Kts G.S. entry | -4.53 | 4.36 | 816.127 |
| 365 | 4200 | 5040 | 0 | 15000 | 0 | VNE autorotation @ NRMAX 30° bank LH turn | -6.09 | -6.3 | -700.882 |
| 368 | 4200 | 5040 | 0 | 0 | 0 | Azimuth 270° at 50 Kts G.S. steady | -3.81 | -6.27 | -856.998 |
| 368 | 4200 | 5040 | 0 | 0 | 0 | Azimuth 270° at 50 Kts G.S. recovery | 5.2 | -7.36 | -849.177 |
| 368 | 4200 | 5040 | 0 | 0 | 0 | Azimuth 060° at 50 Kts G.S. entry | 5.3 | -5.8 | 840.34 |
| 368 | 4200 | 5040 | 0 | 0 | 0 | Azimuth 090° at 50 Kts G.S. entry | 4.6 | -5.28 | 767.336 |
| 368 | 4200 | 5040 | 0 | 0 | 0 | Azimuth 060° at 50 Kts G.S. steady | 5.1 | -4.16 | 743.87 |
| 593 | 4800 | 5460 | 135 | 0 | 0 | Azimuth 060° 20 Kts G.S. | -6.16 | 6.55 | 716.098 |
| 594 | 4800 | 5460 | 0 | 0 | 775 | Azimuth 090° at 50 Kts G.S. recovery | -4.69 | -3.69 | 756.151 |
| 594 | 4800 | 5460 | 0 | 0 | 775 | Azimuth 060° at 50 Kts G.S. entry | -5.88 | 5.57 | 756.151 |
| 595 | 4800 | 5160 | 135 | 0 | 0 | Azimuth 270° at 50 Kts G.S. steady | -6.16 | -5.36 | -709.233 |
| 595 | 4800 | 5160 | 135 | 0 | 0 | Azimuth 060° at 50 Kts G.S. steady | -6.43 | 4.84 | 758.666 |
| 595 | 4800 | 5160 | 135 | 0 | 0 | Azimuth 090° at 50 Kts G.S. entry | -4.38 | -3.47 | 737.807 |
| 601 | 4600 | 5120 | 150 | 10000 | 0 | Azimuth 060° at 50 Kts G.S. steady | -9.46 | 1.63 | 752.82 |
| 603 | 4600 | 5460 | 150 | 10000 | 0 | Azimuth 060° at 50 Kts G.S. steady | -6.62 | -1.45 | 798.685 |
| 647 | 4800 | 5287 | 0 | 0 | 1000 | Azimuth 270° at 50 Kts G.S. entry | -2.57 | 4.49 | -716.367 |
| 647 | 4800 | 5287 | 0 | 0 | 1000 | Azimuth 060° at 50 Kts G.S. recovery | 4.2 | -3.22 | 750.861 |
| 647 | 4800 | 5287 | 0 | 0 | 1000 | Azimuth 060° at 50 Kts G.S. steady | -7.28 | -1.87 | 797.854 |
| 649 | 4800 | 5160 | 0 | 0 | 800 | Azimuth 060° at 50 Kts G.S. steady | -8.75 | 1.91 | 733.36 |
| 654 | 4800 | 5460 | 0 | 0 | 775 | Azimuth 060° at 50 Kts G.S. steady | -4.67 | -2.24 | 744.592 |
| 654 | 4800 | 5460 | 0 | 0 | 775 | Azimuth 090° at 50 Kts G.S. steady | -4.8 | -2.05 | 768.089 |
| 677 | 4800 | 5460 | 0 | 7000 | 500 | Azimuth 060° at 50 Kts G.S. steady | -8.7 | -2.49 | 798.713 |

**Figure 1E**

AW169 flight test points with an axial load of 7 kN or greater and calculated bearing bending moment of 42 Nm or greater

The bearing manufacturer calculated the contact pressures resulting from the flight test points shown in table 2E (Table 3E), the final rig test requested test points (Table 4E) and rig test average actual applied loads (Table 5E) to allow for comparison. This confirmed that the contact pressures for the flight test points and the average actual applied loads during the rig test were a similar magnitude.

| Pressure [MPa] | 1 | 2 | 3 | 4 | 5 | 6 | 7 | 8 | 9 | 10 | 11 | 12 | 13 |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| Row1-IR | 3091 | 2423 | 2996 | 2266 | 3060 | 2418 | 3057 | 2294 | 2881 | 2571 | 2851 | 2661 | 2398 |
| Row1-OR | 2920 | 2214 | 2825 | 2070 | 2889 | 2210 | 2887 | 2095 | 2703 | 2363 | 2672 | 2449 | 2205 |
| Row2-IR | 2423 | 2996 | 2266 | 3060 | 2418 | 3057 | 2294 | 2881 | 2571 | 2851 | 2661 | 2398 | 2763 |
| Row2-OR | 2214 | 2825 | 2070 | 2889 | 2210 | 2887 | 2095 | 2703 | 2363 | 2672 | 2449 | 2205 | 2587 |

| Pressure [MPa] | 14 | 15 | 16 | 17 | 18 | 19 | 20 | 21 | 22 | 23 | 24 | 25 | 26 |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| Row1-IR | 2763 | 2372 | 2286 | 2824 | 2498 | 3017 | 2233 | 2645 | 2792 | 2569 | 2820 | 2770 | 2969 |
| Row1-OR | 2587 | 2201 | 2114 | 2648 | 2295 | 2846 | 2039 | 2436 | 2613 | 2362 | 2643 | 2545 | 2791 |
| Row2-IR | 2372 | 2286 | 2824 | 2498 | 3017 | 2233 | 2645 | 2792 | 2569 | 2820 | 2770 | 2969 | 2583 |
| Row2-OR | 2201 | 2114 | 2648 | 2295 | 2846 | 2039 | 2436 | 2613 | 2362 | 2643 | 2545 | 2791 | 2371 |

| Pressure [MPa] | 27 | 28 | 29 | 30 | 31 | 32 | 33 | 34 | 35 | 36 | 37 |
|---|---|---|---|---|---|---|---|---|---|---|---|
| Row1-IR | 2583 | 2901 | 2766 | 2998 | 2997 | 2095 | 2640 | 2919 | 2477 | 2854 | 2711 |
| Row1-OR | 2371 | 2725 | 2540 | 2821 | 2827 | 1912 | 2425 | 2742 | 2274 | 2679 | 2533 |
| Row2-IR | 2901 | 2766 | 2998 | 2997 | 2095 | 2640 | 2919 | 2477 | 2854 | 2711 | 2567 |
| Row2-OR | 2725 | 2540 | 2821 | 2827 | 1912 | 2425 | 2742 | 2274 | 2679 | 2533 | 2367 |

**Figure 2E**

Calculated bearing contact pressures for the flight test points in table 2E

| Load spectrum for test | | | *Units* | Case_1 | Case_2 | Case_3 | Case_4 | Case_5 |
|---|---|---|---|---|---|---|---|---|
| Contact pressure | Row1 | Inner ring | *[Mpa]* | 3 370 | 3 120 | 2 955 | 2 830 | 2 655 |
| | | Outer ring | *[Mpa]* | 3 205 | 2 950 | 2 785 | 2 660 | 2 485 |
| | Row2 | Inner ring | *[Mpa]* | 1 980 | 2 030 | 2 050 | 2 055 | 2 050 |
| | | Outer ring | *[Mpa]* | 1 795 | 1 850 | 1 870 | 1 880 | 1 880 |

**Figure 3E**

Calculated bearing contact pressures for the final rig test requested test points

| Pressure [MPa] | 1 | 2 | 3 | 4 | 5 | 6 |
|---|---|---|---|---|---|---|
| Row1-IR | 3235 | 936 | 1788 | 3102 | 2945 | 1251 |
| Row1-OR | 3066 | 849 | 1627 | 2932 | 2774 | 1139 |
| Row2-IR | 936 | 1788 | 3102 | 2945 | 1251 | 1732 |
| Row2-OR | 849 | 1627 | 2932 | 2774 | 1139 | 1581 |

**Figure 4E**

Calculated bearing contact pressures for the final rig test average actual applied loads
(Highest three positive and negative loads)

**Appendix F**

# CERTIFICATION SPECIFICATION E.515 - ENGINE CRITICAL PARTS

## E.515 – Engine Critical Parts

The integrity of the Engine Critical Parts identified under CS-E 510 must be established by:

(a) An Engineering Plan, the execution of which establishes and maintains that the combinations of loads, material properties, environmental influences and operating conditions, including the effects of parts influencing these parameters, are sufficiently well known or predictable, by validated analysis, test or service experience, to allow each Engine Critical Part to be withdrawn from service at an Approved Life before Hazardous Engine Effects can occur. Appropriate Damage Tolerance assessments must be performed to address the potential for Failure from material, manufacturing and service-induced anomalies within the Approved Life of the part. The Approved Life must be published as required in CS-E 25(b).

(b) A Manufacturing Plan which identifies the specific manufacturing constraints necessary to consistently produce Engine Critical Parts with the Attributes required by the Engineering Plan.

(c) A Service Management Plan which defines in-service processes for maintenance and repair of Engine Critical Parts which will maintain Attributes consistent with those required by the Engineering Plan. These processes must become part of the instructions for continued airworthiness.

## AMC E.515 Engine Critical Parts

(1) Introduction

Because the Failure of an Engine Critical Part is likely to result in a Hazardous Engine Effect, it is necessary to take precautions to avoid the occurrence of Failures of such parts. Under CS-E 510(c), they are required to meet prescribed integrity specifications.

For that purpose, an Engineering Plan, a Manufacturing Plan and a Service Management Plan are required under CS-E 515.

These three plans define a closed-loop system which link the assumptions made in the Engineering Plan to how the part is manufactured and maintained in service; the latter two aspects are controlled by the Manufacturing and Service Management Plans respectively. These plans may generate limitations which are published in the Airworthiness Limitation Section of the Instruction for Continued Airworthiness. This AMC provides means for the establishment of such plans.



**Appendix F cont**

(2) General

(a) Identification of Engine Critical Parts

The safety analysis required under CS-E 510 identifies Engine Critical Parts that are required to comply with CS-E 515. An Engine Critical Part is a Critical Part, by definition, with regard to compliance with Part 21.

If a part is made of various sub-parts, which are finally integrated in an inseparable manner into a unique part, and any one of the sub-parts is identified as an Engine Critical Part, the entire part is then treated as an Engine Critical Part.

(b) Attributes of a part

'Attributes' include, but are not limited to, material mechanical properties, material microstructure, material anomalies, residual stress, surface condition, and geometric tolerances. Processes such as alloy melting practise, ingot conversion to billet or bar, forging, casting, machining, welding, coating, shot peening, finishing, assembly, inspection, storage, repair, maintenance and handling may influence the Attributes of the finished part. Environmental conditions experienced in service may also affect the Attributes.

(c) Content of a plan

The Engineering Plan, Manufacturing Plan and Service Management Plan should provide clear and unambiguous information for the management of the Engine Critical Parts.

'Plan', in the context of this rule, does not necessarily mean having all technical information contained in a single document. If the relevant information exists elsewhere, the plan may make reference to drawings, material specifications, process specifications, manuals, etc, as appropriate.

It should be noted that these references should be clear enough to uniquely identify the referenced document. The plan should allow the history of the individual part number to be traced.

(3) Means for defining an Engineering Plan

(a) Introduction

The Engineering Plan consists of comprehensive life assessment processes and technologies that ensure that each Engine Critical Part can be withdrawn from service at a life before Hazardous Engine Effects can occur. These processes and technologies address the design, test validation, and certification aspects, and also define those manufacturing and service management processes that should be controlled in order to achieve the Engine Critical Part design intent.

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(b) Elements of the Engineering Plan

The Engineering Plan should address the following subjects:

- Analytical and empirical engineering processes applied to determine the Approved Life.
- Structured component and Engine testing conducted to confirm Engine internal operating conditions and to enhance confidence in the Approved Life.
- Establishment of the attributes to be provided and maintained for the manufacture and service management of Engine Critical Parts.
- Development and certification testing, and service experience required to validate the adequacy of the design and Approved Life. Any in-service inspections identified as critical elements to the overall part integrity, should be incorporated into the Service Management Plan.

(c) Establishment of the Approved Life – General

Determining the life capability of an Engine Critical Part involves the consideration of many separate factors, each of which may have a significant influence on the final results.

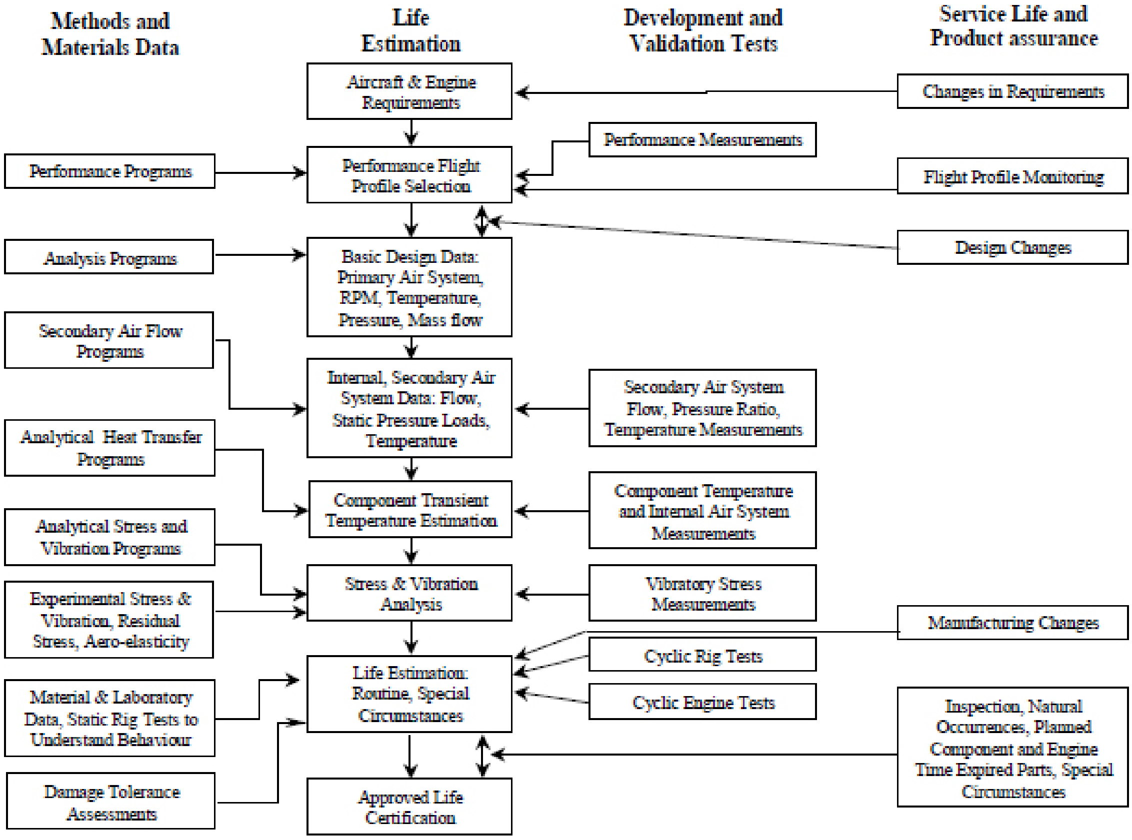
It is possible that the final life calculated may be in excess of that considered to be likely for the associated airframe application.

However, the life, in terms of cycles or hours, as appropriate, should still be recorded in the Airworthiness Limitations Section in order for the usage of the part to be properly tracked.

**Appendix F cont**

(d) Establishment of the Approved Life - Rotating parts

The following describes a typical process for establishing the Approved Life of rotating parts:




The major elements of the analysis are:

(i) Operating conditions.

For the purposes of certification, an appropriate flight profile or combination of profiles and the expected range of ambient conditions and operational variations will determine the predicted service environment. The Engine Flight Cycle should include the various flight segments such as start, idle, takeoff, climb, cruise, approach, landing, reverse and shutdown. The assumed hold times at the various flight segments should correspond to the assumed limiting installation variables (aircraft weight, climb rates, etc).

For Rotorcraft turbine Engines, the representative usage of the 30-minute Power rating should be considered in the Engine Flight Cycle when establishing the Approved Life of each Engine critical part. A maximum severity cycle that is known to be conservative may be used as an alternative.

The corresponding rotor speeds, internal pressures, and temperatures during each flight segment should be adjusted to account for Engine performance variation due to production tolerances and installation trim procedures, as well as Engine deterioration that can be expected between heavy maintenance intervals. The range of ambient temperature and take-off altitude conditions encountered during the Engines' service life as well as the impact of cold and hot Engine starts should also be considered.

The appropriateness of the Engine Flight Cycle should be validated and maintained over the lifetime of the design. The extent of the validation is dependent upon the approach taken in the development of the Engine Flight Cycle. For example, a conservative flight cycle where all the variables are placed at the most life-damaging value would require minimum validation, whereas a flight cycle which more accurately represents some portion of the actual flight profile but is inherently less conservative, would require more extensive validation. Further refinements may be applied when significant field operation data is gathered.

(ii) Thermal analysis.

Analytical and empirical engineering processes are applied to determine the Engine internal environment (temperatures, pressures, flows, etc.) from which the component steady-state and transient temperatures are determined for the Engine Flight Cycle. The Engine internal environment and the component temperatures should be correlated and verified experimentally during Engine development testing.

(iii) Stress analysis.

The stress determination is used to identify the limiting locations such as bores, holes, changes in section, welds or attachment slots, and the limiting loading conditions. Analytical and empirical engineering processes are applied to determine the stress distribution for each part. The analyses evaluate the effects on part stress of Engine speed, pressure, part temperature and thermal gradients at many discrete Engine cycle conditions.

From this, the part's cyclic stress history is constructed. All methods of stress analysis should be validated by experimental measurements.

(iv) Life analysis.

The life analysis combines the stress, strain, temperature and material data to establish the life of the minimum property part. Plasticity- and creep-related effects should also be considered. Relevant service experience gained through a successful programme of parts retirement or precautionary sampling inspections, or both, may be included to adjust the life prediction system.

The fatigue life prediction system is based upon test data obtained from cyclic testing of representative laboratory, subcomponent, or specific component specimens and should account for the manufacturing processes that affect low-cycle fatigue (LCF) capability, including fabrication from production grade material. Sufficient testing should be performed to evaluate the effects of elevated temperatures and hold times, as well as interaction with other material Failure mechanisms such as high-cycle fatigue and creep. The fatigue life prediction system should also account for environmental effects, such as vibration and corrosion, and cumulative damage.

When the fatigue life is based on cyclic testing of specific parts, the test results should be corrected for inherent fatigue scatter. The factors used to account for scatter should be justified. In order to utilise this approach, the test should be designed to be representative of the critical Engine conditions in terms of temperature and stress at the specific features, e.g. bore, rim or blade attachment details, of the part being tested. Appropriate analytical and empirical tools should be utilised such that the fatigue life can be adjusted for any differences between the Engine conditions and cyclic test. In the event the test is terminated by burst or complete Failure, crack initiation for this particular test may be defined using the appropriate crack growth calculations and/or fracture surface observations. It may also be possible to utilise the number of cycles at the last crack-free inspection to define the crack initiation point. This approach requires an inspection technique with a high level of detection capability consistent with that used by the Engine industry for rotating parts.

The test data should be reduced statistically in order to express the results in terms of minimum LCF capability (1/1000 or alternately -3 sigma). The fatigue life should be determined as a minimum life to initiation of a fatigue crack, defined typically as a crack length of 0.75mm.

An alternative way of using the data is to base the fatigue life on an agreed safety margin to burst of a minimum strength part. Typically a 2/3 factor has been applied to the minimum (1/1000 or alternatively -3 sigma) burst life; however, any factor used should be justified for a particular material.

(v) Damage Tolerance Assessment.

Damage Tolerance Assessments should be performed to minimise the potential for Failure from material, manufacturing- and service-induced anomalies within the Approved Life of the part. Service experience with gas turbine Engines has demonstrated that material-, manufacturing- and service-induced anomalies do occur which can potentially degrade the structural integrity of Engine Critical Parts. Historically, life management methodology has been founded on the assumption of the existence of nominal material variations and manufacturing conditions. Consequently, the methodology has not explicitly addressed the occurrence of such anomalies, although some level of tolerance to anomalies is implicitly built-in using design margins, factory and field inspections, etc. A Damage Tolerance Assessment explicitly addresses the anomalous condition(s) and complements the fatigue life prediction system. It should be noted that the 'Damage Tolerance Assessment' is part of the design process and not a method for returning cracked parts to service whilst monitoring crack growth.

The Damage Tolerance Assessment process typically includes the following primary elements:

Anomaly size and frequency distributions.

A key input in the Damage Tolerance Assessment is the size and rate of occurrence of the anomalies. This type of information may be statistical in nature and can be presented in a form that plots a number of anomalies that exceed a particular size in a specified amount of material. Anomalies should be treated as sharp propagating cracks from the first stress cycle unless there is sufficient data to indicate otherwise.

Crack growth analysis.

This determines the number of cycles for a given anomaly to grow to a critical size.

This prediction should be based upon knowledge of the part stress, temperature, geometry, stress gradient, anomaly size and orientation, and material properties. The analysis approach should be validated against relevant test data.

Inspection techniques and intervals.

Manufacturing and in-service inspections are an option to address the fracture potential from inherent and induced anomalies. The intervals for each specified in- service inspection should be identified. Engine removal rates and module and piece part availability data could serve as the basis for establishing the inspection interval. The manufacturing inspections assumed in the Damage Tolerance Assessments should be incorporated into the Manufacturing Plan. Likewise, the assumed in-service inspection procedures and intervals should be integrated into the Service Management Plan and included, as appropriate, in the Airworthiness Limitations Section of the Instructions for Continued Airworthiness.

Inspection Probability of Detection (POD).

The Probability of Detection (POD) of the individual inspection processes, such as eddy-current, penetrant fluid or ultrasonic, used to detect potential anomalies should be based upon the statistical review of sufficient quantities of relevant testing or experience. The relevance of this data should be based upon the similarity of parameters such as:

**Appendix F cont**

- the size, shape, orientation, location, and chemical or metallurgical character of the anomaly;
- the condition of the surface condition and cleanliness of the parts;
- the material being inspected (such as its composition, grain size, conductivity, surface texture, etc.);
- variations in the inspection materials or equipment (such as the specific penetrant fluid and developer, equipment capability or condition, etc.);
- specific inspection process parameters such as the scan index;
- the inspector (such as their visual acuity, attention span, training, etc.).

In addition, the following should be noted with regard to the above:

- appropriate Damage Tolerance Assessments.

In the context of CS-E 515(a), "appropriate Damage Tolerance Assessments" recognises that industry standards on suitable anomaly size and frequency distributions, and analysis techniques used in the Damage Tolerance Assessment process are not available in every case listed in the paragraphs below. In such cases, compliance with the rule should be based on such considerations as the design margins applied, application of damage tolerance design concepts, historical experience, crack growth rate comparisons to successful experience, etc. Anomalies for which a common understanding has been reached within the Engine community and the Authorities should be considered in the analysis.

Material anomalies.

Material anomalies consist of abnormal discontinuities or non-homogeneities introduced during the production of the input material or melting of the material. Some examples of material anomalies that should be considered are hard alpha anomalies in titanium, oxide/carbide (slag) stringers in nickel alloys, and ceramic particulate anomalies in powder metallurgy materials unintentionally generated during powder manufacturing.

Manufacturing anomalies.

Manufacturing anomalies include anomalies produced in the conversion of the ingot-to-billet and billet-to-forging steps as well as anomalies enerated by the metal removal and finishing processes used during manufacture and/or repair. Examples of conversion-related anomalies are forging lapsand strain-induced porosity. Some examples of metal-removal-related anomalies are tears due to broaching, arc burns

Appendices

from various sources and disturbed microstructure due to localised overheating of the machined surface.

Service-induced anomalies.

Service-induced anomalies such as non-repaired nicks, dings and scratches, corrosion, etc., should be considered. Similarity of hardware design, installation, exposure and maintenance practice should be used to determine the relevance of the experience.

(e) Establishment of the Approved Life - Static, pressure loaded parts

(i) General Principles

The general principles which are used to establish the Approved Life are similar to those used for rotating parts.

However, for static pressure loaded parts, the Approved Life may be based on the crack initiation life plus a portion of the residual crack growth life. The portion of the residual life used should consider the margin to burst. If the Approved Life includes reliance on the detection of cracks prior to reaching the Approved Life, the reliability of the crack detection should be considered. If, as part of the Engineering Plan any dependence is placed upon crack detection to support the Approved Life, this should result in mandatory inspections being included in the Service Management Plan and in the Airworthiness Limitations Section of the Instructions for Continued Airworthiness. Crack growth analysis techniques should be validated experimentally.

Some construction techniques, such as welding or casting, contain inherent anomalies. Such anomalies should be considered as part of the methodology to establish the Approved Life. Fracture mechanics is a common method for such assessments.

In determining the life of the part, the temperature of the part, any temperature gradients, any significant vibratory or other loads (for example, flight manoeuvre) should be taken into account in addition to the pressure loads.

Manufacturing and in-service inspections are an option to address the potential for fracture. The intervals for each specified in-service inspection should be identified. Engine removal rates and module and piece part availability data could serve as the basis for establishing the inspection interval. The manufacturing inspections should be incorporated into the Manufacturing Plan. Likewise, the assumed in-service inspection procedures and intervals should be integrated into the Service Management Plan and included, as appropriate, in the Airworthiness Limitations Section of the Instructions for Continued Airworthiness.

(ii) Tests

When using testing as part of the substantiation of the life of the part, the basic load cycle should be from substantially zero differential pressure to a value that simulates the most critical operation stress condition and returning to substantially zero differential pressure. When a test is performed, the test pressure level should be adjusted to include the effects of stress due to thermal gradients in actual operation. When this is impossible, due to over-stress of regions other than the critical location or stress reversal in the Engine Flight Cycle for example, the fatigue capability in operation should be established by an additional analysis.

If the part is subject to loads in addition to those resulting from differential pressure (e.g. flight manoeuvre loads, Engine mounting loads, etc.), an analysis should be made of these additional loads and their effect examined. If the effect of these loads is small, it may be possible to simulate them by an addition to the test pressure differential. However, if the loads are of significant magnitude or cannot adequately be represented by a pressure increment, the test should be carried out with such loads acting in addition to the pressure loads.

The part should be tested at the temperature associated with the most critical stress case or alternatively the test pressure differential may be increased to simulate the loss of relevant properties as a result of temperature.

Any fatigue scatter factors used should be justified.

During pressure testing, the methods of mounting and restraint by the test facility or test equipment of any critical section should be such as to simulate the actual conditions occurring on the Engine.

(iii) Analytical Modelling Methods

An analytical modelling method may be used to determine the adequate fatigue life, provided that the modelling method is validated by testing or successful field experience with parts of similar design.

(f) Establishment of the Approved Life - Other Parts

It is possible that the Safety Analysis required by CS-E 510 may identify Engine Critical Parts other than rotating parts or static pressure loaded parts.

In such instances, a methodology for determining the Approved Life will need to be agreed with the Authority, using the general principles for rotating and static pressure loaded parts as a guideline.

(g) Maintaining the Approved Life

At certification, the Approved Life is based on predictions of the Engine operation, material behaviour, environment, etc., which all can be expected to influence the life at which the part should be withdrawn from service to avoid Hazardous Engine Effects.

After certification, it may be necessary to check the accuracy of such predictions, recognising that many aspects, for example, the usage of the Engine and its operating environment, may change during its operational life, especially with a change of ownership. It is important to use any service feedback to confirm that any assumptions made in the Engineering Plan remain valid, or are modified if required. The Engineering Plan should describe not only the basis of the Approved Life, but also those actions subsequent to certification, which will be necessary to ensure that the Approved Life is appropriate throughout the operational life of the Engine.

A regular review of the assumptions made when establishing the Approved Life may be required, depending on the conservative nature of the assumptions made when determining the Approved Life. The Engineering Plan should detail when such reviews should occur and what information will be required in order to complete the review. Aspects which may be considered include, but need not be limited to:

- the frequency of Approved Life reviews;
- detailed inspection of service run parts, including time-expired parts;
- review of flight plans;
- findings during maintenance;
- Engine development experience;
- lessons learned from other engine projects;
- any in-service events.

(h) Influencing Parts

Engine Critical Parts are part of a complex system and other parts of the Engine can have an impact on the Engine Critical Parts and their life capability. Therefore, the Engineering Plan needs to address these parts, and particularly changes to them. Examples of influencing parts include a turbine blade, a mating part, and a static part that impacts on the environment (temperatures, pressures, etc.) around the Engine Critical Part. Examples of changes to influencing parts include a blade with a different weight, centre of gravity, or root coating; a mating part made of a material that has a different coefficient of thermal expansion; and a static part where changes in geometry or material modify the thermal and/or mechanical response of the component and could, as a result, affect the environment around the Engine Critical Part.

(4) Means for Defining a Manufacturing Plan

(a) Introduction

The Manufacturing Plan is a portion of the overall integrity process intended to ensure the life capability of the part. The Engineering Plan includes assumptions about how Engine Critical Parts are designed, manufactured, operated and maintained: each can have an impact on the part life capability. Therefore, it is essential to ensure that the Attributes required by the Engineering Plan are maintained.

(b) Elements of a Manufacturing Plan

The part specific Manufacturing Plan should consider the Attributes of the part delivered by the manufacturing process from raw material to finished part and should highlight all sensitive parameters identified as being significant with regard to part life which should not be changed without proper verification. Such parameters may include, but may not be limited to: material controls, including any zoned areas for special properties, manufacturing method specifications, manufacturing method order of application, inspection method and sensitivity, and any special part rough machining methods or finishing method(s), especially any methods intended to improve fatigue capability or minimise induced anomalies.

(c) Development and Verification of the Manufacturing Plan

The Manufacturing Plan should be reviewed and verified by the following key Engineering and Manufacturing skills:

- Engineering (Design & Lifing)
- Material Engineering Non-Destructive Inspection
- Quality Assurance
- Manufacturing Engineering (Development & Production)

Hence, this same skill mix should evaluate and approve process validation and the procedures for manufacturing change control and non-conformance disposition to ensure that the product of manufacturing is consistent with the design assumptions of the Engineering Plan. The intent is that:

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- Manufacturing processes are developed and applied with the appropriate level of oversight to ensure the part life capability assumed in the Engineering Plan is consistently achieved. Substantiation programmes are agreed up-front and executed as part of the process validation.
- Changes to such manufacturing processes and practices are visible and are not made without cross functional review and approval.
- When a suspected non-conformance event occurs, it is reviewed with the appropriate skill mix prior to disposition.

The level of detail in the Plan may vary depending on the specific process step being considered, the sensitivity of the particular process step, and the level of control required to achieve the required life capability.

For instance, consider the case where a process specification exists to control the drilling of holes. If the use of this specification produces a hole that meets the life capability specifications for a flange bolt hole, the plan may simply note that the flange bolt hole will be produced per the specification. However, if a rim air hole requires cold expansion, after drilling per the specification, to meet the life capability specifications, it may be necessary to reference the cold expansion process in the plan.

(5) Means for defining a Service Management Plan.

(a) Introduction

The Service Management Plan forms part of the overall process intended to maintain the integrity of Engine Critical Parts throughout their service life.

The Engineering Plan includes assumptions about the way in which the Engine Critical Parts are manufactured, operated and maintained: each can have an impact on the life capability of the part. Therefore, it is essential to ensure that these assumptions remain valid. The Service Management Plan conveys the processes for in-service repair and maintenance to remain consistent with the assumptions made in the Engineering Plan.

(b) Determining the acceptability of repair and maintenance processes

Repair and maintenance processes should be reviewed by the following key skills:

- Engineering (Design & Lifing)
- Material Engineering
- Non-Destructive Inspection

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- Quality Assurance
- Product Support Engineering
- Repair Development Engineering The role of this cross-functional review is consistent with that laid out for the Manufacturing Plan. The review should include process validation, change control and non-conformance to ensure the product of any repair or maintenance is consistent with the engineering specification. The intent is that:
- Repair and maintenance processes and practices are developed with the appropriate level of oversight, and with due regard to their possible impact on the life capability of the part. Substantiation programmes are agreed up-front and executed as part of the validation process.
- Changes to such processes and practices are visible to all parties, and are not made without cross-functional review and approval.
- When a suspected non-conformance event occurs, it is reviewed with the appropriate skill mix prior to disposition.

To achieve the necessary control of the application of those processes and practices, the procedures for repair and maintenance should be clearly articulated in the appropriate section(s) of the engine shop manual.

These procedures should also include clearly delineated limits to these processes and practices that will ensure that Engine Critical Parts maintain attributes consistent with those assumed in the Engineering Plan.

(c) Service Management Aspects of Static Pressure Loaded Parts or Other Parts

The difference in approach to lifing for static pressure loaded parts or other parts means that in addition to the Approved Life, instructions for continued airworthiness may typically contain:

- A defined periodic inspection interval in the airworthiness limitations section.
- The inspection method(s) to be used.
- A detailed description of the area(s) to be inspected.
- Inspection result acceptability limits.
- Acceptable repair methods, if applicable.

- Any other instructions necessary to carry out the required inspection and allowable maintenance procedures.

(6) Airworthiness Limitations Section

(a) To ensure a closed-loop between the in-service parts and the Engineering Plan, the importance of the limits to the repair and maintenance of Engine Critical Parts should be highlighted in the Engine manuals required by CS-E 25. Further, since inappropriate repair or maintenance could impact the integrity of the part in a hazardous manner, visibility should be provided through the airworthiness limitations section (ALS) of instructions for continued airworthiness. Wording as, or similar to, that shown below should be placed in the appropriate section of the ALS.

"The following airworthiness limitations have been substantiated based on engineering analysis that assumes this product will be operated and maintained using the procedures and inspections provided in the instructions for continued airworthiness supplied with this product by the Type Certificate holder, or its licensees.

For Engine Critical Parts and parts that influence Engine Critical Parts, any repair, modification or maintenance procedures not approved by the Type Certificate holder, or its licensees, or any substitution of such parts not supplied by the Type Certificate holder, or its licensees, may materially affect these limits."

(b) For engines with OEI ratings, the airworthiness limitations section should include a method for accounting for the number of cycles used in operation at the OEI ratings. This may be accomplished by adding a finite number of cycles to the expended life of the affected Engine Critical Parts or by using appropriate life reduction factors for each of the OEI power excursions.

**Appendix G**

## NOTICE OF PROPOSED AMENDMENT 2022/01

The following text, relevant to this investigation, was proposed to amend CS 29, Amendment 10 in NPA 22/01122F[1].

**CS 29.602**

(a) Critical part - A critical part is a part, the failure of which could have a catastrophic effect upon the rotorcraft, and for which critical characteristics have been identified which must be controlled to ensure the required level of integrity.

(b) If the type design includes critical parts, a critical parts list must be established. Procedures must be established to define the critical design characteristics, identify processes that affect those characteristics, and identify the design change and process change controls necessary for showing compliance with the quality assurance requirements of Part 21.

(c) As part of the process of compliance with this paragraph, a continued integrity verification programme (CIVP) shall be developed. The CIVP should ensure the continued validity of assumptions made during certification that could affect the integrity of Critical Parts.

**AMC1 29.602 Critical parts**

This AMC supplements FAA AC 29-2C, § AC 29.602 and should be used in conjunction with that AC when demonstrating compliance with CS 29.602.

(a) Explanation

The continued integrity verification programme (CIVP) should address all critical parts. In addition, it may also include other parts the failure of which could have a catastrophic effect upon the rotorcraft and for which no critical characteristics have been identified at the time of certification. Actions arising from a finding in a CIVP could in the future change the certification approach for similar components or lead to a continued airworthiness action.

(b) Procedures

(1) The CIVP should assess the continued validity of assumptions made during certification regarding the condition and operation of critical parts in order to help ensure their continued integrity. This should include but not be limited to demonstration of the continuity of the effectiveness of design, maintenance and monitoring provisions (e.g. health monitoring, usage monitoring and safety devices) developed to comply with CS 29.547(b), CS 29.571,

1 These are just extracts of the sections relevant to the investigation. The full NPA text can be viewed on the EASA website www.easa.europa.eu.



 

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CS 29.573 and CS 29.917(b) through the life of the type design.
(2) The following data can be used to support the CIVP:

(i) analysis of occurrence reports;

(ii) analysis of unscheduled removal rates;

(iii) results of scheduled maintenance;

(iv) strip reports / analysis at overhaul;

(v) post-TC development and maturity tests;

(vi) additional inspection (non-destructive and/or destructive) and testing on selected high time or rejected components;

(vii) feedback from lead customers;

(viii) audits of subcontractors and suppliers of critical parts;

(ix) statistical process control data of manufacturing processes affecting critical characteristics;

(x) review of concessions;

(xi) changes in utilisation and operating environment;

(xii) operator / applicant working group activities;

(xiii) health monitoring data; and

(xiv) usage monitoring data.

(3) The assessments required by the CIVP, as described above, should be performed at suitable periods through the complete life of the subject component types, considering the types of operation, environment and ageing effects expected.

To meet this objective, an evaluation will need to be performed on at least one sample of each component at each major inspection interval or overhaul, and at retirement time, as applicable. In addition, the applicant should consider scheduling early evaluation opportunities to confirm the suitability of the inspection intervals scheduled at entry into service. Consideration should be given to adding new samples and revising the CIVP when changes to the types of operation or environment occur. Where inspections and feedback from service need to be provided by operators or maintenance organisations, the information necessary should be clearly specified by the applicant within the continued integrity verification programme plan (CIVPP) and relevant maintenance instructions.

(4) A CIVPP, defining the tasks and schedule of the CIVP should be agreed during certification. Reports stating the findings of the CIVP during service should be furnished to the Agency. The CIVPP may be revised during the life of the rotorcraft if considered to be appropriate by the applicant and agreed by the Agency. On conclusion of the CIVP, an assessment of all findings should be made by the applicant and reported in the continued integrity verification programme report (CIVPR). The applicant should consider the participation of an operator for review of the CIVPR.

(5) Additionally, the CIVP could be used to verify the continued validity of compensating provisions identified in the design assessments required by 29.547(b) and 29.917(b) and their associated assumptions.

**AMC1 29.571 Fatigue tolerance evaluation of metallic structure**

FATIGUE EVALUATION OF ROTOR DRIVE SYSTEM COMPONENTS

This AMC supplements FAA AC 29-2C, § AC 29.571 and should be used in conjunction with that AC when demonstrating compliance with CS 29.571.

(a) Definitions

(1) Rolling contact fatigue (RCF): a form of fatigue that occurs due to the cyclic strains arising from the loading present during rolling contact between two parts of an assembly, e.g. a bearing race and a rolling element.

Note: For the purposes of this AMC, RCF also includes combinations of rolling and sliding contact phenomena.

(2) Integral race: a bearing race that is an integral part of the transmission structural component such as a gear or shaft.

(b) Explanation

Service experience has shown that RCF can initiate cracks in integral bearing races of rotor drive system structural elements that, in some cases, can propagate to a failure with catastrophic results. It is often assumed that RCF leads first to failure modes such as micro-pitting and spalling that will be detected before more severe failure modes can develop. The procedures of this AMC are intended to help ensure that the effects of RCF are accounted for in the fatigue tolerance evaluations required by CS 29.571.

(c) Procedure

The fatigue tolerance evaluation of rotor drive system principal structural elements (PSEs) should include, when applicable, the combined effect of RCF and other damage threats such as dents, scratches, corrosion, loss of preload in bearings or joints, surface and sub-surface material defects, etc., considering residual stress coming from surface treatments and other manufacturing processes and all other applicable loading conditions. Particular attention should be paid to evaluation of components with integral bearing races.

Steps should be taken to minimise the risk of crack initiation due to RCF in integrated races by minimising contact stresses, specifying high standards for surface finishes, ensuring good lubrication and maintaining oil quality regardless of the fatigue tolerance approach selected. Experience has demonstrated that it can be beneficial for bearings to be designed so that the reliability of the integrated race of the PSE is even higher than the less critical race of the bearing. In this way, degradation of the less critical race can lead to detection of the bearing failure before cracking initiates in the integrated race. The consequences of damage to the integrated race from the debris generated in such scenarios should be considered in the evaluation.

As it is difficult to totally preclude cracking initiated by RCF, a fail-safe approach is recommended wherever possible, such that failure or partial failure due to cracking of the rotor drive system structural element is detected prior to its residual strength capability falling below the required levels prescribed in CS 29.571(f).

This method using analysis supported by test ensures that, should fatigue cracks initiate, the remaining structure will withstand service loads and limit loads without failure until the cracks are detected. Analysis, experience with similar designs and testing should be used to verify any assumptions related to the way the crack or cracks develop in the structure from potential surface and sub-surface origins and whether a through crack may develop and its relationship with other forms of damage including spalling. In addition, the continued safe operation of the gearbox should be ensured for this period considering the effect of the cracking on stiffness, dynamic behaviour, loads and functional performance.

The effectiveness and reliability of means of crack detection for the fail-safe approach, including indirect means of detection such as chip detection systems, and associated instructions for continued airworthiness should be evaluated to show that, if implemented as required, they will result in timely detection and repair or replacement of damaged components. In addition, the instructions for continued airworthiness, prescribing the maintenance actions leading up to and following detection of potential damage should be substantiated sufficiently to ensure timely repair or replacement of damaged components. The substantiation should consider aspects such as threshold criteria on indicators of means of detection for additional investigative actions and removal from service of the damaged parts, the overall clarity and practicality of the instructions for continued airworthiness and human factors aspects.

A continued integrity verification programme (CIVP), as prescribed in CS 29.602(c), should be implemented to monitor critical parts and may be extended to all PSEs (see AMC1 29.602) subject to RCF to ensure assumptions supporting the compliance demonstration remain valid throughout the operational life of the component.

**AMC1 29.1309 Equipment, systems, and installations**

As stated in AMC 29, the AMC to CS-29 consists generally of FAA AC 29-2C Change 7, dated 4 February 2016. This AMC supplements AC 29-2C, § AC 29.1309 and should be used in conjunction with that AC when demonstrating compliance with CS 29.1309.

**Development assurance process**

Any analysis necessary to show compliance with CS 29.1309(b) should consider the possibility of development errors and should focus on minimising the likelihood of those errors.

Errors made during the development of systems have traditionally been detected and corrected by exhaustive tests conducted on the system and its components, by direct inspection, and by other direct verification methods capable of completely characterising the performance of the system.

These tests and direct verification methods may be appropriate for systems containing non-complex items (i.e. items that are fully assured by a combination of testing and analysis) that perform a limited number of functions and that are not highly integrated with other rotorcraft systems. For more complex or integrated systems, exhaustive testing may either be impossible because not all system states can be determined or impractical because of the number of tests that must be accomplished. For these types of systems, compliance may be demonstrated using development assurance.

The applicability of system development assurance should also be considered for modifications to previously certificated aircraft.



 

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ED-79A/ARP4754A is recognised as providing acceptable guidelines for establishing a development assurance process from aircraft and systems levels down to the level where software/airborne electronic hardware (AEH) development assurance is applied.

The extent of application of ED-79A/ARP4754A to substantiate development assurance activities depends on the complexity of the systems and on their level of interaction with other systems.

(a) Software development assurance

This AMC recognises AMC 20-115 as an accepted means of compliance with CS 29.1309 (a) and (b).

(b) AEH development assurance

This AMC recognises AMC 20-152 as an acceptable means of compliance with the requirements in CS 29.1309 (a) and (b).

(c) Open problem report management This AMC recognises AMC 20-189 as an acceptable means of compliance for establishing an open problem report management process for the system, software and AEH domains.

**Appendix H**

## Extracts from EASA Certification Specification 29 Amendment 2

This was the wording of CS 29 at the time the AW169 was certified for regulations referenced in this report.

**CS 29.547 Main and tail rotor structure**

(a) A rotor is an assembly of rotating components, which includes the rotor hub, blades, blade dampers, the pitch control mechanisms, and all other parts that rotate with the assembly.

(b) Each rotor assembly must be designed as prescribed in this paragraph and must function safely for the critical flight load and operating conditions. A design assessment must be performed, including a detailed failure analysis to identify all failures that will prevent continued safe flight or safe landing, and must identify the means to minimise the likelihood of their occurrence.

(c) The rotor structure must be designed to withstand the following loads prescribed in CS 29.337 to 29.341, and CS 29.351:

(1) Critical flight loads.

(2) Limit loads occurring under normal conditions of autorotation.

(d) The rotor structure must be designed to withstand loads simulating:

(1) For the rotor blades, hubs and flapping hinges, the impact force of each blade against its stop during ground operation; and

(2) Any other critical condition expected in normal operation.

(e) The rotor structure must be designed to withstand the limit torque at any rotational speed, including zero. In addition:

(1) The limit torque need not be greater than the torque defined by a torque limiting device (where provided), and may not be less than the greater of:

(i) The maximum torque likely to be transmitted to the rotor structure, in either direction, by the rotor drive or by sudden application of the rotor brake;
and

(ii) For the main rotor, the limit engine torque specified in CS 29.361.

(2) The limit torque must be equally and rationally distributed to the rotor blades.

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**CS 29.561 Emergency landing conditions**

**General**

(a) The rotorcraft, although it may be damaged in emergency landing conditions on land or water, must be designed as prescribed in this paragraph to protect the occupants under those conditions.

(b) The structure must be designed to give each occupant every reasonable chance of escaping serious injury in a crash landing when:

(1) Proper use is made of seats, belts, and other safety design provisions;

(2) The wheels are retracted (where applicable); and

(3) Each occupant and each item of mass inside the cabin that could injure an occupant is restrained when subjected to the following ultimate inertial load factors relative to the surrounding structure:

(i) Upward - 4 g

(ii) Forward - 16 g

(iii) Sideward - 8 g

(iv) Downward - 20 g, after the intended displacement of the seat device

(v) Rearward - 1.5 g.

(c) The supporting structure must be designed to restrain under any ultimate inertial load factor up to those specified in this paragraph, any item of mass above and/or behind the crew and passenger compartment that could injure an occupant if it came loose in an emergency landing. Items of mass to be considered include, but are not limited to, rotors, transmission and engines. The items of mass must be restrained for the following ultimate inertial load factors:

(1) Upward - 1.5 g

(2) Forward - 12 g

(3) Sideward - 6 g

(4) Downward - 12 g

(5) Rearward - 1.5 g

(d) Any fuselage structure in the area of internal fuel tanks below the passenger floor level must be designed to resist the following ultimate inertia factors and loads, and to protect the fuel tanks from rupture, if rupture is likely when those loads are applied to that area:

(1) Upward - 1.5 g

(2) Forward - 4.0 g

(3) Sideward - 2.0 g

(4) Downward - 4.0 g

**CS 29.562 Emergency landing dynamic conditions**

(a) The rotorcraft, although it may be damaged in a crash landing, must be designed to reasonably protect each occupant when:

(1) The occupant properly uses the seats, safety belts, and shoulder harnesses provided in the design; and

(2) The occupant is exposed to loads equivalent to those resulting from the conditions prescribed in this paragraph.

(b) Each seat type design or other seating device approved for crew or passenger occupancy during take-off and landing must successfully complete dynamic tests or be demonstrated by rational analysis based on dynamic tests of a similar type seat in accordance with the following criteria. The tests must be conducted with an occupant simulated by a 77 kg (170pound) anthropomorphic test dummy (ATD), sitting in the normal upright position.

(1) A change in downward velocity of not less than 9.1 metres per second (30 ft/s) when the seat or other seating device is oriented in its nominal position with respect to the rotorcraft's reference system, the rotorcraft's longitudinal axis is canted upward 60°, with respect to the impact velocity vector, and the rotorcraft's lateral axis is perpendicular to a vertical plane containing the impact velocity vector and the rotorcraft's longitudinal axis. Peak floor deceleration must occur in not more than 0.031 seconds after impact and must reach a minimum of 30 g.

(2) A change in forward velocity of not less than 12.8 metres per second (42 ft/s) when the seat or other seating device is oriented in its nominal position with respect to the rotorcraft's reference system, the rotorcraft's longitudinal axis is yawed 10°, either right or left of the impact velocity vector (whichever would cause the greatest load on the shoulder harness), the rotorcraft's lateral axis is contained in a horizontal plane

containing the impact velocity vector, and the rotorcraft's vertical axis is perpendicular to a horizontal plane containing the impact velocity vector. Peak floor deceleration must occur in not more than 0.071 seconds after impact and must reach a minimum of 18.4 g.

(3) Where floor rails or floor or sidewall attachment devices are used to attach the seating devices to the airframe structure for the conditions of this paragraph, the rails or devices must be misaligned with respect to each other by at least 10° vertically (i.e. pitch out of parallel) and by at least a 10° lateral roll, with the directions optional, to account for possible floor warp.

(c) Compliance with the following must be shown:

(1) The seating device system must remain intact although it may experience separation intended as part of its design.

(2) The attachment between the seating device and the airframe structure must remain intact, although the structure may have exceeded its limit load.

(3) The ATD's shoulder harness strap or straps must remain on or in the immediate vicinity of the ATD's shoulder during the impact.

(4) The safety belt must remain on the ATD's pelvis during the impact.

(5) The ATD's head either does not contact any portion of the crew or passenger compartment, or if contact is made, the head impact does not exceed a head injury criteria (HIC) of 1000 as determined by this equation.

$$HIC = (t_2 - t_1)\left[\frac{1}{(t_2 - t_1)}\int_{t1}^{t2} a(t)dt\right]^{2.5}$$

Where – a(t) is the resultant acceleration at the centre of gravity of the head form expressed as a multiple of g (the acceleration of gravity) and t2–t1 is the time duration, in seconds, of major head impact, not to exceed 0.05 seconds.

(6) Loads in individual shoulder harness straps must not exceed 7784 N (1750 lbs). If dual straps are used for retaining the upper torso, the total harness strap loads must not exceed 8896 N (2000 lbs).

(7) The maximum compressive load measured between the pelvis and the lumbar column of the ATD must not exceed 6674 N (1500 lbs).

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(d) An alternate approach that achieves an equivalent or greater level of occupant protection, as required by this paragraph, must be substantiated on a rational basis.

**CS 29.571 Fatigue evaluation of structure**

(a) General. An evaluation of the strength of principal elements, detail design points, and fabrication techniques must show that catastrophic failure due to fatigue, considering the effects of environment, intrinsic/discrete flaws, or accidental damage will be avoided. Parts to be evaluated include, but are not limited to, rotors, rotor drive systems between the engines and rotor hubs, controls, fuselage, fixed and movable control surfaces, engine and transmission mountings, landing gear, and their related primary attachments. In addition, the following apply:

(1) Each evaluation required by this paragraph must include:

(i) The identification of principal structural elements, the failure of which could result in catastrophic failure of the rotorcraft;

(ii) In-flight measurement in determining the loads or stresses for items in sub-paragraph (a)(1)(i) in all critical conditions throughout the range of limitations in CS 29.309 (including altitude effects), except that manoeuvring load factors need not exceed the maximum values expected in operations; and

(iii) Loading spectra as severe as those expected in operation based on loads or stresses determined under sub paragraph (a)(1)(ii), including external load operations, if applicable, and other high frequency power cycle operations.

(2) Based on the evaluations required by this paragraph, inspections, replacement times, combinations thereof, or other procedures must be established as necessary to avoid catastrophic failure. These inspections, replacement times, combinations thereof, or other procedures must be included in the airworthiness limitations section of the instructions for continued airworthiness required by CS 29.1529 and paragraph A29.4 of appendix A.

(b) *Fatigue tolerance evaluation (including tolerance to flaws).* The structure must be shown by analysis supported by test evidence and, if available, service experience to be of fatigue tolerant design. The fatigue tolerance evaluation must include the requirements of either sub paragraph (b)(l), (2), or (3), or a combination thereof, and also must include a determination of the probable locations and modes of damage caused by fatigue, considering environmental effects, intrinsic/discrete flaws, or accidental damage. Compliance with the flaw tolerance requirements of sub-paragraph (b) (1) or (2) is required unless it is established that these fatigue flaw tolerant methods for a particular structure cannot be achieved within the limitations of geometry, inspectability, or good design practice. Under these circumstances, the safe-life evaluation of sub-paragraph (b)(3) is required.

(1) *Flaw tolerant safelife evaluation.* It must be shown that the structure, with flaws present, is able to withstand repeated loads of variable magnitude without detectable flaw growth for the following time intervals:

(i) Life of the rotorcraft; or

(ii) Within a replacement time furnished under paragraph A29.4 of appendix A.

(2) Failsafe (residual strength after flaw growth) evaluation. It must be shown that the structure remaining after a partial failure is able to withstand design limit loads without failure within an inspection period furnished under paragraph A29.4 of appendix A. Limit loads are defined in CS 29.301(a).

(i) The residual strength evaluation must show that the remaining structure after flaw growth is able to withstand design limit loads without failure within its operational life.

(ii) Inspection intervals and methods must be established as necessary to ensure that failures are detected prior to residual strength conditions being reached.

(iii) If significant changes in structural stiffness or geometry, or both, follow from a structural failure or partial failure, the effect on flaw tolerance must be further investigated.

(3) Safelife evaluation. It must be shown that the structure is able to withstand repeated loads of variable magnitude without detectable cracks for the following time intervals:

(i) Life of the rotorcraft; or

(ii) Within a replacement time furnished under Paragraph A29.4 of appendix A.

**CS 29.602 Critical parts**

(a) Critical part - A critical part is a part, the failure of which could have a catastrophic effect upon the rotorcraft, and for which critical characteristics have been identified which must be controlled to ensure the required level of integrity.

(b) If the type design includes critical parts, a critical parts list shall be established. Procedures shall be established to define the critical design characteristics, identify processes that affect those characteristics, and identify the design change and process change controls necessary for showing compliance with the quality assurance requirements of Part-21.

**CS 29.607 Fasteners**

(a) Each removable bolt, screw, nut, pin or other fastener whose loss could jeopardise the safe operation of the rotorcraft must incorporate two separate locking devices. The fastener and its locking devices may not be adversely affected by the environmental conditions associated with the particular installation.

(b) No self-locking nut may be used on any bolt subject to rotation in operation unless a non-friction locking device is used in addition to the self-locking device.

**CS 29.923 Rotor drive system and control mechanism tests**

(a) *Endurance tests, general*. Each rotor drive system and rotor control mechanism must be tested, as prescribed in sub-paragraphs (b) to (n) and (p), for at least 200 hours plus the time required to meet the requirements of sub paragraphs (b)(2), (b)(3) and (k). These tests must be conducted as follows:

(1) Ten-hour test cycles must be used, except that the test cycle must be extended to include the OEI test of sub-paragraphs (b)(2) and (k), if OEI ratings are requested.

(2) The tests must be conducted on the rotorcraft.

(3) The test torque and rotational speed must be:

(i) Determined by the powerplant limitations; and

(ii) Absorbed by the rotors to be approved for the rotorcraft.

(b) *Endurance tests, takeoff run.* The take off run must be conducted as follows:

(1) Except as prescribed in sub paragraphs (b)(2) and (b)(3), the take-off torque run must consist of 1 hour of alternate runs of 5 minutes at take-off torque and the maximum speed for use with take-off torque, and 5 minutes at as low an engine idle speed as practicable. The engine must be declutched from the rotor drive system, and the rotor brake, if furnished and so intended, must be applied during the first minute of the idle run. During the remaining 4 minutes of the idle run, the clutch must be engaged so that the engine drives the rotors at the minimum practical rpm. The engine and the rotor drive system must be accelerated at the maximum rate. When declutching the engine, it must be decelerated rapidly enough to allow the operation of the overrunning clutch.

(2) For helicopters for which the use of a 2½-minute OEI rating is requested, the take off run must be conducted as prescribed in subparagraph (b)(1), except for the third and sixth runs for which the take-off torque and the maximum speed for use with take-off torque are prescribed in that paragraph. For these runs, the following apply:

(i) Each run must consist of at least one period of 2½ minutes with take-off torque and the maximum speed for use with take-off torque on all engines.

(ii) Each run must consist of at least one period, for each engine in sequence, during which that engine simulates a power failure and the remaining engines are run at the 2½ minutes OEI torque and the maximum speed for use with 2½-minute OEI torque for 2½ minutes.

(3) For multi-engine, turbine-powered rotorcraft for which the use of 30-second/2 minute OEI power is requested, the take-off run must be conducted as prescribed in sub paragraph (b)(1) except for the following:

(i) Immediately following any one 5-minute power-on run required by sub paragraph (b)(1), simulate a failure, for each power source in turn, and apply the maximum torque and the maximum speed for use with the 30-second OEI power to the remaining affected drive system power inputs for not less than 30 seconds. Each application of 30-second OEI power must be followed by two applications of the maximum torque and the maximum speed for use with the 2 minute OEI power for not less than 2 minutes each; the second application must follow a period at stabilised continuous or 30-minute OEI power (whichever is requested by the applicant.) At least one run sequence must be conducted from a simulated 'flight idle' condition. When conducted on a bench test, the test sequence must be conducted following stabilisation at take-off power.

(ii) For the purpose of this paragraph, an affected power input includes all parts of the rotor drive system which can be adversely affected by the application of higher or asymmetric torque and speed prescribed by the test.

(iii) This test may be conducted on a representative bench test facility when engine limitations either preclude repeated use of this power or would result in premature engine removals during the test. The loads, the vibration frequency, and the methods of application to the affected rotor drive system components must be representative of rotorcraft conditions.

Test components must be those used to show compliance with the remainder of this paragraph.

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(c) *Endurance tests, maximum continuous run*. Three hours of continuous operation at maximum continuous torque and the maximum speed for use with maximum continuous torque must be conducted as follows:

(1) The main rotor controls must be operated at a minimum of 15 times each hour through the main rotor pitch positions of maximum vertical thrust, maximum forward thrust component, maximum aft thrust component, maximum left thrust component, and maximum right thrust component, except that the control movements need not produce loads or blade flapping motion exceeding the maximum loads of motions encountered in flight.

(2) The directional controls must be operated at a minimum of 15 times each hour through the control extremes of maximum right turning torque, neutral torque as required by the power applied to the main rotor, and maximum left turning torque.

(3) Each maximum control position must be held for at least 10 seconds, and the rate of change of control position must be at least as rapid as that for normal operation.

(d) *Endurance tests: 90% of maximum continuous run*. One hour of continuous operation at 90% of maximum continuous torque and the maximum speed for use with 90% of maximum continuous torque must be conducted.

(e) *Endurance tests; 80% of maximum continuous run.* One hour of continuous operation at 80% of maximum continuous torque and the minimum speed for use with 80% of maximum continuous torque must be conducted.

(f) *Endurance tests; 60% of maximum continuous run.* Two hours or, for helicopters for which the use of either 30-minute OEI power or continuous OEI power is requested, 1 hour of continuous operation at 60% of maximum continuous torque and the minimum speed for use with 60% of maximum continuous torque must be conducted.

(g) *Endurance tests: engine malfunctioning run.* It must be determined whether malfunctioning of components, such as the engine fuel or ignition systems, or whether unequal engine power can cause dynamic conditions detrimental to the drive system.

If so, a suitable number of hours of operation must be accomplished under those conditions, 1 hour of which must be included in each cycle, and the remaining hours of which must be accomplished at the end of the 20 cycles. If no detrimental condition results, an additional hour of operation in compliance with sub-paragraph (b) must be conducted in accordance with the run schedule of sub-paragraph (b)(1) without consideration of sub-paragraph (b)(2).

(h) *Endurance tests; overspeed run.* One hour of continuous operation must be conducted at maximum continuous torque and the maximum power-on overspeed expected in service, assuming that speed and torque limiting devices, if any, function properly.

(i) *Endurance tests: rotor control positions.* When the rotor controls are not being cycled during the endurance tests, the rotor must be operated, using the procedures prescribed in subparagraph (c), to produce each of the maximum thrust positions for the following percentages of test time (except that the control positions need not produce loads or blade flapping motion exceeding the maximum loads or motions encountered in flight):

(1) For full vertical thrust, 20%.

(2) For the forward thrust component, 50%.

(3) For the right thrust component, 10%.

(4) For the left thrust component, 10%.

(5) For the aft thrust component, 10%.

(j) *Endurance tests, clutch and brake engagements*. A total of at least 400 clutch and brake engagements, including the engagements of sub-paragraph (b), must be made during the take-off torque runs and, if necessary, at each change of torque and speed throughout the test. In each clutch engagement, the shaft on the driven side of the clutch must be accelerated from rest. The clutch engagements must be accomplished at the speed and by the method prescribed by the applicant. During deceleration after each clutch engagement, the engines must be stopped rapidly enough to allow the engines to be automatically disengaged from the rotors and rotor drives.

If a rotor brake is installed for stopping the rotor, the clutch, during brake engagements, must be disengaged above 40% of maximum continuous rotor speed and the rotors allowed to decelerate to 40% of maximum continuous rotor speed, at which time the rotor brake must be applied. If the clutch design does not allow stopping the rotors with the engine running, or if no clutch is provided, the engine must be stopped before each application of the rotor brake, and then immediately be started after the rotors stop.

(k) *Endurance tests, OEI power run.*

(1) 30 minute OEI power run. For rotorcraft for which the use of 30-minute OEI power is requested, a run at 30-minute OEI torque and the maximum speed for use with 30-minute OEI torque must be conducted as follows. For each engine, in sequence, that engine must be inoperative and the remaining engines must be run for a 30-minute period.

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(2) Continuous OEI power run. For rotorcraft for which the use of continuous OEI power is requested, a run at continuous OEI torque and the maximum speed for use with continuous OEI torque must be conducted as follows. For each engine, in sequence, that engine must be inoperative and the remaining engines must be run for 1 hour.

(3) The number of periods prescribed in sub-paragraph (k)(1) or (k)(2) may not be less than the number of engines, nor may it be less than two.

(l) Reserved.

(m) Any components that are affected by manoeuvring and gust loads must be investigated for the same flight conditions as are the main rotors, and their service lives must be determined by fatigue tests or by other acceptable methods. In addition, a level of safety equal to that of the main rotors must be provided for:

(1) Each component in the rotor drive system whose failure would cause an uncontrolled landing;

(2) Each component essential to the phasing of rotors on multi-rotor rotorcraft, or that furnishes a driving link for the essential control of rotors in autorotation; and

(3) Each component common to two or more engines on multi-engine rotorcraft.

(n) *Special tests.* Each rotor drive system designed to operate at two or more gear ratios must be subjected to special testing for durations necessary to substantiate the safety of the rotor drive system.

(o) Each part tested as prescribed in this paragraph must be in a serviceable condition at the end of the tests. No intervening disassembly which might affect test results may be conducted.

(p) *Endurance tests; operating lubricants.* To be approved for use in rotor drive and control systems, lubricants must meet the specifications of lubricants used during the tests prescribed by this paragraph. Additional or alternate lubricants may be qualified by equivalent testing or by comparative analysis of lubricant specifications and rotor drive and control system characteristics. In addition:

(1) At least three 10-hour cycles required by this paragraph must be conducted with transmission and gearbox lubricant temperatures, at the location prescribed for measurement, not lower than the maximum operating temperature for which approval is requested;

(2) For pressure lubricated systems, at least three 10-hour cycles required by this paragraph must be conducted with the lubricant pressure, at the location prescribed for measurement, not higher than the minimum operating pressure for which approval is requested; and

(3) The test conditions of sub-paragraphs (p)(1) and (p)(2) must be applied simultaneously and must be extended to include operation at any one-engine-inoperative rating for which approval is requested.

**CS 29.952 Fuel system crash resistance**

Unless other means acceptable to the Agency are employed to minimise the hazard of fuel fires to occupants following an otherwise survivable impact (crash landing), the fuel systems must incorporate the design features of this paragraph. These systems must be shown to be capable of sustaining the static and dynamic deceleration loads of this paragraph, considered as ultimate loads acting alone, measured at the system component's centre of gravity without structural damage to the system components, fuel tanks, or their attachments that would leak fuel to an ignition source.

(a) *Drop test requirements.* Each tank, or the most critical tank, must be drop-tested as follows:

(1) The drop height must be at least 15.2m (50 ft).

(2) The drop impact surface must be non-deforming.

(3) The tanks must be filled with water to 80% of the normal, full capacity.

(4) The tank must be enclosed in a surrounding structure representative of the installation unless it can be established that the surrounding structure is free of projections or other design features likely to contribute to rupture of the tank.

(5) The tank must drop freely and impact in a horizontal position ± 10°.

(6) After the drop test, there must be no leakage.

(b) *Fuel tank load factors*. Except for fuel tanks located so that tank rupture with fuel release to either significant ignition sources, such as engines, heaters, and auxiliary power units, or occupants is extremely remote, each fuel tank must be designed and installed to retain its contents under the following ultimate inertial load factors, acting alone.



 

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(1) For fuel tanks in the cabin -

(i) Upward - 4 g.

(ii) Forward - 16 g.

(iii) Sideward - 8 g.

(iv) Downward - 20 g.

(2) For fuel tanks located above or behind the crew or passenger compartment that, if loosened, could injure an occupant in an emergency landing -

(i) Upward - 1.5 g.

(ii) Forward - 8 g.

(iii) Sideward - 2 g.

(iv) Downward - 4 g.

(3) For fuel tanks in other areas -

(i) Upward - 1.5 g.

(ii) Forward - 4 g.

(iii) Sideward - 2 g.

(iv) Downward - 4 g.

(c) *Fuel line self-sealing breakaway couplings.* Self-sealing breakaway couplings must be installed unless hazardous relative motion of fuel system components to each other or to local rotorcraft structure is demonstrated to be extremely improbable or unless other means are provided. The couplings or equivalent devices must be installed at all fuel tank-to-fuel line connections, tank-to-tank interconnects, and at other points in the fuel system where local structural deformation could lead to release of fuel.

(1) The design and construction of self-sealing breakaway couplings must incorporate the following design features:

(i) The load necessary to separate a breakaway coupling must be between 25 and 50% of the minimum ultimate failure load (ultimate strength) of the weakest component in the fluid-carrying line. The separation load must in no case be less than 1334 N (300 pounds), regardless of the size of the fluid line.

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(ii) A breakaway coupling must separate whenever its ultimate load (as defined in sub-paragraph (c) (1) (i)) is applied in the failure modes most likely to occur.

(iii) All breakaway coupling must incorporate design provisions to visually ascertain that the coupling is locked together (leak-free) and is open during normal installation and service.

(iv) All breakaway couplings must incorporate design provisions to prevent uncoupling or unintended closing due to operational shocks, vibrations, or accelerations.

(v) No breakaway coupling design may allow the release of fuel once the coupling has performed its intended function.

(2) All individual breakaway couplings, coupling fuel feed systems, or equivalent means must be designed, tested, installed, and maintained so inadvertent fuel shutoff in flight is improbable in accordance with CS 29.955 (a) and must comply with the fatigue evaluation requirements of CS 29.571 without leaking.

(3) Alternate, equivalent means to the use of breakaway couplings must not create a survivable impact-induced load on the fuel line to which it is installed greater than 25 to 50% of the ultimate load (strength) of the weakest component in the line and must comply with the fatigue requirements of CS 29.571 without leaking.

(d) *Frangible or deformable structural attachments.* Unless hazardous relative motion of fuel tanks and fuel system components to local rotorcraft structure is demonstrated to be extremely improbable in an otherwise survivable impact, frangible or locally deformable attachments of fuel tanks and fuel system components to local rotorcraft structure must be used. The attachment of fuel tanks and fuel system components to local rotorcraft structure whether frangible or locally deformable, must be designed such that its separation or relative local deformation will occur without rupture or local tearout of the fuel tank or fuel system component that will cause fuel leakage. The ultimate strength of frangible or deformable attachments must be as follows:

(1) The load required to separate a frangible attachment from its support structure, or deform a locally deformable attachment relative to its support structure, must be between 25 and 50% of the minimum ultimate load (ultimate strength) of the weakest component in the attached system. In no case may the load be less than 1334 N (300 pounds).

(2) A frangible or locally deformable attachment must separate or locally deform as intended whenever its ultimate load (as defined in sub-paragraph (d)(1)) is applied in the modes most likely to occur.

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(3) All frangible or locally deformable attachments must comply with the fatigue requirements of CS 29.571.

(e) *eparation of fuel and ignition sources.* To provide maximum crash resistance, fuel must be located as far as practicable from all occupiable areas and from all potential ignition sources.

(f) *Other basic mechanical design criteria.* Fuel tanks, fuel lines, electrical wires and electrical devices must be designed, constructed, and installed, as far as practicable, to be crash resistant.

(g) *Rigid or semirigid fuel tanks*. Rigid or semi-rigid fuel tank or bladder walls must be impact and tear resistant.

**CS 29.1309 Equipment, systems, and installations**

(a) The equipment, systems, and installations whose functioning is required by this CS–29 must be designed and installed to ensure that they perform their intended functions under any foreseeable operating condition.

(b) The rotorcraft systems and associated components, considered separately and in relation to other systems, must be designed so that -

(1) For Category B rotorcraft, the equipment, systems, and installations must be designed to prevent hazards to the rotorcraft if they malfunction or fail; or

(2) For Category A rotorcraft:

(i) The occurrence of any failure condition which would prevent the continued safe flight and landing of the rotorcraft is extremely improbable; and

(ii) The occurrence of any other failure conditions which would reduce the capability of the rotorcraft or the ability of the crew to cope with adverse operating conditions is improbable.

(c) Warning information must be provided to alert the crew to unsafe system operating conditions and to enable them to take appropriate corrective action. Systems, controls, and associated monitoring and warning means must be designed to minimise crew errors which could create additional hazards.

(d) Compliance with the requirements of sub-paragraph (b)(2) must be shown by analysis and, where necessary, by appropriate ground, flight, or simulator tests. The analysis must consider:

(1) Possible modes of failure, including malfunctions and damage from external sources;

(2) The probability of multiple failures and undetected failures;

(3) The resulting effects on the rotorcraft and occupants, considering the stage of flight and operating conditions; and

(4) The crew warning cues, corrective action required, and the capability of detecting faults.

(e) For Category A rotorcraft, each installation whose functioning is required by this CS–29 and which requires a power supply is an 'essential load' on the power supply. The power sources and the system must be able to supply the following power loads in probable operating combinations and for probable durations:

(1) Loads connected to the system with the system functioning normally.

(2) Essential loads, after failure of any one prime mover, power converter, or energy storage device.

(3) Essential loads, after failure of:

(i) Any one engine, on rotorcraft with two engines; and

(ii) Any two engines, on rotorcraft with three or more engines.

(f) In determining compliance with sub paragraphs (e)(2) and (3), the power loads may be assumed to be reduced under a monitoring procedure consistent with safety in the kinds of operations authorised. Loads not required for controlled flight need not be considered for the two engine-inoperative condition on rotorcraft with three or more engines.

(g) In showing compliance with sub paragraphs (a) and (b) with regard to the electrical system and to equipment design and installation, critical environmental conditions must be considered. For electrical generation, distribution and utilisation equipment required by or used in complying with this CS–29, except equipment covered by European Technical Standard Orders containing environmental test procedures, the ability to provide continuous, safe service under foreseeable environmental conditions may be shown by environmental tests, design analysis, or reference to previous comparable service experience on other aircraft.

(h) In showing compliance with sub paragraphs (a) and (b), the effects of lightning strikes on the rotorcraft must be considered.

**Appendix I**

## CHANGES TO CERTIFICATION SPECIFICATION 29 AMENDMENT 11

This contains the new text for regulations relevant to this report, introduced by ED decision 2023/001/R following review of the comments to NPA 2022/01.

**AMC1 29.571 Fatigue tolerance evaluation of metallic structure**

ROLLING CONTACT FATIGUE

This AMC supplements FAA AC 29-2C, § AC 29.571 and should be used in conjunction with that AC when demonstrating compliance with CS 29.571.

(a) Definitions

(1) Rolling contact fatigue (RCF): a form of fatigue that occurs due to the cyclic strains arising from the loading present during rolling contact between two parts of an assembly, e.g. a bearing race and a rolling element. Note: For the purposes of this AMC, RCF also includes combinations of rolling and sliding contact phenomena.

(2) Integral race: a bearing race that is an integral part of the transmission structural component such as a gear or shaft.

(b) Explanation Service experience has shown that RCF can initiate on the surface and below the surface in contact areas of structural elements (typically, but not limited to, bearing races and rolling elements and gear teeth) that, in some cases, can propagate to a failure with catastrophic results. It is often assumed that RCF leads first to non-critical partial failures such as micropitting and spalling that will be detected before more severe failure modes can develop, such as a complete crack through a part. However, experience has shown that, in some cases, critical failure modes can develop shortly after the occurrence of non-critical partial failures. In such cases, analyses and tests are necessary to demonstrate that sufficient time is available, and the performance of the detection system is adequate to ensure the timely detection to prevent a catastrophic failure. The certification specifications in CS 29.571 require the identification and fatigue tolerance evaluation of principal structural elements (PSEs), leading to the establishment of inspection and retirement time or approved equivalent means to avoid a catastrophic failure during the operational life of the rotorcraft.

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In order to complete this evaluation, the impact of threats such as environmental effects, flaws and damages should be considered. Specific characteristics of parts submitted to RCF, such as:

- the difficulty to visually inspect,
- the operating nature of these elements, which can lead to mechanical degradation,
- the variability and susceptibility of the RCF mechanism in the presence of flaws or damages, make the evaluation of the impact of RCT on fatigue tolerance evaluation challenging. The procedures of this AMC are intended to help ensure that the effects of RCF are accounted for in the fatigue tolerance evaluations required by CS 29.571.

(c) Procedure

The fatigue tolerance evaluation of PSEs should include, when applicable, the effect of RCF considering:

- damage threats such as dents, scratches, corrosion, loss of preload in bearings or joints, surface and sub-surface material defects;
- residual stress coming from surface treatments and other manufacturing processes and all other applicable loading conditions. For this purpose, steps should be taken to minimise the risk of crack initiation due to RCF on PSEs (and in particular for integrated bearing races), by minimising contact pressures, specifying high standards for surface finishes, ensuring good lubrication, guaranteeing cleanliness and maintaining lubricant quality regardless of the fatigue tolerance approach selected. The applicant should verify that the selected allowables are suitable to ensure the integrity of the affected components in the operating conditions (temperature, lubrication, cleanliness, etc.) applicable to their design. Experience has demonstrated that it can be beneficial for bearings to be designed so that the reliability of any integrated race subject to the fatigue tolerance evaluation is even higher than the less critical race of the bearing. In this way, degradation of the less critical race can lead to detection of the bearing failure before cracking initiates in the integrated race. The consequences of damage to the integrated race from the debris generated in such scenarios should be considered in the evaluation.



 

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As it is difficult to totally preclude cracking initiated by RCF, a fail-safe approach is recommended wherever possible, such that cracking of the affected structural element(s) is detected prior to its residual strength capability falling below the required levels prescribed in CS 29.571(f). Should fatigue cracks initiate and develop into:

(1) Partial failure, such as spalling: the applicant should demonstrate that this condition will be detected at an early stage to avoid a catastrophic failure due to further fatigue failure, or loss of integrity of the affected part or any surrounding ones. Any assumptions regarding potential surface and sub-surface cracking considering possible damages or flaws, and whether a through crack may develop and its relationship with other forms of damage including spalling should be verified.

(2) Failure, such as through-cracking of a part together with any other associated damage in the system: the applicant should demonstrate that the remaining structure will withstand service loads and design limit loads without failure until the failure is detected and damaged components are repaired or replaced to avoid a catastrophic failure. Any assumptions regarding crack path development (i.e. bifurcation, multicracks, etc.) that could affect this fail-safe demonstration should be verified.

This demonstration should be performed as appropriate using experience from similar designs, functional tests, structural tests and/or reliable analyses to substantiate that the fail-safe design objective has been achieved, including residual strength demonstration. In addition, the continued safe operation of the affected mechanical system(s)should be ensured for this period considering the potential effect of the failure or partial failure taking into account any pre-existing fatigue damage accrued prior to the failure in the affected component and/or surrounding ones on stiffness, dynamic behaviour, loads and functional performance.

The effectiveness and reliability of means of crack detection for the fail-safe approach, including indirect means of detection such as chip detection systems, and associated instructions for continued airworthiness should be evaluated to show that, if implemented as required, they will result in timely detection and repair or replacement of damaged components.

Furthermore, the instructions for continued airworthiness, prescribing the maintenance actions leading up to and following detection of potential failure or partial failure should be substantiated sufficiently to ensure timely repair or replacement of damaged components. The substantiation should consider aspects such as threshold criteria on indicators of means of detection for additional investigative actions and removal from service of the damaged parts, the overall clarity and practicality of the instructions for continued airworthiness and human factors aspects.

In addition to following a fail-safe approach, inspection and retirement times may be needed in order to ensure that the assumptions supporting the fail-safety and detection of failure remain valid throughout the operational life of the component.

**CS 29.1309 Equipment, systems, and installations**

(a) Equipment and systems required to comply with type-certification requirements, airspace requirements or operating rules, or whose improper functioning would lead to a hazard, must be designed and installed so that they perform their intended function throughout the operating and environmental conditions for which the rotorcraft is certified.

(b) The equipment and systems covered by sub-paragraph (a), considered separately and in relation to other systems, must be designed and installed such that:

(1) each catastrophic failure condition is extremely improbable and does not result from a single failure, and for Category A rotorcraft, the occurrence of any failure condition which would prevent the continued safe flight and landing of the rotorcraft is considered as catastrophic;

(2) each hazardous failure condition is extremely remote; and

(3) each major failure condition is remote.

(c) The operation of equipment and systems not covered by sub-paragraph (a) must not cause a hazard to the rotorcraft or its occupants throughout the operating and environmental conditions for which the rotorcraft is certified.

(d) Information concerning an unsafe system operating condition must be provided in a timely manner to the flight crew member responsible for taking corrective action. The information must be clear enough to avoid likely flight crew member errors.

**AMC1 29.1309 Equipment, systems, and installations**

As defined in AMC 29.1, the AMC to CS-29 consists of FAA AC 29-2C Change 7, dated 4 February 2016. AMC 29.1309 provides further guidance and acceptable means of compliance to supplement FAA AC 29-2C Change 7 § AC 29.1309. As such, it should be used in conjunction with FAA AC 29-2C Change 7, but should take precedence over it, where stipulated, in the demonstration of compliance.

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**Single failure and common-cause considerations**

According to CS 29.1309(b)(1), a catastrophic failure condition must not result from the failure of a single component, part, or element of a system. Failure containment should be provided by the system design to limit the propagation of the effects of any single failure to preclude catastrophic failure conditions. In addition, there must be no common-cause failure which could affect both the single component, part, or element, and its failure containment provisions. A single failure includes any set of failures, which cannot be shown to be independent from each other. Common-cause failures (including common-mode failures) and cascading failures should be evaluated as dependent failures from the point of the root cause or the initiator. Errors in development, manufacturing, installation, and maintenance can result in common-cause failures (including common-mode failures) and cascading failures. They should, therefore, be assessed and mitigated in the frame of the common cause and cascading failures consideration.

Sources of common-cause and cascading failures include development, manufacturing, installation, maintenance, shared resource, event outside the system(s) concerned, etc. SAE[1] ARP4761 describes types of common-cause analyses, which may be conducted, to ensure that independence is maintained (e.g. particular risk analyses, zonal safety analyses, common-mode analyses).

While single failures should normally be assumed to occur, experienced engineering judgement and relevant service history may show that a catastrophic failure condition by a single-failure mode is not a practical possibility.

The logic and rationale used in the assessment should be straightforward and obvious that the failure mode simply would not occur unless it is associated with an unrelated failure condition that would, in itself, result in a catastrophic failure condition.

By detecting the presence of, and thereby limiting the exposure time to significant latent failures that would, in combination with one or more other specific failures or events identified by safety analysis, result in a hazardous or catastrophic failure condition, periodic maintenance or flight crew checks may be used to help demonstrate compliance with CS 29.1309(b).

**Development assurance process**

Any analysis necessary to demonstrate compliance with CS 29.1309 (a) and (b) should consider the possibility of development errors and should focus on minimising the likelihood of those errors.



1 SAE International is a company which develops internationally accepted engineering standards.

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Errors made during the development of systems have traditionally been detected and corrected by exhaustive tests conducted on the system and its components, by direct inspection, and by other direct verification methods capable of completely characterising the performance of the system.

These tests and direct verification methods may be appropriate for systems containing non-complex items (i.e. items that are fully assured by a combination of testing and analysis) that perform a limited number of functions and that are not highly integrated with other rotorcraft systems. For more complex or integrated systems, exhaustive testing may either be impossible because not all system states can be determined or impractical because of the number of tests that must be accomplished. For these types of systems, compliance may be demonstrated using development assurance.

(a) System development assurance. The applicability of system development assurance should also be considered for modifications to previously certificated aircraft.

ED-79A/ARP4754A is recognised as providing acceptable guidelines for establishing a development assurance process from aircraft and systems levels down to the level where software/airborne electronic hardware (AEH) development assurance is applied.

The extent of application of ED-79A/ARP4754A to substantiate development assurance activities depends on the complexity of the systems and on their level of interaction with other systems.

(b) Software development assurance. This AMC recognises AMC 20-115 as an accepted means of compliance with CS 29.1309 (a) and (b).

(c) AEH development assurance. This AMC recognises AMC 20-152 as an acceptable means of compliance with the requirements in CS 29.1309 (a) and (b).

(d) Open problem report management This AMC recognises AMC 20-189 as an acceptable means of compliance for establishing an open problem report management process for the system, software and AEH domains. Integrated Modular Avionics (IMA) This AMC recognises AMC 20-170 as an acceptable means of compliance for development and integration of IMA.

**Appendix J**

# FLIGHT MECHANICS ANALYSIS INFORMATION PROVIDED BY THE HELICOPTER MANUFACTURER



## 13 Additional Flight Mechanics Analysis

The scope of the analysis is to understand if a different control input sequence after the failure could have enabled a lower vertical speed at impact.
The simulation follows the event up to the instant of the failure. After this moment a different logic is used: instead of feeding the FDR data, controls following the RFM instructions for loss of tail rotor effectiveness are injected instead.
The simulation was performed using Flightlab software. The latest AW169 model was used (F365-169-XXXC24). Flightlab simulations start from a trimmed condition. As the manoeuvre performed by the helicopter was dynamic, an exact trim point is not available. An almost steady flight condition was found at T=34s after the take-off and used as the trim point for the simulation.
In order to better match the FDR data at the time of the failure, the software was instructed to follow the helicopter attitudes as recorded during the event from T=34s to T=43s (time of failure).
At T=43s a failure consisting in a 2.5 seconds ramp to full right pedal ($\theta TR$ = -10°) is injected.
As shown in Table 1, depending on the flight phase, different response times applies. The take-off manoeuvre is an active flight segment and a 0.5 s response time may be used, but, conservatively, a 1.5 s time is considered instead.

| Flight Segment | Recognition Time [s] | Reaction Time [s] | Pilot Response Time [s] |
|---|---|---|---|
| Active | - | 0.5 | 0.5 |
| Attentive–Hands-On | 1.0 | 0.5 | 1.5 |
| Attentive–Hands-Off | 1.0 | 1.0 | 2.0 |
| Passive | 2.0 | 1.0 | 3.0 |

**Table 1 - AC29 - Pilot response time**

In the 1.5 seconds after the failure the control logic is switched in order to try to limit pitch and roll motions. At T=43+1.5 s the control logic follows the RFM prescriptions:

In Hover

- Lower collective to LAND IMMEDIATELY while maintaining attitude and minimizing lateral translation with the cyclic control;
- Select both ENG MODE knobs to OFF if time is available.

The following actions are taken:

- The collective lever is fully lowered
- The cyclic control is used to minimize pitch and roll motion
- At an appropriate altitude above ground, the collective lever is fully raised to cushion impact





 

Due to the low altitude at which the event takes place, the engine is not switched off as the manoeuvre develops rapidly and there is little time available before impact. Moreover this action implies that the pilot has to takes the hand of the collective lever in order to reach the engine knobs on the central console.
The result of this analysis (Figure 1) shows that following the RFM prescriptions it is possible to:

- Limit the Yaw Rate and maintain pitch and roll attitudes within manageable limits
- Avoid the disengagement of the AFCS and thus maintain at least the basic stabilisation along the entire event (by avoiding the saturation of the Yaw Rate channel above 128°/s).
- Decrease the rate of descent at impact to values around 1950 ft/min.





 

## Appendix J cont

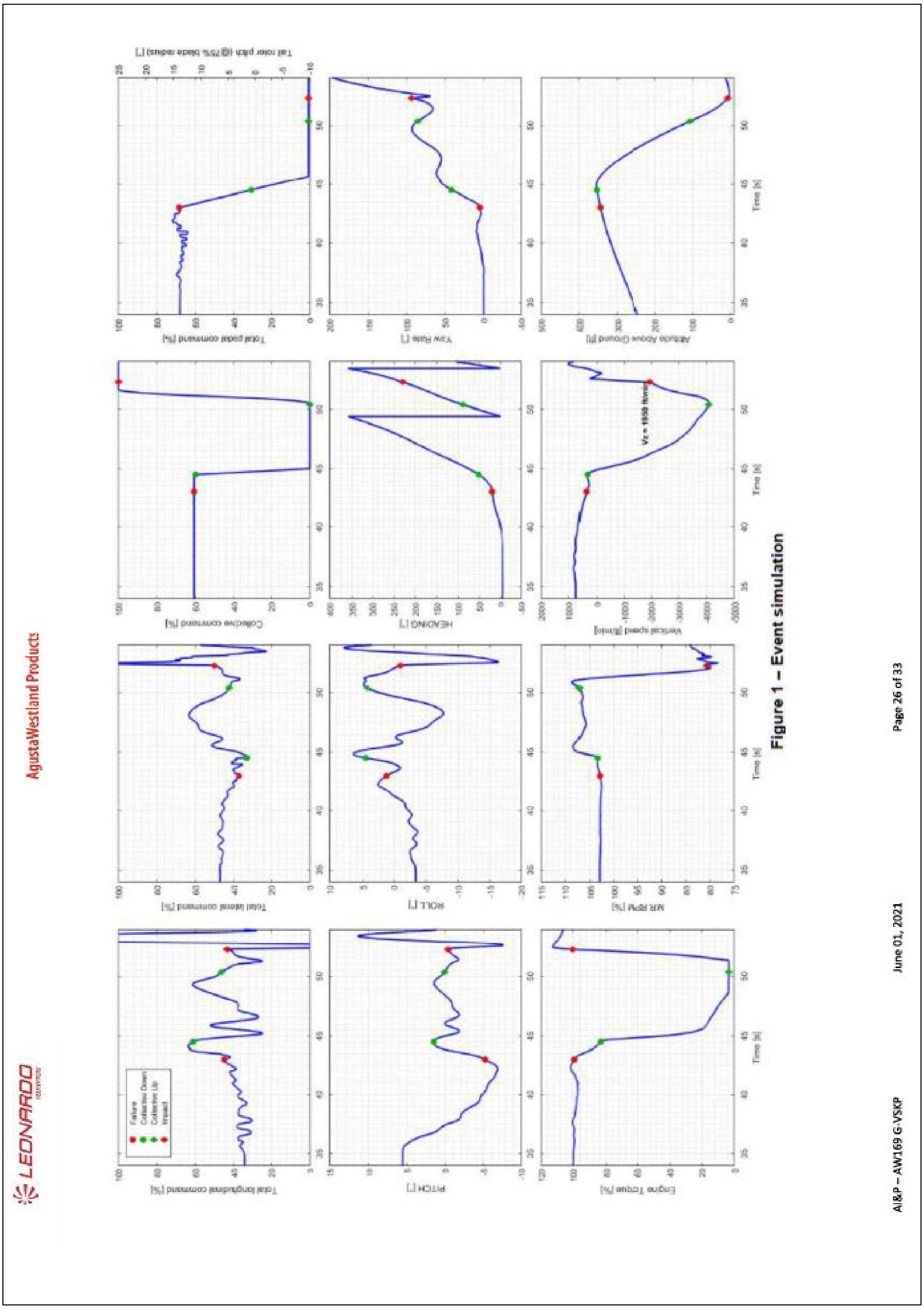


Figure 1 – Event simulation



 

**Appendix K**

**Comments of the ANSV representing the State of Design and Manufacture**

Chapter 6.3 of Annex 13 to the Convention on International Civil Aviation provides that the State conducting the investigation shall send a copy of the draft Final Report to all States that participated in the investigation, inviting their significant and substantiated comments on the report as soon as possible. If the State conducting the investigation receives comments within the period stated in the transmittal letter, it shall either amend the draft Final Report to include the substance of the comments received or, if desired by the State that provided comments, append the comments to the Final Report.

ANSV AGENZIA NAZIONALE PER LA SICUREZZA DEL VOLO

**Subject:** **ANSV, LH and EASA comments on the draft final report about the AW169 registration marks G-VSKP accident.**

Dear Sir,

thank you for having invited the ANSV to participate as Accredited Representatives in the investigation to the Accident which occurred to AW169 registration marks G-VSKP, King Power Stadium, Leicester on October the 27$^{th}$ 2018, and for the opportunity to comment on the final report.

The ANSV together with its Technical Advisers, Leonardo Helicopters and the European Union Aviation Safety Agency, have extensively reviewed the final report which brings some important improvements in the safety of flight through safety recommendations, together with the safety actions stimulated by the discussions made along the investigation process.

Nonetheless, throughout the course of the investigation, a lot of information have been provided to the UKAAIB and the final report does not correctly reflect some of that, as well as the analyses and conclusions. Therefore, during the consultation phase relevant to the draft report a large number of comments were provided to the UKAAIB. These were clearly outlined and explained. However, the final report relevant the AW169 marks G-VSKP does not reflect some of the inputs provided.

Therefore, in view of the release to the public, in order to make as simple as possible to understand the key areas of disagreement, they are summarized as follows:

**Appendix K cont**

1. **Internal contact pressure calculation.**

The UKAAIB Report concludes in the causal factors paragraph «*the tail rotor duplex bearing likely experienced a combination of dynamic axial and bending moment loads which generated internal contact pressures sufficient to result in lubrication breakdown and the balls sliding across the race surface. This caused premature, surface initiated rolling contact fatigue damage to accumulate until the bearing seized*».

A large number of analyses were made in the attempt to understand what could actually have caused combination of dynamic axial and bending moment loads which generated internal contact pressures sufficient to result in lubrication breakdown and the balls sliding across the race surface.

The investigation points out that the bearing may have experienced external loading in service, which was probably enough to initiate the sequence of events resulting in the bearing seizure. However, there is no evidence that the seizure of the bearing was solely due to in-flight loads capable of generating high enough internal contact pressure within the bearing. This is also clear when such scenario is presented as "likely".

In addition, the process itself used to get to the abovementioned considerations seems questionable:

- The investigation used the AW169/189 flight test data relevant the axial loads but performed an independent calculation of bending moments. This calculation was made by the UKAAIB alone using a professional software; details of this calculation were not shared for joint evaluation with the investigation Parties (namely ANSV, LH and EASA). While the UKAAIB may have used powerful tools and instruments to perform the calculation, the manufacturer involved is one of the leading helicopter manufacturers worldwide and would have added some experience and knowledge in this process. On the other hand, if something was possibly missed in the design/certification process, sharing the UKAAIB independent calculation would have explained in detail what exactly could be improved.
- The results in terms of magnitude of the internal contact pressures as calculated with the UKAAIB bending moments (appendix E) are same order of magnitude, in some cases lower, than those calculated in the certification of the load spectrum for the tail rotor duplex bearing (pag. 41, table 4). The latter internal contact pressures were considered anyway acceptable by the bearing manufacturer during certification. Therefore, it is unclear which combination of dynamic axial and bending moment loads generated internal contact pressures sufficient to

result in lubrication breakdown and the balls sliding across the race surface. This may have been the result of the concurrence of other possible causal/contributory factors, not assessed as such by UKAAIB, as described in point 4 below.

**2 Helicopter usage vs internal contact pressure.**

Any theory based upon the combination of dynamic axial and bending moment loads capable of generating internal contact pressures sufficient to result in lubrication breakdown and the balls sliding across the race surface appears to be very debatable. This is especially true when considering why the bearing in the accident helicopter should have been the first in the AW169 and AW189 fleet to fail, considering the AW189 is a helicopter with higher loads on the tail rotor, sharing with the AW169 the same certification of the duplex bearing. In addition, as indicated in the report, this helicopter was operated in a VIP role and had completed just 330 FH at the time of the accident. After the accident, other bearings were analyzed, some of them showing signs of damage similar to an initiation phase of what has been seen on the G-VSKP bearing. Out of all these bearings, excluding one of them, which showed clear evidence of manufacturing defect and failed early, all the others have been in operation for a longer time than the accident bearing; most of them for a much higher number of flight hours. This framework compares favorably with many other aircraft in the AW169/AW189 fleet, which had higher flight hours and were engaged in far more strenuous roles (e.g. HEMS), requiring them to operate in more difficult conditions (both dynamic and meteorological) for longer periods and in a configuration that, for its nature, involves higher loads as demonstrated by certification tests (Hoist, Cargo Hook).

In order to explain this, the UKAAIB's report focuses on hypotheses not confirmed by evidence on different aspects such as:

- «[…] flexibility in helicopter manoeuvres and diversity in atmospheric conditions in which they operate, results in significant potential variability in the duration, magnitude and frequency of exposure to the potentially damaging contact pressures […]»
- *«Some helicopters in the AW169 and AW189 fleet may never have been subject to manoeuvres which generated contact pressures sufficient to cause premature damage […]»*

  Nonetheless, the investigation was not able to confirm in the report which manoeuvres induced unexpectedly high internal contact pressures.

  Also, the usage of the G-VSKP was discussed and the report seems to suggest the manoeuvre associated to the stadium where the accident occurred may have generated such external loading capable of generating extreme internal contact pressures within the bearing:

*«Due to the shape of the football stadium, takeoffs could only be done in one of two directions orientated along the long axis of the pitch. The helicopter could potentially have been exposed to adverse wind directions as it emerged above the stadium roof, but this was not recorded in the flight data or journey logs. The helicopter was locked into this specific routine during the football season, differentiating it from other roles, such as offshore transport or Helicopter Emergency Medical Services (HEMS). »*

It is important to highlight there is no evidence such manoeuvre, even at the stadium of the occurrence, may actually induce axial and bending moment loads able to generate within the bearing internal contact pressures sufficient to result in lubrication breakdown and the balls sliding across the race surface.

**3 Alleged missed mitigating opportunities through in-service experience.**

The text of the final report infers that, since damaged bearings were removed following the accident as a result of the Continued Airworthiness measures put in place through a Service Bulletin and Airworthiness Directive, damaged bearings existed also before the accident. While it is possible in theory that some defective bearings were not identified, the reason for that shouldn't bring the UKAAIB to identify as a contributing factor to these potential missing reports of damaged bearings the fact that «the manufacturer of the helicopter did not require bearings removed from service to be returned to facilitate an inspection of their condition; nor was there any regulatory requirement or guidance that required them to do so». Indeed, this appears not to be consistent with the following evidence:

- a maintenance plan was in place for the Tail Rotor Double Bearing (TRDB), including both recurrent inspections and discard time.
- At 1st November 2018, none of the AW169 in-service accumulated 2,400 FH and no occurrences had been reported from the service on the AW169 fleet (on top of the two occurrences affecting the AW169 Prototypes and quoted in the UKAAIB report). The two bearings removed from the AW169 prototypes were found, following the relevant inspection, to be serviceable (no fault found).
- The only duplex bearing occurrence on the AW189, reported to the OEM, was associated with a condition not considered applicable for the purpose of this investigation.
- According to Annex II, Paragraph 3 of the Commission Implementing Regulation (EU) 2015/1018 of 29 June 2015 laying down a list classifying occurrences in civil aviation to be mandatorily reported according to Regulation (EU) No 376/2014



 

of the European Parliament and of the Council, any defect in a life-controlled critical part causing retirement before completion of its full life must be reported.

- The Discard Requirement of the Bearing was included in Chapter 5 of the H/C Maintenance Manual and the Technical Publication "DM 69-A-00-60-00-00A-010A-A Critical parts - General data"[1] provides the H/C operator with the list of the critical parts including the P/N of the duplex bearing.

Therefore, indication from service experience nor the lack of the postulated routine inspection for bearings removed from service, can be considered as contributory factor for this accident, unless further investigation is conducted on the organizations responsible for the continuing airworthiness of the AW169 and AW189 helicopters questioning possible lack of adherence to regulations addressing the reporting, analysis and follow-up of occurrences in civil aviation.

**4. Other possible hypotheses**

The report widely discusses the topic of internal contact pressures related to the external loading and its definition during the design/certification process, while provides little relative importance to other possible theories that may explain to the observed variability in the final performance of a bearing.

In more detail, the effects associated to non-conformities in manufacturing (bearing and/or helicopter components interfacing with the bearing), bearing design, environmental degradation, any possible issue related to grease and/or bearing installation were not considered capable to either cause or contribute to the event (on their own or in combination), being some of the above hypotheses briefly discussed in paragraph 2.9.3.

For example, bearings S/N 13123 and 17115 were both investigated by UKAAIB and recognized to have production issues, although showing evidence not considered to be similar to those of the accident bearing. This latter, S/N 14126, was anyway part of the same production batch of bearing S/N 14125 and 14134, both returned from service during the post-accident repetitive inspection programme. The information about the same production batch is not even mentioned in the text. Nonetheless, it is factual that 6 bearing have been investigated by either UKAAIB or LH or both and S/N 13123 and S/N 17115 were probably affected by production issues, while 3 out of the remaining 4 bearing investigated are part of the same production batch.

1 Within the Data Module there is a dedicated paragraph instructing the Operator to tell the Manufacturer about any unusual wear or deterioration of a critical part (If necessary, send the critical part to the Manufacturer for inspection).

**Appendix K cont**

In addition, purely as an additional example, the report states: «*The bearing material properties and dimensions were assessed in the bearings inspected and any variations confirmed to be a consequence of the damage process».* However, during the investigation the dimensional analysis together with its assessment are not showed in the report, beside being commented. On the other hand, the accident bearing was heavily damaged and an accurate assessment of the races dimensions is supposed to be very complex, being even more difficult to ascertain the original dimensions of the parts before the initiation of the failure mechanism. A non-conformity in the dimensions of the races, would easily induce high contact pressures.

It is understood the above are only examples and they would provide theories for which supporting evidence would have been difficult to collect. However, they are listed in this comment to support the idea the final report seems to neglect some important possibilities and interesting evidence, while focusing mainly on one single theory, not proven as well, supported by debatable assumptions.

The ANSV applies the right to append this letter of comments the final report, as permitted by ICAO Annex 13 section 6.3.

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Unless otherwise indicated, recommendations in this report are addressed to the appropriate regulatory authorities having responsibility for the matters with which the recommendation is concerned. It is for those authorities to decide what action is taken. In the United Kingdom the responsible authority is the Civil Aviation Authority, Aviation House, Beehive Ringroad, Crawley, West Sussex, RH6 0YR.

Aircraft Accident Report 1/2023

Report on the accident to
**Leonardo AW169, G-VSKP**
at King Power Stadium, Leicester
on 27 October 2018