APPLICATION OF FLAW TOLERANCE METHODOLOGIES TO ROTORCRAFT FATIGUE QUALIFICATION

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Abstract

AgustaWestland implemented and successfully applied Flaw Tolerance methodologies to the fatigue qualification of the new AB139 Twin Engine Medium Helicopter, which achieved the Type Certificate by ENAC in June 2003 and by FAA in December 2004. Full compliance was given to the rule FAR/JAR § 29.571, Amendment 28, which prescribes to evaluate the rotorcraft fatigue tolerance considering the effect of environment, intrinsic flaws or accidental damage.

A combination of conventional Safe-Life methodologies and Flaw Tolerance approaches were applied to the Principal Structural Elements, addressing different design features, materials and manufacturing processes. Either testing or analysis, or a combination thereof, was used as appropriate to each specific critical component. The results of the substantiation consisted in a combination of retirement times and inspection intervals aimed at preventing catastrophic failure due to fatigue cracking during the operational life of the rotorcraft.

This paper goes through the whole fatigue qualification process and presents the different criteria and approaches that were adopted for each critical item depending upon its specific features, such as material, geometry, type of installation and type of loads. Some practical application cases to Main Rotor and Tail Rotor components are described in detail.

Introduction

The current airworthiness regulations (Ref 1) require that a fatigue tolerant design, including the effects of environment, flaws or accidental damage, is accomplished in order to avoid catastrophic failure due to fatigue cracking during the operational life of the rotorcraft. A number of different methodologies and guidelines have been recognized and discussed so far, supporting the establishment of replacement times or inspection intervals.

AgustaWestland successfully met the Flaw Tolerance requirements on the AB139 Medium Helicopter, which was certified by ENAC in June 2003 and by FAA in December 2004. This goal was accomplished by a low stress design approach that included the implementation of design features, such as ForceMate® bushings, shot peening and protective coatings, which inherently provide protection against flaw effects. The substantiation of fatigue tolerant design was achieved by a combination of conventional Safe-Life methodologies and new Flaw Tolerance approaches leading to the definition of retirement times and/or mandatory inspection intervals for all the principal structural elements.

Considerations of component characteristics such as design practice, geometry limitations, weight limitations and manufacturing procedures lead to the identification of principal structural elements for which the flaw tolerance requirements were considered to be not applicable. This type of components included Landing Gear, Drive System Gears, M/R and T/R Shafts, M/R and T/R Controls such as Swashplates and Pitch Rods. For these parts a full safe-life qualification was provided.

The fatigue substantiation went through a detailed Threat Assessment aimed at the determination of type and size of flaws, and their relevant locations, that can be realistically expected during the life of the rotorcraft. The assessment took into account the specific manufacturing and quality controls processes, operational environment and previous company experience on such events. The identified damage scenario included scratches, dents and fretting for metal, impacts and manufacturing discrepancies for composite.

The fracture mechanics based “no crack growth” approach was adopted for all metallic dynamic parts, considering that in case of crack growth the severe loading environment would lead to very short inspections intervals, which are impractical from a maintenance point of view.

This approach allowed demonstrating that, in presence of a flaw, the stresses experienced by the part under the operative loads are below the threshold level at which a crack of the same size

would propagate. This means that no scheduled inspections are required, providing at the same time improved fatigue tolerance capabilities.
This type of substantiation was accomplished by means of both an extensive testing program at coupon level and a detailed FEM analysis.

For a few components the “no crack growth” capabilities were demonstrated by means of full-scale tests; in this case the specimen under test was artificially pre-cracked and subjected to the operational load spectrum.

The substantiation of composite parts was mainly based on experimental activities; full-scale fatigue tests were carried out considering typical manufacturing flaws and impact damages at both Barely Visible and Clearly Visible level. The substantiation allowed establishing both a safe retirement time and a scheduled inspection for clearly visible damage.

A full substantiation was performed for the metallic fuselage as well; a combination of testing and crack propagation analysis was carried out leading to the establishment of practical inspection intervals.
The testing activity, which is still in progress, includes typical design features and structural elements representative of the full-scale structure like riveted joints and sandwich panels with impacts and cracks.

Fatigue Qualification Process

The whole fatigue substantiation process was based upon the following activities:

- Detailed threat assessment aimed at the definition of the rotorcraft damage scenario
- Definition of type of damage, size and critical locations, based on threat assessment results, preliminary stress analysis and safe-life test results
- Damage vulnerability assessment and susceptibility tests, in particular for composite parts
- Safe-life substantiation, by means of traditional full-scale testing
- Flaw tolerance assessment, either by analysis or by test depending upon the specific structure
- Definition of retirement times and/or inspection intervals or additional provisions

Depending upon the specific characteristics of each critical part, such as material, design features and type of loads, different approaches were chosen and implemented in order to meet the flaw tolerance requirements.

In particular the following methods were applied to AB139 principal structural elements.

- Crack growth based methods:

  - No crack growth: demonstrates that, in presence of a flaw, the stresses experienced by the part under the operative loads are below the threshold level at which a crack of the same size would propagate; no inspection is therefore required.
    This method was applied to all rotor components and parts subjected to dynamic loads either by test or by analysis.

  - Crack growth: defines an inspection interval based on the time for a detectable damage to grow to critical size or for residual strength of the structure.
    This method was applied to Fuselage parts and Elastomeric components by analysis and tests respectively.

- Crack initiation based methods:

  - Flaw Tolerant Safe-Life: determines a retirement life addressing the effect of damage that could remain undetected for the life of the part.
    This method was applied to all composite parts by full-scale fatigue tests.

Table 1 summarises all the principal structural elements that, in addition to the conventional safe-life evaluation, were subjected to flaw tolerance substantiation; the specific adopted approach is also reported.
A big effort was put in detailed analytical assessments, experimental FE model validations and experimental material characterisation for addressing crack growth behaviour.

Dedicated experimental activities were also carried out on structural elements, as representative of the real structure application, in order to establish the effect of flaws or cracks. This type of investigation included lug coupons with FTI ForceMate® bushings (Ref 2), sandwich panels loaded in tension and in shear.

The following paragraphs present details of the Threat Assessment and some of the approaches used for the Flaw Tolerance substantiation, in particular for dynamic metallic components, fuselage critical elements and composite parts.

Part	Substantiation method	Analysis/Testing
M/R Blade	Flaw Tolerance Safe-Life	Testing with manufacturing defects + impacts at both BVID and CVID level
M/R Hub	No crack growth Fail-safe	Analysis Testing
M/R Elastomeric Bearing	No crack growth for metal Crack propagation for elastomer	Analysis + Testing Testing
M/R Pitch Control Lever	No crack growth	Analysis
M/R Tension Link	Flaw Tolerance Safe-Life	Testing with manufacturing defects + impacts at both BVID and CVID level
T/R Blade	Flaw Tolerance Safe-Life	Testing with manufacturing defects + impacts at both BVID and CVID level
T/R Hub	No crack growth	Analysis
T/R Elastomeric Bearing	No crack growth for metal Crack propagation for elastomer	Analysis + Testing Testing
T/R Blade Damper Attachment	No crack growth	Analysis + Testing
T/R Hub Damper attachment	No crack growth	Analysis
T/R Elastomeric Damper	No crack growth Crack propagation for elastomer	Analysis Testing
Transmission cases	Flaw tolerance safe-life	Testing
T/R Drive Shafts	No crack growth + Flaw Tolerance Safe Life	Analysis + Testing
Main Gearbox Fittings	No crack growth	Analysis
Main Gearbox Mounting Rods	No crack growth	Analysis
Anti-torque beam	No crack growth	Analysis
Tail Rotor Gearbox Fitting	No crack growth	Analysis
Fittings backup structure and airframe	No growth + Crack growth	Analysis
Fin	No growth + Crack growth	Analysis
Tail Cone	No growth + Crack growth	Analysis
Tail/Rear Fuselage Attachments	No growth + Crack growth	Analysis
Engine attachments	No growth + Crack growth	Analysis
Tailplane	Flaw Tolerance Safe-Life	Testing with impacts at both BVID and CVID level
Tailplane attachment fittings	No crack growth	Analysis

Table 1: Critical parts evaluated for flaw tolerance

<u>Threat assessment</u>

The flaw tolerance substantiation started from a detailed "threat assessment", whose purpose was to define the helicopter potential damages that could occur during manufacturing and service life and that could modify the fatigue behaviour of the structure.

The assessment was made taking into account that a combination of safe-life and flaw tolerance evaluations was adopted for the helicopter fatigue clearance; for this reason threats like fatigue cracking in service and fretting phenomenon were considered to be already covered by the retirement times as determined by traditional safe-life tests.

The threats accounted for were the worst case of material flaws, manufacturing flaws, maintenance and service induced damages that were expected to remain undetected for the whole operational life of the part and the worst case flaws which would not be expected to remain in place without corrective action for a significant period of time. Damages resulting from birdstrike or lightning strike were the object of a separate substantiation.

The threat assessment was based on the analysis of the type of materials used, manufacturing procedures, production quality checks, acceptance tests and in service inspections techniques. In addition previous company experience and collection of data on similar events supported the choice of the damage size likely to occur. The following diagram (Fig 1), that quotes the statistics data of scratches on Aluminium parts, due to an abuse in manufacturing, confirms that a 0.25 mm deep scratch is the worst recurring case and that a 0.38 mm deep flaw can be considered the appropriate size to address an upper limit of accidental damage on metal parts.

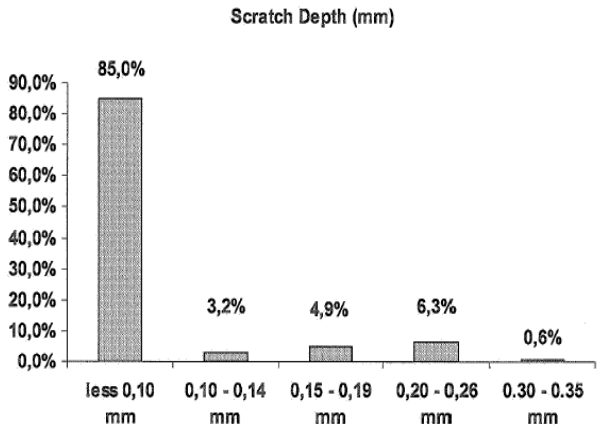

Fig 1: Statistics of scratched components

Based on the above evidences the threat assessment resulted in the following assumptions for the initial damages to be considered in the flaw tolerance analysis.

Metal rotor parts and fixed components subjected to high frequency loads:

- 0.25 mm for all the locations which remain protected after assembly, typically by cowlings or fairings
- 0.38 mm for the other locations
- Smaller values of damage could be applicable only with additional provisions or special inspections
- No defect was considered for parts not exposed to accidental damage after assembly, like bolts and FTI ForceMate® lugs, based on the evidence that proper inspection procedures are in place prior to the final assembly and considering that a crack of 0.05 mm is already covered by the safety factors taken into account for the safe-life assessment

Metal airframe parts:

- 1.27 mm as in the standard practice of fixed wing damage tolerance assessments

Flaw Tolerance of Rotor Components

The fatigue and flaw tolerance substantiation of rotor parts and fixed components subjected to high vibratory loads was accomplished by means of a combination of conventional safe-life and new approaches to take into account the presence of flaws.

The fracture mechanics based "no crack growth" approach was adopted by AgustaWestland as a suitable route for the fatigue and flaw tolerance substantiation of metallic dynamic parts, considering that in case of crack growth the severe loading environment of helicopters would lead to the definition of very short inspection intervals.

The methodology consists in demonstrating that a crack of a certain size, assumed to be present at the component most critical locations, does not propagate under the service loading spectrum. A stress safety factor of 1.3, when using bibliographic threshold data, or 1.15 when using company data, was taken into account in the evaluation.

The methodology was developed in cooperation with Politecnico di Milano, Dipartimento di Meccanica (Ref 3), based on the work carried out by Y. Murakami (Ref 4, 5, 6) and co-authors on the effect of small defects in metal fatigue.

The following procedure was implemented for the assessment of dynamic metallic parts:

- Definition of ΔK_{th} for small cracks using fatigue limit tests on micro-notched specimens under axial or bending loads at different stress ratios and for different crack sizes.
- Performance of a detailed FEM analysis of the critical part.
- Validation of the FE Model by either a strain survey or other methods on a full-scale specimen.
- Verification that ΔK_{max} of the service load spectrum is below ΔK_{th} for a specified crack size consistent with the threat assessment of the part.

When evaluating the flaw tolerance capabilities of complex rotor components the "no crack growth" approach resulted to be not suitable to cover all the critical areas, in particular specific design features where the analysis was considered to be not reliable; in such a case experimental activities were needed.

<u>Flaw Tolerance of a Main Rotor Component</u> The AB139 M/R Hub represents a suitable example of a complex component for which different flaw tolerance approaches were adopted.

The M/R Hub has the function of transferring the torque from the mast to the blades and to transfer the shears arising from the lift and drag of the blades to the mast. The hub is restrained to the mast by means of a spline while the shear loads of the blades are transferred to the hub by means of the elastomeric bearings.

The M/R Hub is made of Ti6Al4V plus a Graphite-Epoxy band that provides a passive multiple load path system, as it is unloaded until failure of the metal part.

The qualification plan of the M/R Hub consisted in a combination of experimental and analytical assessments aimed at satisfying both safe-life and flaw tolerance requirements:

- Safe-Life

1 High Frequency Test
2 Low frequency Tests

- Flaw tolerance

No crack growth analysis
1 Fail-safe Test

Both stress analysis and safe-life testing allowed highlighting the most critical locations (see Fig 2); depending upon their features a different approach was adopted for fulfilling the flaw tolerance requirements.

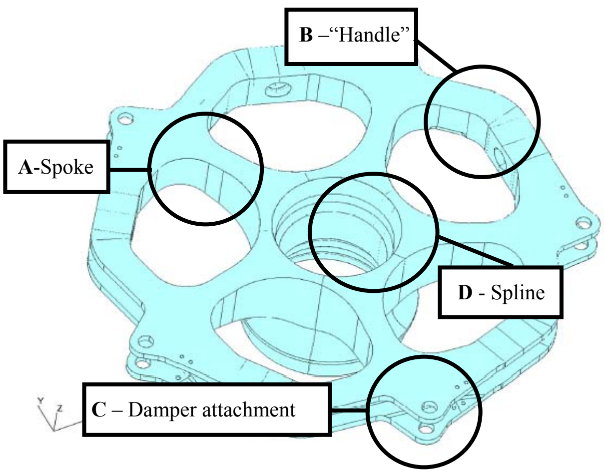

Fig 2: M/R Hub critical locations

- The location **A**, which is mainly subjected to high frequency loading, was highlighted as one of the most stressed areas; this location was covered by analysis demonstrating the “no crack growth” of a 0.38 mm crack under the worst loading case of the applicable usage spectrum.
- The location **B,** which is affected by both start-stop loading due to centrifugal force and high frequency loading**,** was covered by test demonstrating the fail-safety of the hub metal/composite system.
- The location **C** was inherently covered by the presence of FTI ForceMate® bushings.
- The location **D** was covered by design similarity with the EH101 M/R Hub for which a flaw tolerant safe life was performed assuming several spline teeth missing.

A detailed FEM analysis was carried out using ABAQUS Solver highlighting the areas subjected to the highest stresses. With reference to Fig 3, the Zone “Est” and “Int”, corresponding to Location B of Figure 2, were covered by test; the Zone “Sup” is mainly subjected to compression stresses and the area “Inf”, corresponding to Location A of Figure 2, was covered by the “no crack growth” approach.

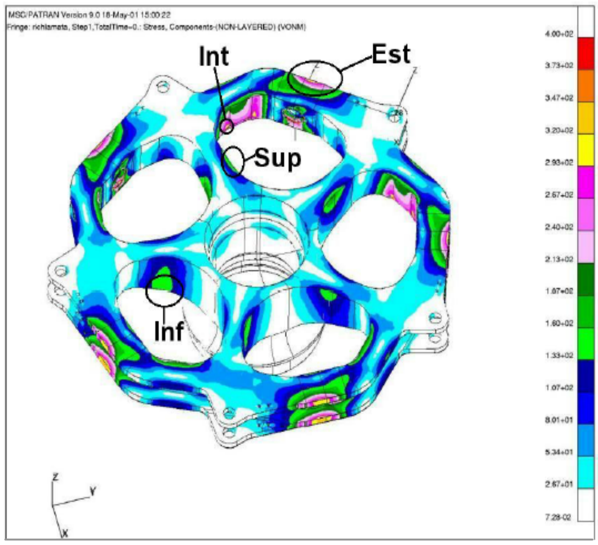

Fig 3: M/R Hub FE Model Stress Analysis

The FE model was validated by means of an experimental strain survey, showing a good correlation between analytical predictions and strain data; however for the flaw tolerance assessment of Location A it was decided to use the stresses directly measured on the part during the Certification Load Survey, considering that a strain gauge was installed exactly in the most stressed area.

Although it was decided to use the experimental data for the assessment, the FE model was essential for the following aspects:

- The FE model allowed to identify the areas where a flaw tolerance assessment was required
- The FE model allowed to demonstrate that the Zone "Inf" was subjected only to a monoaxial stress
- Since the strain gauge S1 was not exactly positioned in the most stressed point, the model allowed to calculate a corrective factor to be applied to the measured data in order to cover the max $\Delta\sigma$ value.

The highest measured stresses of the fatigue spectrum were then compared to the material allowables for no propagation of a 0.38 mm corner crack. As shown in Table 1, the calculated reserve factors, $\eta = \Delta K_{th} / \Delta K$, were higher than the 1.15 safety factor (applicable as company derived data were available) for all the load cases, demonstrating full compliance with the "no crack growth" criteria.

Corner crack					
Load case	$\Delta\sigma$ [MPa]	R	ΔK [MPa√m]	ΔK_{th} [MPa√m]	η
1	141.1	0.00	3.90	5.72	1.47
2	156.6	-0.2	4.33	6.66	1.54
3	143.7	-0.4	3.97	7.64	1.93
4	191.7	-0.3	5.30	7.10	1.34

Table 1: M/R Hub "no crack growth" results

The location B was instead covered by a dedicated test, aimed at demonstrating the fail-safe capabilities of the hub assembly (metal + composite band) when assuming a failure in the metal part.
The test was run on a pre-cracked specimen, namely the same specimen that was previously subjected to a safe-life low frequency test demonstrating a life greater than 40000 hours. A 44 mm crack (Fig 4) was therefore present at the critical location of the hub prior to the start of the fail- safe test.

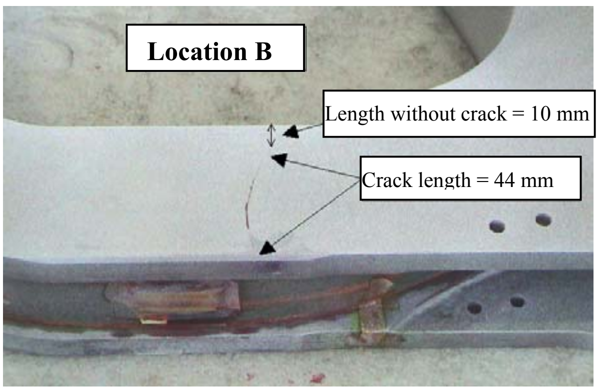

Fig 4: M/R Hub pre-crack for fail safe test

A reduced test spectrum comprehensive of both Start-Stop and High Frequency cycles was applied in order to demonstrate a 2500 hrs inspection interval, assuming a life safety factor of 3. The specimen successfully sustained the spectrum loading and cycles with a very slow growth of the pre-existing crack and was finally subjected to a residual static strength up to ultimate load.

Based on both safe-life and flaw tolerance evaluations the fatigue assessment of the M/R Hub lead to the definition of Unlimited life plus a detailed inspection for one critical area every 2500 hours.

Flaw Tolerance of a Tail Rotor Component Only a few cases occurred where the analysis was not sufficient to demonstrate the "no crack growth" capabilities under the most severe loading conditions of the rotorcraft usage spectrum. Hence the flaw tolerance substantiation was accomplished by means of an experimental activity on full-scale pre-cracked specimens.

This methodology was successfully applied to a Tail Rotor component, namely the T/R Blade Damper Attachment.

This component, which is made of Al7475, has the function of connecting the T/R Damper to the T/R Blade; it is fixed to the root area of the blade by means of two standard bolts and to the damper rod end by means of a bolt (Fig 5). The part is mainly affected by the T/R Damper load, characterised by both high frequency vibratory conditions and low frequency cycles due to the damper centrifugal force range from rotor start to rotor stop.

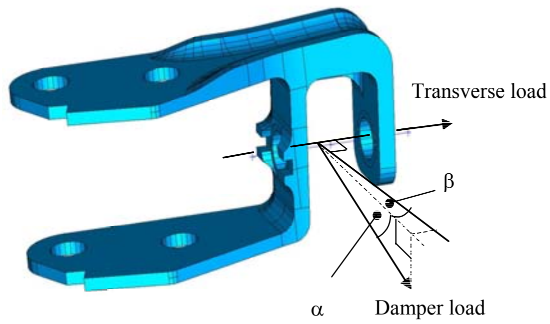

Fig 5: T/R Blade Damper Attachment

The component was firstly subjected to a complete safe-life test campaign that resulted in different failures at the most stressed areas as shown in Fig 6.

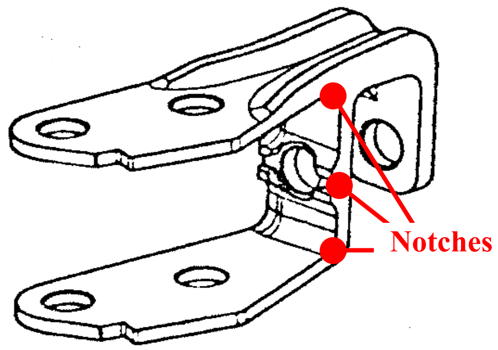

Fig 6: Failures location

A no crack growth analysis was then carried out assuming the presence of a 0.2 mm crack in the most critical areas as highlighted by both the detailed FEM analysis and the safe-life test results. A 0.2 mm crack was assumed considering that accidental damage in service is unlikely to occur, due to the high position of the rotor, and that a dedicated inspection was prescribed in the Maintenance Manual after any maintenance operation with heavy tools performed on any of the tail rotor parts.

The analysis showed negative reserve factors, i.e. the maximum load cycle of the usage spectrum was higher than the material allowable for no growth of a crack of the same size. Therefore a test was run on a pre-damaged full-scale component with the aim of demonstrating experimentally its no growth capabilities. Three 0.2 mm radius semi-circular flaws were introduced at the most critical locations (see Fig 7 as an example) by means of the electrical discharge machining process, which has the characteristic of creating at the bottom of the flaw several micro-cracks without residual stresses, as evidenced by S.E.M. analysis (Ref 2). For this reason the flaws introduced in the part can be realistically considered cracks.

The test was run applying, conservatively, the damper load cycle ranging from minimum to maximum of the whole usage spectrum and resulted in no propagation of the initial flaw after the application of 10^7 cycles, as confirmed by a final NDT inspection with dye penetrant method.

Based on both safe-life and flaw tolerance evaluations, the fatigue substantiation of this component resulted in Unlimited life and no mandatory inspections.

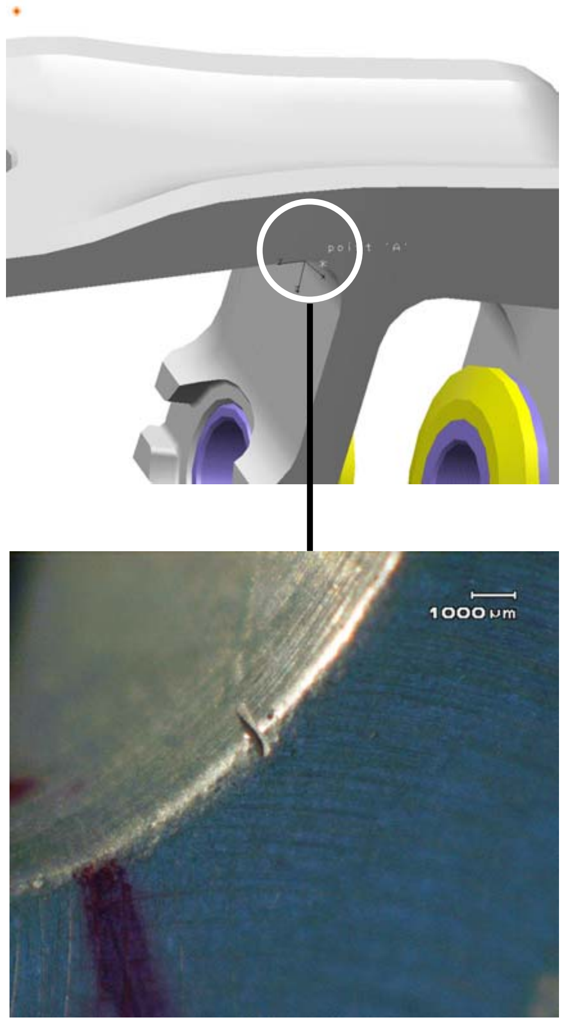

Figure 7: T/R Blade Damper Attachment 0.2 mm initial flaw

Flaw Tolerance of Fuselage Components

Compliance to fatigue and flaw tolerance requirements was also achieved for the fuselage by a combination of safe-life and damage tolerance assessments.

A full conventional safe-life analysis was carried out for all the airframe principal structural elements, highlighting critical areas and allowing determining the most critical loading conditions to be applied in the fuselage full-scale test. The full-scale safe-life test, which is currently in progress, consists of two different phases on two separate specimens: the first one is devoted to the tail structure clearance while the second one to the fatigue clearance of the upper deck critical locations.

The helicopter fuselage structure was classified into different parts depending upon the material selected, the manufacturing process, the type of loading and the stress level, highlighting the areas where a detailed flaw tolerance assessment was deemed necessary. The most critical areas include Main Gearbox Attachments and Upper Deck backup structure, Engine Attachments and backup structure, Tail Rotor Gearbox Fittings, Fin, Tailcone and Tail Assembly to Rear fuselage joint. Two different analytical approaches were used to fulfil the compliance with flaw tolerance requirements, for a total of about 2000 analysed locations.

As for rotor parts, all the well controlled machined components, therefore with a better quality control in manufacturing compared to airframe parts, were analysed using the "no crack growth" approach starting from defect sizes of 0.38 mm or 0.25 mm.

For all the other parts, typically riveted holes at joints, crack growth propagation analysis were performed starting from crack sizes of 1.27 mm. Material fracture mechanics properties were derived experimentally by means of a dedicated coupon testing campaign, showing a good agreement with existing bibliographic data for Aluminium 2024. The crack growth analysis was run using the AFGROW software (Ref 7) and a residual static strength analysis also supported the final results of the assessment.

For some components the analysis were performed starting from the loads directly measured during the certification Load Survey on the instrumented prototypes, for other locations the fatigue spectrum was derived from a dedicated FE Model of the complete airframe.

The full flaw tolerance analysis, supported by the safe-life analysis, lead to the establishment of scheduled inspection intervals, only for the most critical locations, ranging at present from 100 to 300 hrs but suitable of substantial improvement based on completion of the fatigue tests and refined analysis.

Flaw Tolerance of Composite Parts

All the composite parts, namely the Main Rotor Blade, the Tail Rotor Blade, the Main Rotor Tension Link and the Horizontal Tailplane, were substantiated by means of dedicated and extensive experimental activities, in accordance with AC 20-107A (Ref 8) guidelines and on the basis of the experience gained during the EH101 qualification.

The safe-life methodology defined in AC 20-107A is actually a flaw tolerant safe-life method as it prescribes to consider inherent manufacturing flaws and impact damages.
Fatigue tests were indeed performed using specimens representative of the actual manufacturing process, including therefore production defects as accepted by the standard technological process; impact damages at Barely Visible Level were also added in order to cover damages which are reasonably expected during the life of the rotorcraft and could remain undetected.
The aim of the tests was to demonstrate the "no growth" of the introduced damages under cyclic loading up to the desired life and the residual static strength up to ultimate load.

An additional assessment was carried out in order to clear the maximum defect likely to occur during the life of the part but whose effect is clearly detectable. In this case dedicated tests were carried out in order to determine a suitable inspection interval at which the damage could be safely detected and the part replaced or repaired. Residual static strength was demonstrated in this case up to limit load.

Threat assessment and susceptibility evaluation
As for metal parts a detailed hazard assessment was carried out in order to establish the most likely damage scenario each component could be subjected to during its whole operational life. A variety of threats were taken into account and it was established that a 50 J impact was a conservative estimate of energy cut-off value for BVID (Barely Visible Impact Damage).

For CVID (Clearly Visible Impact Damage) the upper realistic energy of 30 J was assumed as cut-off. Lower energies could be associated to locations not exposed to maintenance operations with heavy tools.

The actual energy level for the different areas was established on the basis of dedicated susceptibility tests where different specimens were subjected to impacts at different energy levels with both blunt and sharp impactors at the most critical locations.

Fatigue tests The specimens were subjected to fatigue loading representative of the helicopter usage spectrum, including a few blocks of both high frequency and start-stop loads. Amplification factors were applied to the oscillating components of the loads, taking into account material and manufacturing process variability and environmental effects as the test was performed under Room Temperature and Dry (RTD) conditions. Typical factors used were 1.5 for the material variability and 1.1 for the environment effect.

A separate evaluation was carried out to clear the Impact Damage at Clearly Detectable level; to this purpose additional tests were conducted on impacted specimens with CVID that were then subjected to a reduced fatigue spectrum, followed by a static test up to limit load. Lower scatter factors were used in this case.

The experimental activities resulted in the establishment of a retirement time plus an inspection interval for M/R and TR Blades and an unlimited life with no scheduled inspection for the M/R Tension Link and the Horizontal Tailplane.

Conclusion

Fatigue and flaw tolerance requirements were successfully met with the Certification of the new AB139 Twin Engine Medium Helicopter. The fatigue substantiation was achieved by a combination of conventional Safe-Life methodologies and Flaw tolerance approaches leading to the definition of retirement times and/or inspection intervals for all the principal structural elements.

Several methodologies were identified and implemented addressing different design features, materials and manufacturing processes.

In particular a reliable and cost effective approach was implemented for the assessment of all dynamic components, providing an improvement of fatigue reliability and allowing defining a practical inspection plan for the operative rotorcraft.

Useful guidelines and criteria were defined and areas of improvement were highlighted for future applications of rotorcraft structures, from the early design phases to the final qualification process.

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