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APPLICATION OF DAMAGE TOLERANCE PRINCIPLES TO THE DESIGN OF HELICOPTERS

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Abstract. *Helicopters are very specific aircrafts, for which the design against fatigue phenomena is a particularly important and complex problem. The peculiarity lays in the load spectrum that is composed by a high number of low-amplitude cycles, which result from the mechanical rotation of the rotor blades.*
The fatigue design methodology most commonly applied by the helicopter community was based on the Safe-Life philosophy, applied anyhow with a particular approach.
Since 1989, the Airworthiness Regulations evolved towards the application of Damage Tolerance principles also to rotorcraft. This change has forced the helicopter industries to review their design methodologies, and to face new problems, linked with fracture mechanics applications to their typical structures. Flaws, accidental damages and manufacturing discrepancies must be accounted for, in addition to the retirement life based on Safe Life. Two approaches are mainly used to improve the fatigue assessment: establish the retirement life of parts considering the possibility of initial flaws or cracks of realistic size, or supplement retirement lives by inspection plans defined on the basis of test and analysis.
If safety is assured through inspections, with intervals defined by means of crack growth analysis, it is obvious that more refined crack growth models are necessary, because most of those currently used in the fixed wing industry are based on semi-empirical basis and contain hidden tuning factors that are related to the typical aeroplanes spectra. Moreover, unnecessary conservatism gives rise to very short inspection intervals, that cannot be practically implemented by operators in an inspection program.
The paper reviews the current requirements and presents and discusses the methodologies that the helicopter industry adopts for demonstrating compliance with such regulatory requirements. In addition, recommendations are given on research and development activities required for refining the defined methodologies and for their better implementation.

1 INTRODUCTION

Helicopters are very specific aircrafts, facing very demanding situations, particularly for what concerns the fatigue load environment, as a consequence of the high frequency dynamic loading, induced by the aerodynamic field of the rotor blades. The fatigue spectra have a very high number of cycles per flight, much more than in a fixed wing aircraft, and so the methodologies used by the helicopter structure manufacturers are directly linked with such feature. Historically, the Safe-Life methodology has been the traditional design philosophy used to accomplish safe operation, defining a retirement life. Notwithstanding built-in conservatism and large scatter factors, fatigue failure sometimes occurred, before the designated safe life. Obviously in the years consideration has been taken for the possible presence of defects, which may strongly reduce the safety of the structure: a very specific design methodology, called Enhanced Safe-Life, has been defined, and will be described in the following. Manufacturers used to specify in-service inspections, based on experience and judgment not always rationalized.

Under the impulse of the FAA[1], the Regulatory Change to FAR 29.571 was approved in 1989 (Amdt. 29-28), requiring tolerance to flaws and damages and opening the way to the application to rotorcraft of the Damage Tolerance (DT) design philosophy, developed for fixed-wing aircraft and successfully applied for a significant period of time. Three different options are addressed to demonstrate the safety of fatigue critical elements (Principal Structural Elements, PSE) belonging to the helicopter mechanical parts and to the airframe. At the same time, interest for the application of Damage Tolerance (DT) principles comes also from the many operators of aging helicopter fleets (mainly military), who are increasingly facing fatigue related maintenance issues and the pressure for structural life extension programs. The DT approach could significantly contribute to the solution of these problems, conjugating safety with economy.

It is therefore important to describe the various options offered by the current Regulations to perform the design against fatigue of any PSE, keeping also into account the effects of environment, intrinsic/discrete flaws, or accidental damage:

1. Flaw-Tolerant Safe Life or Enhanced Safe Life, an approach that is adopted only in the helicopter community; it is based on a traditional Safe-Life design methodology, that uses special S-N curves, obtained from specimens containing some defects (flaws). Such test results allow to estimate the life of components containing a defect, which may have been introduced during the manufacturing (scratch) or generated during the operation (impact, pitting), etc. i.e. not a fatigue crack. It must be shown that a PSE containing such defects will be able to sustain the prescribed fatigue load cycles for the whole design life or the replacement time, without any fatigue crack nucleation;
2. Damage Tolerance, as it is called by the fixed wing community, while it is more often called Flaw Tolerance by the helicopter community; and indeed the Regulation uses the term “Fail-safe (residual strength after flaw growth) evaluation”. It requires that the structure can retain, after a partial failure, the capability to withstand the limit load; it is based on the use of redundant solutions (fail-safe) or on the slow crack propagation approach (or, preferably, on the no-growth approach). The use of redundant solutions is often possible in stiffened shell structures, and sometimes multiple load paths can be easily introduced. Nevertheless, most dynamic components cannot be other than single load path. In this case, the slow crack growth (or the no-growth) option is the recommended one. Inspection intervals must be determined, in order to ensure that, if a defect is present, this is timely detected and consequent repair actions taken.
3. Safe Life, i.e. the old, traditional design methodology based on the use of S-N curves of pristine specimens, that defines a period of safe operation of a given component

(retirement life). The high number of cycles that are typical of helicopter spectra are such to require a specific design procedure: the crucial point is that the experimental results necessary to trace an S-N curve with adequate confidence level, particularly in the 10^7 cycles range, are difficult (and expensive) to obtain due the closeness with the endurance limit, which causes a number of lower value run-out results. The helicopter community has established a specific approach to the problem of defining a mean S-N curve using a shape curve, i.e. a curve of a given shape, that is fitted to the experimental data, in order to weight differently the data coming from long life test results with respect to those coming from the shorter life region.

The Regulations state clearly that the Safe Life approach can be utilised only after that the application of Damage Tolerance concepts has been demonstrated not to be possible for practical reasons (limitations due to geometry, inspectability, other valid reasons). Typical examples may be landing gear, drive system gears, main rotor and tail rotor shafts, etc..

In addition to the safety issues specified in the Regulations, helicopter manufacturers must also consider the Durability requirement, commonly applied by means of a traditional Safe-Life approach, which so defines a service life. Shorter retirement lives based on the Flaw Tolerance approach or inspections based on fracture mechanics analysis or tests are additionally mandated to manage tolerance to defects, environment or accidental damages.

The paper discusses the fatigue design methodologies and the application of damage tolerance principles to helicopter structures, limiting the attention to metallic components.

2 TOWARDS DAMAGE TOLERANCE

The debate about the introduction of the DT design philosophy to rotorcraft structures started many years ago. Lincoln[2] first showed some practical examples of how it was possible to apply damage tolerance principles, revisiting the traditional Safe-Life design of some components of the HH-53C and HH-60A helicopters. He concluded that application of damage tolerant design concepts required a re-design of the critical parts examined, with a re-assessment of the safety margins of the various components, measured with a different meter with respect to the standard safe-life design. In most cases, the stress levels must become lower, in order to have viable inspection intervals.

It should anyhow be considered that Lincoln conclusions were somewhat influenced by the success of the ENSIP (Engine Structural Integrity Program) program, that is based on the detectability of very small defects, both in the pre-service and in-service phases. Typical sizes of initial crack for damage tolerance qualification of helicopter machined components are 0.125 mm for “initial quality” cracks and 0.38 mm for “rogue” cracks[1]. This accuracy is not necessarily attainable with the required reliability for helicopters[3].

The general objection from the helicopter community was that, due to the typical features of the helicopter spectra, the inspection interval was often (if not almost always) so short that unacceptable costs were charged to the operators, which made it an impractical solution.

Indeed, owing to the particular load spectra, the implementation of the DT approach is more difficult than for fixed-wing aircraft. In particular, recent research has shown that currently available models do not accurately predict fatigue crack growth under helicopter spectrum loading: they show large scatter and tend to be unconservative when applied to the prediction of crack growth lives in helicopter components[4]. Moreover, load interaction effects for the special characteristics of helicopter spectra (many high R-ratio cycles, sometimes interspersed by few underloads) are not properly accounted for[5-6]. In addition, various authors have pointed out that insufficient (or even inappropriate) crack growth data are used for the near-threshold regime in the ΔK vs. da/dN diagram, that is of decisive importance for the high-cycle (vibratory) fatigue that occurs in helicopters[7]. These problems must be studied in depth, since any DT-based design and maintenance concept needs reliable crack growth predictions. A considerable research effort is devoted in this direction, and some recent development, based on the use of a strip yield model, have been presented[8,9].

A last, but obvious observation, is that also the load spectrum, i.e. the stress history in a given location of a PSE, must be known with higher accuracy and research in this direction is highly important. The following sub-paragraph presents some peculiarities of such spectra.

2.1 Load spectra

A few examples are appropriate in order to better describe the peculiarities of a rotorcraft spectrum. In the framework of preliminary load surveys on a new helicopter developed by Agusta-Westland, acquisitions were made of load acting on various parts. Fig. 1 shows the minimum and maximum stress of the various cycles on thin panels belonging to the rear fuselage, in a passenger transportation mission. In the early part of the flight, maneuvers are performed, with abrupt changes of mean and amplitude, followed by a long cruise segment.

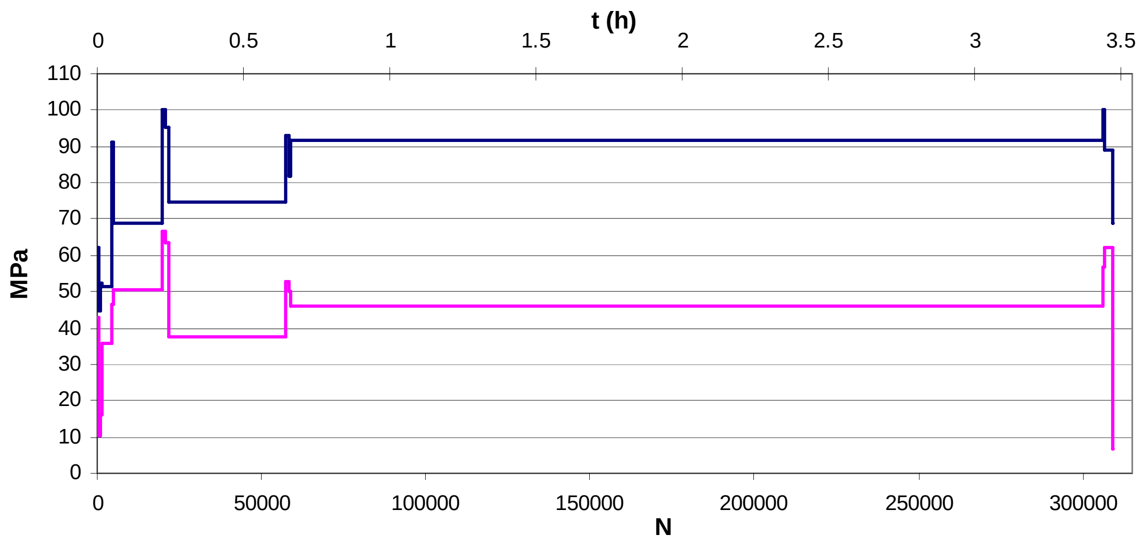

Fig. 1 - Typical spectrum (maximum and minimum stress) on rear fuselage panels in a AB 139 Pax mission

Rotor components are subjected to other type of spectra, dominated by many high R ratio low-amplitude cycles, interspersed with a number of low values minima, consequence of the start-stop cycle (equivalent to the Ground-Air-Ground cycle for fixed wing aircraft structures). Standard load sequences have also been defined, like Helix and Felix[10], test load sequences for articulated or semi-rigid rotors, respectively (Fig. 2). The R values in Helix and Felix range between 0.4 and 0.6 (rotor blades and rotor roots); more recent work indicated that more than 80-90% of the total number of cycles have high R values (0.7-0.9), which is reflected also in other recent standard sequences, such as Rotorix[11] and Asterix[12].

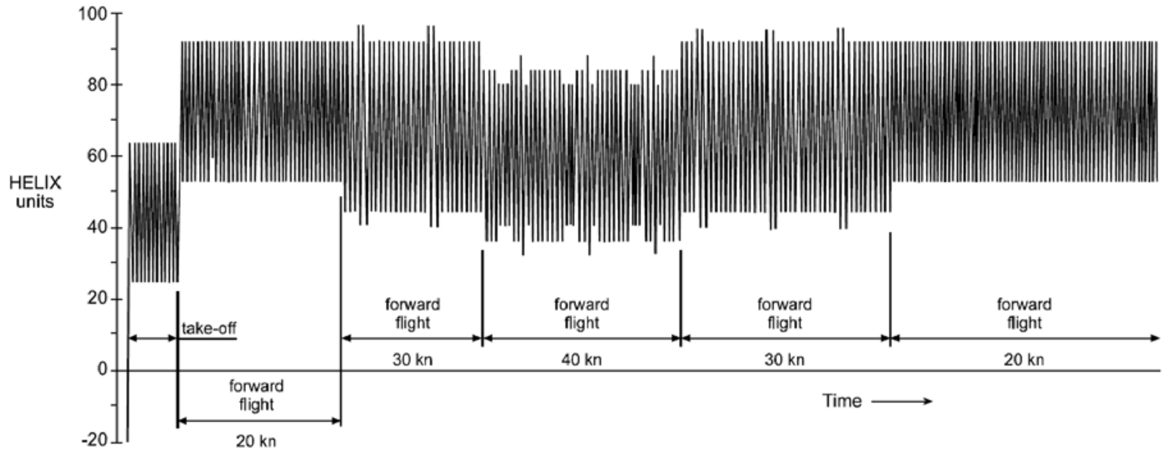

Fig. 2 - Example of the load-time history of the first phase of a training flight in Helix

Great emphasis is being given to the development of Health and Usage Monitoring Systems (HUMS): the FAR regulation indeed points out how the determination of the real operative usage is a fundamental issue, as it should be for the design of every aircraft, but even more critical in the case of helicopters. Just to quote a possible source of variability, in addition to multi-role machines (particularly the military ones, used by different customers in different roles), it is worth mentioning that many versions of helicopters can operate with an external cargo; the occurrence of this situation and its frequency have a large influence on the load spectrum in various components of the rotorcraft. Therefore, many efforts are currently dedicated to develop simple but reliable HUMS, aiming at identifying the time spent in the various flight regimes, with particular emphasis on the most damaging for a given component.

3 COMPLIANCE WITH REGULATIONS

The Advisory Circular AC 29-2C provides suggestions and gives more information, in the Miscellaneous Guidance 11 section, about methods suitable to demonstrate Flaw Tolerance of metallic parts.

3.1 Flaw Tolerant Safe Life

The objective of a Flaw Tolerant Safe Life (FTSL) design is to establish a retirement life, i.e. a period of operation of the structure, that may be affected by flaws of expected types and sizes (as defined in the Threat Assessment, defined below), with only routine inspections necessary. More recent discussions suggested to use it also to define an inspection interval for accidental damages; this assumes that residual life to crack initiation is the FTSL. Anyhow, this point is still the subject of a wide debate.

The analyses and/or testing methodologies used in the FTSL approach are similar to those used in the Safe Life approach, with the substantial difference that the test results used in the design phase come from structures or coupons with flaws of the expected size, rather than from structures or coupons in the as-manufactured condition. This implies that a "Threat Assessment" must be preliminarily performed, i.e. an analysis of the possible defects that can be tolerated by the quality assurance system for the component under examination, also considering the manufacturing route. A flaw is defined as an intrinsic imperfection (such as inclusions, forging laps or porosity) or discrete damage (such as gauges, nicks, corrosion, fretting, wear and impact) that could be expected during the manufacturing operations or in operations. Machined parts or cast parts have evidently different defect typologies.

The size of the Flaw has always been a subject of debate, from the cradle of the Damage Tolerance philosophy. It is based on a Threat Assessment, that has the purpose of defining the potential damages, consequences of both manufacturing processes and in-service operation. The USAF specified the requirements for initial damage sizes in MIL-A-83444 and soon after some debate about their arbitrariness arose, together with comments about its supposed level of conservatism. At present, these dimensions are normally used in fuselage structures, which are typically stiffened shell structures, with an inherent level of redundancy that places them in the Fail Safe solutions. A number of statistical analysis of the size of defects found by production quality controls have been presented in various papers by helicopter manufacturers, and are a good basis for estimating the pre-service damage. An example of Agusta-Westland findings[13] in machined aluminium parts is shown in fig. 3, but data with similar statistical distribution have been published also by Sikorsky[14].

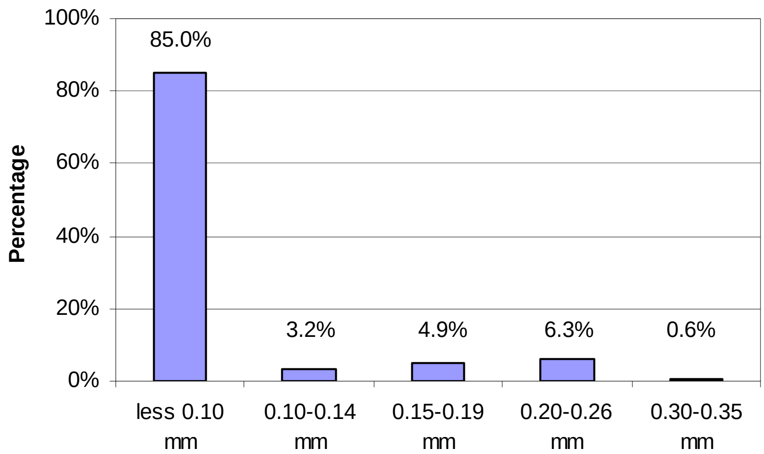

Fig. 3 - Distribution of scratch sizes in aluminium parts (source: Agusta-Westland).

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The common design practice for FTSL is to establish a S-N curve, based on fatigue tests performed on flawed coupons, by fitting with traditional least square technique a curve shape to the experimental data. The mean curve so obtained must be scaled to a working curve by means of appropriate scale down factors; due to the field of cycles that are of interest, it is more meaningful to apply a factor in stress at high number of cycles and a factor in life (similarly to the fixed wing aircraft community) at low number of cycles (typically, for $N < 10^5$). Normally, the reduction between mean and working curves is smaller than in the pure Safe Life analysis, and equivalent to μ-2σ, in order to avoid a dual penalization.

Anyhow, Airworthiness Authorities would like this methodology, considered grounded on less sound basis, to be abandoned progressively and privilege the adoption of Fail Safe solutions.

3.2 Fail Safe solutions

Whenever possible, adoption of redundancy is recommended; multiple elements are such that a possible failure is confined to just one load path, while all the others remain active. This situation relaxes the need for identification of cracks when they are very short, and so it is highly preferred by Airworthiness Authorities and by the operators. Safety is accomplished by means of inspections and so it is fundamental to perform a crack growth analysis, by means of fracture mechanics tools, in order to define inspection intervals: first (or threshold) interval and repetitive interval.

The first inspection interval is assessed performing a crack growth analysis, starting from an initial defect of the size defined by the accuracy of the manufacturing quality control; in this case, it is assumed that the flaw behaves like a crack, immediately ready to propagate, without any nucleation time. This interval is divided by a safety factor, which is at minimum 3, while the repetitive inspection interval is obtained dividing by at least 4 the time between the moment when the crack becomes detectable with the NDI specified in the field inspection and the moment when it reaches the critical value.

In practice, the repeated inspection interval for mechanical parts tends to be too short and so the inspection program is too expensive, in such a way that the first inspection is used to scrap the component.

The procedure summarized is typical for single load path elements, while for multiple load path elements the determination of the repetitive inspection interval follows a slightly different method.

Anyhow, even if the procedure is quite simple and clearly defined, there are a lot of concerns within the helicopter industry, because first applications were extremely disappointing, with inspection intervals that resulted to be unacceptably short. Even if more refined NDI methods are developed, the nightmare for a designer is a crack that remains dormant for a long time, propagated only by the GAG cycle, and that suddenly “awakes” when the dynamic loading cycles become progressively more and more significant, until in a few minutes of flight the crack explodes, reaching the critical value. The problem is indeed also the shape of the crack growth curve, that tends to be quite “unfair” for the inspectors.

The only possibility for the designers is to lower the stress levels in those areas where unacceptably adverse crack growth behaviour is obtained; in other words, the safety of the structure is measured with a different meter, more close to fracture mechanics meter than the traditional meter used before, close to the fatigue endurance meter.

It is therefore evident how in some cases the designers prefer to apply the so-called “No / benign crack growth” concept (preferred to the “Safe Crack Growth”), that takes place when an element with a given flaw does not experience crack growth at all or in a reduced measure within the design life. The objective of this design option is to assure airworthiness for the given PSE by determining a replacement time, rather than an inspection plan. In this case, the analysis and/or tests should consider a more severe initial crack, i.e. the “rogue” crack, defined as the most severe crack consistent with manufacturing, maintenance and operation environment.

The preferred choice is to have a replacement life equal to the design life, i.e. the no-growth option. In this case, operators accept willingly the weight penalization associated with the reduction of stress levels.

4 IMPLEMENTATION OF A FLAW TOLERANT DESIGN METHODOLOGY

The previous paragraph has given a rather detailed picture of the difficulties in certifying helicopter structures according to the Flaw Tolerance principles, particularly following the “Fail Safe (residual strength after flaw growth)” methodology. There are various open questions, or issues where research and development efforts are strongly required to make application of DT principles easier. Indeed, the FAA has in the past years presented a “Roadmap for Application of Damage Tolerance Principles to Rotorcraft”, [15], where various topics where research was devised were illustrated. Among others, there were spectra measurements and usage monitoring, threshold investigation in long and short crack regimes, developments in NDI technology, life enhancement, residual stresses, corrosion control.

In the second paragraph, the need for the development of refined and reliable crack growth methods was pointed out, and shortly discussed.

Another open question is linked to the threshold data bases and to the concern associated with the so-called “short crack effect”: in their early stages of life, cracks that nucleate naturally at notches can propagate at lower stress intensities ranges than the long crack threshold and at higher growth rates.

In many crack growth data bases, such as NASGRO or AFGROW, the lower part of the curve is described by the so-called NASGRO equation or assigned by the user, who gives a number of points, for a given stress ratio, through which the program interpolates to find the crack growth rate corresponding to a given ΔK. In general, the values of the threshold are deduced from tests on long crack (LC) specimens, obtained by the load shedding technique. In general, the threshold value for $R=0$ is assigned, and the values for the other stress ratios can be deduced by a given relationship. The stress ratios that are of major interest for the rotorcraft community are around 0.7-0.8, which are not well treated by this procedure. Moreover, these are LC thresholds, and to consider the short crack effect, the LC values must be elaborated by means of another relationship, where a material intrinsic constant is present.

This procedure does not produce reliable values.

To circumvent these difficulties, Agusta-Westland has developed a data base, extended to the most common metallic materials used in the rotorcraft construction, relevant to the study of growth - no growth conditions. The approach utilized is the one proposed originally by Kitagawa and ri-elaborated by Murakami for the study of fatigue behaviour of high strength steel: in a stress amplitude vs. crack size diagram, regions of no-growth are identified. In a standard Kitagawa plot (fig. 4), the endurance limit and the long crack threshold are combined together to identify such no-growth conditions.

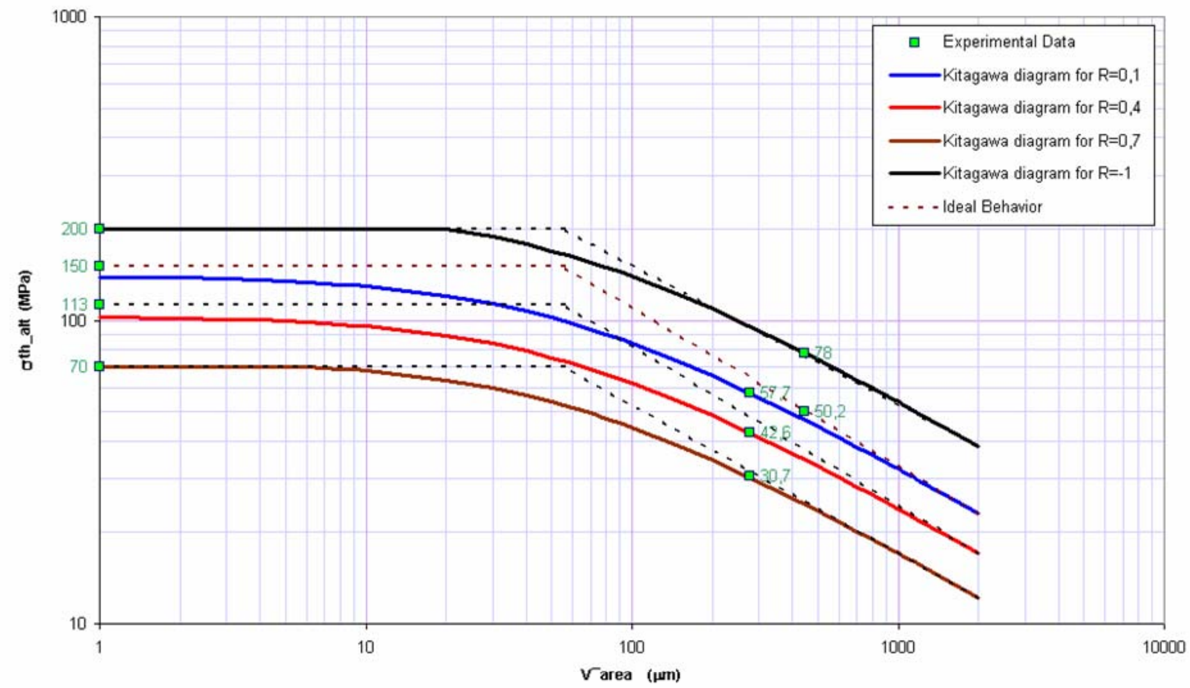
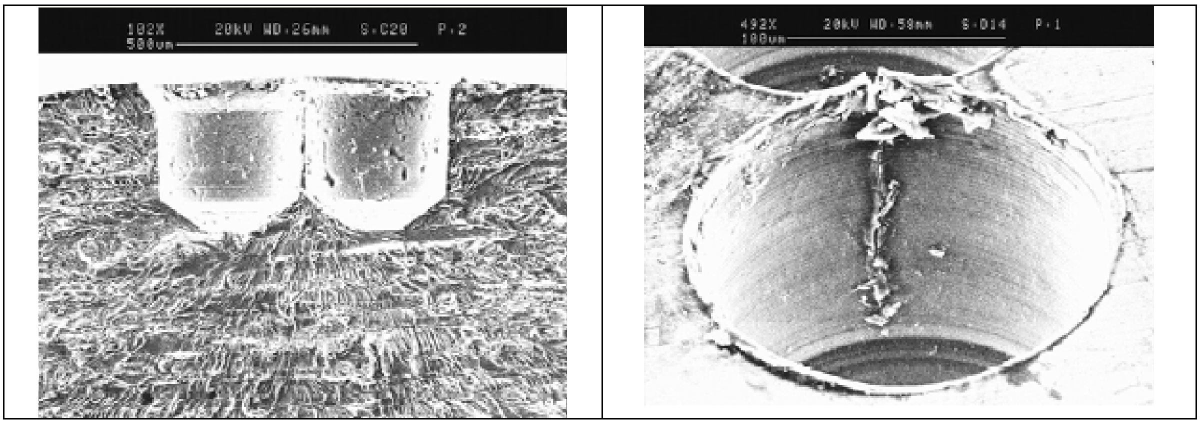


Fig. 4 - Kitagawa diagram for 7475-T7351 aluminium alloy, for various stress ratios.

Details about the experimental procedure have been presented [16,17]; it is important to point out that the methodology is based on the study of a realistic problem, i.e. the nucleation of a short crack in a notched detail. The only difficulty is to introduce a small defect, that must be as realistic as possible. There has been a long debate about the most appropriate method to introduce this defect and the choice has at last fallen on the use of a very small drill (< 2 tenths of mm in diameter) and to introduce two small holes, very close one to the other (fig. 5a). The thin diaphragm between the holes requires a very small number of cycles to break (fig. 5b), so that the defect has the dimension of the projected area of the envelope of both holes. The stress intensity factor is linked to the applied stress and to the defect geometry by simple relationships, and so it is possible to determine the short crack threshold for the material under examination with a reduced number of simple tests: a staircase method is adopted to determine the threshold, defining the no-growth condition. This is a bridge between the endurance test and the fracture mechanics world, an explanation of the endurance limit in presence of defects given on more rigorous basis of fracture mechanics.



a) b)

Fig. 5 - High magnification images of mechanically drilled small holes for threshold study.

An alternative process, based on the use of Electro Discharge Machining, has been studied and qualified for introducing the defect[18]; after that the process has been optimized, a simpler and more robust manufacturing process is obtained, associated with a better control on the defect size.

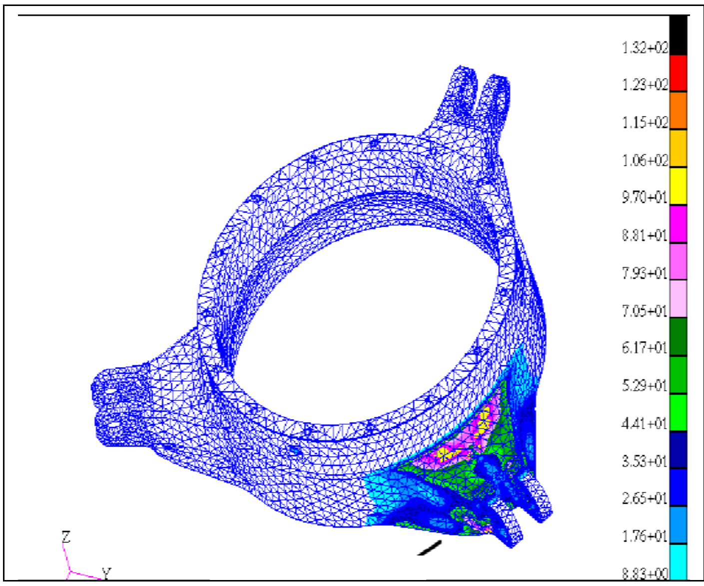


Fig. 6 - Swashplate - The “no-growth” was proved by means of a comparison of a detailed (and validated) stress analysis with proprietary material data.

For the application of the presented methodology to a PSE, all what is required is a detailed stress analysis, as shown in the fig. 6, for the relevant load conditions. The stress in the critical location is found and the comparison with data of the type reported in fig. 4 allows

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the designer to take a decision: if the no-growth condition is demonstrated for the dynamic component of the spectrum and only the start-stop cycle is capable to induce crack growth, the inspection interval can be determined with only reference to the start-stop cycle and so a life longer than the design life is generally assured. Safety does not depend on a specific inspection program, so reducing the efforts of the operators and the maintenance costs, while only routine inspections must be performed (for corrosion, impact damage, etc.).

5 CONCLUSIONS

Application of Damage Tolerance principles to rotorcraft design is becoming progressively more extensive, under the impulse of the Airworthiness Authorities, who after the successful application of such philosophy to the military and commercial aircraft world, as well as to the engine community, are convinced that this is the best way to ensure safety against fatigue cracking and accidental damages.

Specific difficulties have been evidenced by the helicopter industry and a Roadmap with a list of issues where research effort and improvement must be addressed has been defined by the FAA; government and industrial agencies have invested in these directions.

The major difficulties are linked with the peculiar spectra characterized by a very high number of cycles per flight hour, at high stress ratios, interrupted by excursions to low stress levels (e.g. start-stop cycles); this situation makes very problematic to have a slow crack growth and to control safety by means of inspections, at least using current NDE techniques. This view seemed to have been accepted already by the industry, since Eurocopter, Sikorsky, Agusta-Westland and Bell have all done work which indicates that the only practical way to move forward is to demonstrate that defects of a defined size (usually 0.38 mm) in mechanical parts would not propagate. Inspection and quality control during manufacture should ensure that larger flaws do not exist in new components.

For redundant structures, such as (but not limited to) airframe components, Fail Safety can be demonstrated with similar approaches to those developed by the fixed-wing world.

REFERENCES

[1] B. Dickson, J. Roesch, D. Adams and B. Krasnowski, "Rotorcraft fatigue and damage tolerance", Paper 10 in: *Twentyfifth European Rotorcraft Forum*, Rome, Italy, (1999).

[2] J. Lincoln, "Damage Tolerance for Helicopters", Proceedings of the 15th ICAF Symposium, Jerusalem, EMAS Publ., pp. 263-290, (1989).

[3] M.L. Overd, "Damage Tolerance in Helicopter Airframes - Is Crack Growth Practical ?", Proceedings of the 20th ICAF Symposium, Seattle, EPIC Publ., Dayton (Ohio), USA, pp. 521-535, (1999).

[4] P.E. Irving, J. Lin and J.W. Bristow, "Damage Tolerance in Helicopters. Report on the Round Robin Challenge", presented at the *American Helicopter Society 59th Annual Forum*, Phoenix (Arizona), USA, (2003).

[5] V. Zitounis and P.E. Irving, "Fatigue crack acceleration effects during tensile underloads in 7010 and 8090 aluminium alloys", *International Journal of Fatigue*, Vol. 29, pp.108-118, (2007).

[6] P.E. Irving, V. Zitounis and R. Buller, "Prediction of Fatigue Crack Growth Rates for Damage Tolerant Helicopter Design - the Role of Load Spectrum and Material", *2003 ICCES Conference*, Corfu, Greece, (2003).

[7] R.E. Vaughan and J.H. Chang, "Life Predictions for High Cycle Dynamic Components Using Damage Tolerance and Small Threshold Cracks", *American Helicopter Society 59th Annual Forum*, Phoenix (Arizona), USA, (2003).

[8] J.C. Newman, jr., P.E. Irving, J. Lin and D.D. Le, "Crack Growth Predictions in a Complex Helicopter Component Under Spectrum Loading", *American Helicopter Society 59th Annual Forum*, Phoenix (Arizona), USA, (2006).

[9] M.J. Bos, "Development of an Improved Model for Predicting Fatigue Crack Growth in Aluminium Alloy Helicopter Airframe Components", presented at the *24th Symposium of the International Committee on Aeronautical Fatigue*, Naples (Italy), (2007).

[10] P.R. Edwards and J. Darts, "Standardised fatigue loading sequences for helicopter rotors - Helix and Felix - Part 2: Final definition of Helix and Felix", RAE Technical Report 84085, Royal Aircraft Establishment, Farnborough, UK, (1984).

[11] Anon., "A Standardized Fatigue Loading Sequence for Helicopter Main Rotorhead Structures (Rotorix)", Damage Tolerance Group Report CU/927/N/27, Cranfield University, UK, (1996).

[12] Anon., "A Standardized Fatigue Loading Sequence for Helicopter Airframe Structures (Asterix)", GKN Westland Helicopters, Report RP980, UK, (1996).

[13] U. Mariani and M. Vicario, "Flaw Tolerance Assessment of Rotorcraft for Civil Certification", Proceedings of the *23rd ICAF Symposium*, Hamburg, pp. 55-66, (2005).

[14] D.O. Adams, "Development of Flaw Tolerance Methodology and Application to the S-52 Helicopter", *Workshop on Fatigue Design of Helicopters*, University of Pisa, (2002).

[15] D.D. Le, "A Roadmap for Damage Tolerance Implementation in Rotorcraft", *Workshop on Fatigue Design of Helicopters*, University of Pisa, (2002).

[16] S. Beretta, M. Giglio, U. Mariani and M. Vicario, "Flaw Tolerance Evaluation of Helicopter Rotor Parts", Proceedings of the *22nd ICAF Symposium*, Lucern (Switzerland), pp. 689-702, (2003).

[17] U. Mariani and M. Vicario, "Application of Flaw Tolerance Methodologies to Rotorcraft Fatigue Qualification", *European Rotorcraft Forum*, (2005).

[18] M. Ciprandi, U. Mariani, G. Ratti, S. Beretta and M. Giglio, "Evaluation of Fatigue Threshold of Metallic Materials with Cracks", *Aerospace Testing Expo,* Hamburg, (2006).