HELICOPTER FUSELAGE CRACK MONITORING AND PROGNOSIS THROUGH ON-BOARD SENSOR NETWORK

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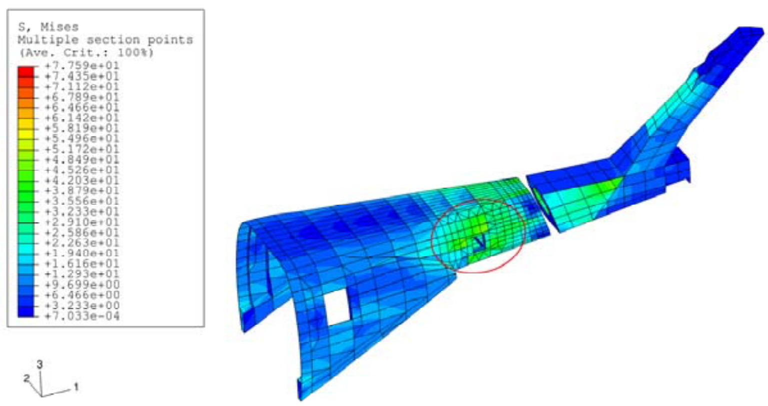
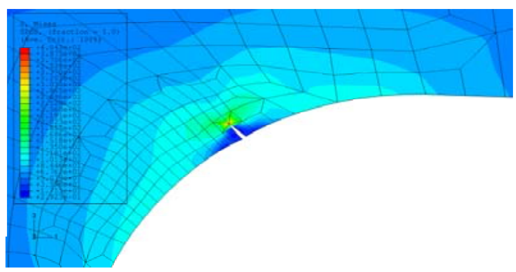
An advanced method for maintenance cost reduction and structural safety improvement of helicopters frame is presented. Specifically residual life evaluation is based on real time acquisition of the crack damage, by means of dedicated sensor network, and focused FE model of the fuselage.
Recently, the European Defence Agency has proposed a Joint Investment Programme on Innovative Concepts and Emerging Technologies (JIP-ICET). Following this call, the HECTOR proposal has been submitted for evaluation; the purpose of this proposal concerns the application of methods for identification, monitoring and prognosis of potential damages (like cracks) in the fuselage of a helicopter.
The key issues of this activity will be related with structural assessment, real time acquisition and advanced data fusion process. In particular advanced models for the stress assessment, will be used for the identification of the most critical area for crack nucleation and growth; these area will be monitored with a on board network of last generation sensors (Comparative Vacuum Monitoring, Optical Fiber Sensors, Crack Propagation Gauges, etc). Moreover the final research aim is to obtain a reliable method to assess the damage accumulated in the fuselage by means of on-line advanced prognostic models that allow the real time definition of schedule for periodic and special inspections.
Thus an increase in the safety of the aircraft by prognostic and monitoring of the frame is the main objective; in addition, the results achieved will be useful in the cost reductions regards maintenance operations and life extension for aged aircraft.
In this paper, a review of the state-of-the-art concerning the Structural Health Monitoring applied on helicopter fuselages will be describe In particular the review will be focused on the numerical FE models of cracked structures and on residual life evaluation as will be applied on HECTOR project.





Helicopter fuselage crack monitoring and prognosis through on-board sensor network

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Abstract

An advanced method for maintenance cost reduction and structural safety improvement of helicopters frame is presented. In this approach, residual life evaluation is based on the real time acquisition of the crack damage, by means of dedicated sensor network, and a focused FE model of the fuselage.
Recently, the European Defence Agency has proposed a Joint Investment Programme on Innovative Concepts and Emerging Technologies (JIP-ICET). Following this call, the HECTOR proposal was submitted for evaluation; the purpose of this proposal concerns the application of methods for identification, monitoring and prognosis of potential damages (like cracks) in the fuselage of a helicopter.
The key issues of this activity will be related with structural assessment, real time acquisition and advanced data fusion process. In particular advanced models for the stress assessment, will be used for the identification of the most critical area for crack nucleation and growth; these area will be monitored with a on board network of last generation sensors (Comparative Vacuum Monitoring, Optical Fibre Sensors, Crack Propagation Gauges, etc). Moreover the final research aim is to obtain a reliable method to assess the damage accumulated in the fuselage by means of on-line advanced prognostic models that allow the real time definition of schedule for periodic and special inspections.
In this paper, a review of the state-of-the-art concerning the Structural Health Monitoring applied on helicopter fuselages is described. Moreover an example of application of a numerical FE models of cracked structures and on residual life evaluation is presented.

1. Introduction

The structural safety of helicopters is guaranteed with a deeply fatigue analysis in the design phase[1,2,3] and a clear schedule of inspection during life. However the design and maintenance of helicopters are particularly important and complex respect to the aircrafts. The peculiarity lays in two ways: the load spectrum that is composed by a high number of low-amplitude cycles, which result from the mechanical rotation of the rotor blades (severe vibratory loads), and the low velocity impact damage. This type of loads can lead to high fatigue damage accrued in short time or rapid crack propagation from accidental flaws or damages. Due to this the condition based maintenance for fatigue is not the most typical approach for helicopter fatigue, since frequently requires short inspection intervals. The FAR (Federal Aviation Regulations) standard 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. Actually, adoption of redundancy and low stress level is recommended but cannot be always adopted. Furthermore failures need to be monitored also due to not directly predicted damage like accidental damage, environmental damage, life extension[4].

Considering these issues the development of Health and Usage Monitoring Systems (HUMS) has received considerable attention from the helicopter community in recent years[5,6] with the declared aim to increase flight safety, increase mission reliability, extend duration of life limited components and of course reduce maintenance cost[7].

Structural Health Monitoring SHM seems capable to help in reducing the maintenance cost, which is about 25 per cent of the direct operating cost of the helicopter[5] and plays an important role especially in the case of the ageing helicopters.

With the aim to reduce the direct maintenance costs of the airframe without compromising any safety or reliability issues, in the last years some vibration monitoring systems were integrated into the gears and rotating shafts of the helicopter transmission, and now the HUMS is an integral part of the new generation helicopters. However is important to underline that HUMS is mainly based only on monitoring vibrations generated in components critical for the flight performance. Data are continuously recorded using accelerometers based on piezoelectric, processed and compared to a threshold value that describes the allowable damage. Actually HUMS is installed on various helicopters such as the Sikorsky SH-60B Black Hawk, S-61, and S-76, the AgustaWestland EH-101 and AW139, the Boeing CH-47D Chinook, MH-47E,WAH-64 Apache, UH-60A Black Hawk, and AH-64D Apache but are based only on the principle of usage monitoring acquiring loads and vibration without any direct indication of incipient failure. An exception is the on-board diagnostic systems like the structural life monitoring of the V22.

Thus respect to the structural helicopter fuselages, only some attempt to apply reliable methods to monitoring directly on-line the damage nucleation, accumulation and propagation during life were carried out. In this field, and in particular in the military applications, an integrated and reliable system for monitoring the damage in the fuselage and for evaluating the time inspections and remaining life (prognosis) is missing.

For what concern the aircraft scenario, Boeing and Airbus are starting to use HUMS, in particular approximately 150 Comparative Vacuum Monitoring CVM sensors were used in the qualification of GLARE on the A380 full scale fatigue test rig[8].

2. Sensor technology for SHM

The main aim of the sensor network directly dedicate to the health monitoring is the real time evaluation of the degradation mechanism. Thus candidate sensors would identify when a fatigue crack has initiated or when an existing crack grows, and monitor crack growth in the most stressed hotspot.
Considering also the future requirement to be embedded in real operating frame, sensor reliability becomes important when measurements are required over a long period of time, as long as costs, environmental resistance and weight.
Specifically for the SHM purpose of aerospace frames the potential sensor types are crack gage, comparative vacuum monitoring CVM, Acoustic-Ultrasonic AU, Acoustic emission AE, Lamb waves as well as fibre-optic strain in particular, Fibre Bragg Gratings (FBG), Eddy Current, MEMS etc. A change in the material local behaviour (and hence a damage) can be picked up and localized by an array of such sensors.
The crack gage is a well known sensor for crack monitoring and consists of a thin coupon that can be bonded to a structural member in the vicinity of a known stress raiser. The gage utilizes the indirect potential drop method for measuring the crack growth. Comparative Vacuum Monitoring CVM[8,9] technique provides a novel and interesting method for crack initiation detection, and long term monitoring of fatigue cracks in aircraft structures. Open cracks generate leaks in a series of galleries bonded to the structures. Pressure changes in a system of hair-fine capillaries provide an indication of structural defects (cracks, corrosion and loss of bonding contact), tracked with a remote monitoring device. CVM has the ability to monitor external surfaces of materials for crack initiation, propagation and corrosion. In addition, CVM sensors can also be embedded between components (e.g. lap joints) or within material compounds such as composite fibre. Airbus and Boeing are using CVM technology both for laboratory and structural tests. In particular Airbus has developed sensors for early detection of fatigue cracks within riveted lap and butt joints. This is achieved by placing the sensor between the lap/butt joint components. The sensors are inert and may be left in-situ on the structure for real time or periodic monitoring. Further, the sensors do not suffer the restriction of wire crack gauges, which have a significant probability of failure if a bending moment is repeatedly applied to the sensor. CVM sensors can be placed in fatigue critical hotspots and are sensitive enough to pick up cracks as they initiate Test with CVM on helicopter has been executed on for Sea King helicopters being operated by the British Royal Navy and Royal Air Force.
AE/AU technology can detect structural defects long before possible catastrophic failures[8]. This is possible because discontinuities will produce detectable emissions, long before structural integrity is compromised and structural failure occurs. Acoustic Emission (AE) are based on elastic radiation generated by the rapid release of energy from sources within a material (impacts, crack initiation, crack growth, delamination). AE sensors are the passive small piezoelectric sensors mounted to a convenient surface of the material. The sensor response and front end filters can remove frequencies below about 100 kHz, which includes most audible noise. The result is that acoustic emission can be used to monitor a structure for active damage even when ambient noise levels are extremely high. Acoustic emission is sensitive enough to detect newly formed cracks. AE helicopter HUMS has been used as device for detecting damage in SH-60 helicopter drive trains. Acoustic-Ultrasonic (AU) is a technique that sends acoustic waves into the structure and intercepts them when they emerge on the other side. Deviations from the

expected wave pattern indicate the presence of discontinuity like cracks. The European project AISHA Aircraft Integrated Structural Health Assessment aimed to contribute in realizing an aircraft monitoring technology by using ultrasonic Lamb waves as the basic sensing principle(11). The special potential of Lamb waves for damage detection arises from their propagation capabilities. Lamb waves are guided acoustic waves propagating in plate-like structures. In the case of damages, the propagation of an ultrasonic Lamb wave will be disturbed resulting in a characteristic reflection and attenuation pattern. Experiments performed within AISHA on lab-scale and on selected full-scale parts showed the ability of Lamb waves or other guided waves to give information on correlations between acoustic parameters and damage in structural parts.
Fibre Bragg Gratings (FBG) are fibre-optic sensors with elastic properties that mirror those of the tested material, and can be used to monitor temperature, thermal and mechanical stress, damage caused by collision or impact, and delaminations. These sensors have the advantage of being light weight, having all passive configurations, low power utilization, immunity to electromagnetic interference, and bandwidth, compatibility with optical data transmission and processing, long lifetimes, low cost and high sensitivity, comparing with typical electrical strain gauges, such as resistive type, piezoelectric, semiconductor, and capacitance gauges. These sensors in fact generally have a small dynamic range, gauge factors of less than 5, and are affected by environmental conditions such as moisture and temperature.
Principle of Eddy Current Testing (ECT) originates from the electromagnetic induction phenomena. A primary alternating electromagnetic field generated by a sensor induces eddy currents in a tested conductive object. A secondary electromagnetic field created by eddy-currents counterworks to the primary exciting electromagnetic field. When the pattern of eddy-currents flow is altered by the presence of a discontinuity, the resultant electromagnetic field reflects those changes. Thus, it is possible to observe structural changes, i.e. cracks and corrosion patches, in a tested object in this way. ECT is the contact-less method with very high sensitivity for surface breaking defects. Moreover, the method provides a possibility to evaluate dimensions of a defect. Eddy Current Testing Foil Sensors (ETFS) are especially suitable to be integrated in an on-board sensor network for on-line monitoring. However, it can only be used for metallic structures.
Using of MEMS is also an interesting option. Microelectromechanical systems (MEMS) are miniature electromechanical sensor and actuator systems. Advances in MEMS technologies have led to dramatic reductions in size, power consumption, and cost for wireless communications. Their small size allows them to be used in applications where conventional sensors and actuators would be intrusive. Because of the economies of scale achievable from the conventional chip manufacturing processes, they can be mass produced and copiously applied in a cost-effective manner. These types of sensors have been used on aircraft structures due to their minimal aerodynamic disruption(12). MEMS sensing technologies are appropriate for local SHM applications, such as those that identify crack initiation, propagation and corrosion, deploying a large number of MEMS devices over a large area in a cost-effective manner is a difficult problem.
However, can be noted that reliability and measurement accuracy are still problems that must be addressed for successful implementation of MEMS technologies together with wireless data transmission and connection to a power source.

3. Crack propagation in helicopter fuselages

The current state of the art is mainly represented only by devices that monitor vibrations generated in components critical for the flight performance. The purpose of the HECTOR proposal is the creation of a on-board demonstrator for helicopters that permits, on the basis of a complete knowledge of the stress state on the fuselage (advanced FE model) and the data collected by a low power sensor network positioned in the most critical areas, to perform the Embedded Structural Health Monitoring and the on-line evaluation of the degradation mechanism based on advanced damage criteria. This package could be mounted on new helicopters to define a more reliable, efficient and economic program of inspections, or on ageing airframes to perform a life extension of the machines.

Traditionally, cracks must be monitored by conducting visual and non destructive testing on a large area of the aircraft during the operating life. Often this can require significant downtime to get access to such an area with a handheld testing machine. The advantage of the method here described is that once the sensors network is installed and set up, inspection is possible not only without disassembling parts but also in continuum using a real time strategy to detect failures (existence, location, type, extent) and for the prognosis of their effect on the overall reliability of the structure. The application of low-power sensors will bring these benefits with low cost in terms of overall energy consumption of the helicopter.

The base of this advanced prognostic program is the availability of finite element models (FEM) of the structure with and without damage. This is the key step for the extraction of information from sensor data, i.e. the identification of the damage-sensitive properties, derived from the measured dynamic response, which allows distinguishing between the undamaged and damaged structures. Different types of sensors will be considered during the research activity, selecting the more suitable to perform a full structural approach in the helicopter fuselage. The main goal is to make non-destructive testing technology to become an integral part of the aircraft structure itself (Embedded Structural Health Monitoring – ESHM). In case of damage, the system directly identifies the location and follows-up actions that can be taken. The ESHM approach on helicopter structure can have a strong impact as a means of possibly revolutionizing the current structural monitoring and design process.

Thus the application of ESHM on helicopter fuselage will develop the following objectives, in which an innovation in methods, systems and/or hardware is considered:

- the creation of a complete FE model of the fuselage, both in healthy and damaged situation, considering the different stress state caused by a progressive crack in the most critical areas;
- the definition of an integrated network of sensors of different types (crack gages, comparative vacuum monitoring, fibre sensors, eddy current, etc.) chosen on the basis of the facility of dismantling and the data acquisition time (on-line or off-line mode);
- the construction of an advanced communication system, choosing the most reliable between wireless and traditional one;
- the acquisition of an automated advanced signal processing with filtering techniques applied such that the signal resulting from damage can be clearly identified from any other noisy signals being around.

3.1 Helicopter fuselage FE structural analysis

The purpose of this investigation is to describe a procedure to evaluate the crack propagation into a helicopter fuselage using an advanced FE model. This method requests to know the behaviour of stress intensity factor K_I versus crack length *an* introduced for hypothesis on the most stressed zone of the helicopter fuselage.
In particular the following methodology is developed:

- definition of the stress distribution in the helicopter fuselage with the operating loads applied on tail rotor;
- choosing of most nominal stressed zone to introduce the crack;
- elaboration of submodel with crack to determine the curve stress intensity factor K_I vs. crack length *a*.

3.1.1 Helicopter's FE model

To find the stress on the whole helicopter fuselage, a finite element model using the ABAQUS/Standard 5.8 program was developed; in Figure 1 the full FE model and the detail of the skin thickness is shown.

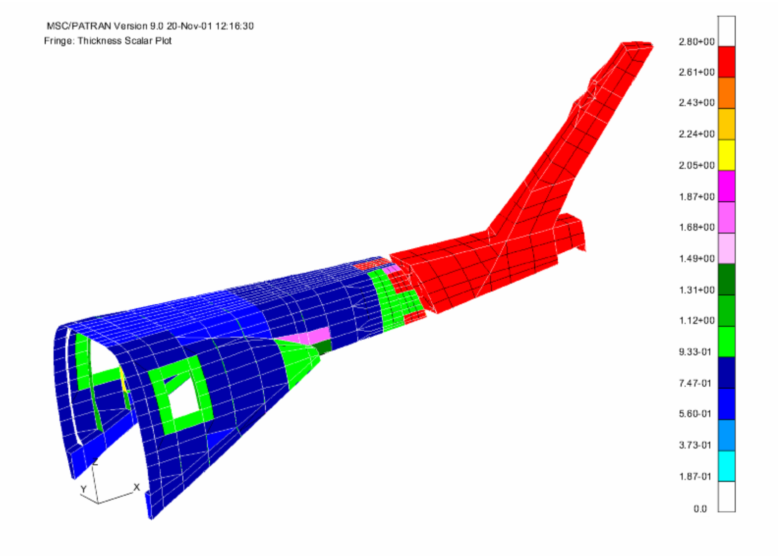

Figure 1. Elements thickness on FE global model of helicopter fuselage

This part of the fuselage is built following the normal aeronautical design; in fact it consists in panels, stringers and ribs.
The panels and the ribs' cores are modelled with two-dimensional shell elements with four nodes. The ribs are strengthened with beam elements to give out *of the plane* stiffness (like the schematisation seen for the stringers). The stringers are simply represented with beam elements. The folding beam schematisation is obtained with infinity rigid *beam*. Both skin panels of helicopter fuselage and the stringers are made in aluminium alloy.

3.1.2 Most stressed elements

The analysis of helicopter fuselage and tail global model aims to determine the most stressed zone to introduce the crack configuration (Figure 2).

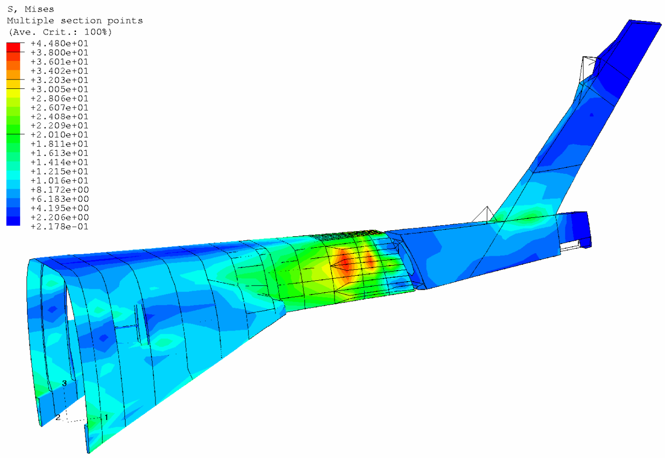

Figure 2. von Mises stress on helicopter fuselage with reference load

From analysis done with different loads is possible to find that, in the case of crack propagation, the more stressed load in terms of entity and quantity is the tail rotor traction, F_y. In the first time of submodeling (Figure 3), the stress for reference load of $F_y = 10000$ [N] is carried out.

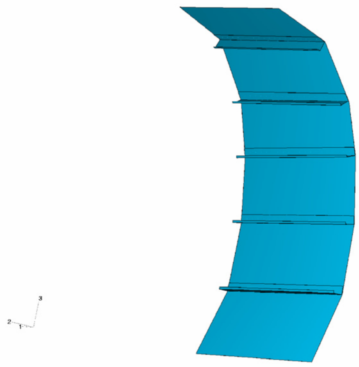

Figure 3. Submodel used for fatigue crack growth propagation

The applied loads are defined in three manoeuvre loads and many vibratory loads, at high and low frequency. The applied loads consists of two forces and one torque that represent tail rotor traction (indicated with F_y), tail lift force (F_z), and tail rotor reaction torque (M_y).
Considering that loads have constant applied direction and way and all skin fuselage is made in the same manner, it is possible to define univocally the most stressed zone about crack skin propagation with crack starting from a hole in the fastener's row between the skin and the lower part of a stringer.

3.1.3 Panel submodeling and crack configuration

The stress intensity factor K_I calculation, for any configuration, cannot be carried out with the whole original model because of too large differences in the scale: the model of the helicopter's tail has a nominal dimension of thousand of millimetres while the crack propagation analysis, at least at the beginning, has a nominal dimension of a few millimetres. It is thus practically impossible to use the same model to analyse the two problems together.

In this case the use of mesh submodeling option available in FE software permits to easily define a model with small geometrical dimensions and to make an analysis with it starting from the results obtained with a much larger model, applying to the submodel nodes (*driven nodes*) the displacement field obtained from global model.

In the most stressed helicopter fuselage zone a crack has been positioned, obtaining the most dangerous condition for crack propagation; in particular, a crack on the skin, starting from a hole in the fastener's row between the skin and the lower part of a stringer, with stringer's failure, has been considered.

To determine the curve stress intensity factor K_I vs. crack length a, for the configuration in analysis, many submodel with different crack length and for all applied loads (F_y, F_z, M_y) have been created. Practically, the K_I values for some crack length and therefore the $K_I = f(a)$ function with an interpolation of the values, for each tail rotor applied load have been calculated (Figure 4, Figure 5, Figure 6).

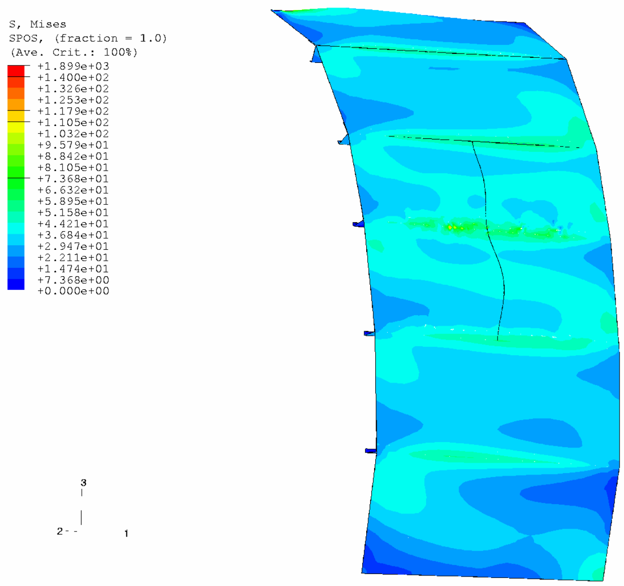

Figure 4. von Mises stress for crack with a = 3.25 [mm], load F_y = 10000 [N]

3.1.4 FE analysis results

The aim of the FE analysis is to define a relationship among any general load combination, the crack size and the stress intensity factor K_I. With all these analysis twelve laws of variation for the K_I factor are obtained, one for each configuration, load and direction of crack propagation (Up & Down). These laws give the value of K_I versus crack size.

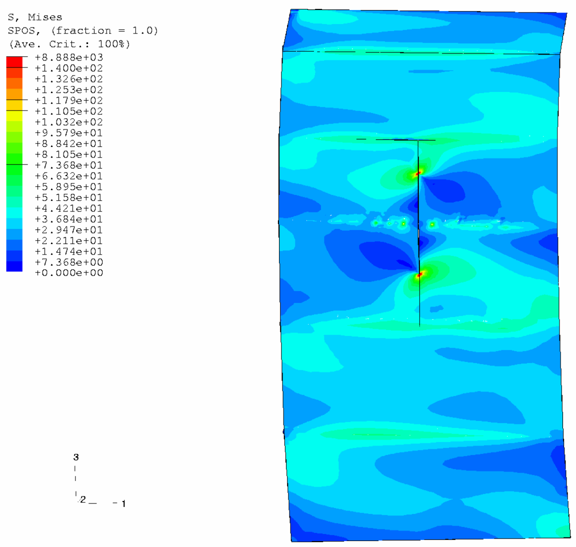

Figure 5. von Mises stress for crack with a=115 [mm], load F_y=10000 [N]

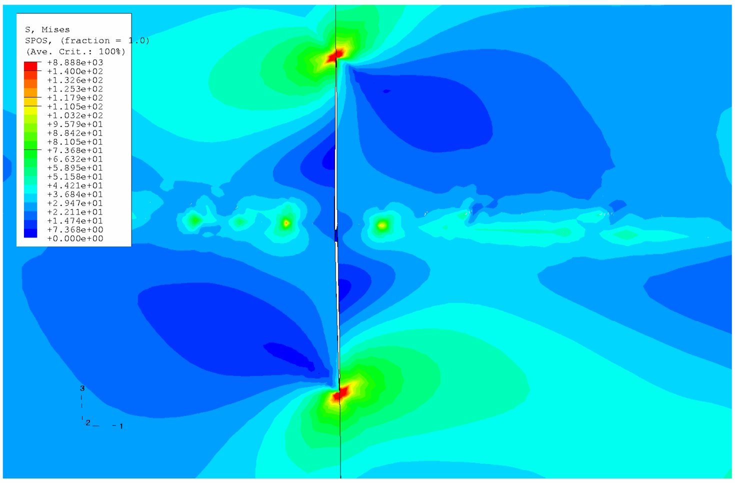

Figure 6. Detail of crack with a=115 [mm] with riveting local effect

To be really sure that in the adopted procedure is correct to apply the superposition of effect's principle for J-Integral calculation, several analyses with contemporary load combinations are carried out. Comparing the results of these analyses and directly from a linear combination of results the difference are so smaller that they confirm the goodness of method.
Starting from the *J-Integral* value (with correspondent K_I) calculated for the load combination and different crack length, by means of a least square interpolation the stress intensity factor K_I in function of crack length *a* for applied loads have been obtained (Figure 7)
.The obtained results are interpolated by polynomial or exponential function determined with least square method separately for each load; more importance is given to the tail rotor traction F_y, which turns out to be the most important load for the propagation.

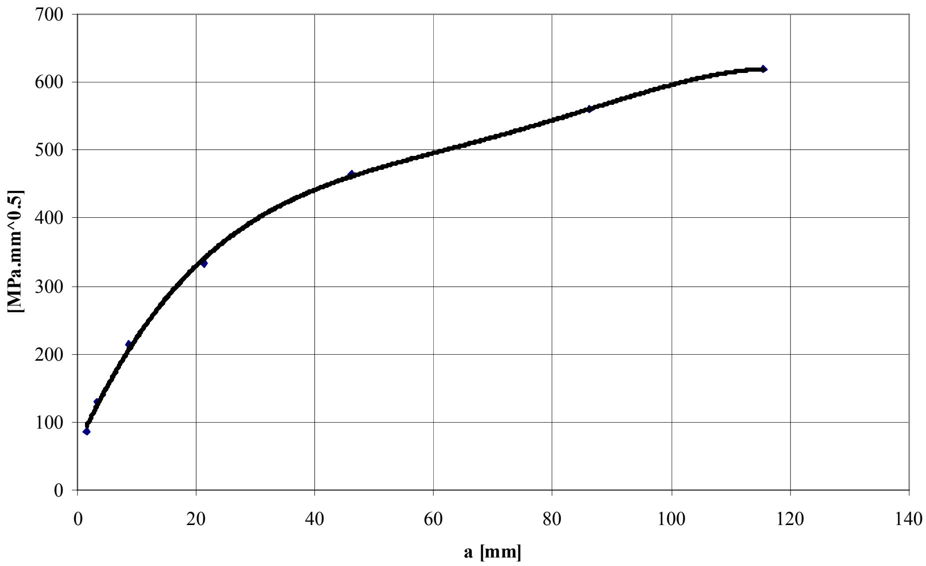

Figure 7. K_I vs. crack length a, load F_Y =10000 [N]

3.2 Crack propagation analysis

The activity described here has its main purpose in the requirement to apply *Damage Tolerance* design methodology to helicopters.
A *Damage Tolerance* design, in contrast with a *Safe Life* methodology (that in the past was the common design approach for helicopter) has essentially two main aspects:

- It is assumed that there is some degree of damage in the structure (cracks). Apart from the reasons that caused this presence (possible sources can be, for example, intrinsic material flaws or manufacturing errors), generally there is a defined initial cracks length (based on agreements between the constructor, the customer and the authority) for each part of the structure where there is a logical possibility that a defect may occur.
- The crack propagation time assessment for the vehicle's life is required. Basing on this assessment and a proper maintenance schedule, the safety of the structure is granted, in spite of cracks presence.

Generally there are three main parameters controlling the problem: the crack's initial size, the minimum crack's size that can be normally observed in practice and the critical crack's size tolerable by the structure. These three quantities are utilised to define two different time periods:

- The first period is from the minimum crack size to the crack of the minimum detectable length.
- The second period is from the point where the crack is detectable to the length when the remaining strength of the structure will drop below an acceptable limit (theoretical fracture point).

Once these periods are defined (it is possible to have different periods depending on which part of the structure is considered), the operator can arrange proper inspection times basing on his requirements (bearing in mind both safety conditions and economical reasons), starting from a reliable crack propagation times.
As already mentioned in other parts of this paper, the first assumption of the analysis is that crack propagation times are obtainable from analyses (experimental ones or analytical ones and with FE) which define some parameters necessary for the application of classical methodologies from the fracture mechanics (material properties, nominal stress in the critical zones, stress intensity factor K_I variation with crack size).

3.2.1 Load history

The loads used to calculate crack propagation are defined on the basis of the loading spectra for the helicopter under investigation. The load history is based on the definition of three different flight types (high, medium and low loading level) and on the definition of vibratory loads (Figure 8). The vibratory loading cycles represent those loading cycles not directly linkable to specific flight manoeuvres but generally present because of the interaction between the rotor's downwash and the tail structure. The different loading cycles, both the deterministic ones from manoeuvre spectra and the vibratory ones, are defined referring to the value of two forces (the tail rotor traction F_y and the tail plane negative lift F_z) and to a couple (the reaction torque from the tail rotor M_y) for each cycle. The application point and the direction of all these components are known and so, using the global FE model it is possible to pass from the known three values (F_y, F_z, M_y), which define each loading level, to the relative stress intensity factor K_I for each crack size.

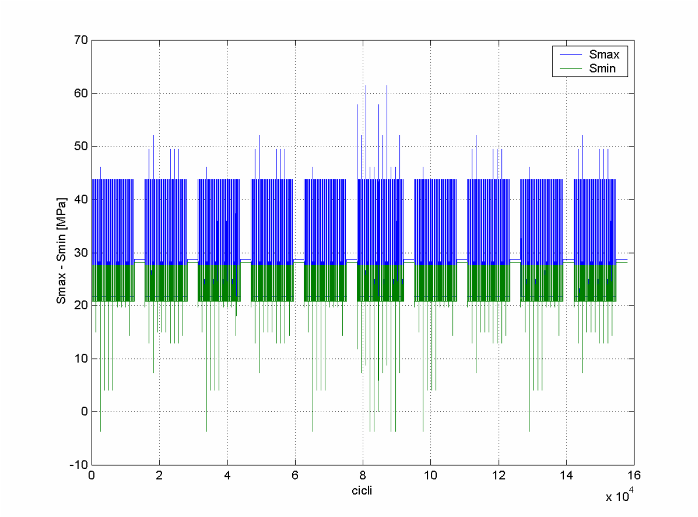

Figure 8. Typical load spectrum

3.2.2 Results of the crack propagation

To simulate the crack growth a calculation program rightly set, based on previously scheme to interpolate the specimen experimental tests, has been used; in particular, a crack propagation behaviour of the material used (aluminium alloy) has been obtained with specific standard tests (ASTM E647).
This behaviour is well defined with a crack propagation function (Nasgro law) which links the crack growth rate da/dN with the stress intensity factor range ΔK:

$$\frac{da}{dN} = C \cdot \left[\left(\frac{1-f}{1-R} \right) \cdot \Delta K \right]^n \cdot \frac{\left(1 - \frac{\Delta K_{th}}{\Delta K} \right)^p}{\left(1 - \frac{\Delta K_{\max}}{K_c} \right)^q} \qquad (1)$$

For this crack configuration the simulation begin from a initial crack size (0.5 inch) using the linear model. The simulation carried out permits to evaluate the flight hours associated to the following events, linked to a specific NDT operation or a minimum safety condition:

- Crack reaching a nominal length of 7.5 mm;
- Crack reaching a nominal length of 15 mm;
- Crack reaching a crack propagation rate of 10^{-5} m/cycle for maximum limit loads.

In Figure 9 the curves of crack growth, different for the two apexes of the crack, are reported.

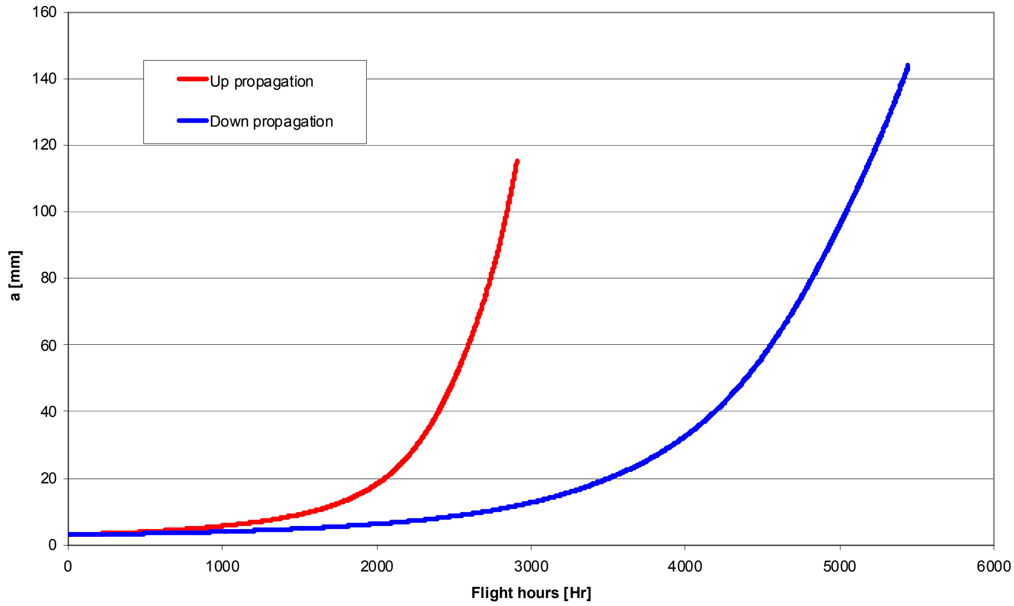

Figure 9. Crack propagation indicated in flight hours

4. Conclusions

In this paper a brief description of the EDA project HECTOR is carried out. This project consists of the definition of a reliable method to monitor the crack propagation in helicopter fuselages by the using of a sensor network, with a complete and detailed FE model needs to select the sensor position (most stressed zones) and to simulate the damage evolution.

A complete description of the numerical method, considering a FE model able to calculate the crack propagation parameters, is presented. This method permits to monitor the increasing of the crack and to define the residual life of the component.
Therefore it is clear that the great advantage of ESHM is that the degradation mechanism evaluation could be performed on line and in real time. Once the data from sensors are collected, elaborated and integrated with the results from FE models the prognosis (remaining life) can be executed in real time. The system directly identifies the location and follows-up actions that can be taken.
This is fundamental not only for a drastic reduction of inspection time and maintenance cost but also as a real time compliance/performance index of the machine respect to the contingent mission.

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