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REVIEW OF AERONAUTICAL FATIGUE INVESTIGATIONS IN FRANCE DURING THE PERIOD MAY 2007 - APRIL 2009

Compiled by

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INTRODUCTION AND ACKNOWLEDGMENT

The present review, prepared for the purpose of the 31th ICAF conference to be held in Rotterdam (The Nertherlands), on 25-26 May 2009, summarises works performed in France in the field of aeronautical fatigue, over the period May 2007-April 2009.
Topics are arranged from basic investigations up to full-scale fatigue tests.
References, when available, are mentioned at the end of each topic.
Correspondents who helped to collect the information needed for this review in their own organisations are :

- Delphine Alléhaux, Sophie Gourdet, Fabrice Congourdeau and Bertrand Journet for EADS Innovation Works
- Renaud Héron, Stéphanie Turchi and Alain Santgerma for Airbus France
- Jacques Argoud, Frédéric Desbordes, Benoit Morlet and Lionel Le Tellier for Dassault Aviation
- Nicolas Baréa and Gilles Garrigues for Latécoère
- Mathieu Fressinet, Pierre Madelpech and Pascal Hamel for CEAT.
- Gilbert Hénaff for LMPM (Poitiers)
- Jean-Louis Chaboche for ONERA

They will be the right point of contact for any further information on the presented topics.
Many thanks to all of them for their contribution

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6.1. MATERIAL FATIGUE BEHAVIOUR

6.1.1. Fatigue crack propagation behaviour of new generation aluminium lithium alloys (LMPM)

Due to their high specific mechanical properties, aluminium alloys are widely used in aircraft structures. The dramatic increase in fuel prices and the environmental considerations lead the aircraft designers to get interested in new lighter materials, in particular in the third generation of aluminium lithium alloys. An optimization of the damage tolerance properties of these alloys is currently studied in the LMPM. The current works [1,2] deal with 6 mm thick plates of 2198, 2196 and 2050 alloys, provided by Alcan. It is particularly shown that the 2050 alloy, containing T1 precipitates, presents a retarded crystallographic propagation for ΔK ranging between 7 and 11 MPa.m1/2 in ambient air (Figure 1), leading to the best resistance against fatigue crack propagation. Influence of environment and crack closure is also examined in order to improve the analysis of the crack propagation mechanisms in these alloys.

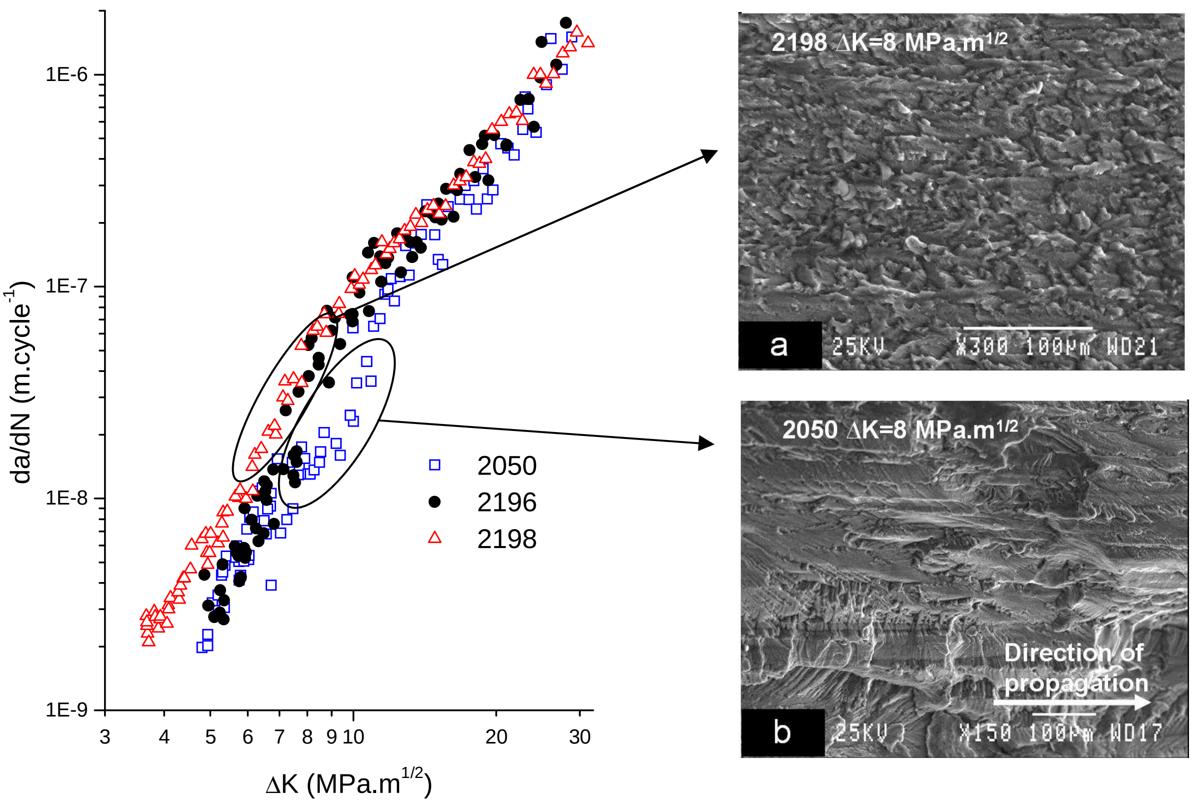


Figure 1 : da/dN vs. ΔK curve in ambient air at R=0.1 of three third generation aluminium-lithium alloys, and associated SEM micrographs at ΔK=8 MPa.m$^{1/2}$ for a) 2198 alloy and b) 2050 alloy [1].

References :

1. Richard S, Sarrazin-Baudoux C, Petit J. Fatigue crack propagation in new generation aluminium-lithium alloys. TMS 2009, 138th Annual Meeting and Exhibition. San Francisco: The Minerals, Metals & Materials Society; 2009. p. 69-76.
2. Richard S, Sarrazin-Baudoux C, Petit J. Near threshold fatigue crack growth behavior of a third generation aluminium-lithium alloy. 12th International Conference on Fracture. Ottawa. To be published.

6.1.2. Fatigue crack propagation of friction stir welded joints in 2139 Al-Cu alloy [1] (EADS IW)

EADS Innovation Works dealt with the fatigue crack growth behaviour of a new generation aluminium-copper alloy, developed by ALCAN and graded 2139, welded by the Friction Stir Welding (FSW) process. Considering the best performances of this age formable alloy in its T8 temper, the weld assessment has been conducted on the T8 as welded condition. This new generation alloy is especially investigated as a higher performance fuselage candidate, with respect to the conventional and flying 6056 alloy, presenting higher crack growth resistance and good weldability in laser beam and friction stir welding. For the fuselage application, two joining configurations using FSW have been developed and mechanically tested: butt joining using the bobbin tool technique (Figure 1) for the longitudinal junction of fuselage panels and lap stringer-skin joining using the retractable pin tool technique (Figure 2), both in replacement of riveting. The first one was a part of the national funding project, AFFISA, the second one was developed within the European project, WEL-AIR.

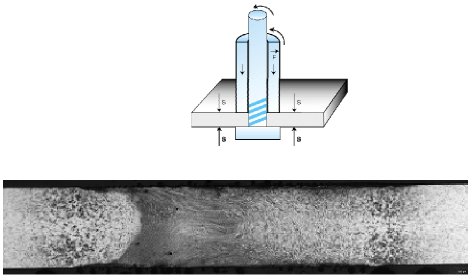


Figure 1: 2139 2,5 mm thick butt joint

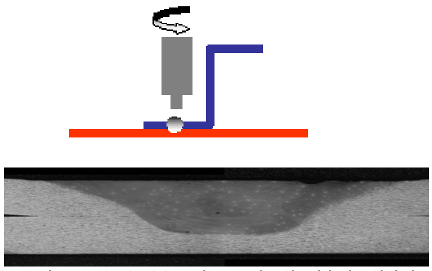


Figure 2: 2139 stringer (1,6)-skin(1,6) joint

For both configurations, metallurgical investigation and microhardness measurements lead to identify the lowest hardness point of the welded area as the weld nugget. This critical location has then been selected to localize the mechanical notch of the fatigue crack growth specimens, M(T) 200 mm wide specimens. Due to the main loading direction representative of the targeted applications, cracking along the weld direction has been investigated. To avoid any bending effect in the lap joint configuration, the stringers were fully machined. For both configurations, fatigue crack growth behaviour was assessment under a constant amplitude loading of 0,1 and the results were analyzed in term of crack growth rates (da/dN) as a function of ΔK value including a comparison with the parent metal performance in T-L direction (Figure 3).

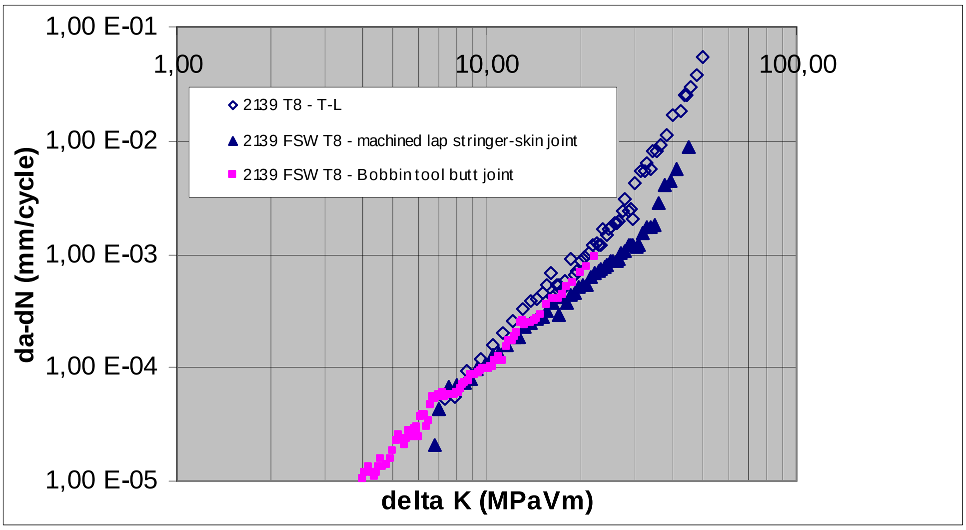


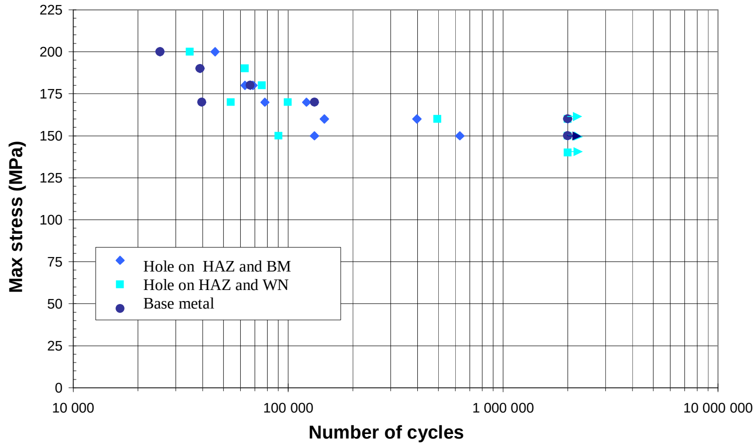
Figure 3 : fatigue crack growth curves (parent metal, butt joint, stringer-skin joint)

References :

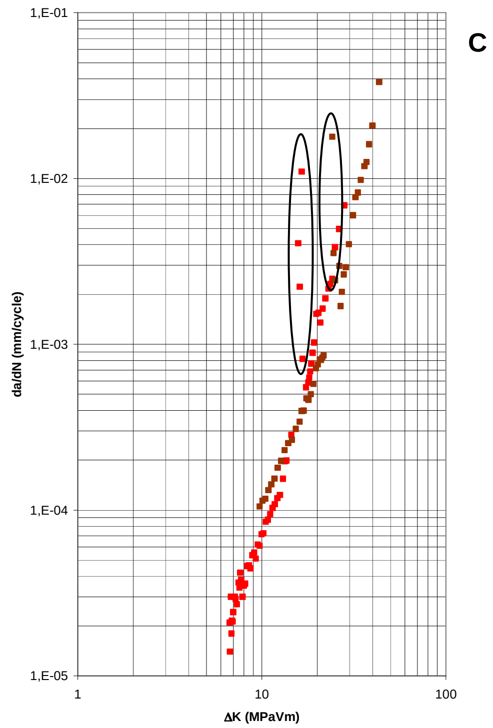
1. D. Alléhaux : "Mechanical and corrosion behaviour of the 2139 aluminium-copper alloy welded by the Friction Stir Welding using the bobbin tool technique". ICAA 10, July 9-13 2006,

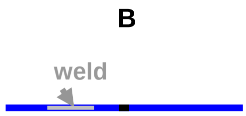
6.1.3. Fatigue endurance and fatigue crack propagation behaviour in friction stir welded coupons [1] (EADS IW)

The mechanical properties of thick (>6 mm) FSW coupons of high strength 2xxx and 7xxx series aluminium are under investigation. Some of the works performed at EADS IW are illustrated below. Holed fatigue specimens were tested with 3 different positions of the hole (in the base metal, in the heat affected zone and in the welded nugget). No difference is noticeable between the 3 S-N curves (**A**), which indicates a good fatigue behaviour of the weld. The fatigue crack behaviour was investigated with dissymmetrical CCT specimens (**B**): one crack propagates only in the base metal whereas the other one will cross the weld. A very high propagation rate has been observed when the crack is in the welded nugget (**C**). This effect is strongly reduced when improved welding parameters are applied (**D**). The post welding residual stresses are under investigation, to evaluate their effect on the increase of the propagation rate, which is also favoured by the small grain size of the nugget.

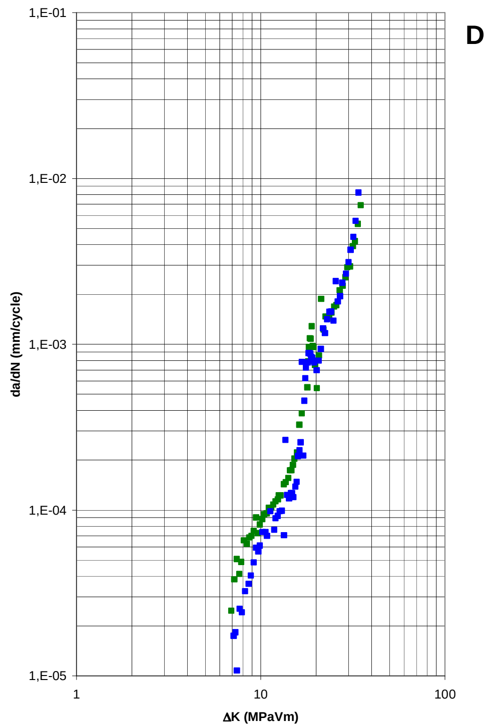


A - 7055 T7 FSW on 8.5 mm thick coupons. S-N curves R=0.1 for holed specimens (Kt=2.3) parallel to the welding direction with various locations of the hole with regard to the weld position. Note that the fatigue limit is similar for all tested configurations





7055 T7 FSW on 8.5 mm thick coupons, first welding campaign. Note the sudden and momentary increase of the propagation rate (circled points), when the crack crosses the FSW nugget.



7055 T7 FSW on 6 mm thick coupons, second welding campaign with improved parameters. Note that the crack propagation rate in the nugget is strongly reduced compared to the graph at the left.

References :

1. S. Gourdet, French national DTP C-Best (2003-2006), presented at ALUFORM 2006 CETIM Senlis, France

6.1.4. Creep-fatigue crack growth rates in an aluminium alloy for supersonic aircraft fuselage panels (LMPM)

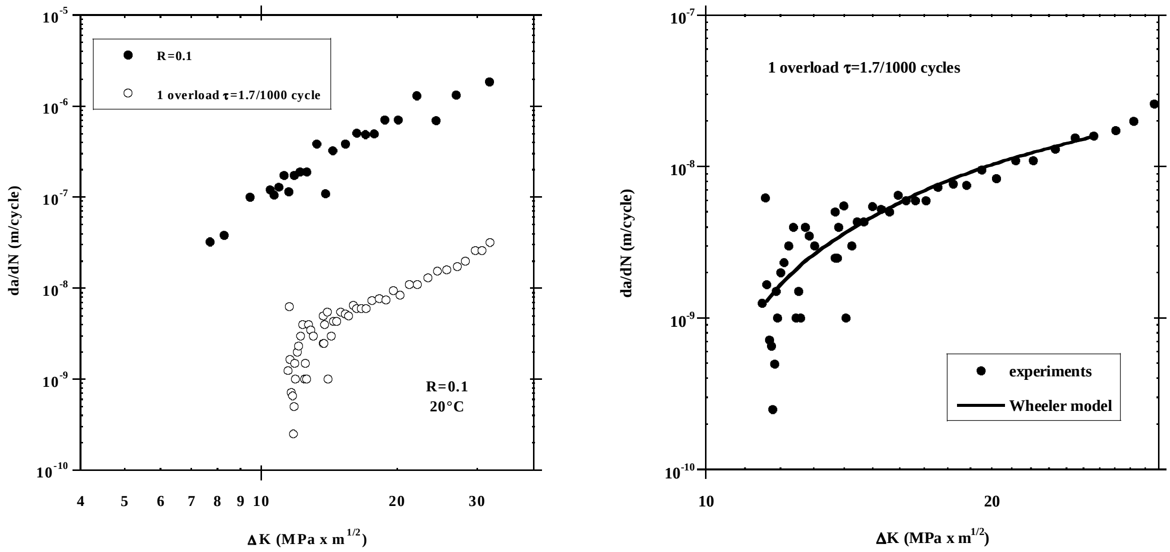
The creep-fatigue crack growth resistance of a precipitation hardened 2650 T6 aluminium alloy selected for fuselage panels of a future civil supersonic aircraft has been studied. The objective was to develop a methodology to predict crack growth under very low frequency loading at elevated temperatures. With this aim, creep crack growth rates (CCGRs), fatigue crack growth rates (FCGRs), creep-fatigue crack growth rates (CFCGRs) have been measured at 130°C and 175°C in laboratory air and in vacuum at R=0.5 under different load frequencies and wave shape signals. It is shown that, for a given temperature, CFCGRs are unaffected by frequency below a critical value of the load period Tc. Above this value CFCGRs are directly proportional to the load period. This time-dependent crack growth regime is assisted by a significant creep damage as indicated by the large amount of intergranular decohesions induced by cavitation on fracture surfaces. CFCGRs are calculated under the assumption that fatigue damage and creep damage can be linearly summed. In vacuum the predictions are in good agreement with experimental data at both temperatures. In air however a discrepancy is observed for low frequency loading, suggesting the occurrence of a creep-fatigue-environment interaction. As a consequence the time-dependent crack growth behaviour affected by this interaction is different from creep crack growth behaviour, although the reasons for this are still unclear. A methodology is then proposed to predict CFCGRs in air. This methodology, if assessed by very low frequency experimental results, could be extended to different structural components made of aluminium alloys operating at elevated temperatures, provided that the mechanisms are unchanged.

References :

1. G. Odemer, G. Henaff, and B. Journet, Creep crack growth resistance of an age hardened aluminium alloy for supersonic applications, Scripta Materialia, 54 (2006) 51-55.
2. G. Henaff, G. Odemer, and A. Tonneau-Morel, Environmentally-assisted fatigue crack growth mechanisms in advanced materials for aerospace applications, International Journal of Fatigue, 29 (2007) 1927-1940.
3. G. Odemer, G. Benoit, E. Koffi, G. Hénaff, and B. Journet, Prediction of creep-fatigue crack growth rates in inert and active environments in an aluminium alloy, International Journal of Fatigue, in press (2009).

6.1.5. Damage tolerance of a FeAl alloy prepared by mechanical alloying and forging for high temperature structural application (LMPM)

The improvement of aerostructure performances requires the development of innovative materials and processes. In particular in many applications stiff and light materials based on intermetallic compounds are considered to replace conventional alloys such as steels or titanium alloys. Thus the Feal 40 grade 3 alloy developed by the CEA/CEREM in France could meet the requirements for the replacement of these materials in the fabrication of ribs in engine pylons. However the damage tolerance of these components has to be assessed and with this respect fatigue crack growth data are required. Fundamental mechanisms governing Fatigue Crack Growth (FCG) in the FeAl grade 3 alloy were studied in a previous study, mostly in the as-hipped condition. Upscaling of the process in order to produce actual components was one the main issue tackled in the framework of the ALFA program sponsored by the National Agency for Research (ANR). This upscaling implies mechanical alloying of larger batches of powders and forging of hipped compacts in order to produce large plates. In the present study, it was demonstrated that upscaling in the production of the mechanically alloyed FeAl 40 grade 3 alloy has no significant impact on the fatigue crack growth resistance. In the domain of low ΔK values of, it is also comparable with that of as-hipped material. It is furthermore noteworthy that in the forged products (R0 and MK25), probably thanks to a higher toughness, a high ΔK propagation domain can be explored. This domain is furthermore characterized by a gross reduction in the Paris law exponent. In addition, in order to prevent over-conservative predictions, load history effects have to be taken into account and the point is that the issue of possible retardation effects in intermetallics has not been addressed so far. However, due to the brittle behaviour of these materials, only limited retardation effects could be expected. Now a strongly retarded propagation is observed under the application of periodic overloads. It is probably one of the very first evidences of this effect, which is well-known in conventional metallic alloys, in the case of an intermetallic alloy. Complementary studies are still necessary to improve the understanding and the description of this highly beneficial effect from the viewpoint of the damage tolerance of the FeAl 40 grade 3 alloy.



(a) (b)

Figure 1: comparison of FCGRs under periodic overloads and constant amplitude loadings (b) comparison with Wheeler's model.

6.1.6. Fatigue damage modes of duplex titanium alloys: crystallographic aspects at different scales (LMPM)

The elaboration techniques of polycrystalline titanium alloys, especially those used in the aeronautic industry for critical parts, lead frequently to very complex and fine grained microstructures. In presence of a homogeneous material in terms of size and shape of the grains, a strong scattering of the HCF fatigue lifetime can still be observed within some forged titanium alloys. The crystallographic orientation distribution and particularly some meso-textures revealed in these alloys may explain this fatigue strength scatter.

A large experimental study has been conducted to investigate the influence of the crystalline orientation at different scales on the micromechanical fatigue behaviour of an industrial duplex forged Ti-6Al-4V titanium alloy. The local microstructural conditions favouring fatigue damage have been determined at different scales: microscopic scale of crack initiation and mesoscopic scale of crystallographic textures. A micromechanical analysis based on the influence of the resolved shear stress (Schmid's factor) and normal stress (basal plane inclination, elastic stiffness) has been proposed. The relevant factors for crack initiation in basal, prismatic and pyramidal planes have been determined.

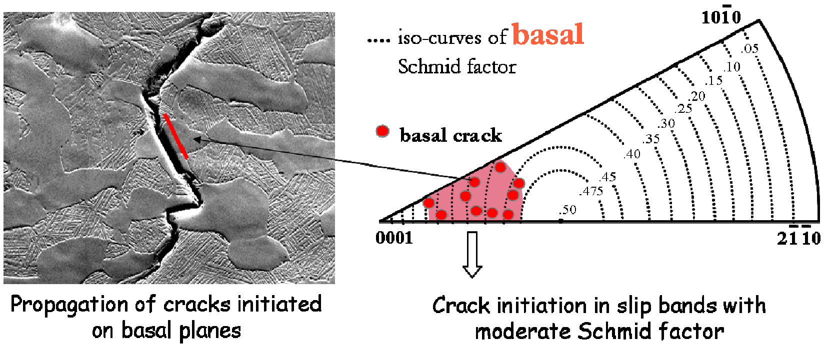


Figure 1 : Crack initiation in basal planes as a combination of plasticity and quasi-cleavage processes. Crystallographic domain (Inverse pole figure) concerned by this process.

Deformation micro-mechanisms of a Ti-6Al-4V alloy under fatigue loading at room temperature have also been studied using a three-dimensional crystal plasticity constitutive model. The model employs a minimum set of fitting parameters based on the experimental data obtained for Ti-6Al-4V. Cyclic deformation behaviour at the macroscopic scale and at the local scale of grains is analysed through the simulation of 20 cycles of fatigue on a polycrystalline structure of 900 randomly oriented grains. The progressive activation of slip (basal, prismatic, and pyramidal) is analysed and compared to experimental observations. Some elongated textures are then modelled to evaluate the influence of the spatial distribution of crystallographic textures on the cyclic plastic deformation.

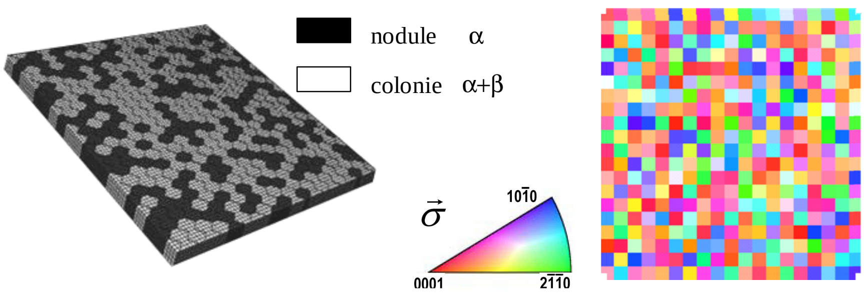


Figure 2 : Finite element mesh used for the crystal plasticity model taking into account the local nature of the phase (nodules αp / lamellae colonies ($\alpha+\beta$)) and their crystalline orientation.

References

1. F. Bridier, P. Villechaise and J. Mendez, “Analysis of the different slip systems activated by tension in a alpha/beta titanium alloy in relation with local crystallographic orientation,” *Acta Materialia*, vol. 5, no. 3, 2005, pp. 555-567.

2. F. Bridier, P. Villechaise and J. Mendez, “Slip and fatigue crack formation in a alpha/beta titanium alloy in relation with crystallographic texture at different scales,” *Acta Materialia*, vol. 56, no. 15, 2008, pp. 3951-3962.

3. F. Bridier, D.L. Dowell Mc, P. Villechaise and J. Mendez, “Crystal Plasticity modeling of slip activity in Ti6Al-4V under heigh cycle fatigue loading,” *International Journal Of Plasticity*, vol. doi 10.1016/j.iplas., no. in press, 2008.

6.1.7. Interaction between creep and low cycle fatigue at high temperature for nickel base superalloy Udimet 720 (LMPM)

Nickel-base superalloys employed for components in the hot parts of aircraft engines are subjected to both steady and fluctuating stresses due to high temperature, centrifugal force and high frequency vibrations.

The present study concerns the polycrystalline cast and wrought Udimet 720 elaborated and forged by Aubert & Duval corp. and provided by Turbomeca. It focuses on the interaction between creep and fatigue at 700°C. For that, tests have been performed with a trapezoidal signal and a dwell time ranges from 1 to 50 seconds. Results are compared to "pure" creep and fatigue tests.

The damage process relative to the different loadings is described from SEM observation. A transition between an intergranular crack initiation and a transgranular propagation is observed depending on the hold time. Deformation mechanisms are studied too. The correlation of dislocations structures (TEM investigations) with the macroscopic behaviour is made from fatigue to creep conditions. A special attention is paid on the role of the hold period to investigate the transition of mechanical behaviour and durability from the "pure" fatigue to the creep testing conditions. These processes are studied from results obtained on two different metallurgical states: a homogeneous coarse grains structure and a bimodal one composed of alternating bands of coarse and fine grains. In addition the finest γ' precipitation that is quite different in terms of size and density has to be taking into account. The comparison of results obtained on both material help us to analyse creep –fatigue interactions.

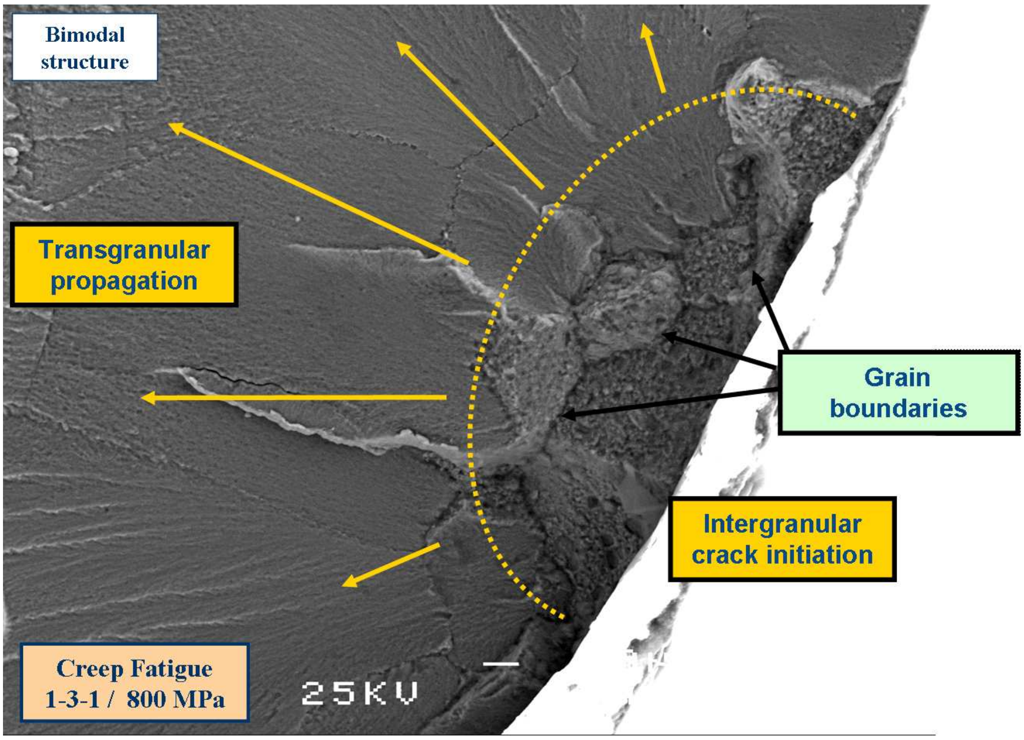


Figure 1 : Intergranular crack initiation and inter-transgranular transition of crack propagation for creep – fatigue conditions.

References

1. T. Billot, P. Villechaise, M. Jouiad and J. Mendez, “Interaction between creep and low cycle fatigue at high temperature for a nickel-base superalloy Udimet 720,” *Proc. Sixt International conference on Low Cycle Fatigue*, 2008.

2. T. Billot, P. Villechaise, M. Jouiad and J. Mendez, “High temperature interaction between creep and low cycle fatigue on a nickel-base superalloy Udimet 720" *Proc. TMS Annual Meeting*, 2009.

6.1.8. Corrosion Fatigue Crack Propagation Mechanisms in 2024-T351 aluminium alloy (LMPM)

This study tackles the issue that was widely addressed in the past, especially in the U. S. But that still needs clarification, of the corrosion fatigue crack propagation in aluminium alloys used in aircraft structures. The industrial concern is related to aging air fleets, but also to the modification of the regulation on the use of corrosion inhibitors. From a fundamental point of view, the main objective of this study was to identify the domains where synergistic effects between fatigue and corrosion might exist, and to analyse the influence of various factors on these synergistic effects. In this aim a 2024 T351 aluminium-copper alloy was selected. CT specimens were machined at the centre of a thick plate where slower cooling rates during quenching induce a high sensitivity to intergranular corrosion.

The experimental results indicate a “negative” frequency effect in saline solution under sinusoidal signal. Indeed the highest growth rates are observed at high frequency (> 2.5 Hz) while the low frequencies growth rates are comparable to those obtained in air or distilled water. It is however noteworthy that these latter environments in turn significantly enhance growth rates as compared to high vacuum by inhibiting the crystallographic cracking mode. It has been demonstrated that the fatigue crack growth rate enhancement observed at high frequency in saline solution under sine wave is actually controlled by the load rising time, and not by the total duration of the exposure to the corrosive medium. Additional test results have shown that this load rising time effect is observed only under conditions of high stress corrosion sensitivity, that means under in the S-L orientation (not in L-T), in the T351 temper (not in the T851). However the enhanced growth rate regime does not correspond with a fully intergranular crack path as observed in stress corrosion cracking. This is only observed during crack growth under cyclic loading and immersion/emersion exposure cycles.

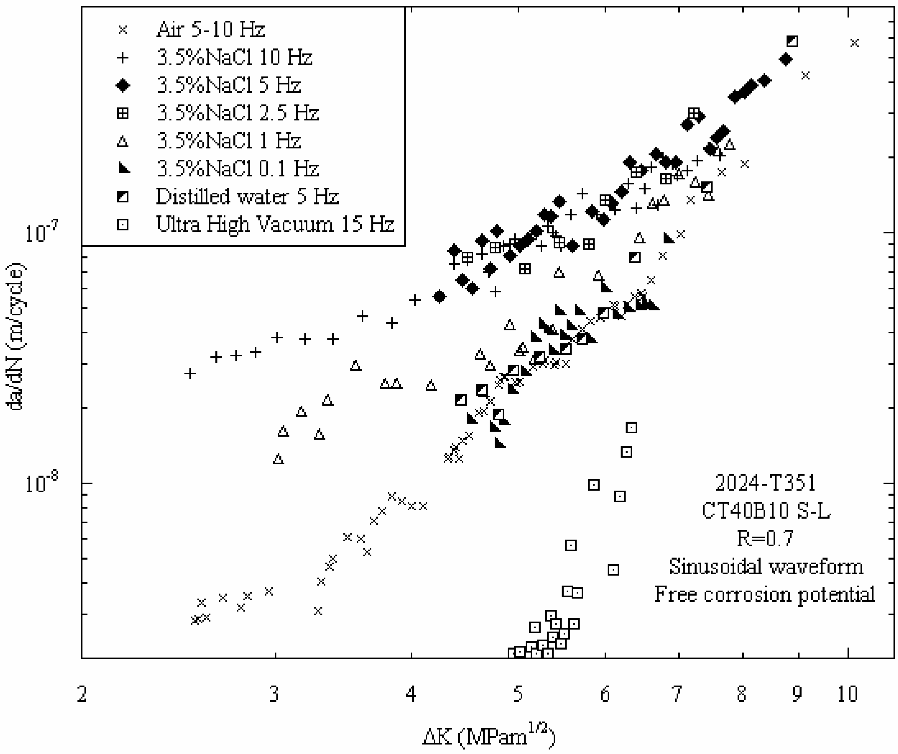


Figure 1. Fatigue crack propagation rates in saline solution at different frequencies under sinusoidal waveform as compared to air and distilled water.

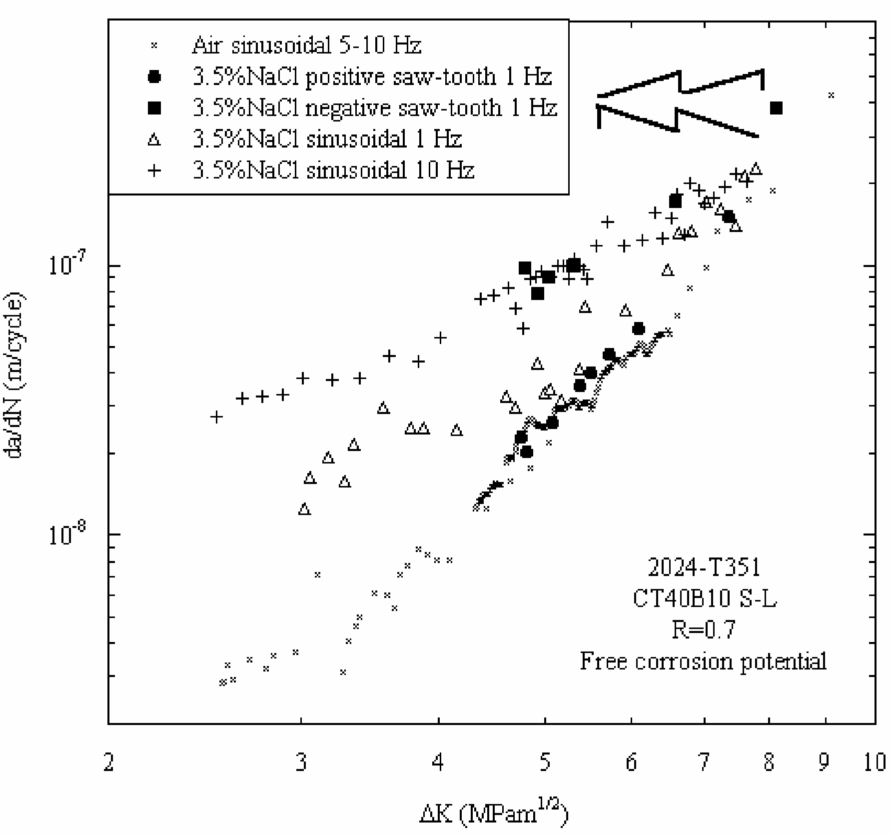


Figure 2 : Influence of loading waveform on fatigue crack propagation rates in saline solution.

References

1. F. Menan, “Influence de la corrosion saline sur la tolérance aux dommages d’un alliage d’aluminium aéronautique 2XXX”, thèse de doctorat de l’université de Poitiers, 2008.
2. F. Menan et G. Hénaff, “Influence of Frequency and Exposure to a Saline Solution on the Corrosion Fatigue Crack Growth Behavior of the Aluminum Alloy 2024”, accepté pour publication dans *International Journal of Fatigue*, 2009.
3. F. Menan et G. Hénaff, “Influence of Frequency and Waveform on Corrosion Fatigue Crack Propagation in the 2024-T351 Aluminium Alloy in the S-L orientation”, accepté pour publication dans *Materials Science & Engineering A*, 2009.

6.1.9. Effect of a severe pitting corrosion on static and fatigue performance of 2024 aluminium (CEAT)

Recent corrosion problems on the wing skin of military aircrafts induced a research program on the effect of the pitting corrosion on the static and fatigue performances of 2024 T351 aluminium. This program, preliminary to a more extensive common program (DGA and industry) expected soon, have been completed by CEAT in 2008. Main achievements were :

- Acquisition of a know-how on an accelerated corrosion process, able to produce typical corroded specimens.
- Assessment of the effect of a severe pitting corrosion (depth till 40% of the thickness) in a plain, an holed and an assembly specimens on the static strength, fatigue and propagation performances.
- Assessment of the effect of a repairing of the corroded area.
- Development of a basic understanding for the modelling of this type of corrosion.
- Development of a basic understanding of the process of corrosion fatigue crack initiation (transition area).

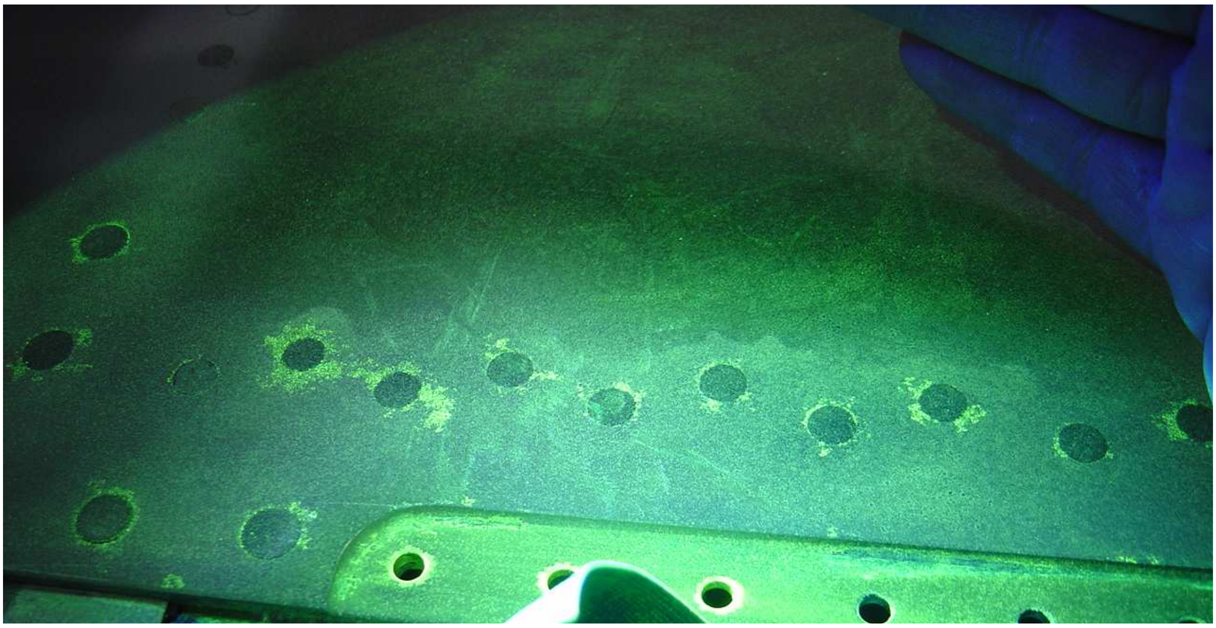


Figure 1 :Pitting corrosion around rivets of a wing

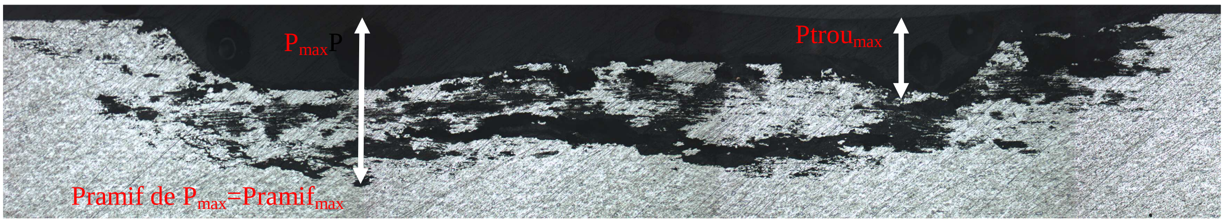


Figure 2 : Typical 0.8 mm depth pit of corrosion obtained by the corrosion process

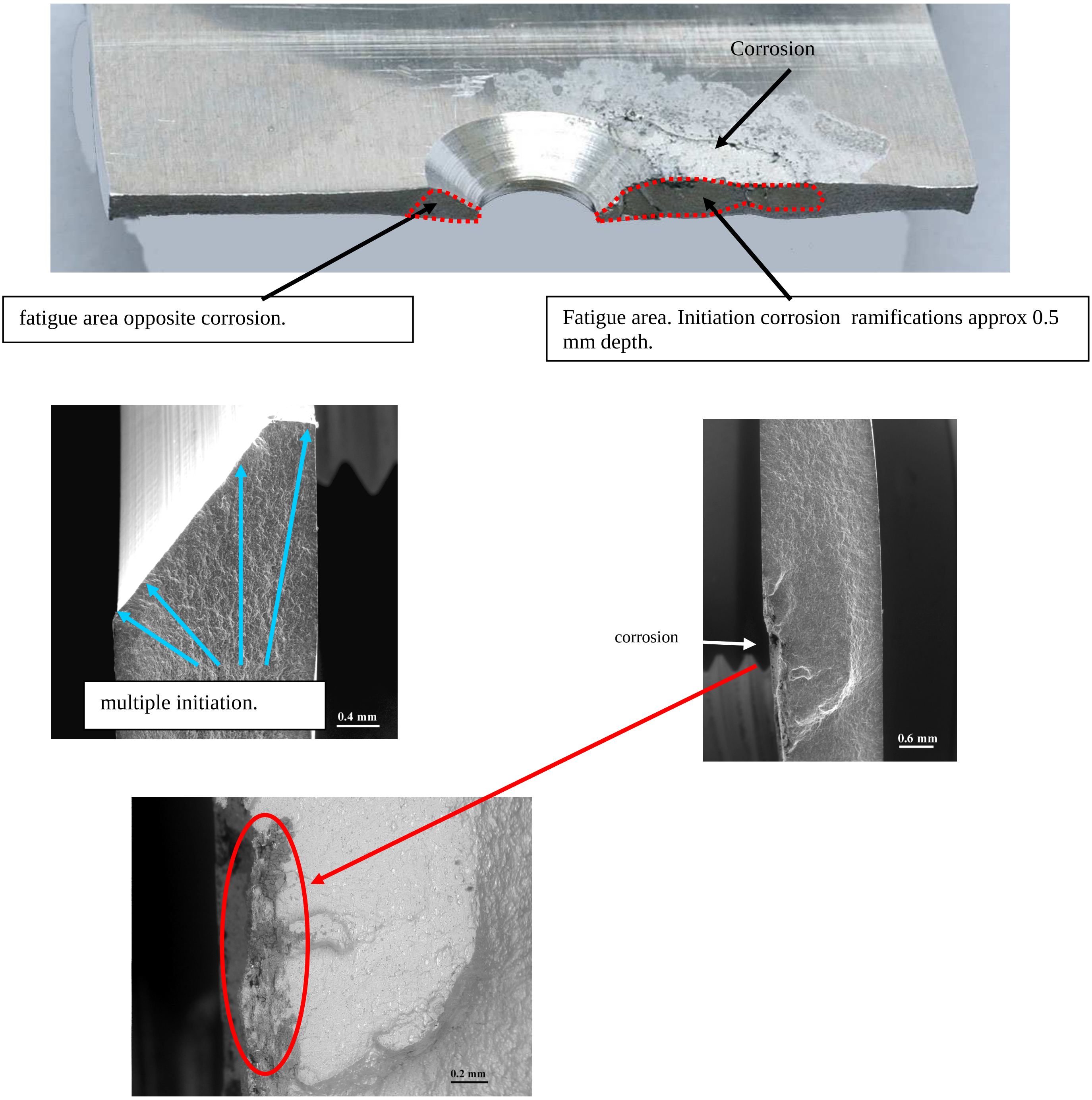


Figure 3 : Analyse of the fatigue initiation areas on holed specimen with countersink

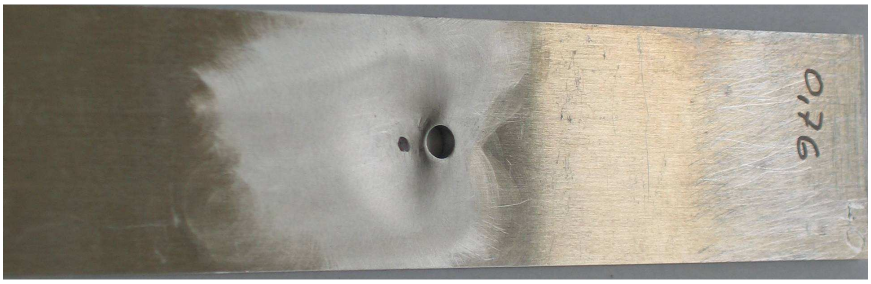


Figure 4 :specimen with a repairing of the corroded area

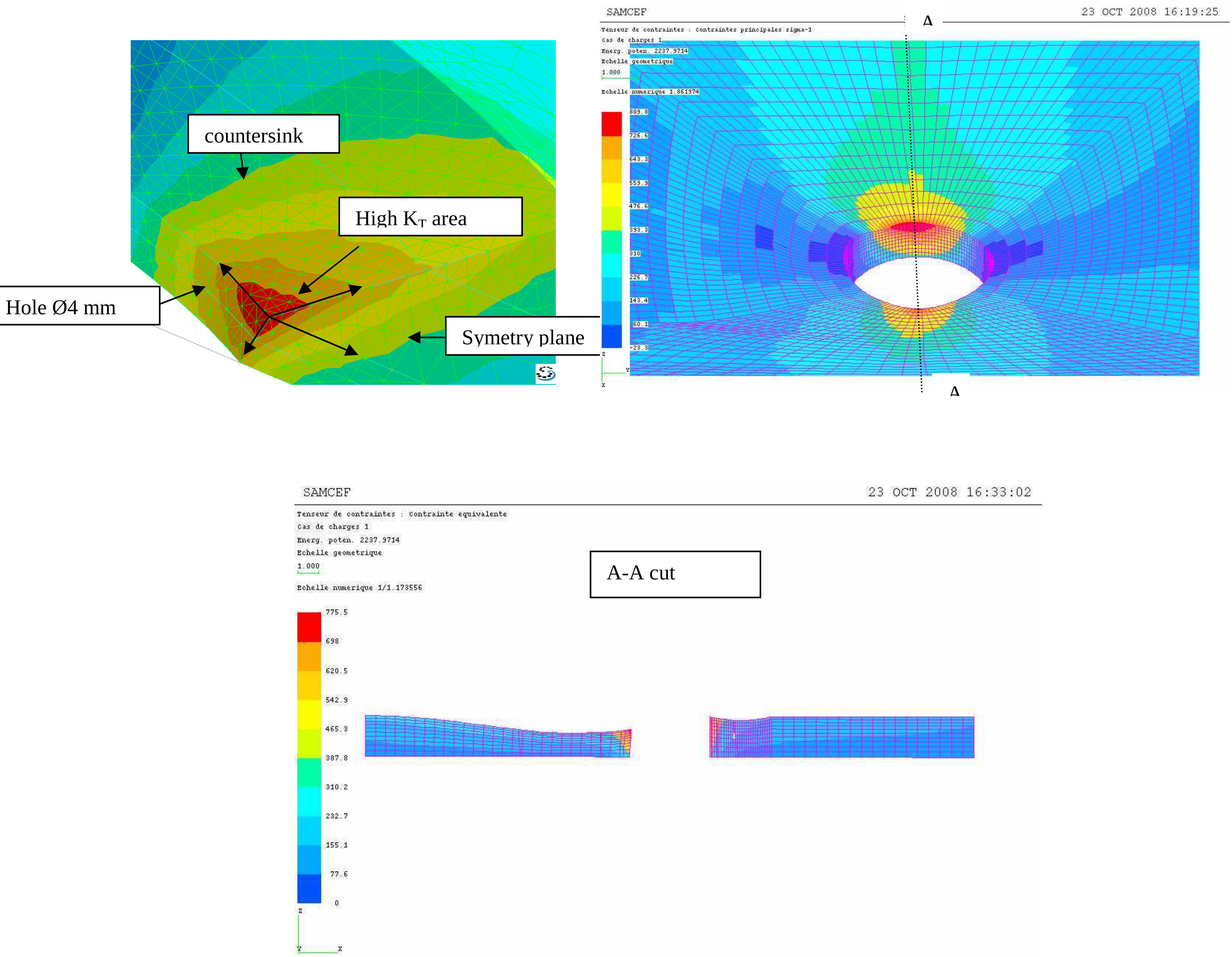


Figure 5 : Modelling of the specimens for fatigue life calculation

6.1.10. Evaluation of fatigue life models on aeronautical aluminium and steels (CEAT)

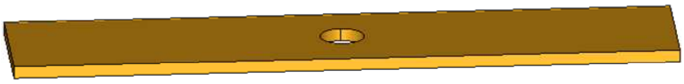
The objective of this study is to develop, in CEAT, a know-how on basic and multiaxial high cycle fatigue criterions for notched aeronautical structures. Fatigue calculations with basic methods, based on the comparison of the local stress ($Kt.\sigma^{\infty}$) to a S-N curve on smooth specimens, and Crossland criterion have been compared to the experimental results of 2024, 2214, 7075 and 35NCD16 notched specimens (R=0.1) for various fatigue lives. Due to the conservatism of the predictions, critical distances methods have been tested with those criterions.

The main lessons learned are:

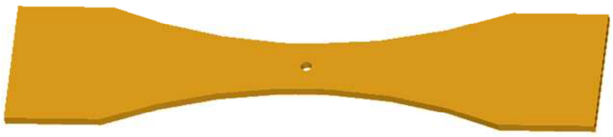
- The inability of most of fatigue criterions to take into consideration the notched phenomenon.
- The restriction of Crossland's criterion $\sqrt{J_{2,a}} + \alpha \cdot \sigma_{H,\max} \leq \beta$ to a Von Mises criterion $\sqrt{J_{2,a}} \leq \beta$ for aluminium 2024 and 7010: the identification of Crossland coefficient α with alternate torsion and traction/compression fatigue data lead us to α=0.
- Relatively good results on aluminium 2024 for the Crossland criterion with the critical distance d_0 = 0.08 mm and for the various fatigue lives tested

- S-N curves for 4 specimen types
-

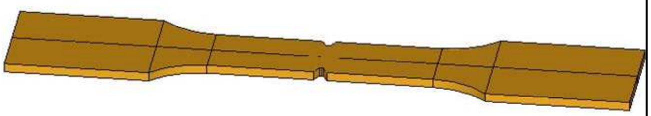
T type: W=30 Ø=10



T_NR type: W=30 Ø=4



FPE20 type: W=20 r=1,36



FRE type:

- FRE50A radius = 56 mm Kt = 1.035
- FRE50D radius = 2 mm Kt = 1.7
- FRE50F radius = 0,84 mm Kt = 2.4
- FRE50I radius = 0,36 mm Kt = 3.3

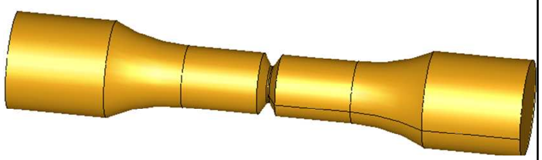


Figure 1 : Fatigue data used for 2024 T351 aluminium

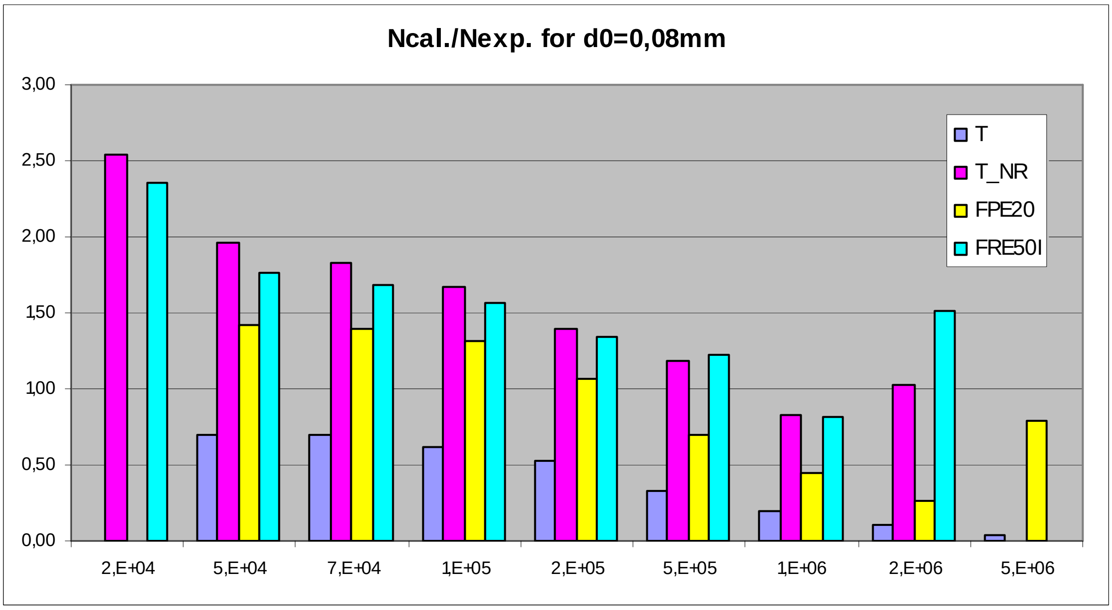
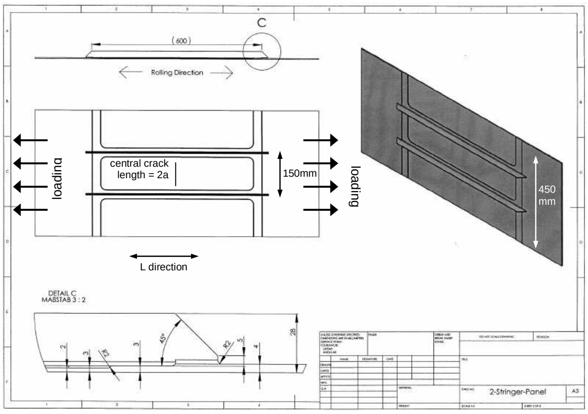


Figure 2 : Results of the fatigue prevision with Crossland criterion and critical distance method

6.2. FATIGUE LIFE PREDICTION STUDIES AND FRACTURE MECHANICS

6.2.1. Spectrum loading on welded panels : PREFFAS model (in CRACK-KIT® software platform) (EADS IW)

The reported work has been carried out in the frame of FP6 DaToN project [1]. The PREFFAS model which has been developed by EADS IW [2] has been evaluated in the case of a welded stiffened panel containing a central crack. The panel has two welded stringers and, due to welding, the bay area between the stringers is loaded with compressive residual stresses. The panel geometry is shown in Figure 1 together with the residual stress due to laser beam welding in the loading direction as determined by Pisa University in the project [3] using the cut compliance method. The width of the bay area is 150mm. The skin and stringers of the panel are made out of 6056T6 alloy. The panel is tested in the as welded condition. The planned fatigue crack growth tests deal with R=0.1 constant amplitude loading and Airbus A330 fuselage spectrum loading.



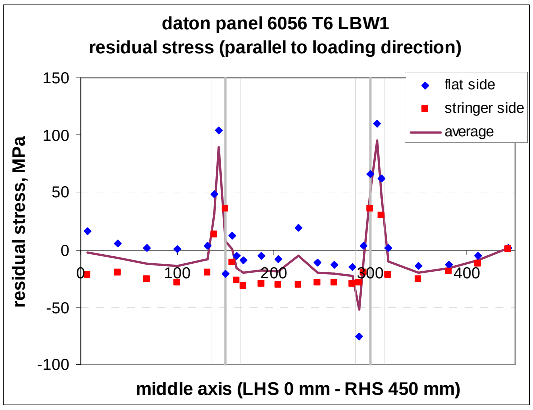


Figure 1 – Panel geometry and residual stresses. The panel has two welded stringers. The residual stresses in the loading direction due to LBW were determined by Pisa University.

The compressive residual stress in the bay area will slow down the fatigue crack growth. It will improve the damage tolerance capability of the panel. The objective of the work was to assess this benefit under constant amplitude loading and under spectrum loading. Both LBW (laser beam welding) and FSW (friction stir welding) technologies aer addressed. The research work dealt with K calculation with residual stress and the implementation of a scheme in PREFFAS to analyse the spectrum effects with the residual stresses.

The residual stress profile was imported into ABAQUS code in order to calculate the stress intensity factor K. The effect of residual stress is to change the R ratio of the fatigue cycles. Prediction of fatigue crack growth (FCG) under R=0.1 constant amplitude loading (CAL) was made using the PREFFAS parameters and formulae to predict FCG under any R ratio. Figure 2 shows a comparison between the predictions and the tests results concerning the LBW panel. This validates the approach. Predictions without any residual stress were made. There is a clear benefit from the residual stress.

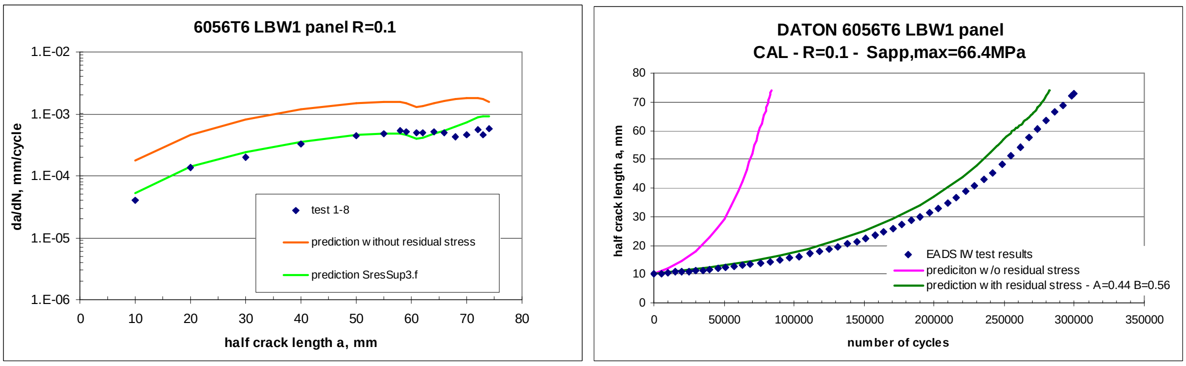


Figure 2 – Fatigue crack growth under R=0.1 CAL on LBW panel. Note the beneficial effect brought by the residual stresses due to welding.

The A and B parameters of PREFFAS model were determined through two baseline fatigue crack growth tests: a CAL test (R=0.1) and a periodic overload test (1.4 overload every 1000 baseline R=0.1 cycles). The obtained values are: A=0.44, B=0.56.

The stress spectrum was modified to include the residual stress. Then PREFFAS was implemented with the compression option in order to make the prediction of crack growth. The comparison to the experimental data shows some fairly good agreement when the crack is within one half of the bay area on the LBW panel (Figure 3). In terms of number of flights to reach the stringer, the prediction is 15% conservative. Additional work will be done to tackle the issue when the crack is approaching the stiffener area and refine the modelling approach. Predictions without residual stress were also carried out to show the benefit brought by the residual stress.

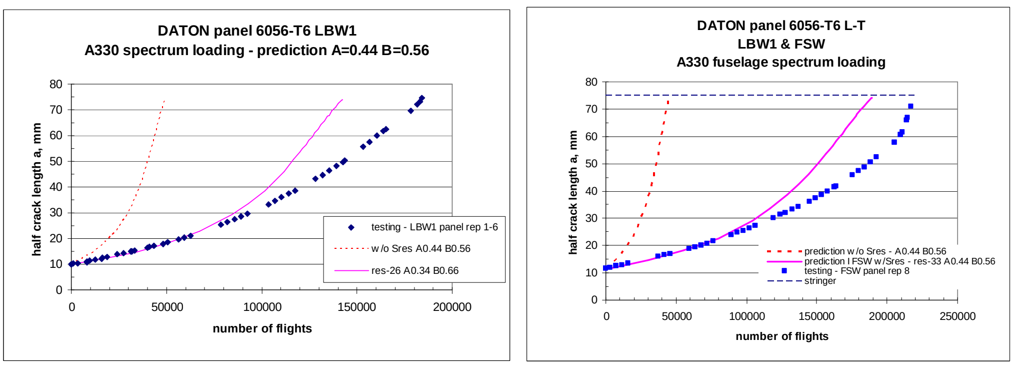


Figure 3 – Fatigue crack growth under A330 fuselage spectrum loading on LBW and FSW stiffened panels. Testing results, predictions with and without residual stresses. Note the beneficial effect brought by the residual stresses due to welding.

The same exercise has been carried out on panels with two FSW stringers and the results are shown in Figure 3. There is a dramatic benefit from the FSW technology. This benefit is even greater than that of LBW. The predictions are fairly good when the crack is in the first half of the bay area. In terms of number of flights to reach the stringer, the prediction is 20% conservative. Additional work will be done to refine the understanding when the crack is approaching the stringer.

Prospective calculations were made in order to assess the expected benefit from LBW and FSW technologies over integrally machined panels (Figure 4). The damage tolerance capability was assessed in terms of inspection intervals calculated under the A300 spectrum loading. The FSW technology offers an improvement by a factor of 4 over the machining technology.

	LBW1 over M	**FSW over M**	**FSW over LBW1**
expected increase in inspection interval	+ 189%	+ 323%	+ 46%

M: machined stringers (no residual stress)

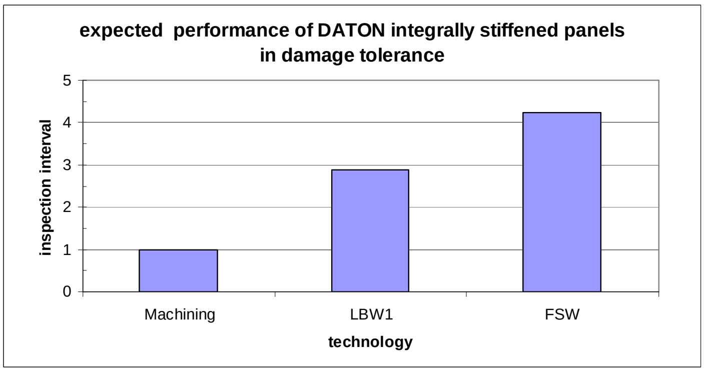


Figure 4 – Expected performance of integrally stiffened panels (DATON geometry and A330 flight spectrum).

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6.2.2. Multiaxial crack initiation fatigue life prediction in rotor components (ONERA)

An important program has been performed in cooperation between Snecma, Turboméca and Onera, involving also the participation of CEAT and Armines. The experimental part involves series of classical fatigue tests under stress or strain control, on TA6Vpq and INCO 718, but also multiaxial fatigue testing, in tension-compression-torsion and in tension-internal pressure on tubular specimens, together with biaxial fatigue tests on specially designed cruciform specimens. All these tests are designed to assess multiaxial crack initiation fatigue criteria under loadings that involve large stress ration, as needed for several typical components lifing (it is a macrocrack initiation, of a significant size, like 0.3mm in depth.

The greatly improved knowledge generated by this experimental program has demonstrated, among other results, that classical multiaxial fatigue criteria are all with some limitations when we simultaneously take into account pure shear and equibiaxial regimes together with a correct description of uniaxial mean stress effects.

Such defects in the standard modelling capabilities lead to the development of a new multiaxial criterion and a corresponding lifing model. This model calls TOS (like Turboméca-Onera-Snecma). Figure 1 summarises fatigue results under uniaxial and multiaxial conditions on TA6Vpq, using a representation of iso-life lines in the biaxial stress plane. Tubular specimens are used for reversed tension-torsion (R = -1) and for repeated uniaxial and equibiaxial conditions (R = -0.3) [1], though cruciform specimens (tests performed at Onera) cover intermediate ratios (R = -0.15). The left figure shows classical Sines criterion and Gonçalvès et al. [2] criterion, though the right one shows the new proposed criterion [3]. On the other hand, Crossland criterion gives totally uncorrect results, both for pure shear conditions (very unconservative) and for equibiaxial ones (very conservative).

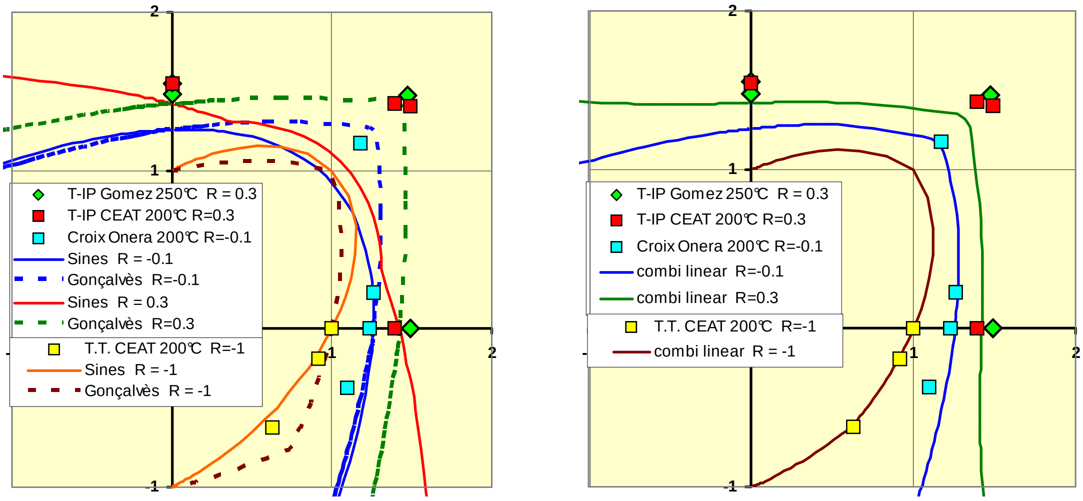

Figure 1 : experimental results for TA6V at two temperatures in biaxial maximal stresses, with stress ratio of R = -1, -0.1, 0.3 and interpretation with 3 criteria : (a) Sines and Gonçalvès criteria ; (b) new proposed criterion

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6.2.3. Study of fatigue propagation of short cracks: influence of plastic wake and closure (LMPM)

Crack closure is widely used to rationalize various fatigue crack growth characteristics and has become one of the most intensively studied phenomena associated with fatigue crack growth. However the characterization of closure is mostly based on long crack concept, while only few studies have been performed on short crack. Yet, the behaviour of short cracks is shown to be different from that of long cracks. In long crack regime, the fatigue crack growth rate, da/dN, can be characterized by the SIF range, ΔK, as a dominant driving force. On the other hand, when using ΔK for describing short crack growth rate characterization, it usually exhibits a faster growth rate than prediction made under long crack methodology; moreover short cracks even grow below the threshold where long cracks are assumed to be dormant. The effect of short crack and characterization of short fatigue crack growth are, therefore, fairly critical for accurate lifetime prediction of aeronautical components and structures.

This study aims to investigate the effect of crack length and of the size of the plastic wake on crack closure in a 304L austenitic stainless steel [1-5]. In a first step, acute identification of the cyclic behaviour law has been carried out, up to a ±2.4% cyclic strain amplitude.

Then, the effect of tensile prestrain on crack growth behaviour has been investigated. Fatigue crack propagation tests were performed on single-edge notched tension (SENT) raw specimens (0% of prestrain) and on prestrained specimens (2% and 10%). On one hand, it has been found that the different levels of prestrain exhibited no significant influence on crack propagation in the high range of Stress Intensity Factor (SIF), where there is no detectable crack closure. On the other hand, a clear effect of prestrain on crack growth rate can be observed in the near threshold region where closure is detected. Thus, it can be concluded that the prestrain mainly affects the crack growth rate through its influence on the crack closure.

Moreover, CT specimens have been initially pre-cracked under different constant applied stress intensity factor (SIF) ranges at a load ratio of R = 0.1. Crack closure is detected by means of a back face strain gage at different crack lengths obtained from a progressive electric discharge machining of the crack wake, in order to obtain residual crack lengths between 0.1 mm and 2-3 mm. A numerical tool is developed to analyze the load-displacement curves after signal analysis by means of data filtering. For different applied SIF ranges, the SIF for crack opening Kop is shown to increase with the residual crack length ΔC and reaches a steady state Kop level for ΔC longer than 1.2 mm. An analytical expression of the evolution of closure versus the crack length is given.

Associated three-dimensional FEM calculations with Abaqus are under progress.

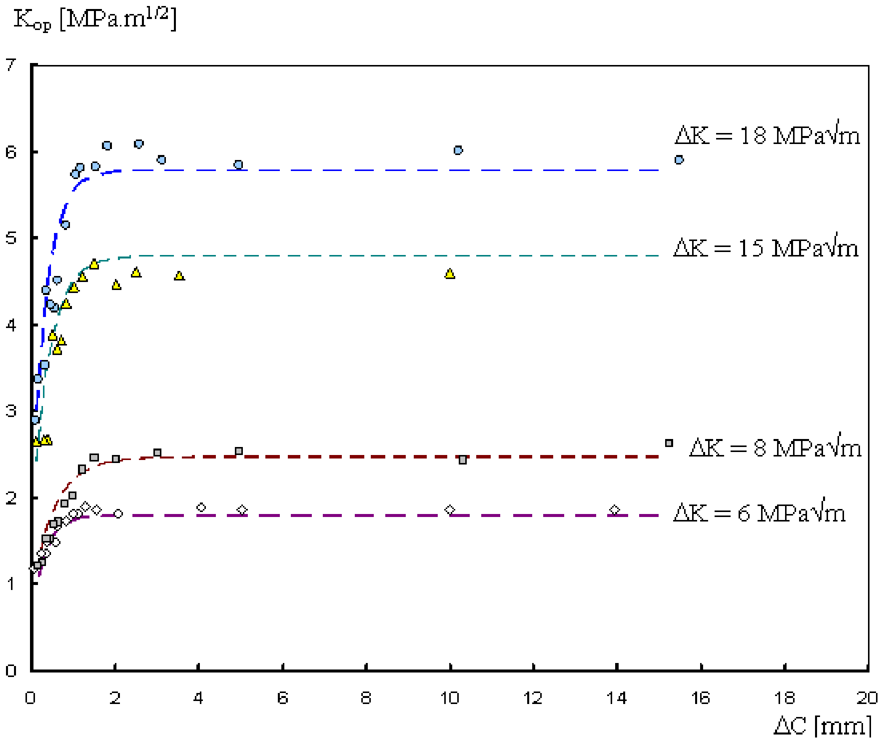

Figure 2 : Comparison of closure behaviour between current model and previously obtained experimental results (Dot for experimental data, dashed lines for modelling).

References

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6.2.4. Influence on cast defects on the Fatigue Quality Factor in a Ti6AlV alloy (LMPM)

The fatigue design of cast components requires taking into account the presence of defects induced by the process. One of the basic data used by Airbus is the Fatigue Quality Factor (IQF) which corresponds with the allowable stress for a fatigue life equal to 105 cycles. The present study has many objectives:

- the evaluation, on the basis of experimental results, of the decay on the IQF when a surface or internal defect is present in the material;
- the comparison of this decay with the predictions of conventional engineering rules and the proposal of advanced rules in relation with NDT techniques;
- the development of a model accounting for the observed decay in an engineering component

At this stage, this study has mainly been concerned with experimental developments. Figure 1 presents a small surface defect machined at the surface of a fatigue testpiece and chemically machined in order to reproduce actual service conditions.

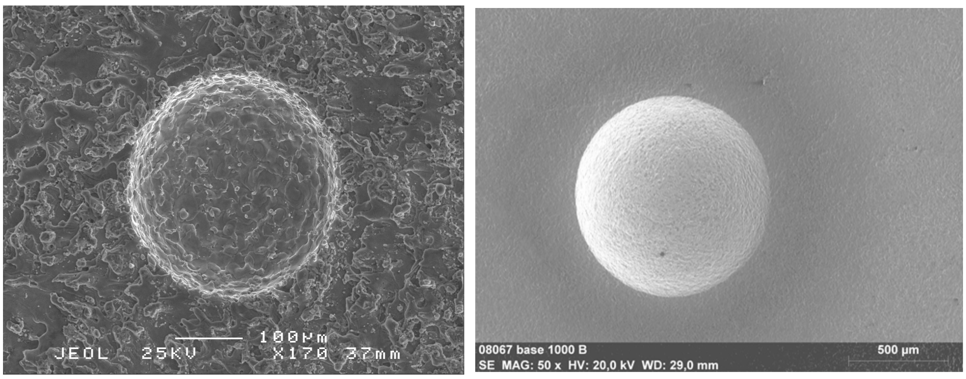

Figure 1: representative artificial defect at the surface of a test specimen before and after chemical machining.

References

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6.2.5. Comparison of different numerical approaches in the evaluation of the nocivity of internal defects in fatigue life at 20K of cast Inconel alloy (LMPM)

The main objective of this preliminary study is to evaluate the nocivity of internal casting defects in an Inconel alloy fatigued at 20 K. As the available experimental data are very limited, different methods have been compared to the conventional approach which consists in considering a casting defect as a circular internal crack. The diameter of this crack is equal to the larger dimension of the defect and the propagation of this defect is controlled by the threshold in mode I. The present work has demonstrated that such an approach is conservative and confirmed the interest of the crack initiation approach developed in LMPM. The basic principles of this approach are presented in figure 1.

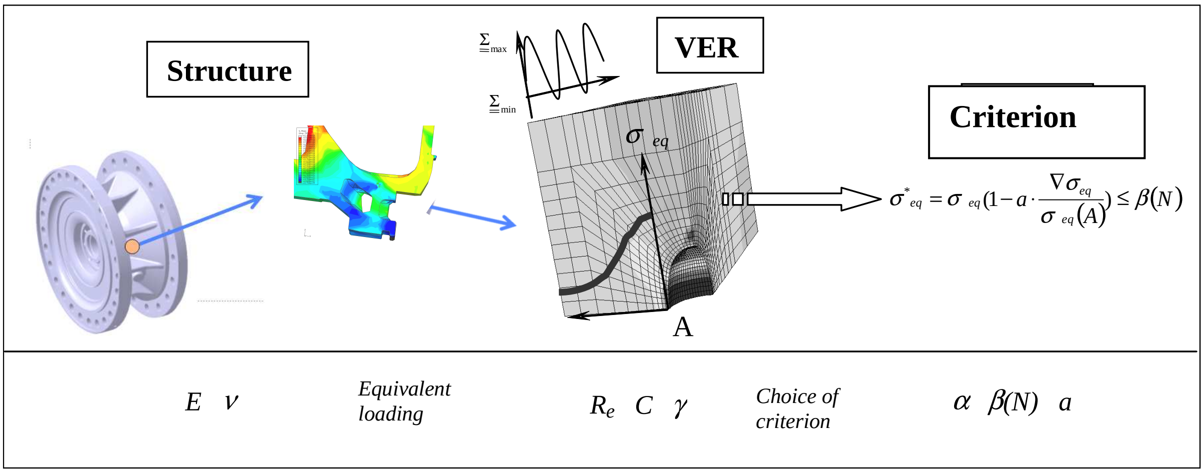


Figure 1 : Methodology of the evaluation of the nocivity of an internal defect.

3D FEM calculations are carried out in order to obtain principal stresses that are then applied on the edges of a cubic model containing a spherical cavity in its centre. This geometry is less severe than a crack and more representative of a casting defect. The stress gradient in the vicinity of the defect in then introduced in a multiaxial crack initiation criterion. This criterion requires the identification of only three parameters.

6.2.6. Multiaxial fatigue life prediction: an approach by invariants (LMPM)

The objective is to propose a criterion based on invariants of the stress tensor. This criterion depends only on the macroscopic loading allows a fair prediction of fatigue life for complex multiaxial (tension – torsion) loadings. In particular, phase effects are correctly taken into account by distinguishing the influence of the second invariant of deviatoric stress and its mean value. The influence of mean stress is also captured by involving explicitly its amplitude. This criterion is validated by comparison with a wide database (119 tests, cf. Figure)

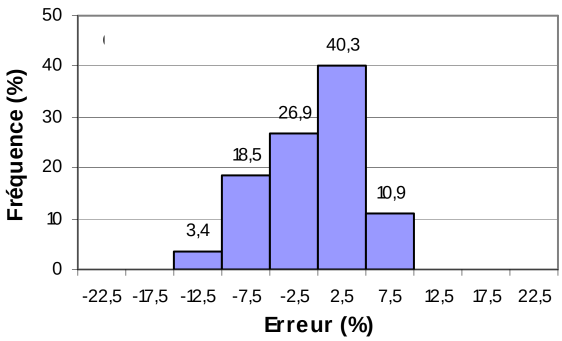


Figure 1: Error prediction of the criterion

The domain of limited life time ($10^5 – 10^6$ cycles) is dealt with by coupling this criterion with a damage model built in the rigorous framework of the Thermodynamics of Irreversible Processes. The life time is obtained by controlling the evolution of a scalar damage variable until it reaches a critical value. The model is identified and validated from tests on steel (cf. Figure 2 and Figure 3), for different values of phase and ratio between tension and torsion stress.

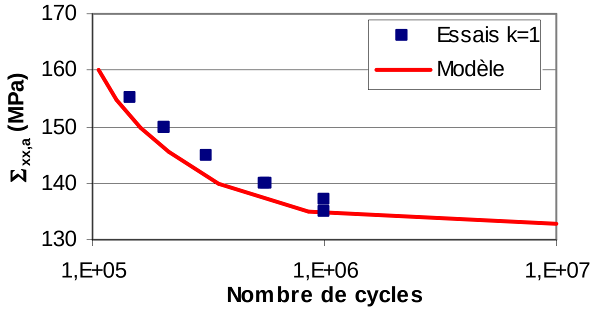


Figure 2: In-phase loading

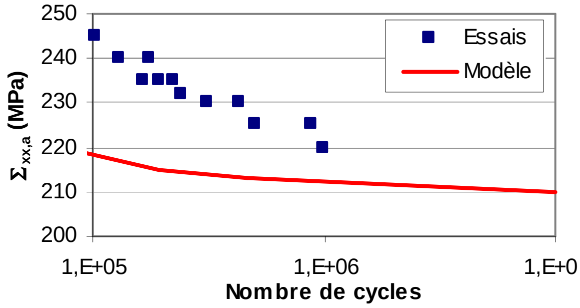


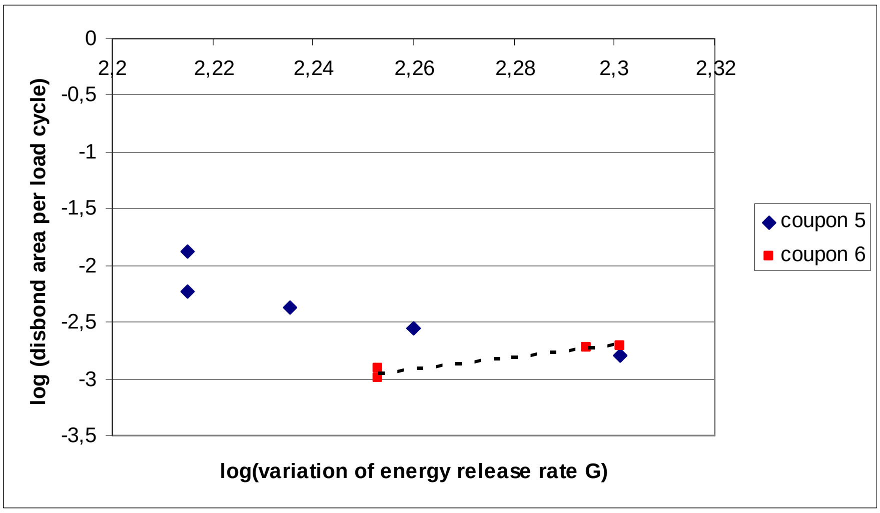
Figure 3: Out-of-phase loading

6.2.7. Repair of metallic structures with bonded composite patch : disbond propagation testing and modelling (CEAT)

The general objective is to progress in the certification of metallic structure repair process by bonding a composite patch on the damaged part. On military aircraft, some countries already use that repair and its mechanical efficiency is now well established. The composite patch acts as a stress bypass avoiding any supplementary hole drilling and associated stress concentration. Nevertheless the process is not used in France because of a lack of confidence in bonding quality and durability.

Besides, a disbond can be detected by classical non destructive techniques. Therefore, disbond propagation can be controlled if a structural inspection program is defined and followed. Thus, damage tolerance requirements may be fulfilled. For this purpose, disbond propagating must be predictable with sufficient confidence. In fact, this study addresses the problem of predicting the growth of an initiated disbond between the metallic structure and the repair.

The finite elements modelling of the disbond growth has been performed, based on total energy release rate. The energy release rate related to disbond growth has been determined for different disbond sizes and load levels. Besides, fatigue testing was performed on coupons in order to characterize the linear elastic fracture parameters of the adhesive. In that purpose, the fatigue test is regularly stopped and the specimen is controlled by ultrasonic non destructive techniques. The results, give few points which represented on the following figure.



Unfortunately, too few test results were obtained to give a law with sufficient accuracy. It is due to the fact that, on the one hand, the shapes of the occurred disbond were not exactly the one assumed for the energy release rate computation and, on the other one, that the precision of the visual measures are poor. To complete the propagation prediction tool, a method is proposed to estimate the disbond size growth under a composite patch and therefore the fatigue life.

Significant gain in confidence in the propagation law and extension to downgraded bonding conditions are necessary to complete the assessment of bonded repair but the difficulty to obtain debonding and the stable growth rate of the observed disbond are globally reassuring for this repair technique.

6.2.8. Vibratory fatigue : gigacycle fatigue exploration (CEAT)

Experience feed back has shown for many years that some cracks could appear and propagate in vibratory environment even if they were not predicted by standard tools usually used to design aircraft structure. As a matter of fact, after having drawn a state of the art, CEAT launched a study on the exploration of the gigacycle fatigue domain, a study which is part of a more general project concerning vibratory fatigue. Indeed, if it is well establish that fatigue limit does not always exist, fatigue test are commonly stopped at 107 cycles while Wöhler's curves are extended thanks to models like the Haibach ones or thanks to statistical approaches. Unfortunately, when the life range between 107 and 109 cycles is explored, the difference between the fatigue limit at 106 and 109 is sometimes huge and the statistical tools, usually used to extrapolate the conventional curve, do not take into account such a gap. It explains why some cracks might appear even if all the precautions have been taken.

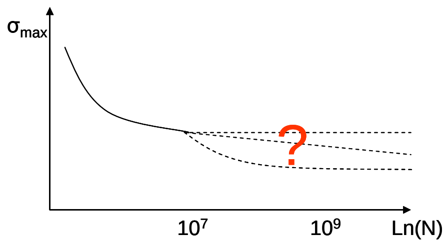


Figure 1 : Wöhler's curve extension

As reaching 109 cycles at 100 Hz lasts more than 16 weeks with traditional fatigue test devices, which is not economically acceptable, many efforts have been undertaken to increase test frequencies and to set up new test processes. To investigate such phenomenon, CEAT developed its own test procedure.

To accelerate the test frequency, the idea was to use hydraulic shakers usually employed to test the robustness of electronic devices in vibratory environment. The philosophy, inspired from [1] and [2], consists in exciting a sample clamped on an electrodynamic shaker near one of its resonance mode.

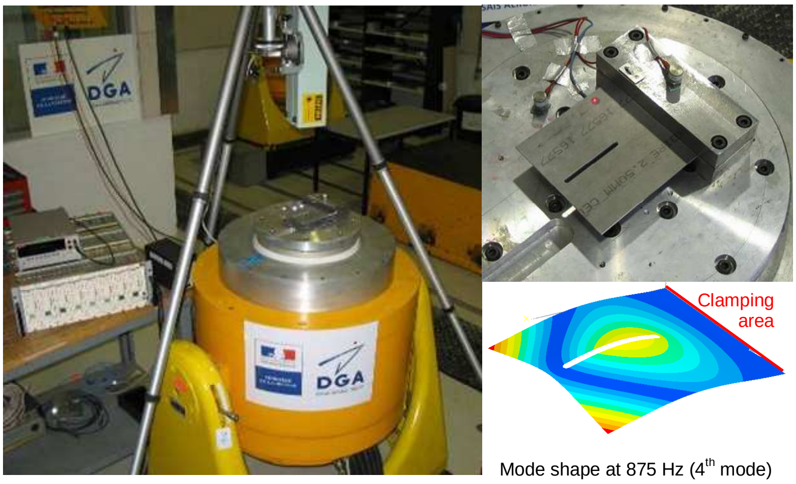


Figure 2 : test set up and test specimen

The efficiency of such a test was proved thanks to a correlation with a fully reversed bending fatigue test at 2Hz. While no damping heating was detected during the test, life durations obtained at more than 860Hz were similar to these at 2 Hz. Finally first tests of an ongoing test campaign demonstrated that it was possible to explore the gigacycle fatigue domain with this new test procedure.

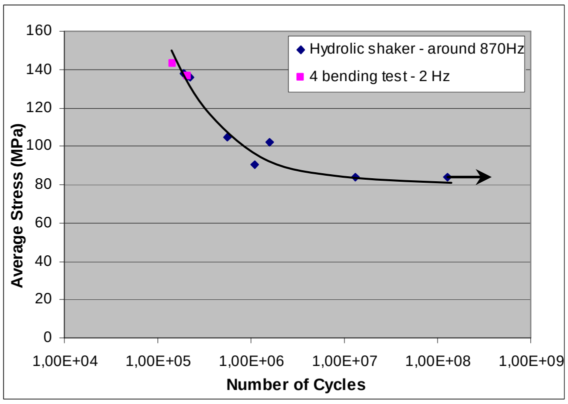


Figure 2 : test set up and test specimen

The next step will be to use this new fatigue testing methodology to study crack initiation on pitting corrosion and to demonstrate that very short cracks could propagate even subjected to a very low stress level.

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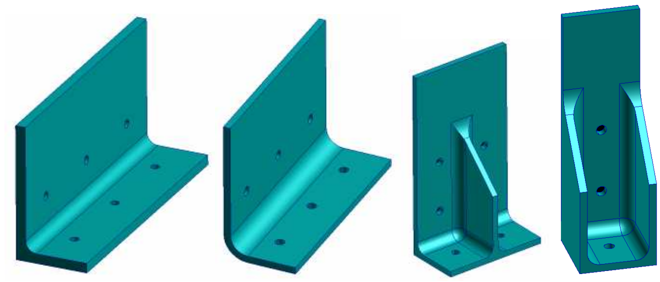
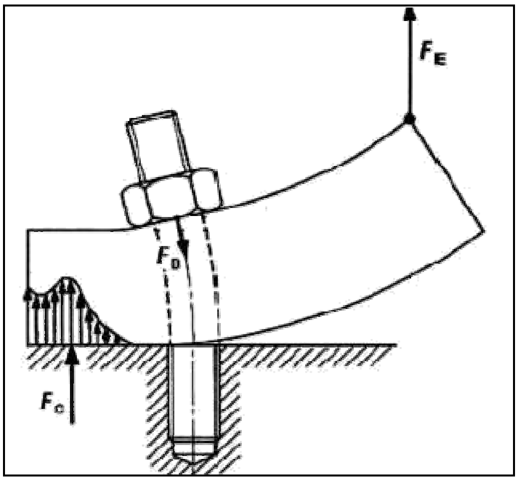
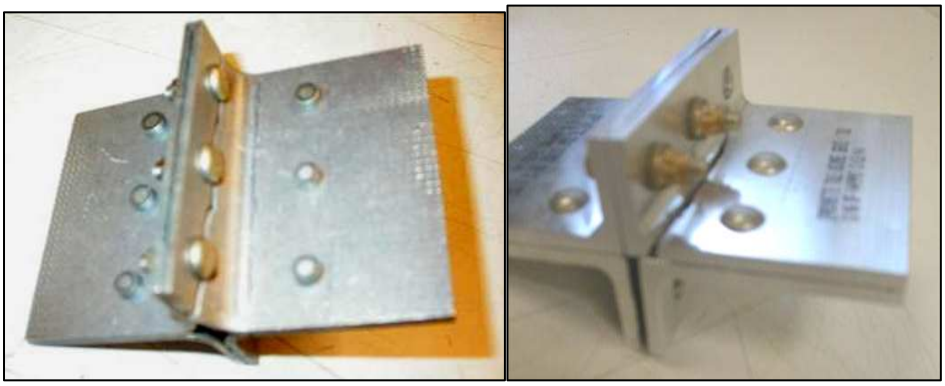
6.2.9. Fatigue life evaluation on tension type fitting (Latécoère)



Latecoere presents during ICAF2009 Poster Sessions an “in house” methodology developed for predicting fatigue live for unfolding fitting. This approach is based on combination between a basic analytic approach and tests results. By analysis, the fitting equilibrium is defined, including “prying effect”. A simple “fictive peak stress”, picture of the unknown real peak stress, is determined.



Correlated by tests results, it is able to determine a law which allows obtaining fatigue lives.



Many unfolding fitting configuration are studies (1 to 3 walls, pretension fasteners or without pretension, folded sheets or machined fittings).

More information could be found during ICAF Posters Sessions.

6.3. COMPUTATIONAL TECHNIQUES

6.3.1. Modelling of crack initiation and propagation in single crystal turbine blades (ONERA)

With the objective to better predict and assess these two stages of the component life, two studies have been made in parallel:

- the classical crack initiation model of Onera (for this kind of single crystal, it is the FatOxFlu model, Fatigue-Oxydation-Creep model [1]) and a new incremental version have been used to predict crack initiation at small cooling holes. From an experimental study performed for single crystal AM1 at 950°C on plates of various thicknesses and containing holes of different diameters, the need for taking into account gradient effects in the damage rules has been evidenced. Various different approaches have been studied in order to take into account the gradient effects, and to experimentally identify a consistent “material length” parameter. A strategy has been developed at the post-treatment stage of the full cyclic inelastic analysis, that allows to correctly average the fields before application of the damage model. Figure 3 illustrates the results obtained and the corresponding predictions, including the possibility to have in the same framework the prediction of initiation at large porosities (of about 50 microns size), also treated by the full 3D inelastic analysis [2].

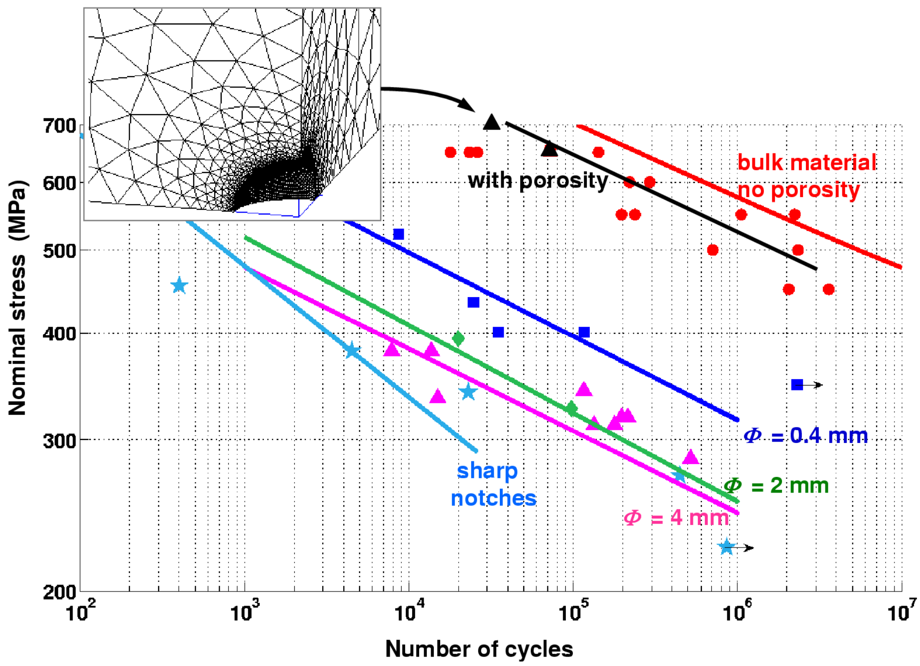


Figure 3: modelling of the stress gradient effect at various notches and hole diameter on a single crystal superalloy (simple Basquin law adjusted on the smooth specimens of bulk material with no porosity. Lives at two observed porosities on smooth specimens are also correctly predicted after the full inelastic analysis (with 2 successive cycles only).

- In the context of the European Program SOCRAX, a numerical methodology has been developed, that allows to simulate the simultaneous evolution of a fatigue crack and of cyclic elasto-viscoplasticity at the crack tip, in the active plastic zone. The method uses cohesive elements all along the considered crack path, and allows a continuous crack growth under cyclic conditions, consistent with the predicted fatigue crack growth rate. The cohesive model involves a continuum damage type of description, including an incremental rate independent part and a creep part (consistent with the creep damage modelled at the continuum level in the bulk material). The application is made on a single crystal superalloy used in turbine blades, where the bulk material is described by crystal viscoplasticity constitutive equations (considering both octahedral and so called “cubic” slip systems). Figure 4 illustrates the overall axial plastic strain component after a fatigue crack propagation under a constant maximum load, including 20 cycles with an overload [3].

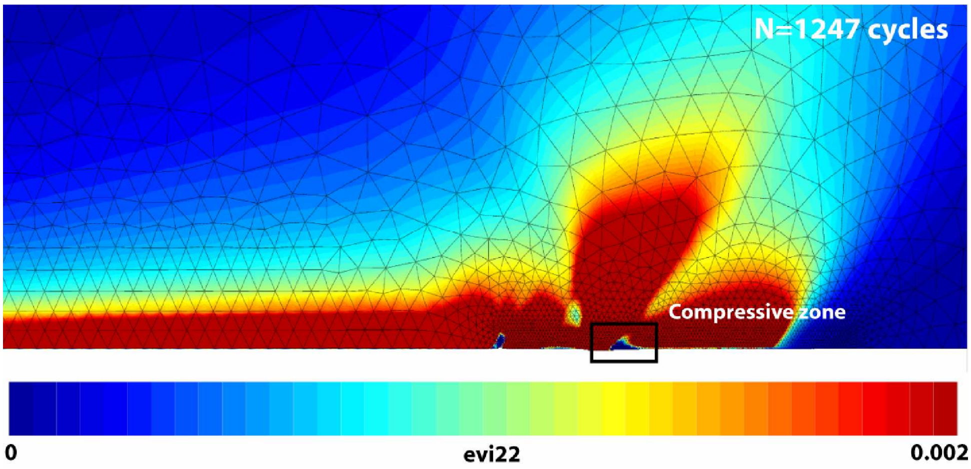


Figure 4: simulation of fatigue crack propagation and associated plastic strain fields on a single crystal plate for a periodic loading that includes 20 successive overload cycles

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6.3.2. Automatic 3D fatigue crack growth simulation in structural components (ONERA)

Another well advanced activity concerns the finite element implementation of crack propagation simulation methods both for monotonic fracture and for fatigue, in the context of LEFM and "Damage Tolerance" analyses [1]. The most recent techniques for evaluating SIF along the crack front were developed (based on energy release considerations together with so-called $G\theta$ method, a French development [2]). The 3D crack propagation is simulated automatically, using these techniques, together with a conforming automatic remeshing (at each step of the non-linear crack growth analyses) or with XFEM method.

Some advantages of the implemented tools are to address any kind of local fatigue crack growth equation (local along the front) and to be usable either in an explicit incremental way or with a more stable implicit scheme. Taking into account residual stress fields and the possible continuous bifurcation of the crack through a generalised energy framework (in 3D) are additional capabilities of the method.

Figure 2 shows simulated fatigue crack propagation on a turbine disk in the attachment zone of the blades, and a crack propagation in a cruciform specimen (N18 powder metallurgy Nickel base superalloy at 600°C) on which the biaxial loading has been an evolutive one in order to promote a continuous crack bifurcation.

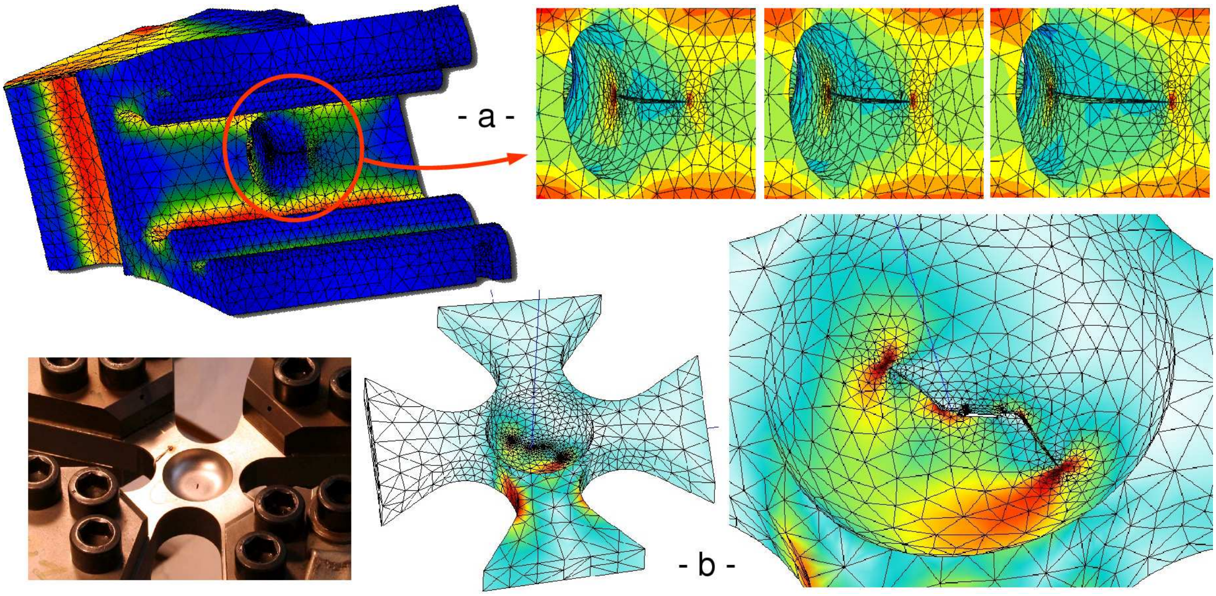


Figure 2 : simulation of fatigue crack propagation, on turbine disk attachment part and on a laboratory cruciform specimen with a varying load ratio

References

1. G. Edeline, V. Chiaruttini, M. Bonnet, F . Feyel, Approche énergétique implicite pour la propagation de fissures en 3D, 8ème Colloque National en Calcul de Structures, Giens, 2007.
2. P. Desruynder, M. Djaoua, S. Lescure, Quelques remarques sur la mécanique de la rupture élastique, Journal de Mécanique Théorique et Appliquée, Vol. 2, N° 1, pp. 113-135, 1983.

6.3.3. CDM approach applied to fatigue crack propagation on aeronautical metallic alloys (LMPM)

This paper deals with a work conducted in collaboration with AIRBUS France. The aim is to develop a numerical model to predict the behaviour of cracked panels, in place of expensive and complex experiments. In particular, the model has to simulate R-curves and fatigue crack propagation at high level of ΔK for a wide variety of metallic alloys including aluminium alloys, titanium alloys and steels.

The proposed model is based on thermodynamic approaches within the framework of Continuum Damage Mechanics which provide numerical robustness as well as a precise description of the physical micromechanism of fracture. Up to now, CDM models have been only applied separately to low cycle fatigue and high cycle fatigue. This study examines the possibility to use such models to simulate fatigue crack propagation. Instead of classical CDM theories, using a single damage variable, two distinct scalars called cyclic damage and static damage have been introduced to take into account experimental observations. Scanning Electron Microscopy has been used to differentiate the two types of damage which appear on fatigue crack propagation. While the static damage evolution law is consistent with the classical Lemaître's model, the cyclic damage is assumed as a consequence of cumulated plastic strain and does not influence plasticity. Then, an energetic equivalence criterion has been inserted to model crack propagation and to reduce the mesh dependency. This method, originally developed by J. Mazars and G. Pijaudier-Cabot, fills the gap between damage and fracture and avoids the introduction of additional boundary conditions or complex code developments associated with non-locality. The last improvement is the transposition of results from a small specimen to a large one.

This model has been implemented in the commercial finite element code ABAQUS/CAE. Implicit and explicit computations have been performed to highlight drawbacks and advantages of the method. Meanwhile, all the parameters of the model have been identified by simple tests like tensile and cyclic stress-strain tests.

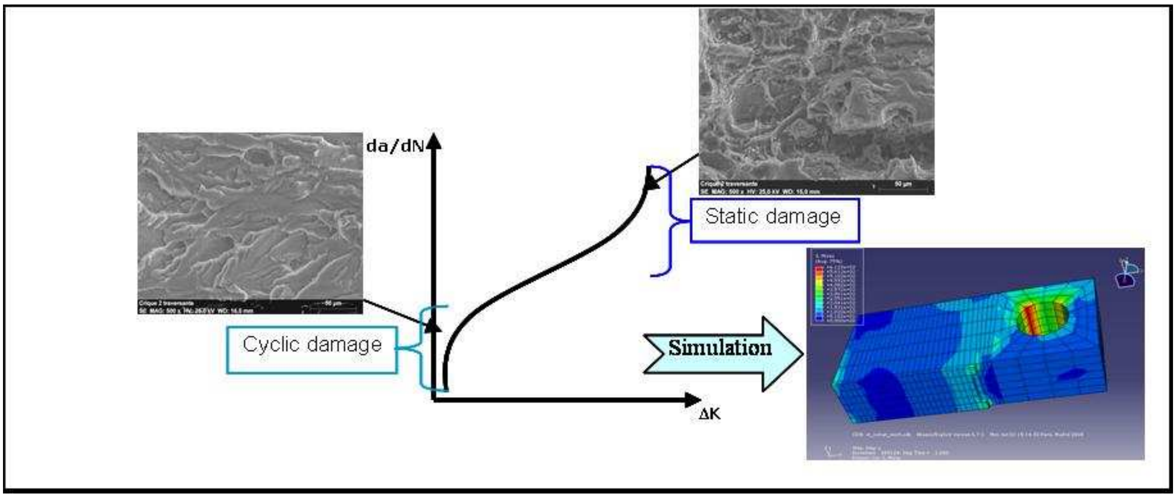


Figure 3 : Differentiation of two type of damage with a Scanning Electron Microscopy and implementation in ABAQUS/explicit to model fatigue crack propagation.

6.3.4. Virtual Testing: fatigue analysis and justification of a complex structural node (Airbus France)

Aircraft certification according to Fatigue & Damage Tolerance (F&DT) requirements is made by analyses supported by test. F&DT criteria are more and more taken into account at design / sizing stage of new aircraft (A/C) models. Full-Scale Fatigue Tests (FSFT) is performed to support analyses. The test results allow identifying fatigue hot spots in structure that will require design modifications and/or additional inspections in service. The test identification of fatigue sensitive areas could occur at the end of the aircraft development step.

Virtual Testing consists in a refined fatigue analysis of the structure based on a hierarchy of successive fatigue analyses and associated finite element models from the global aircraft review to the local one.

The virtual testing objective is twofold:

- Improve the fatigue prediction of hot spots during the design stage (before FSFT starts), in order to improve the design, improve the in-service maintenance and avoid later retrofit modifications.
- Improve the justification approach of modifications applied on derivative A/C, based on a better extrapolation from basic fatigue assessment and/or test results.

To validate the virtual testing approach, a first investigation is performed by a 'round robin' exercise for a known complex structure to identify the locations of the fatigue sensitive sites and their fatigue life potential.

The Virtual Testing consists here in a detailed fatigue analysis focusing on a complex A/C structure. It is based on the development and the analysis of efficient finite element models. A single detailed model is not efficient for the complete aircraft area to be analyzed as :

- Such model would require a detailed FE model with a huge number of elements in order to enable fatigue analysis based on the local stress analysis at all riveted and non-riveted areas of the focused structure.

- The needed numerical capabilities and processing time are not realistic today, regarding the fatigue criteria

The fatigue objectives of the Virtual Testing are reached by establishing **a hierarchy and a prioritization of successive fatigue analyses and associated models**, regarding fatigue criteria.

A first **global finite element model (GFEM) of the complete aircraft** is used to determine the internal fatigue loads at the focused complex area. Several load cases (steady load cases and disturbances during ground and flight steps) are analyzed to define the complete loading spectrum,

A second **global-local model (FEM1) of the focused complex area** is used to identify and select the possible areas with potential risks of damages :

- FEM1 model does not include contact elements and local fastener stiffness.
- The local stress analysis is not possible with FEM1 model in riveted areas.

Selection criteria are defined to determine the areas to be detailed in FEM2 model. The selection criteria are based on fatigue considerations and load cases:

- Criteria based on tensile and/or compression stresses at non-riveted areas.
- Criteria based on by-pass stress and transferred forces at riveted areas.

A third **detailed model (FEM2)** is refined in selected areas to extract the local stress:

- FEM2 model shall include local contact between assembly parts, local re-meshing of the most loaded fasteners, and possible re-meshing of the generic areas of structural parts (machining radius…).
- It enables to post-process the local stresses regarding the fatigue criteria.

A **detailed fatigue analysis** is performed for the focused structure:

- To determine the fatigue capability of the structure, according to Airbus fatigue methods:
  - Location and fatigue life of each fatigue sensitive site.
- To establish the justification report compliant with fatigue requirements:
  - Fatigue optimization and justification of the structural design, regarding the F&DT requirements (during the design process),
  - Determination of additional inspection tasks in service (during certification process) or fatigue justification of structural design changes for derivative aircraft.

A first investigation is performed by a ‘round robin’ exercise for a known complex structure. The results of the virtual testing approach are compared to the damage findings during a full-scale fatigue test.

Using the prioritization of the successive analyses on a known complex structure:

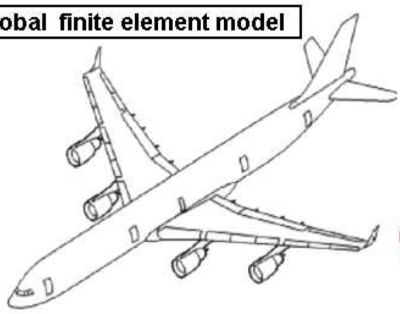
- The virtual testing analysis and its associated finite element models of local areas enable to focus the most significant locations of damaged areas.
- For these areas, the fatigue life calculations performed at Airbus are accurate or more conservative than the life from test results.

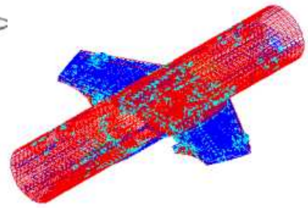
Based on the fatigue analysis of known complex structures, directives and recommendations are identified for the use of the Virtual Testing on non-tested structures.

Numerical analysis of fatigue spectra with

Global finite element model

(partial view from GFEM for the centre fuselage sections)





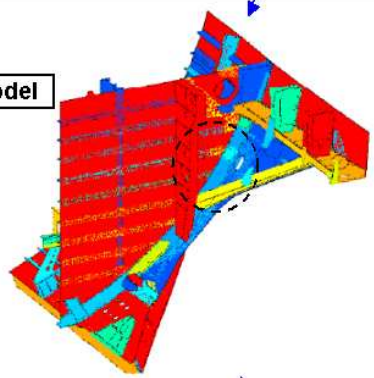
Fatigue spectrum analysis:
- steady and disturbance load cases at ground and flight steps.

--> around 50 load cases

- determination of fatigue equivalent stresses for each significant local areas, according to Airbus fatigue methods.

'Blind' numerical analysis with

overall FEM1 model



Overall un-detailed model:
- shell elements
- some volume elements
- rigid fasteners
- no contact element

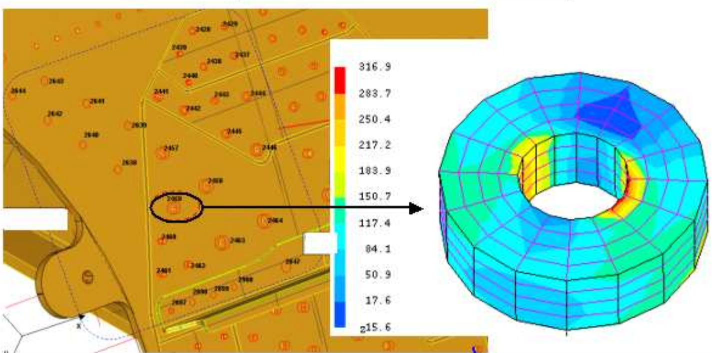
Number of assembly parts:
60 thin structural parts
70 thick structural parts
3 000 rigid fasteners

--> FEM1 un-detailed model:
> 600 000 nodes
> 350 000 elements
> 2 000 000 degrees of freedom

Definition of selection criteria to identify areas with potential risk of fatigue damages:
- Un-notched areas in tension: tensile stress
- Un-notched areas in compression: compression stress
- Riveted areas with low or high shearing transfer: shearing transferred force & by-pass stress
- Riveted areas with high tensile transfer: tensile transferred force

'Blind' numerical analysis with

Detailed local FEM2 model



Local detailed model:
- 3D fasteners and hole surroundings
- refined meshing in riveted areas
- contact elements for the assembly
- Detailed model connected to the overall model with a specific transition areas

Number of assembly parts:
Un-detailed FEM01 model including detailed FEM02 local area:

+ 5 local areas of structural parts (volume elements)
+ Number of 3D-fasteners: 40 maximum.
+ 2000 volume elements per fastener

--> FEM2 local detailed 3D-model:
> 4 000 000 degrees of freedom

Extraction of local stress values.

Detailed fatigue analysis

At fastener n° xxx surrounding:

Principal stress 1	Stress value [MPa]	d fastener [mm]	E	M	T	C	CS ref.	TF Test Factor	N calculation [FC]
S1 max / Doubler	Fatigue parameters, from Airbus fatigue methods.								231 599
S1 max / Stiffener									58 774
S1 max / Panel									46 791

	N test [FC]
Crack detected at panel level	48 782

Detailed fatigue analysis:
- Location and fatigue life of each fatigue sensitive site

--> - Optimization of the structural design regarding the fatigue requirements.

figure 1

The lessons learnt and the resulting fatigue directives improve Airbus capability to predict fatigue potential of complex structures at an earlier step of the aircraft development.

Up to now, the Virtual Testing could be successfully performed:

- At the design step:

Fatigue justification of the structural design (location and fatigue life of each fatigue sensitive site) & Design optimization of complex structural areas regarding fatigue criteria are improved at an earlier step (before fatigue certification tests), to avoid later retrofit modifications.

- At the certification step:

Determination of additional inspections in service, for specific fatigue sensitive sites. The inspection intervals are better optimized.

Justification of structural design changes for derivative aircrafts based on efficient extrapolation from fatigue assessment and/or test results of the basic aircraft.

- At the in-service step:

Regarding the improvement of the fatigue prediction due to virtual testing process at the design step and certification step, benefits exist for the maintenance program.

For the near future, more experiences and feedbacks from fatigue tests and numerical simulation results will improve the virtual testing approach and associated directives and methods to predict with accuracy the fatigue capability of aircraft structures.

6.4. FATIGUE OF COMPOSITES

6.4.1. Fatigue of aeronautical CFRP materials (LMPM)

Despite the exceptional mechanical specific properties of polymer matrix – carbon fibre composites structures, their current range of application is rather limited. The aeronautical industry, in particular, feels the need to enlarge this range, employing such materials in structural applications that involve severe long term thermo-mechanical solicitations under aggressive environments, in which the material is exposed to thermal-oxidation effects. Deep understanding of all these phenomena is crucial to the building of non empirical models for predicting the material behaviour, the onset and the development of damage and the durability of composite structures.

In our laboratory, campaigns of thermal cycling tests have been performed on various carbon/epoxy laminates in more or less oxidative or neutral atmospheres (dry air, pure oxygen and nitrogen). These studies put in light the influence of the environment of the thermal cycling test on different damage mechanisms :

- Superficial matrix shrinkage due to the oxidation process (Figure 1)
- Matrix cracking, quantified through microscopic observations of the edges of the samples and X-radiographs
- Weight loss

The comparison of these damage parameters according to the environment of the thermal cycling test emphasises the accelerating effect of an oxidative atmosphere. Moreover, the higher the oxygen concentration, the more significant the acceleration of the damage processes. Two different amplitudes of thermal cycling have been considered: these experiments highlighted a coupling between matrix cracking and oxidation processes, the degradation rates as well as the ultimate damage states depending on both the ply-stresses and the thermal ageing conditions.

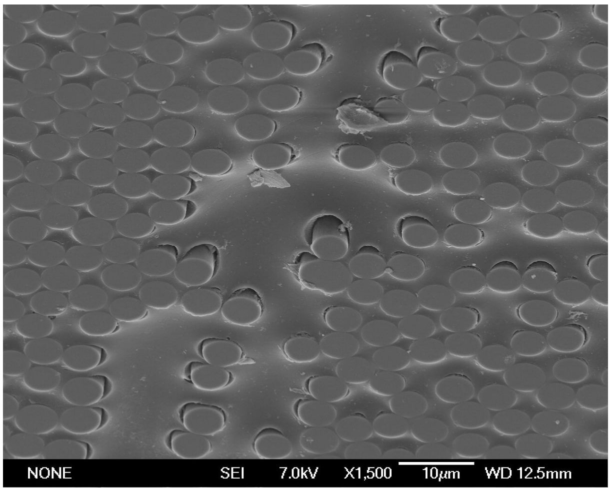


Figure 1: Carbon/epoxy composite: SEM observation of damages due to 1000 thermal cycles (-50℃/150℃) in air

References :

1. Lafarie-Frenot MC. Damage mechanisms induced by cyclic ply-stresses in carbon-epoxy laminates: Environmental effects. International Journal of Fatigue. 2006;28(10):1202-1216.
2. Lafarie-Frenot MC, Rouquie S, Ho NQ, Bellenger V. Comparison of damage development in C/epoxy laminates during isothermal ageing or thermal cycling. Composites Part A: Applied Science and Manufacturing. 2006;37(4):662-671.

6.4.2. Coupled reaction-diffusion-stress model for CFRP durability (LMPM)

Thermal oxidation has generated less interest in the last 30 years. Most of the published data during the 1970–1990's period are relative to weight changes or mechanical property changes with highly questionable kinetic treatments (for instance use of Arrhenius law, etc …). In the 80's, authors begin to take into account the fact that oxidation kinetics is diffusion controlled. The first models of diffusion–reaction coupling contain a correct formulation for diffusion terms, but have an exaggeratedly simplified approach for chemical reaction, described with a single elementary mechanism. Then, Verdu et al. (LIM-ENSAM Paris) develop a model based on a realistic mechanistic scheme for polymer thermal oxidation.

At the present time, as far as polymer matrix composites for aeronautical applications are concerned, only few research groups are preoccupied about this original phenomenon. In the US, there is a renewed interest towards the understanding of oxidation processes for PMR-15 matrices. In France, LIM-ENSAM and LMPM-ENSMA are the two main research groups working on the thermal oxidation of thermoset matrices, mainly epoxies.

Recently, at LMPM, a coupled reaction-diffusion-stress model has been developed: it employs the mechanistic scheme proposed by LIM-ENSAM within the formalism of the thermodynamics of irreversible processes. The model is able to catch stress localisation occurring during thermal oxidation and to predict eventually the oxidation induced fibre-matrix debonding ().

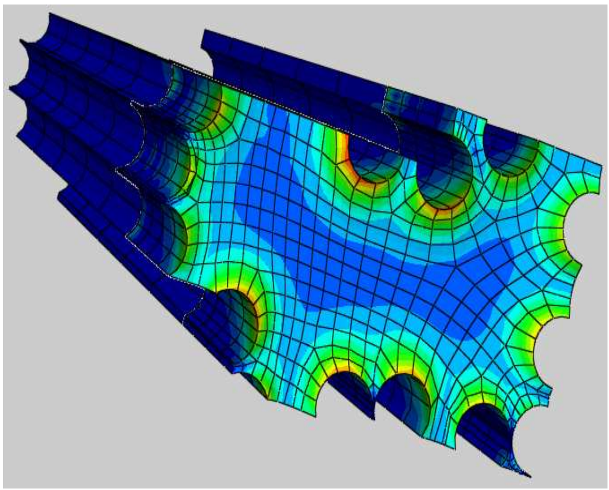


Figure 1: Carbon/epoxy composite: numerical computation of VM stresses due to isothermal ageing at 150℃ in air

References :

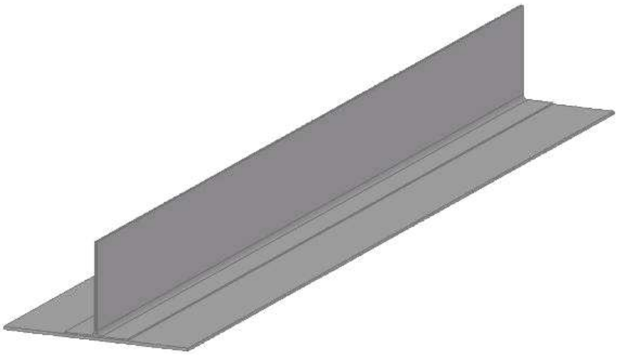
1. Olivier L, Baudet C, Bertheau D, Grandidier JC, Lafarie-Frenot MC. Development of experimental, theoretical and numerical tools for studying thermo-oxidation of CFRP composites. Composites Part A: Applied Science and Manufacturing. 2008;In Press, Corrected Proof.
2. Olivier L, Ho NQ, Grandidier JC, Lafarie-Frenot MC. Characterization by ultra-micro indentation of an oxidized epoxy polymer: Correlation with the predictions of a kinetic model of oxidation. Polymer Degradation and Stability. 2008;93(2):489-497.

6.4.3. Composite door : the COMDOR project (Latécoère)

In the scope of Aerospace Valley labelled projects, Latecoere is working on a new-concept of a composite carbon fibers' stiffened panel. Basically, the main technologies used for making a stiffened panel, except "black metal concept", are the co-curing or co-bonding process. But these processes do not have an efficient strength for damage tolerance requirements.

The aim of the new technology launched by Latecoere, is an innovative process applied on dry preforms for assembly, and injected following RTM technology (Resin Transfer Molding). This process could increase damage tolerance behaviour instead co-curing or co-bonding process Disbonding and delaminations areas must be significantly reduced after BVID impact and the pull-off resistance of the samples improved.

A tests campaign is currently running to demonstrate the expected effect. The tests program started by basic samples (T and I shapes) to validate the technology and initial sizing. Residual Static after fatigue tests with BVID impact were achieved with success.



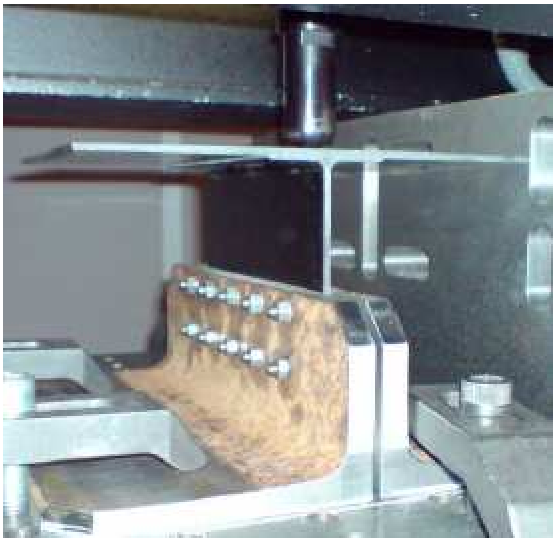
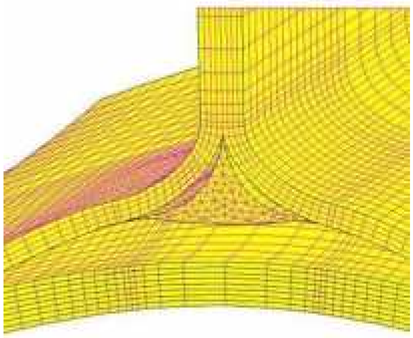
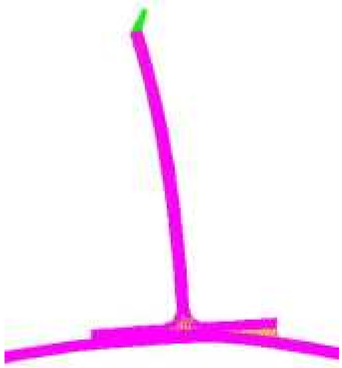


Figure 1 :Initial tests campaign principle

An innovative method of delamination analysis in composites has been applied to study and size the pull-off resistance after impact. VCCT techniques and cohesive elements have been incorporated in a detailed model of the stiffener base.





Now, the next step scheduled is on a complex coupon representative of a stiffened fuselage panel loaded by pressure. The test coupon dimensions are roughly around 1 meter square. The specimen is currently in production phase. After produced and controlled, it will be impacted, and tested in fatigue by clusters load introduction points simulating pressure load.

These tests are scheduled in middle of 2010.

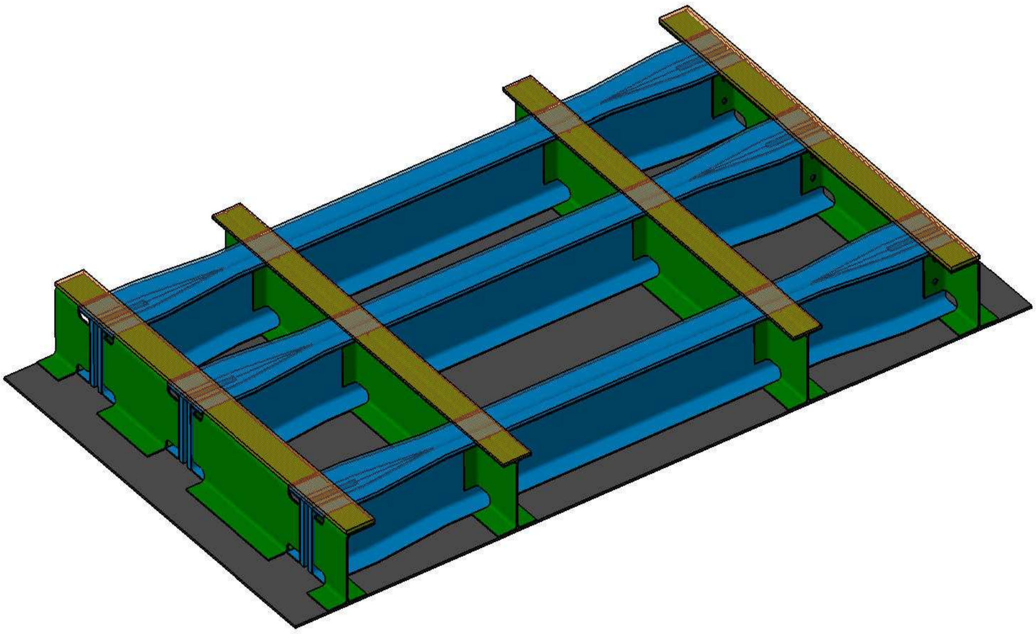


Figure 2 : Panel test coupon

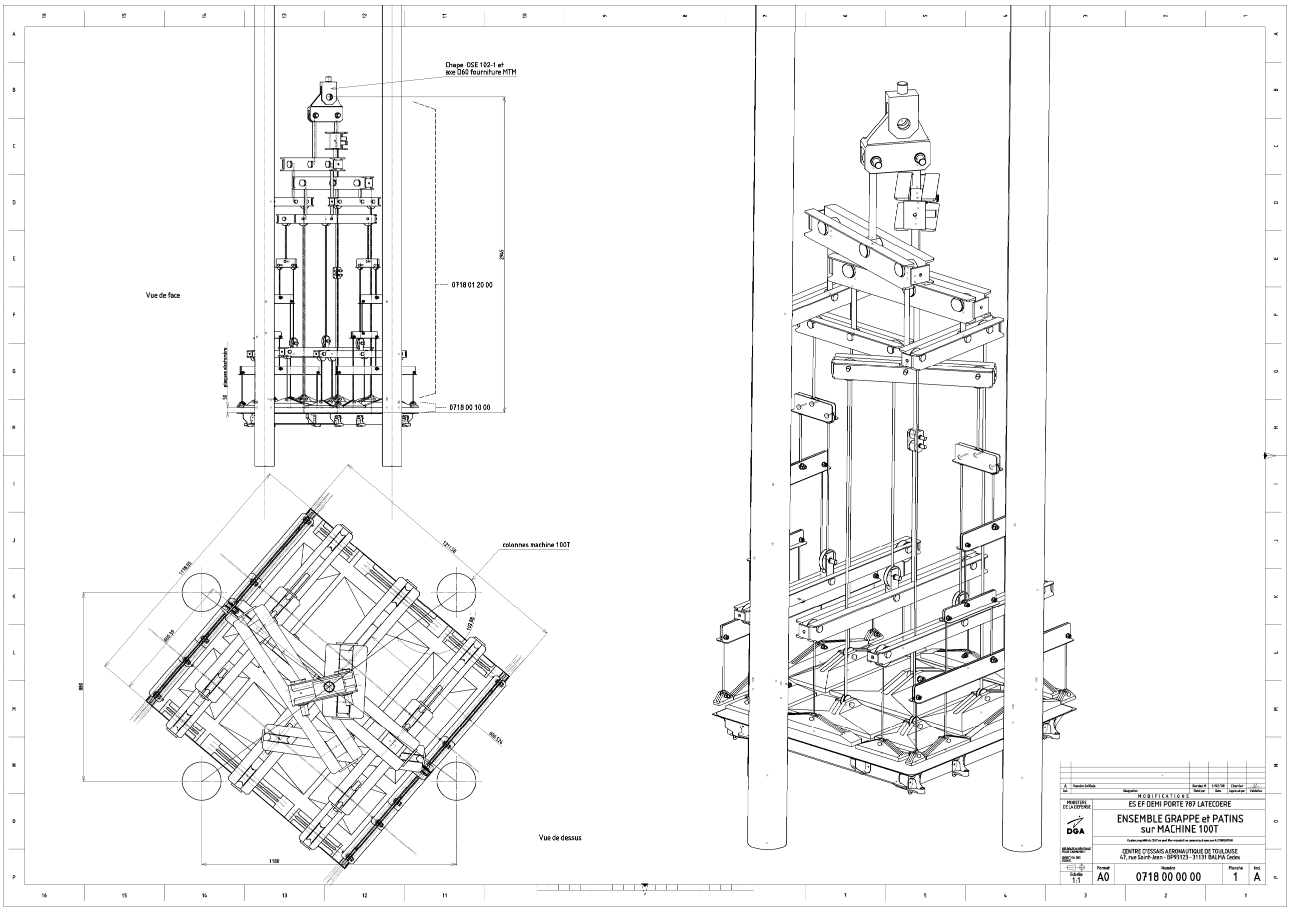


Figure 3 : Test machine for panel

6.5. LIFE EXTENSION & FULL-SCALE FATIGUE TESTS

6.5.1. Alpha Jet Trainer Life Extension Program (Dassault-Aviation and CEAT)

From the origin of the aircraft, the Alpha-Jet fatigue strength has been assessed according to the safe life principles. The structural design was based on a lifetime of 10000 flight hours with a design load spectrum resulting in a fatigue damage used to define the reference value of 100 for the aircraft fatigue index (IF).

A first fatigue full-scale test has been carried out by the IABG test center (1975-76) on an airframe as manufactured up to 540 IF and, considering a scatter factor of 3, an allowable cumulated fatigue damage of 180 IF could be retained for the whole of the airframe.

In the frame of the Alpha-Jet life extension program, a second full-scale fatigue test has been performed by CEAT (1994-2007) using as a test cell the airframe of an aircraft retired from the FAF and implementing loading spectra updated in consistence with the recorded service history of the aircraft.

This test has evidenced a safe life of 325 IF for the fuselage of the aircraft, notably taking advantage of the decrease of the scatter factor allowed by testing of 2 airframes instead of only 1 (from 3 down to 2.74) without prejudice to flight safety.

Unfortunately spare wings had to be used to pursue this second test for the benefit of the fuselage enhanced life substantiation so that the wing safe life could not be extended beyond 180 IF. It has thus been decided to enforce the damage tolerance approach with the aim to extend the service life of the wings up to half of the fatigue damage substantiated by the previous full-scale tests (up to 245 IF) in accordance with the usual rule that for damage tolerant designs of aircraft, a fatigue test must cover the service life by a factor of at least 2.

The damage tolerance evaluation has included the damages, due to fatigue, that have happened during the fatigue tests as well as damages due to corrosion or accidental damages liable to occur in service.

Instructions for inspections have been established for every damage notably based on extents of damage consistent with the residual strength under limit loads or with fuel-tightness for fuel tank boundary elements.

As regards areas poorly solicited on the fatigue test cells, detail inspections are to be carried out on a sample of wings presently in service and commensurately with the lessons learned dedicated additional inspections be prescribed if needs be.

To make the best possible use of this extended life potential, it is also envisaged to proceed to cold working of a number of fastener holes in the course of the scheduled maintenance operations on the wings.

Apart from the fuselage central part and of the wings life potential , the second fatigue test has allowed to increase the safe number of repeated operating events such as:

- 16 400 landings (after enforcement of a scatter factor of 5)
- 7 000 cabin pressure cycles (after enforcement of a scatter factor of 4)
- About 13 000 flight hours for the rear fuselage (after enforcement of a scatter factor of 4), the stabilizer and the vertical fin.

6.5.2. Super-Etendard Navy Aircraft Fatigue Evaluation (Dassault-Aviation and CEAT)

The Super-Etendard structural design is very similar to the Etendard one, the former being an upgraded version of the latter. This explains why fatigue requirements were not addressed in the original specifications of the Super-Etendard aircraft, whereas fatigue was not a concern for combat aircraft when the Etendard aircraft was developed.

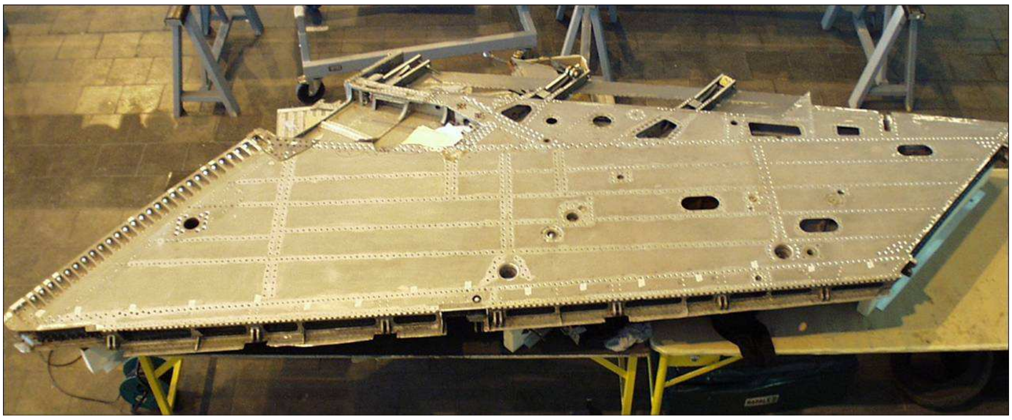
With the aim to perform a fatigue evaluation of the primary structure and also of some preventive reinforcements implemented on the aircraft in service, a fatigue full-scale test has been carried out by CEAT from 1989 up to 2006. 2 different Etendard fuselages and a number of Etendard wings have been successively tested.

For the test loading spectrum that had been derived as the average spectrum expected in service from an OLM campaign on 2 aircraft, this test has substantiated, based on a safe-life approach, a life potential of 6755 flight hours for the fuselage and 6060 flight hours for the wings.

Some additional evaluations have therefore been undertaken to allow for the wings the same life potential as for the fuselages.

The most damaged wing that had experienced the largest number of load cycles in the fatigue test has been entirely dismantled. Apart from the damages having caused the removal of that wing from the test cell, the subsequent thorough inspections didn't reveal any damage liable to impair the structural integrity of the wing.

In parallel, the detail design points as identified by the previous fatigue tests have been submitted to fatigue and damage tolerance analyses, the main outcome being that the wing life potential may be increased to fit to the fuselage one.



6.5.3. M2000 Full Scale Fatigue Test (Dassault-Aviation and CEAT)

Since 3 years, a full-scale fatigue test has been carried out in the CEAT Test Plant on the aged airframe of an aircraft retired from the French Air Force fleet.

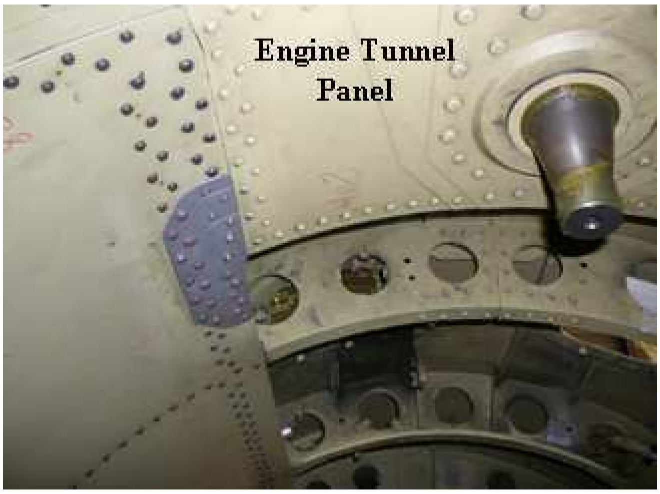
The loading spectrum simulates the French Fleet Fatigue Index (FI) Consumption Rate expected in the future on all the key elements of the structure, with updated in-flight and ground loads.

The objective to demonstrate a significant extension of the Mirage 2000 fatigue life (in a "Safe Life" approach, with punctual damage tolerance) is almost achieved:

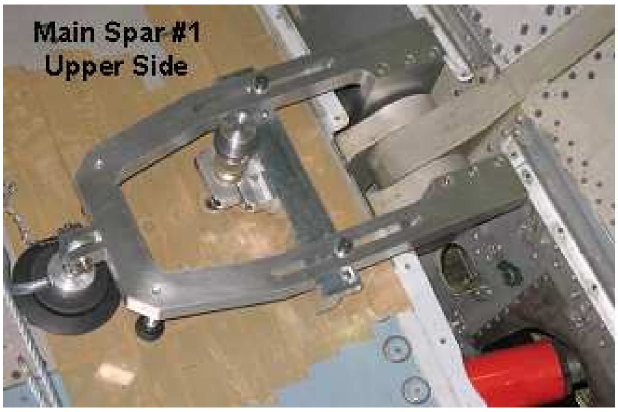
- 240 FI for the fuselage and 230 FI for the wing box have already been substantiated.

None of the fatigue damages that have been disclosed until now have to be rated as critical, in terms of static strength or reparability. Only "light" repairs have been implemented and will be validated through the end of the test.

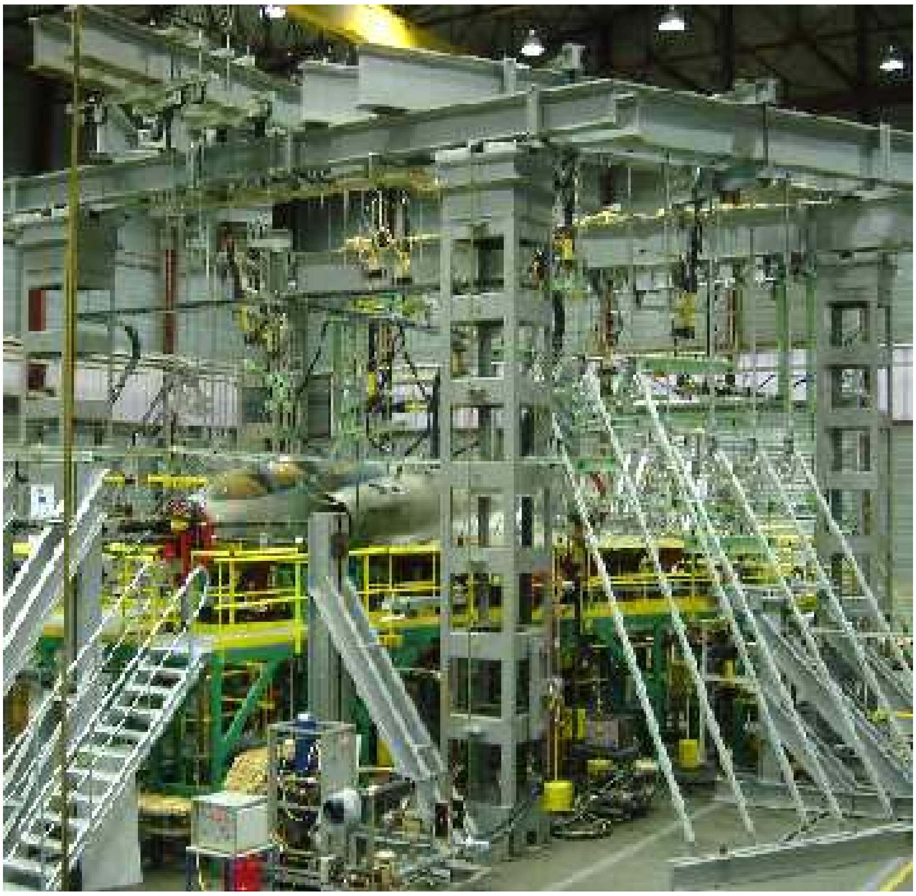
A fuselage repair is presented hereafter.



Wing main spar bore repair operations on this cell are pictured below.



An upgraded inspection and structural maintenance plan of the aircraft with improved cost effectiveness has been issued to the Mirage 2000 operators.



6.5.4. Fatigue and damage tolerance certification tests of the FALCON 7X (Dassault Aviation and CEAT)

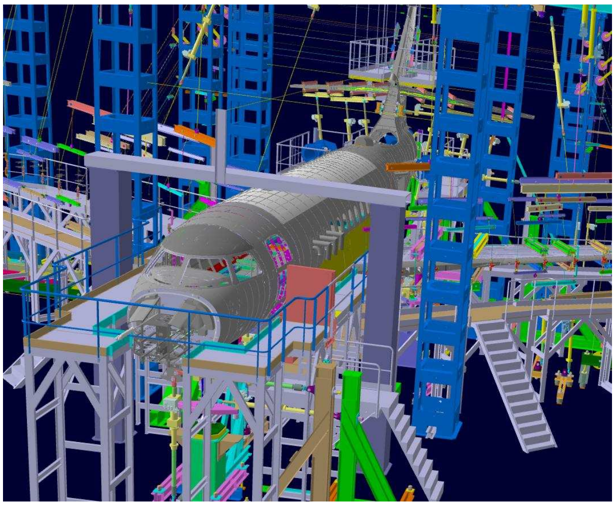
The F7X Mechanical Test Cell (MTC) has been tested at the CEAT facilities in Toulouse. Since December 2004, a preliminary static test (at 0.7 times the limit loads corresponding to the maximum fatigue loads), 2 lives in fatigue , complete static tests including limit and ultimate loads tests and 1 life of damage tolerance followed by residual strength tests have been successfully conducted on one single cell.

In addition, a margin search test has been performed until failure of the wing before disassembly in June 2008.

The MTC was composed of the fuselage, the wing box, the flaps, the composite vertical fin, the engine mounts, the horizontal tail box, dummy engines, dummy landing gear and dummy slat tracks, other structural elements being separately tested on dedicated specimens.

The test rig and loading device comprised 64 hydraulic jacks and 3 pressurization channels (passenger cabin, luggage compartment and fuel tanks).

F7X mechanical test cell



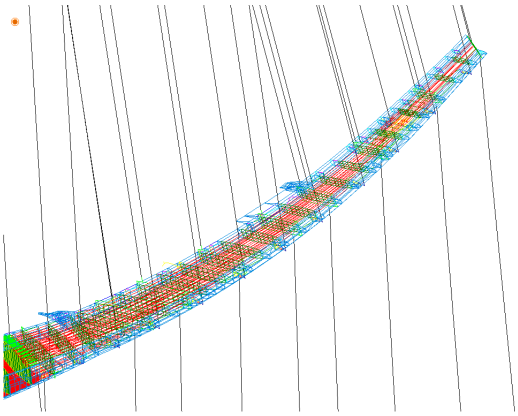
Associated CATIA V5 model

The fatigue loads and the test spectrum have been defined based on fatigue damage calculations of significant structural items as performed on a dedicated FE model of the MTC; for the composite fin, the loads have been enhanced by 13% so that fatigue testing during 2 lives, in presence of low energy impact damages, can substantiate one life as the safe life.

After the fatigue test, about 95 damages have been observed among which only a dozen are serious, some of them leading to structural modifications which were implemented in the definition of the aircraft before type certification and retrofitted to the aircraft already assembled.

Static tests have been successfully performed under calibrated in-flight and ground envelope loads. Eventually, the linear and non-linear numerical models of the F7X structure have been validated showing good correlation levels.

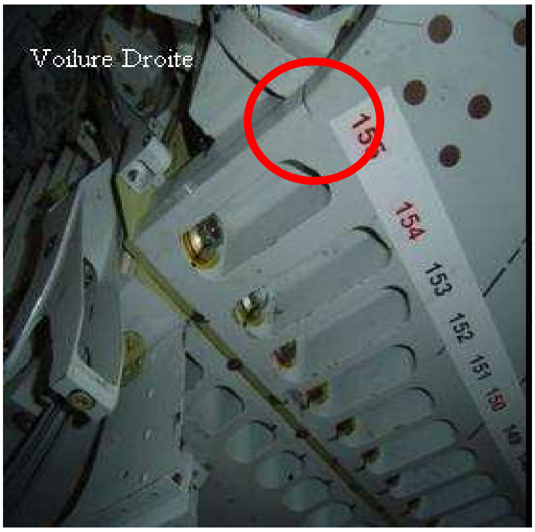
Wing bending at UL



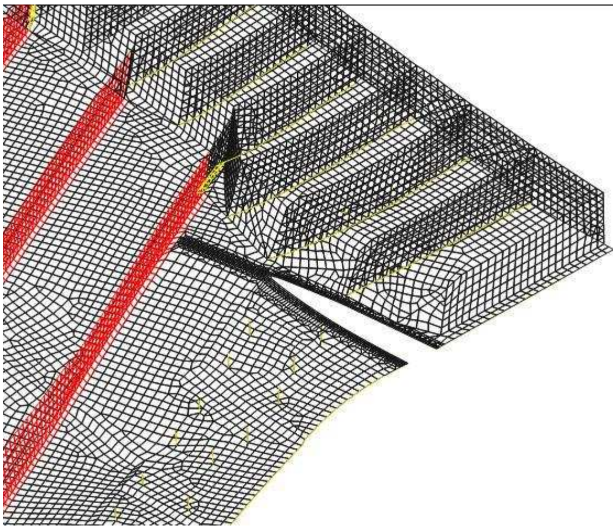
Associated non linear FE computation

70 Artificial damages have then been inserted for damage tolerance testing with a size larger than the plastic zones locally induced by the prior static tests under ultimate loads, so that the crack growth rates are not affected by the residual strains developed about the plastic zones. The damage tolerance tests have been carried out under fatigue calibrated loads, the test spectrum applied, as defined and even as measured, being shown, through crack growth analyses on the most significant structural items, resulting in the same crack growth rates than the genuine load spectrum.

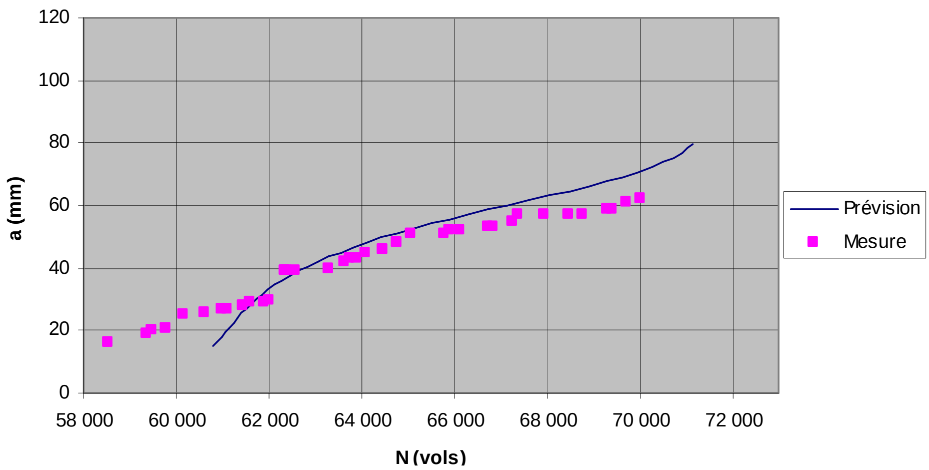
Comparison of measured and computed crack growth rates demonstrates the quality of the damage tolerance computational methodology used throughout the development of the aircraft as shown by the following example of the crack growth analysis of the artificial damage located in the lower panel rear edge aft of the wing root bolt recesses.



Wing root lower panel



Associated FE model



Measured and computed crack growth rate comparison

After completion of one life of damage tolerance test, 108 natural damages have been observed. In addition to the artificial damages inserted, they have successfully passed the residual strength tests.

After structural repair regarding the main damages, the F7X test cell has been submitted to margin search tests: the main static loadings were successfully applied to the test cell up to 1.7 times the limit loads; collapsing of the wing eventually occurred, under upward wing bending loads, at 1.74 times the limit loads.

With Dassault and CEAT mutual work, the ambitious aims of these tests have been reached on time. The MTC observed behaviour and measurements have been shown very close to the predicted ones highlighting a fair accuracy of the Dassault's global sizing methodology as well as the high robustness of the F7X structure.

6.5.5. Transall structure life extension (AIRBUS & CEAT)

Developed in the 60’s in a French-German co-operation, the TRANSALL C160 aircraft is a military transport aircraft. 67 of them are used by the French Air Force for tactical and humanitarian missions, basically. The service life extension for the Transall from 20,000 to 22,500 Flight Cycles, corresponding to 5 more years in service, was decided in 1996 for the following major reasons:

- the advanced fleet age of aircraft in service,
- the estimated date of entry into service of the new generation military aircraft.

The definition and substantiation of the life extension programme have been based on the analysis of both :

- an extensive in-service damage collection with the establishment of a damage data bank,
- a full scale fatigue test on an aircraft retired from service.

The major participants in this process were: A.I.A (Atelier Industriel de l'Armement) from "Clermont-Ferrand", CEAT, SPAé (Service des Programmes Aéronautiques) and AIRBUS.

A big concern was to define the load spectrum to be applied to the test airframe. A usage monitoring campaign was launched in 1996 for this purpose based on:

- collection of general information about each flight of each TRANSALL aircraft (paper form containing the type of mission, flight duration, take-off and landing weights, door openings for droppings…),
- in-flight recording of flight parameters and stresses on 4 aircraft of the fleet, to derive the loads associated with each type of mission.

A large amount of data have been collected over 3 years, exchanged by the different partners of the programme and analysed.

At the same time, the assembly of the test fixture at CEAT was completed. It consists of 112 hydraulic jacks + fuselage pressurisation, and around 600 strain gauges.

After a few last adjustments, the test began in November 1999.

First major damages, which appeared on the lower wing panels around man holes, have already induced a specific maintenance programme for the fleet. Some additional coupon tests on specific details were performed to evaluate the more adequate preventive modification solution. New major damages appeared in October 2002 on the lower wing panels and concerned doublers around fuel pumps (cf ICAF 2003 French National Review). In November 2003, further new major damages (cf ICAF 2005 French National Review) were encountered after 34,000 simulated flights on panels inner surfaces of the centre wing box leading to eleven months of complete stop of cycling. Investigations, including fractographic analysis have been carried out to deduce repercussions for the fleet.

In March 2006, during an inspection at 44,750 flight cycles, new major damages were discovered at rib #1560 especially in the rear spar of the centre wing box leading to stop the test again. After analysis, AIA has designed a repair which has been applied on the structure. Due to the complexity of this repair, test only started again in 2007.

In September 2008, 58,356 simulated flight cycles were completed, delivering 22,500 landings and 27,900 flight hours in the most critical area. A final static test at Limit Load in November 2008 was performed allowing to clear the additional margin taken initially by Airbus rising the final potential to 23,204 flights (landings) and 28,773 flight hours.

The test rig is currently being dismantled and the aircraft is subjected to a tear down for specific areas.