ICAF 2003 Conference – Congress Center KKL, Lucerne, Switzerland - 5-6 May 2003

REVIEW OF AERONAUTICAL FATIGUE INVESTIGATIONS

IN FRANCE DURING THE PERIOD JUNE 2001 - APRIL 2003

Compiled by

Jean Rouchon

DIRECTION DES CENTRES D'EXPERTISE ET D'ESSAIS
Centre d'Essais Aeronautique de Toulouse
31131 BALMA - FRANCE

DGA

<u>CENTRE D'ESSAIS AERONAUTIQUE DE TOULOUSE</u>

DIRECTION DES CENTRES D'EXPERTISE ET D'ESSAIS

INTRODUCTION AND ACKNOWLEDGEMENTS

The present review, prepared for the purpose of the 28th ICAF conference to be held in Lucerne (Switzerland), on 5-6 May 2003, summarises works performed in France in the field of aeronautical fatigue, over the period June 2001 - April 2003.
Topics are arranged from basic investigations up to in-service reporting.
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 :
Mr Yves Nadot for ENSMA,
MM Franck Gallerneau and Alain Déom for ONERA,
Mr Louis Anquez for Dassault Aviation,
Mr Alain Sangerma for Airbus,
Mr Bertrand Journet and Miss Odile Pétillon for EADS, Joint Research Center (CCR),
Mr Florian Sérouart for Messier Dowty,
MM Christophe Simon, Pascal Hamel, and Alexandre Lahousse for CEAT.
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.

TABLE OF CONTENT

6.1 FATIGUE LIFE PREDICTION STUDIES AND FRACTURE MECHANICS

6.1.1 Fatigue from defects (ENSMA Futuroscope)

ENSMA has completed a study to propose a fatigue criterion for metallic materials with defects. Defects considered can be casting defects (shrinkages), corrosion pits, small impacts or inclusions with a size varying from about 50 µm to 1 mm, and with various geometries as shown figure 1.
In order to elaborate the fatigue criterion, fatigue tests have been conducted on two materials and for various multiaxial loadings. The first material is a nodular cast iron containing shrinkage porosity and the second one is a C36 steel containing artificial surface defects with various geometries. Fatigue mechanisms are observed by SEM close to the fatigue limit : for both materials, only a very few non-propagating cracks have been observed and they are very small compared to the size of the defect, most of the samples do not contain cracks at the fatigue limit. It is therefore proposed to define the fatigue limit for such material as the threshold stress that creates a crack around the defect. In order to take into account the 3D stress concentration effect at the tip of the defect, 3D elastic-plastic analysis are conducted for each defect. The Crossland endurance criterion has been applied using the local values of the stress tensor at the tip of the defect : results are very poor and lead to the conclusion that local values at the tip of the defect are not enough to describe size and shape of the defect. It is therefore proposed to take into account the stress gradient (G) at the tip of the defect. The endurance criterion is proposed as follows :

$$\sqrt{J_{2,a}} + a\ J_{1\max}\left(1 - a\,\frac{G}{J_{1\max}}\right) \leq b \qquad \text{with} \qquad G = \sqrt{\left(\frac{\partial J_{1\max}}{\partial x}\right)^2 + \left(\frac{\partial J_{1\max}}{\partial y}\right)^2 + \left(\frac{\partial J_{1\max}}{\partial z}\right)^2}$$

Results are now in very good agreement with experimental results as shown on figure 2. It can be therefore concluded that the stress gradient around the defect is able to describe both size and shape effect of a defect on the fatigue limit of a metallic material. This result is interesting because only one parameter ‘a’ needs to be identified to describe the whole geometry (size and shape) of the defect while the two other parameters ‘α’ and ‘β’ describe the multiaxial stress state.
Contact : Y. Nadot, Laboratoire de Mécanique et de Physique des Matériaux - UMR CNRS n°6617
ENSMA, Téléport 2, BP 40 109, 86961 FUTUROSCOPE cedex, FRANCE
Tel: (+33) 5 49 49 80 43 Fax: (+33) 5 49 49 82 38 e-mail: yves.nadot@lmpm.ensma.fr

6.1.2. Crack growth rate in welded joints (EADS CRC)

The objective is to determine the fatigue growth rates of cracks which are located at the interface between the base metal and the weld seam. The application is for the determination of inspection threshold on welded structures.
The fatigue crack propagation behaviour of welded stiffened coupons (figure 3), made out of 6056, with the loading axis perpendicular to the stringer, was investigated with cracks at the interface between the weld and the base metal. It turns out that the crack growth rate is 1 to 3 times as much as that of the base metal. The crack front is partly in the weld and in the base metal.

6.1.3 Initial flaws (EADS CRC)

The objective is to assess the influence of manufacturing flaws or service damage on the fatigue life of alloys used for structural parts.
As regards applications to helicopters, the investigation was carried out on titanium alloy Ti-10-2-3, with scratches (0.1, 0.2 and 0.4 mm deep) and impacts (0.125 and 0.5 mm deep). The series of fatigue tests (see figure 4) showed that the 0.2/0.4 mm deep scratch flaws were the most detrimental to the fatigue performance of the alloy (knock down factor of 2, for both, on the fatigue limit).
As regards applications to aircraft, studies were continued on the influence of rogue flaws for various types of joints and pocketed coupons. For joints, the large detrimental effect of incorrect clamping was demonstrated and quantified (see figure 5 for example). Fractographic analysis are in progress to verify the initiation mechanisms.

6.1.4 Modelling stable tearing in fatigue crack growth (CEAT in co-operation with AMRL)

A research programme on the modelling of stable tearing in fatigue crack growth is in progress at AMRL (Australia) and CEAT (France).
Fracture surfaces sometimes display large bands of tearing, as shown figure 6 on a CCT100 aluminium alloy 7050 specimen with a 9 mm thickness, which are caused by rapid, but stable, growth of a crack under application of a single high load and, today, no fatigue crack growth model is properly taking into account this phenomenon. To improve aircraft structural integrity and safety, this programme is aimed at understanding the conditions under which tear bands appear, and to develop a prediction capability for this phenomenon under operational conditions.
The research programme involves two Work Packages (WP):

- WP1: a study of stable tearing under controlled laboratory conditions. Analyses of tear bands with the empirical tearing models of Forsyth and Schivje and the R-curve concept.
- WP2: Modelling of stable tearing using incremental crack growth/FE models, currently in progress at AMRL.

WP1 results and lessons:
Numerous stable tearing have been produced on aluminium alloy 7050 CCT100 specimens with thickness variations: 3.6 and 9 mm.
The experimental results have shown that the tearing models of Forsyth and Schijve give a good prediction of the stress intensity (K_{final}) which causes the stable tearing, these results confirm previous AMRL's results obtained on aluminium alloy 7050 CT specimens. The R-curve concept has also been used for the prediction of K_{final}, with better results.
Although these tearing models are only valid for the post fracture analysis, the model of Schivje and the R-curve concept have been experimented, without success, as prediction models for predicting of the crack tearing propagation Δa.
An analysis of the influence of the precracking and loading conditions (ΔK, ΔK_{eff}, loading speed, displacement/load control…) have showed the difficulty to point out the main parameters involved in the tearing phenomenon. The scatter of the experimental results may be the main cause.
On these specimens, an analysis has showed that the fatigue life was not significantly reduced by the presence of stable tearing.

6.2 COMPUTATIONAL TECHNIQUES

6.2.1 Metallic riveted repair calculation (Airbus)

In order to achieve a significant reduction of the maintenance and operating costs of civil transport aircraft – keeping the standards of safety and including repercussion of new trends in airworthiness regulation – improvements of existing repairs for primary metallic structures are required. These improvements can be achieved through the development of new calculation methods and the use of improved repair processes.
Regarding riveted repairs, AIRBUS has therefore decided to create new methods/tools to calculate static, fatigue and damage tolerance issues.
It appears that a set of parametric FE models could provide accurate results and a well-suited answer to the numerous repair configurations, which are included in Structure Repair Manual's or justified through Repair Approval Sheet. On the other hand the integration of this model in an ergonomic graphic user interface as SAFE, which is the Fatigue and Damage tolerance tool developed within AIRBUS, could facilitate the use of complex FE solutions.
This solution has required to develop original FE models (figure 7) combining shells, volumes and crack tip elements to deal at the same time with static (pinloading calculation), fatigue (stress calculation) and propagation (SIF calculation) aspects and to avoid prohibitive CPU solving time. These models take into account biaxial loading and stress distribution including secondary bending (geometrical non-linearity).
In a first step, skin repairs models have been created in order to be compared to fatigue test results for the validation of modelling hypothesis and methods. Then more complex configurations, including stiffeners, have been processed. Finally, the crack propagation assessment has been added to models and allows AIRBUS to contribute to the improvement of numerical predictions for repairs.

6.2.2 Influence of thermal effects on fatigue and damage tolerance (Airbus)

Due to a mix of materials with different thermal expansion coefficients, and to temperature gradients, stresses due to thermal effects are applying to aircraft structures. This phenomenon is of particular importance on the new aircraft programmes, with an extensive use of composite structures.
In this field, studies have been performed to improve the way to take them into account both in fatigue analysis and in tests. Stresses due to thermal effects are taken into account in Finite Element Analysis, and then in fatigue spectrum calculation. Full-scale experiments for thermal effects have been carried out as part of the A340-600 fatigue test. A lot of work will be performed in the next few years in order to improve the way in which thermal effects are taken into account in fatigue analyses. Which conditions representative of the day-to-day usage have to be considered at sizing phase, which deviant conditions have to be taken into account in parametric studies, how to include thermal stresses systematically in fatigue stress spectra, how to improve test demonstration are some key issues that will be developed.

6.2.3 Fatigue crack growth on riveted joints (EADS CRC)

The objective is to validate an approach to predict the fatigue crack growth rates in the presence of residual stresses induced by interference fitting. The application is to assess the damage tolerance of fastener holes in joints, with cold expansion and interference fitting.
The method which was developed for the damage tolerance of cold worked holes, and was presented at the ICAF 2001 Symposium [1], has been extended to interference fit holes. The method proceeds with finite element calculations to simulate the fastening process, in order to set up the residual stress field, then derives the stress intensity factor when

the service load is combined with the residual stress field. Then the prediction of the crack growth rate is made using the baseline crack growth behaviour of the material.

The method was applied and validated on holes with 1% interference fit pin, on 2024 and 7075 Al alloys [2]. Main results are presented in the following. Figure 8 shows a comparison between the experimental data and the predictions on 2024 aluminum alloy, 2 mm thick. The crack is a corner flaw at the beginning and turns into a through crack when the surface crack length is 5mm. The crack aspect ratio a/c goes from 1 (at the beginning) to 0.4 (5 mm surface length) and the predictions were carried out with different values of a/c. The prediction agrees fairly well with the experimental data.

Crack retardation mastercurves were then derived, as shown on figure 9 as a retardation ratio between the CGR with residual stress over CGR without residual stress vs surface crack length. The minimum crack growth rate occurs within the first millimetre of crack growth, with a retardation ratio of 1/30. This type of mastercurve enables the design office to apply it directly using its crack growth calculation tools, without the need for modelling the retardation effect.

Prospective retardation mastercurves were calculated on cases representative of joints, namely 4% cold expanded holes with or without 1% interference fitting, load transfer coefficients ranging from 5% to 50% and several service loads. On 7075 alloy, 3.17 mm thick, the minimum retardation ratio is 1/20, within the first millimetre, and the figure 10 shows that cold expansion by itself is more efficient.

[1] B. Journet, F. Congourdeau, F. Vinhas, C. Ithurralde, E. Lucien, D. Duprat , Damage tolerance of cold worked holes". ICAF'2001, Design for Durability in the Digital Age, proceedings of the 21st symposium of the International Conference on Aeronautical Fatigue, vol. I, Toulouse, France, 27-29 june 2001.

[2] B. Journet, F. Congourdeau, F. C. Ithurralde, Prévision de la fissuration par fatigue des alésages de jonctions rivetées. Colloque MECAMAT, Aussois, France, 21-24 janvier 2003.

6.2.4 Residual strength of welded stiffened panels (EADS CRC)

The objective is to predict the residual strength of welded stiffened panels. The usual methods based on R-curve are not conservative. EADS CCR has developed an approach which deals with a local approach of fracture, namely Rice & Tracey criterion (refer to the national review of ICAF 2001). This is an elastoplastic finite element analysis, which implements the local failure criterion ahead of the crack tip in order to simulate the crack advance under the applied load. This method does not suffer from any limitations such as those met by the usual methods which are conducted within the frame of linear elastic fracture mechanics. Furthermore, this approach accounts for the cracking of the stiffeners when the main crack passes by these latter ones.

This approach has been applied to new cases of welded stiffened panels designed by Airbus France. The results are the purpose of a paper presented at the symposium of ICAF 2003. Figure 11 shows a result of crack growth simulation in the case of a flat panel with 7 welded stringers, and an initial centre crack, the length of which is two-stiffener pitches. The predicted failure (maximum) load is within 5% of the test result.

6.2.5 Tools integration (EADS CRC)

EADS CCR research work is valued in an in-house BEM code named CRACK-KIT®. This tools is displayed on EADS CCR internet site www.eads.net/crack-kit or crack-kit@eads.net . This tools is based on BEM to make fatigue crack growth calculations. The crack growth models are integrated (for instance PREFFAS model) and the presented research area is computational mechanics.

The programme was improved for efficiency, achieving a reduction factor of CPU time ranging from 10 to 100. To achieve such an improvement, several programming techniques were used, among which a tailored Gauss inversion. New functionalities were added to the code, such as the capability to account for initial stresses or the effect of stiffeners. On the figure 12, the normalised stress intensity factor for a crack symmetrical with respect to a single broken stiffener is plotted as a function of the crack length to stiffener area ratio (see [1] for the reference case). The current implementation is accurate for moderately stiffened panels (i.e. for cracks large enough with respect to the stiffener area). A modified bolt model is being implemented to extend the capability to highly stiffened structures.

Reference : N. K. Salgado, M. H. Aliabadi, The application of the dual boundary element method to the analysis of cracked stiffened panels, Eng. Frac. Mech., vol. 54, n°1, pp. 91-105, 1996.

6.3 EXPERIMENTAL TECHNIQUES

6.3.1 Development of an Axial-Torsional Thermo-Mechanical Fatigue Test Device (ONERA)

Structural components, such as Aircraft Gas Turbine blades, working at high temperature under cyclic conditions, are the seat of many complex mechanical phenomena. The strong requirements in design procedure and the superalloys evolution (polycrystals to single crystal i.e. isotropic to anisotropic materials) have involved the development at ONERA of sophisticated lifetime prediction models, together with complex experimental multiaxial test devices for the validation, taking account the cyclic viscoplastic behavior of anisotropic materials as well as crack initiation under creep, oxidation and fatigue conditions.

ONERA perfected in his laboratory an axial-torsional thermo-mechanical fatigue test device able to generate specified complex thermal and multiaxial mechanical loading to a coated single crystal superalloy specimen to be as representative as possible to the real blade in service.
The principle of a thermo-mechanical test is to control the mechanical strain, that is to subtract the thermal strain from the total one. The specific test procedure includes the thermal cycling, the thermal compensation and the realisation of the fatigue test with a mechanical strain control. Two techniques (potential drop technique and internal pressure technique) are simultaneously applied to detect a crack initiation in order, first to deduce the life duration corresponding to the initiation of a macroscopic crack in the structure, and second to stop the test before the complete specimen breaking to avoid the deterioration of the extensometer rods. Figure 13 shows an example of the material response which has been obtained, in term of stress, under a realistic thermo-mechanical fatigue cycle in tension, in comparison with the simulation.
To simulate the response of the anisotropic material due to a combined thermal and mechanical cycle, a 3D finite element simulation is required as soon as a multiaxial loading is generated in the specimen.
Figure 14 shows the mesh of the problem with four elements in the specimen wall. The difficulty of such calculation concerns the application of specific boundary conditions to obtain locally, in the gage length, the strain imposed experimentally by the axial-torsional extensometer. There is a good correlation between the stress response of the material recorded during the test and the computation results performed in cyclic viscoplasticity reached to stabilisation. The fatigue life prediction is then given by a creep-fatigue-oxidation interaction model, developed for coated single crystal superalloy, applied as a post-treatment to the finite element analysis.
Contacts : Vincent BONNAND, Franck GALLERNEAU, Didier PACOU, Daniel POIRIER.

6.3.2 Investigation on R curve precision (CEAT)

For ductile fractures, a commonly used design criteria is based on the R-curve concept. The experience accumulated by design offices, over the past several years on this concept, pointed out the difficulty to extrapolate with success the results obtained on samples of small dimensions to large structures, in spite of the independence of geometry stated by one of the Krafft's hypotheses. This problem could be attributed to a limitation of the concept, but also to an imprecision of the R-curve.
The evaluation of the R-curve precision is a hard problem due to the difficulty to analyse the results with statistical methods (a curve is a series of quantities derived from a single sample) and the complexity of testing and processing methods which need to be well controlled to guarantee a low scattering of the results. Being involved in R-curve tests, the CEAT carried out a study to evaluate the influence of some testing and processing parameters on the R-curve determined and to optimise its methods. The main testing parameters evaluated were, the displacement rate, the number and duration of loading steps, the width of the specimen and the type of anti-buckling device. This study, based on 21 CCT200 and 4 CCT700 samples made of 2024T3 material, mainly showed :

- A great variability of the results, due to the compliance method used to calculate the effective crack size. The graph figure 15 shows the influence, on the R-curve, of the slope obtained on the linear part of the curve displacement vs load.
- The necessity to advantage, for given material and thickness, R-curves issued from a large specimen.
- The necessity to use a well-chosen equation type to fit experimental data and to mention the validity domain, notion which tends to vanish with the use of the fitting equation.
- A low influence of the test parameters evaluated.

6.3.3 Statistical reduction of endurance fatigue test data (CEAT)

The major issue when dealing with fatigue initiation tests is the scattering of the results and thus the way to perform a statistic processing of experimental data. CEAT has developed a new processing method based on an iterative Monte-Carlo simulation. The method takes into account the final status of samples after testing (broken or not) when determining the endurance curve and supplies confidence curves around the mean one and a B-value curve, corresponding to a maximal 10% failure probability with a confidence of 95%.
The statistic variable is supposed to be the applied maximum stress for a given lifetime. The statistic curves supplied are adaptive, as the confidence and B-value take into account the number of test results obtained within the considered lifetime. It has also been demonstrated that the method based on an uniform translation of the endurance mean curve by a value calculated from the tabulated B-value coefficients of the normal law is not conservative (see figure 16).

6.4 MATERIALS AND TECHNOLOGY TESTING

6.4.1 Influence of temperature on the residual strength (Dassault Aviation)

Tom Swift pointed out in "Verification of methods for damage tolerance evaluation of aircraft structures to FAA requirements". (12th ICAF Toulouse 1983) that low temperatures expected in service can considerably reduce the residual strength of cracked structures (particularly in the case of 7000 series aluminium alloys and some 2000 series).

This particular point was further investigated in D.A. Laboratories on four materials, one of the 2000 series and three of the 7000 series.
The results obtained are presented in table 1 where the apparent stress intensity factors are given, material by material, measured at 20°C and at - 50°C.
As it can be seen, Kapp (-50°C) appears higher than Kapp (20°C) in the case of the 2024 T351 while it is lower in the case of the 7475 and of the 7050.
But the most interesting information is provided by the results obtained for the age formed 3-7050. It appears that the ratio of Kapp (-50°C) to Kapp (20°C) varies with the size of the specimen for a given thickness.
Equal to 0.87 in the case of a CCT 100 x 300, the ratio falls to 0.82 in the case of a CCT 300 x 900.
Considering that the evolution of the ratio can be linked to the crack length, in a first approximation, we have then a value of 0.87 when a = 14 mm and 0.82 when a = 40 mm.
It is then highly probable that in the real case (long crack in a structure), the reduction of residual stress can be significantly lower than 0.82.
It is therefore necessary to take account of this phenomenon all along the damage tolerance demonstration for any new structure.

Material	Specimen	Kapp 20°C (MPa√m)	Kapp - 50°C (MPa√m)	$\frac{\text{Kapp}(-50°\text{C})}{\text{Kapp}(20°\text{C})}$
2024 T351	CT 40B12	58.9	62.5	1.06
age formed 7475	CT 40B12	65.8	57.9	0.88
age formed 1-7050	CT 20B10 T1	32.8	28.2	0.87
age formed 2-7050	CT 20B10 T2	40.7	36.8	0.90
age formed 3-7050	CT 10B10	27.2	26	0.96
age formed 3-7050	CT 20B10	30.5	28.5	0.93
age formed 3-7050	CT 40B10	40.2	36.4	0.91
age formed 3-7050	CCT 100x300x5	53.4	46,6	0.87
age formed 3-7050	CCT 300x900x5	65.3	53.6	0.82

Table 1.

6.4.2 Fatigue behaviour of fuselage panels with welded stiffeners (AIRBUS)

With the goal of saving manufacturing cost, the welded technology for fuselage panels is very interesting for aircraft designers. This is why the knowledge and the prediction of structural behaviour of welded stiffeners becomes a major preoccupation.
The development of this technology for fuselage induces new sizing methods taking into account the specific behaviour of integral structures under pressurisation loads. Particularly, the local fatigue of the weld line (and weld line run-out) in a complex stress field must be studied finely.
The purpose of this study is to present a numerical methodology to predict the fatigue behaviour of such structures.
Most of literature fatigue prediction methods for integral structures use 3D representations. However, for an industrial usage, these methods can hardly be employed because the computation time is very high and the shape of the weld line is highly process conditions dependant (welding parameters, thickness etc.).
This is why AIRBUS developed a numerical approach based on shell modelling: the shape of the weld line is not represented. Obviously, because of this shell modelling, the stress concentration factor value cannot be used directly: it needs a preliminary calibration of the model by test results on basic coupons.
The validation development phase had two main goals (that will be addressed in a paper presented at the symposium):
- To demonstrate the effectiveness of such an approach for fatigue life predictions.
- To put in stress the specific behaviour of integral (welded) stiffeners under complex loads in order to give recommendations for designers.

The numerical methodology has been assessed through correlation to small scale test specimens and fuselage panels. The effectiveness of this method depends very closely on the preliminary calibration. Particularly, it has been demonstrated that it was very important to calibrate the model with coupons that have exactly the same weld line shape than the panel to predict.
As for integral structures, the run-out of the weld line must be observed carefully and has been extensively studied. Because of the local bending at the stiffener arrest, the fatigue performance is lower for uniaxial longitudinal loads (stiffener direction) in comparison with transverse uniaxial loads. But, this performance can be significantly increased using a coupling (which design is very important).
For typical fuselage applications, the influence of the stress field (biaxiality) should be particularly investigated. Biaxility

is mainly induced by pressure loads. It has been successfully predicted and validated that the fatigue performance of the panel may be significantly lower than in uniaxial loading conditions. Indeed, the pressure load induces a skin pillowing effect and a folding / unfolding phenomenon along the stiffener. As a consequence, specific design precautions such as the protection of the weld line by an over thickness need to be taken into account when using integral (welded) technologies in pressurised areas.
It has been finally concluded that this test correlated numerical approach can be used with confidence to predict the fatigue life of integral (welded) panels for fuselage application.
In addition, to its easy implementation, the calibration by tests on basic coupons restricts errors due to pure numerical study

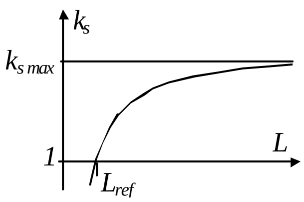
6.4.3 Proposition of a bounded law for scale effects in fatigue (Messier Dowty)

The so-called scale effect in fatigue corresponds to the experimental observation that, at a given stress, the fatigue strength of a component decreases when its size increases, as shown figure 17. This effect has extensively been demonstrated by tests, and is widely represented by a power law of the following form :

$k_s = \left(\frac{L}{L_{ref}} \right)^a$, where L and L_{ref} are representative length or surfaces of the component and specimen respectively.

However, it is obvious to notice that this law is not bounded. This means that k_s can rise to infinity when L increases. This somewhat contradicts the physic, and the observation that the fatigue strength of a very long bar for example would never be reduced to zero.
In fact, we can easily imagine that any part will fail on the weakest "defect" included within the surface of highest stress (strain), following a logical weakest link law. In particularly, the larger the surface, the higher the probability to find the most critical defect (at a given stress) leading to the shortest life. On the contrary, the smaller the surface, the lower the probability to get this defect, and this results in the specimen having a longer average life. However, if the surface continues to be increased, the most critical defect will, at a time, simply appear twice, and the part will fail on one or the other, but still at the same number of cycles.
A logical consequence of this is that testing small specimens will exhibit more scatter than bigger ones (figure 18), but the minimum life (at a given stress) or the minimum stress (at a given life) will simply be the same whatever the specimen size, corresponding to the most critical defect at a given stress or life. The scale effect does represent the reduction of the mean S/N curve (50/50), but it has to be bounded so that the reduced curve does not fall below the minimum SN curve (dashed). This latter can reasonably be taken as the -3σ one (99/95), provided that a sufficient number of specimens are tested.
The proposition above can be summarised in the following formula :

$$k_s = k_{s\max} + (1 - k_{s\max}).\exp\left(1 - \left(\frac{L}{L_{ref}} \right)^a \right)$$



L_{ref} is the same as the previous formula and depends on the reference specimen. $k_{s\,max}$ can be calculated as $\sigma/\sigma_{-3\sigma}$ at any suitable life (1E5 for example for aircraft structures) and depends on the material, including possible effect like chemical treatment, plating or grain flow (as they can potentially modify the standard deviation of data).
It is to be noted that, according to the proposition above, there would be no need to take scale effect into account if the 99/95 SN curves is conservatively used for fatigue analysis, in particularly for safe life design.

6.5 NON DESTRUCTIVE TESTING

6.5.1 Detection of corrosion by three imaging techniques (ONERA)

The detection of corrosion in aluminium alloy aircraft structures is made difficult by the complex and varied nature of the structures to be inspected. Any inspection system must be easy to use in the field, for example, in a maintenance hangar. The techniques commonly used (X-ray radiography, Eddy current and Ultrasonic inspection) are still undergoing development in an attempt to improve their performance. They are all local methods and therefore require long inspection times.
The DMSE section at ONERA has investigated the performance and limitations of three other corrosion detection techniques, presently little used in France, or not at all. These potentially rapid inspection processes are all based on image formation: pulsed infrared thermography, laser ultrasonics and magneto-optical imaging.

Pulsed infrared thermography consists in monitoring temperature changes in the region of interest after illumination with flash lamps. First of all, a theoretical study involving finite difference simulations revealed the potential and limits of the method in the case of generic defects (flat-bottomed holes). Experiments were then performed on a wide range of specimen types, including real corroded aircraft components. Figure 19 shows the detection of circular corrosion zones, obtained by chemical attack on the rear side of a Falcon 900 under-surface panel. Corrosion damage as shallow as 1/10 mm, (relative thickness variation of 5%) could be detected.
In order to study the feasibility of magneto-optical imaging, which combines eddy currents with the Faraday effect, a commercial apparatus was employed forming 50 x 50 mm images of the structure to be analysed. An image processing software was developed to optimize the detection of corroded zones (figure 20). Work is currently in progress to improve the sensitivity and enable observations in the second sheet with sufficient resolution.
The so-called laser ultrasonic inspection technique (generation of thermoelastic vibrations due to the impact of laser pulses and detection with an interferometer probe) is already used by EADS. The originality of the present approach lies on the production of Lamb waves, characterized by a large distance or propagation without attenuation and by a remarkable sensitivity to variations in thickness. A laboratory system has been built enabling the examination of 1 m long panels and the analysis of Lamb modes using wavelet transforms. The detection of incipient corrosion is presently based on analysis of the disappearance of the S1 mode (figure 21). The possibility of using other modes is currently being explored.

Contact : A. Déom, J.-C. Krapez, M. Lemistre, C. Titry and F. Lepoutre
ONERA 29, Avenue de la Division Leclerc
F92320 Châtillon France
Phone 33.1.01.46.73.48.76 fax 33.1.01.46.73.48.91 email : deom@onera.fr

6.5.2 Benefit from NDE techniques for test (EADS CRC)

The increasing value of structural testing led EADS Corporate Research Centre to implement non-destructive testing (NDT) techniques in order to monitor structures during mechanical tests. Indeed available techniques allow to have more data at one's disposal and thus to draw more profit from individual test results. Three main fields for using NDT techniques have been identified and implemented from the sample scale to the full structure : damage detection, damage sizing, full-field strain and stress measurement.
The information and value provided by a large range of these techniques have been now clearly assessed at EADS CCR. Ultrasonic waves suit the detection of cracks or delaminations, Eddy current are sensitive to cracks or fibre rupture while acoustic emission or infrared thermography have a potential for detecting microdamages. Optical methods have known a significant growth for the last years. Main uses in NDT concern the detection of debonding or delamination damages. However, most techniques enable to achieve a quantitative measurement. Among available techniques, projection Moiré or fringe projection enable to characterise out-of-plane deformation. Other methods based equally on triangulation enable to get in-plane deformation, and possibly in complement out-of-plane deformation. Photogrammetry but preferably image correlation techniques offer this potential. Moreover, acoustic emission is used on a regular basis to ensure the safeguarding of high value structures.
Specific systems based on ultrasonic or electro-magnetic techniques have been developed in order to provide maps of the damage at regular intervals during mechanical tests. One very first application aimed at better understanding the fatigue behaviour of longitudinal joints. Indeed the respective influences of initiation and propagation phases on fatigue resistance are not well known. A better understanding would help in selecting the most favourable properties of joints to improve the life. An in-situ system based on ultrasounds has been developed to record the crack profile for all joint holes during a fatigue test. The analysis and exploitation of the records are much easier than it would be using a destructive approach and cycle counting. The profile of the crack for rivet n°10 is shown in the figure 22. The results present a good correlation with destructive examination.
Reference : O. Pétillon and D. Simonet, “Non destructive investigation tools value creation for structural test monitoring”. ICAS2002 CONGRESS

6.5.3 Monitoring techniques for maintenance (EADS CRC)

6.5.3.1 New NDT tools

New tools, based on the use of multi-elements sensors, are currently under evaluation for maintenance application. Either Eddy current or ultrasounds, multi-elements, allow more sophisticated implementation. It is possible to continuously control the ultrasonic wave field in a component, in depth and under an angle of incidence, with a linear electronic scan at the same time. Studies are under progress to improve the evaluation of lap joints, using eddy current multi-elements, and to carry out NDT of welded stringers using ultrasonics multi-elements.
Reference : H. Voillaume & G. Ithurralde, « Utilisation des réseaux multi-éléments pour le contrôle ultrasonore de pièces métalliques et composites ». EADS CCR report, 2002-28025/F-DCR/SE, June 2002, Convention CEAT N°00/22212/00/227/31/48 (MOD contract).

6.5.3.2 New NDT concepts

In order to face growing economical constraints to reduce operating costs, Structural Health Monitoring concepts appear to be a source of maintenance cost savings for the following reasons:
- permanent in situ-NDT could significantly reduce costs and time devoted to manual NDT control during scheduled maintenance,
- detection of damages at an early stage could limit repair complexity and consequently repair costs,
- time reduction for maintenance operations could increase commercial availability.

Nevertheless, market success for such a new technology will exist only if the price for a complete implementation of the system does not cancel expected maintenance operations cost savings. Consequently, research effort shall be focussed on the reduction of the more costly scheduled and non-scheduled maintenance operations. Three main critical damage types are identified as the main sources of maintenance costs:
- fatigue cracks on metallic riveted junctions (mainly for existing aircraft fleet),
- corrosion damages,
- impact damages on thin CFRP composites (significant increase of CFRP composites expected for the next decade).

For each critical damage, maturity of various candidate technologies has been assessed (Bragg fibers network, corrosion sensor, acousto-ultrasonic methods, Integrated Acoustic Emission sensors network,…). Trade-off between expected cost savings and necessary research developments was discussed in order to identify the most promising technologies for industrialization.
For instance, EADS Corporate Research Center is investigating the potential of optical fibers for corrosion detection. This type of sensors has already been tested in the US by Boeing, and is now also developed in France by Ecole Centrale de Lyon (ref. [2]) under MOD contract. The sensors are based on the following principle: a multimode optical fiber is unclad along a given portion of its length. A specific metallic layer is applied on this sensitive region, directly onto the fiber core. The transmitted light intensity is measured. The numerical aperture of the fiber has very different features in the case of the uncorroded and corroded sensors. The corrosion sensor can be used in transmission or reflection mode.
Figure 23 shows an experimental result for which an accelerated chemical corrosion attack was used. In this particular case, a copper deposit was made on the fiber core, and the fiber was immersed into an acid solution. The index of corrosion level used here is the inflection angle of the transmitted light intensity vs angle of incidence curve.
One can observe that this particular index is very well correlated to the corrosion state of the sensor. Micrographic pictures are shown at the various stages of acid attack. The first one shows the fiber in its initial uncorroded state, the second exhibits a moderate corrosion pit. In the third and last picture, the metallic coating is strongly damaged.
As for other optical fiber sensors, this corrosion sensor benefits from an array of advantages : long distance detection, no sensitivity to electromagnetic perturbations, low cost, no electrical power is needed for the sensor to operate (passive sensors). In addition, the nature of the metallic coating can be tailored to the specific environment of interest. Further tests are still necessary to assess reproducibility of such promising sensors before first integration attempt on structures.
References : [1] D. Guedra-Degeorges, J. Saniger, JP. Dupuis and B. Petitjean, « Emerging Health monitoring technologies : a route towards a flexible maintenance of aerospace structure".
[2] S. Abderrahamne, A. Himour, R. Kherrat, E. Chailleux, N. Jaffrezic-Renault, G. Stremsdoerfer, "An optical fibre corrosion sensor with an electroless deposit of Ni-P", Sensors and Actuators B 75, pp. 1-4, 2001.

6.6 RULEMAKING SUPPORTING ACTIVITIES

6.6.1 Equivalent flaw size (EADS CRC)

This chapter deals with the application of future certification principles. The presented research area is the EIFS concept. It has to be noted that the above reported study on initial flaws on titanium alloy will serve as input for the flaw tolerant safe life approach of helicopters. New certification rules (see AC 25-571 1c and AC 29-2c for large transport aircraft and rotorcraft respectively) have prompted a significant number of studies on the Equivalent Initial Flaw Size (EIFS) methodology. It is quickly obvious that such a methodology leads to non intrinsic quality flaw (QF) sizes, i.e. the flaw sizes depend largely upon the geometry, stress and load ratio. In this context, a reasonable approach consists in back-calculating the QF size from S-N curves of representative details. This approach was applied, in collaboration with Airbus France, to the calculation of the inspection threshold of repairs. The study demonstrated a strong sensitivity of the threshold to the S-N curve chosen for the back-calculation. Specifically, it was demonstrated that the use of upper bound values for a family of similar configurations may lead to largely conservative estimates of the threshold, and that the probability of failure attached to the S-N curves used for back-extrapolation is equivalent to a safety factor.

6.7 FULL-SCALE FATIGUE TESTS

6.7.1 Completion of the Gazelle helicopter fatigue test (EUROCOPTER & CEAT)

The gazelle SA341 is a combat light helicopter produced by Eurocopter for observation, linking and anti-tank missions. This helicopter currently equips the French Army and, with machines built under license by Westland, the United Kingdom Forces.
The structure tests of this helicopter were performed in CEAT in the period 1974-78 with, in particular, a full scale fatigue test that could release a 7,500 flight hours (FH) safe life time, extended by analysis to 10,000 FH later on. Taking into account the remaining potential of their own fleets, French and UK Forces need to prolong the utilisation up to 15,000 FH. For this purpose, a second fatigue test, co-funded (25%) by the UK Forces, has been performed in CEAT (figure 24).
The test article, service aged and removed from the fleet, is made of a tail boom coming from the French Forces (5,440 FH accumulated) and a fuselage coming from the UK ones (7,500 FH accumulated).
The year 1996 was dedicated to design and build up the test facility while settling the airframe in the test desired configuration.
The fatigue test started in December 1996.
Twenty seven load channels are used to simulate the three phases of the test programme :
- unarmed (smooth) version : training and linking (28,125 FH),
- armed version without VIVIANE sight : day-time tactical flight (3,750 FH),
- armed version with VIVIANE sight : day and night-time flight (5,625 FH).
Data from 16 strain gages and 12 bridges are continuously acquired during the test.
The test was achieved in September 2002.
Fifty two damages were detected on the test article, most of them had been already found in service or during the first fatigue test. The most important ones are localized as follows :
- canopy welded frame,
- bulkhead at 2,571.5 FH,
- tailboom and fuselage junction bores,
- quarter of hull skin,
- left longitudinal beam.
Some identified damages were repaired using definite solutions already applied in service, new damages needed the design of specific repairs which have be validated during the test.
Before dismantling the test rig, it is necessary to check if the operational flight missions, used to elaborate the fatigue spectrum, have not been modified since the beginning of the test. For an other part, the global objective of 15,000 FH and the number of flight hours in each version need to be confirmed. Calculation rules have been proposed by Eurocopter to convert flight hours simulated in one version into flight hours corresponding to another version.

6.7.2 - Alphajet fatigue test (Dassault Aviation & CEAT)

In service in the French Air Force (FAF) since 1979, the Alphajet training and aerobatics aircraft has been the subject, since the end of 1994, of a life extension programme with the aim to define the actions to be taken such as to continue both economical and safe operations beyond the original service life. The resulting extended life should allow to fulfil the FAF's needs beyond 2001 and up to 2016 with 150 aircraft operating about 38,000 flight hours per year.
This programme mainly relies on a durability test (figure 25) currently performed on the airframe of the first aircraft having reached the original safe life limit of 180 Fatigue Index (FI) which was substantiated at the beginning of the serial production through an initial fatigue test carried out up to 540 FI (equivalent to 54,000 hours of the expected usage). Up to now, in addition to the 180 FI experienced in service, the test airframe has accumulated 430 FI.
Should the scatter factors usually considered for French military aircraft to promulgate service lives from single full scale fatigue tests be applied, the test progress should hardly be sufficient to release a significant life extension. Nevertheless, probabilistic considerations have permitted to make allowance for both present and initial full scale test results and to reduce the value of the scatter factor to be used while keeping unchanged the provided safety level. Like this an increased safe life limit of 220 FI could be adopted.
In addition the present test has been shown very useful :
- to determine growth rates for cracks revealed by periodic inspections on the test cell and also, in some cases, on the aircraft in service,
- to develop and/or adapt appropriate NDI methods,
- to develop repair solutions ranging from interim repair to more definitive ones, themselves either preventive or curative, and to assess the fatigue potential of the repaired structural parts.

6.7.3 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, 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 program will be 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 are: 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 to that end, 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 has 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 (figure 26).
After a few last adjustments, the test began in November 1999. Up to now, 26,355 Flight Cycles have been simulated in addition to the Flight Cycles experienced in service.
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 (figure 27). Analysis of this problem, including fractographic investigations (figure 28), is in progress to deduce eventual repercussions for the fleet.
Tear-down inspections are expected at the end of the test to complement the fatigue test and in-service damage data, in order to establish the life extension conditions and the updated maintenance program necessary to operate the fleet beyond 20,000 flights.

6.7.4 Fatigue Tests of the Rafale landing gears (Messier Dowty & CEAT)

The first Rafale M are already in service in the French Navy and the Rafale B and C will be soon in operation in the French Airforce. After assessment of structural resistance by full-scale ground testing of an airframe, specific fatigue tests are being performed on the landing gears (static tests are completed).
Four different landing gears are tested : the main and the nose landing gears of the Rafale M and the main and the nose ones of the Rafale B. For each one, a specific load spectrum, including: taking off, landing, breaking, turns, and, for the Rafale M, catapult launches and landings on aircraft carrier, are applied.
To certify, for the Rafale B, 4,200 Flight Cycles (corresponding to 7,000 Flight Hours), 21,000 Flight Cycles are going to be simulated. These figures for the Rafale M are respectively 4,900; 7,000 and 24,500.
NDT inspections are performed at given intervals, and will lead to subsequent analysis if needed.
Up to now, 15,000 Flight Cycles have been simulated on the landing gears of the Rafale B and 4,400 FC on the landing gears of the Rafale M without any damage.

6.7.5 Mirage 2000 life extension (Dassault Aviation & CEAT)

Mirage 2000 is an aircraft operated by the French Airforce for different kinds of mission (air battle, bombings, nuclear deterrent power…). Each aircraft is equipped with a g-counter (Mirage 2000 B, C, N, -5F) or "MICROSPEES" device (Mirage 2000 D) to follow its fatigue consumption.
The analysis of the damaging speed compared with the use scheduled by the French Air Force pointed out the need to extend the life of this aircraft.
Several actions are being conducted with various schedules.
For one part, the load calculations have been re-evaluated with new and more accurate aerodynamic numerical codes developed for the Rafale. The results are being supported by in-flight tests.
For an other part, a new and more extensive inspection program is going to be developed to extend the life of the aircraft considering damage tolerance analysis.
Last but not least, a new fatigue test is going to be conducted to identify new potential critical points in fatigue, to support the damage tolerance analysis and to validate repair solutions. The test will be performed on an aircraft retired from service. One of the main difficulties consists in taking into account the different uses and versions of the aircraft with a single full scale test. The beginning of the flight simulation on the aircraft is scheduled in June 2005.
Life extension of the landing gear (taking into account the quite important increasing of the weight of this aircraft) is conducted with specific fatigue tests. These tests are run to substantiate the increase of the number of landings from 4,200 (already proved for the initially specified load level) to 9,000. Dassault Aviation used all flight measurements available in order to update the stress levels to be applied during testing. These data are parts of the experiments run on

the ground : rolling, braking, overloading during take off, touchdown, pivot, etc. The first cycles have been applied on the nose landing gear in February 2003.
As part of the life extension programme of the Mirage 2000, a subcomponent test is running to approve the repairing of a spar attachment by bush setting. This test should confirm the feasibility and determine the following : the expanding bush parameters, the corrosion (galvanic and stress corrosion), the strength and the resistance to fatigue initiation with bush. Before bush setting, the velocity of crack propagation in the spar attachment lug is measured, in order to take into account the stress induced by die forging.
Along with this test, CEAT also performs a numerical analysis of fretting initiation in the junction lug of the wing spar under the FEM code ANSYS and a crack propagation analysis under complex spectrum with the analytical code NASGRO (figure 29). This action will help in determining the inspection schedule for maintenance purpose. Furthermore, the comparison of calculations with test results over an unrepaired piece will allow determining the influence of residual stresses due to the drop forging process.

6.8 IN-SERVICE AIRCRAFT FATIGUE MONITORING

6.8.1 System general description and data collection (CEAT)

Most French military aircraft and a few ones of the same type operated by German and Italian Forces, are equipped with "fatiguemeters" g-counters. This device is able to count the number of times the vertical acceleration reaches or exceeds some predetermined levels during each flight. A debriefing form has to be filled in for each flight. The values of the counters, the mission code and the takeoff store configuration of the flight are written on the same form. Data collection, analysis and calculations on these individual aircraft recordings are performed by CEAT.
Simple calculation rules are used to provide, at any time during each aircraft lifetime, the cumulative fatigue damage at some points of the structure (pointed out as critical during the full-scale fatigue test for instance).
The cumulative fatigue damage is calculated on the basis of Miner's rule, assuming a representative Kt and an appropriate S-N curve . The relations between stresses and vertical acceleration are fitted to give a damage equal to one for one lifetime in the condition of the full-scale fatigue test.
First calculations were performed in 1974. Today, more than 800 aircraft are monitored by CEAT, representing the processing of more than 150,000 flight hours a year.
The load monitoring is not carried out on a sample of each fleet : each aircraft is equipped with a g-counter. Today the following fleets are monitored :
- Mirage 2000, Mirage F1 and Mirage IVP,
- Alphajet,
- Jaguar,
- Atlantique ATL1 (French, Italian, German),
- Falcon 10 Mer,
- Epsilon,
- Mystère 20.

For those fleets approaching their potential limitations, periodicity of processing and feed back to the Air Forces has been recently increased. This is the case for the Alphajet (every month) and the Mirage 2000 (every other month).

6.8.2 Evolution of the system : MICROSPEES (Dassault Aviation & CEAT)

Early production line Mirage 2000 for French Air Force (FAF) are equipped with the fatiguemeters (g counters) described § 6.8.1, the data of which are mostly representative of the loads supported by the wing root. With the development of new external load inventories, the M. 2000D aircraft, being now delivered to the FAF, are equipped with the "MICROSPEES" device.
This device - 5 analogic channels + 7 binary parameters data (no strain-gauge) - is able to record the history of various flight parameters such as : Mach, altitude, static and dynamic pressures, control deflections, fuel consumption. The analysis of the records sent to CEAT consists in deriving the stress history at different points of the structure and implementing these local stress spectra into models for fatigue crack initiation and propagation :
- fatigue crack initiation prediction is very similar to that performed on the other Mirage 2000 (on the basis of load exceedance records from g-counters) : a fatigue index (FI) is derived from the damage calculated on the basis of Miner's rule assuming an appropriate S-N curve. The only difference lies on the accuracy of the spectrum and the stress history which is taken into account by use of a RAINFLOW process,
- fatigue crack propagation prediction uses a model developed by ONERA : this model is based on the crack closure concept and accounts for stress history.

The first inspection is derived from the crack detection on the durability test airframe, by application of a reduction factor of 3 to the corresponding calculated Fatigue Index.
The crack growth investigation under various spectra and stress level outlines that the above mentioned "Fatigue Index" (FI), is penalizing with regard to the definition of inspection intervals when considering more severe utilization of

aircraft. Thus, the subsequent inspection intervals are defined by a specific "Propagation Index" (PI) value corresponding to the application of a reduction factor of 2 to the calculated time necessary to reach the critical crack size.
The first processing of the MICROSPEES records was carried out in April 2000. Processing are now performed four times a year.

6.8.3 New generation of the system : HARPAGON (Dassault Aviation and CEAT)

HARPAGON is the name of the Health and Usage Monitoring System on the RAFALE aircraft. Part of it (SIESTRA) is dedicated to fatigue damage calculation.
The approach is similar to that applied to the Mirage 2000-D fleet and is described in § 6.8.2. Fatigue and Propagation Indices (FI and PI) are derived from in-flight recorded parameters. The main improvements are the following :
- more parameters recorded (around 30),
- more structural points tracked (around 20),
- day-to-day assessment of the fatigue remaining potential and inspection interval performed by the Air Force and Navy themselves. These values are periodically updated by feed back and comparison to more extensive calculations performed by CEAT.

Up to now, the calculation code has been developed and is going to be validated. In the same time, the data necessary to performe FI and PI calculation for each point are being defined.

6.9 IN-SERVICE FINDINGS AND CONTINUED AIRWORTHINESS

6.9.1 A300 frame 47 forward fitting cracking problem (Airbus)

Some A300 experienced service cracking, in a radius of forward fitting at frame 47. This fitting is carrying loads from Outer Wing Box to Centre Wing Box rear spars (see figure 30). The crack initiated and developed in an area that is mostly under high compression loading. Static jumps were observed in the crack surface during fractographic analysis.
In order to ensure safety, several service actions have to be defined, including inspection programmes, repair solutions and retrofit modification of this structural element. Different means have been developed and used in order to understand the mechanical phenomenon, and then to define repair and modification solutions. Firstly, fractographic analyses of service cracks provided data on crack initiation and growth. Then, strain gauges have been installed on an aircraft, and flight tests have been performed to evaluate the local stresses due to the different flight phases, in the vicinity of the fitting radius. Theoretical analysis of the phenomenon has been done, in order to explain the initiation and growth of this crack in an area mainly loaded by compression. Existing and new coupon test results have been used to support this analysis. Finally, a Finite Element Model has been developed to investigate this problem.
Thanks to these three information sources, the cracking phenomenon seems to be now explained. The initiation and first part of the propagation is due to a residual traction stress field. Then, outside this plastic area, vertical negative gusts lead to low level of traction loading, which are nevertheless sufficient to cause static jumps.
Dealing with the FE model, it has been built to understand the phenomenon, to carry out comparative studies and get to the certification of the repair and modification. The area being difficult to investigate by local FE analysis, a global/local FE model represented a significant part of this Wing/Fuselage attachment area (figure 31). The size of this zone is approximately a two-meter diameter sphere. Shell and beam elements have been used to represent most of the structural items, while volume elements where used for the fitting. Around a third of the 110,000 elements are dedicated to the refined volume modelling and the study of the sensitive area. Moreover fasteners (around 1 thousand) located in the vicinity of the critical radius have been also represented in order to calculate and justify fastener load transfers. At the same time, local stresses in the radius have been investigated for different load cases both with linear (comparative studies) and non-linear analyses (determination of the size of the plastic area). In addition cracks growing in 2 different directions can be represented in this global/local model, in order to predict horizontal and vertical crack growth according to these various load cases. These extensive FE analyses have been successfully used to support the design of efficient repair and modification solutions in this area.

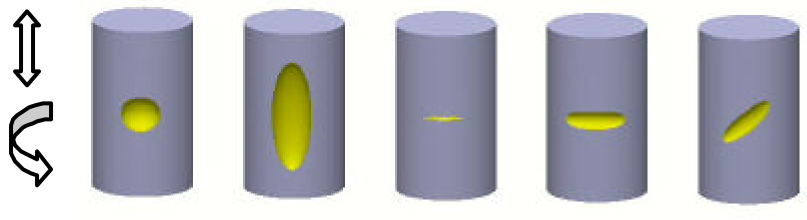


Figure 1 : Investigation carried out on *various loadings* : tension (R = -1 and 0.1), torsion (R = -1 and -0.4), combined tension-torsion and *various defects* : size from 50 to 1000 µm, geometry : spherical defect, elliptical horizontal, vertical or 45° tilted, natural defects (shrinkage)

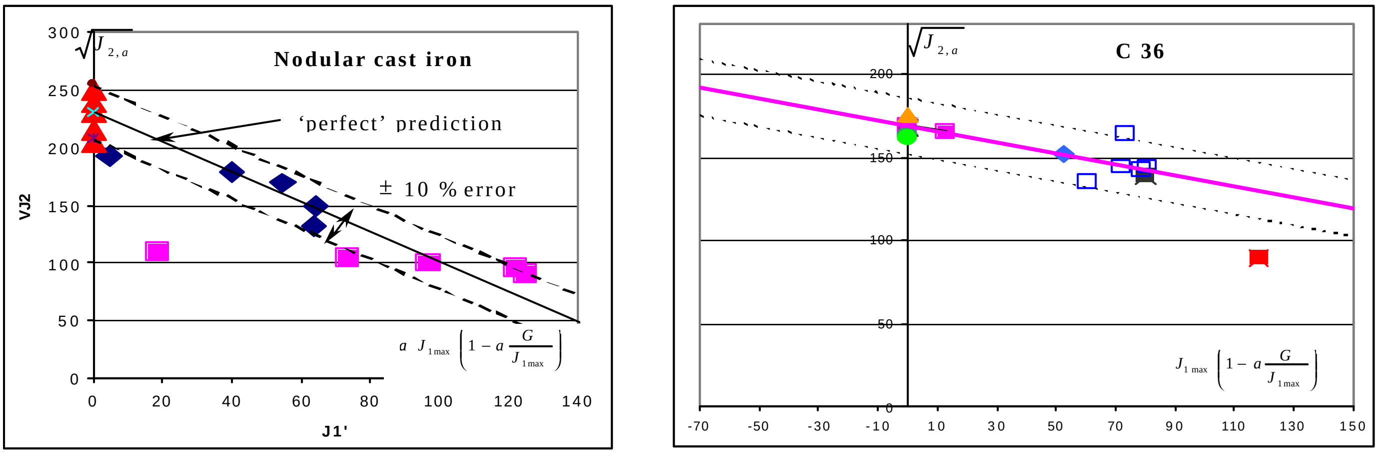


Figure 2 : Test results on a cast iron and C36 steel

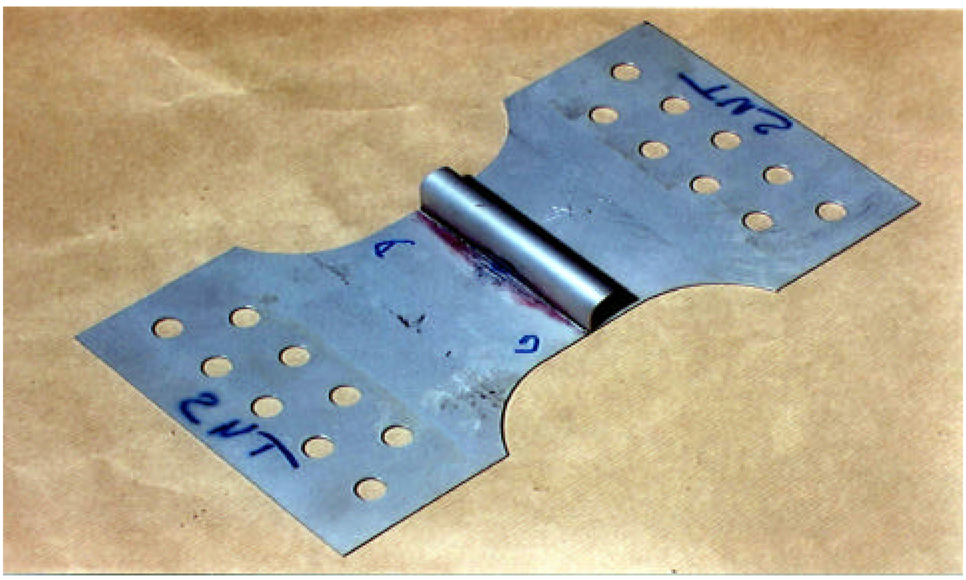


Figure 3 : Specimen used to investigate the fatigue crack growth in welded panels.

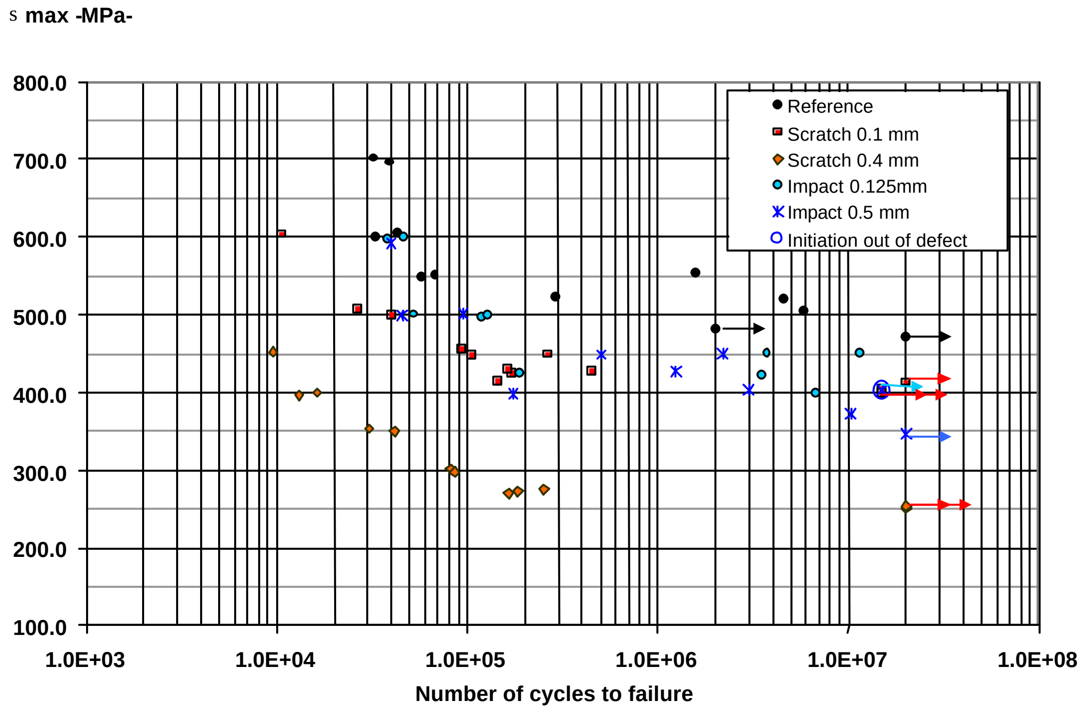


Figure 4 : Initial flaw influence on the fatigue performance of Ti-10-2-3

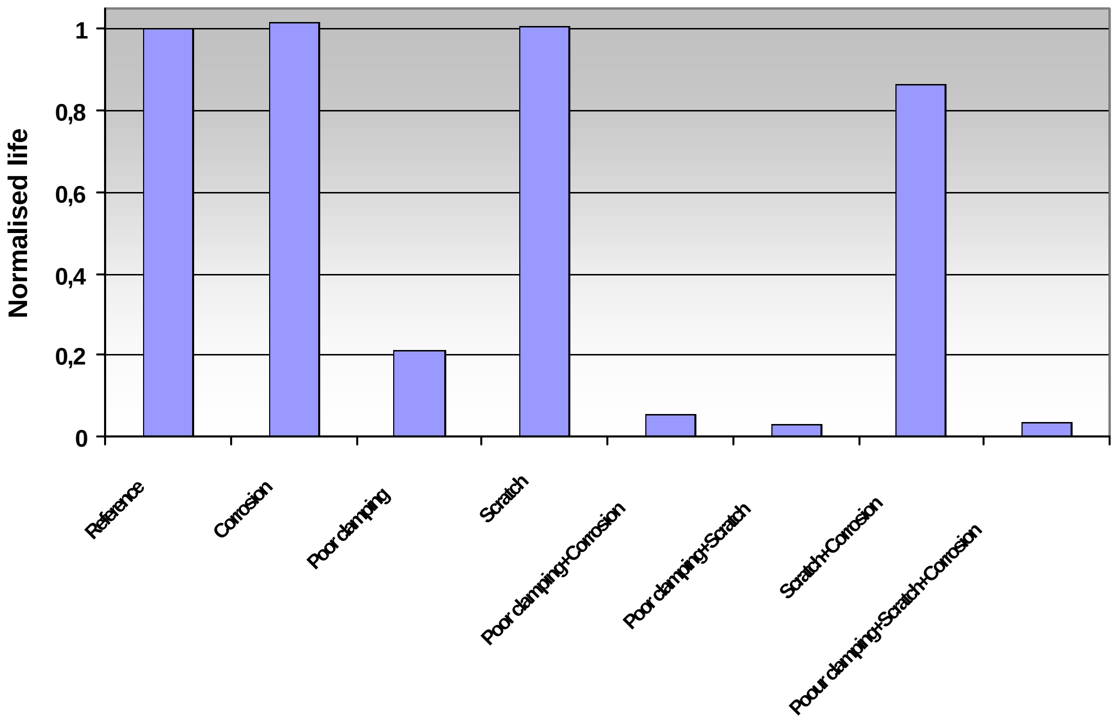


Figure 5 : Detrimental effects of poor clamping

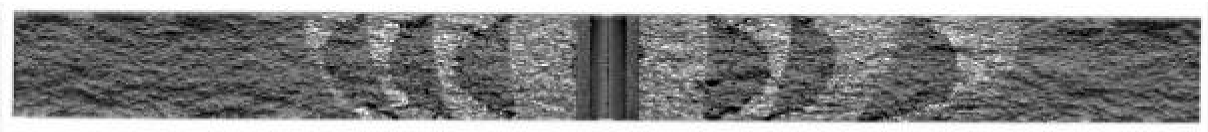


Figure 6 : Stable tearing on a CCT100 aluminium alloy 7050 specimen with a 9 mm of thickness

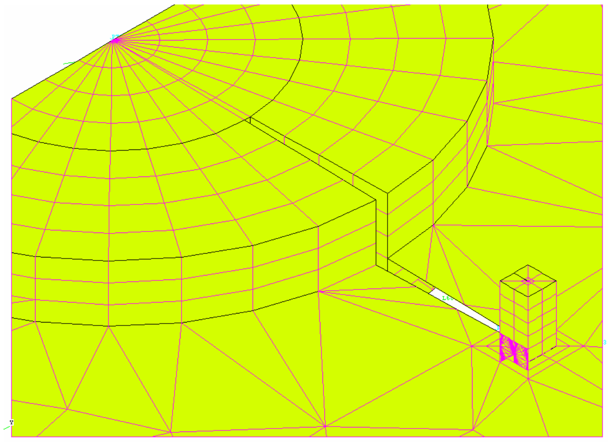


Figure 7 : Detail of a repair model in a FE model mixing shells, volumes and crack tip elements at a cracked bore

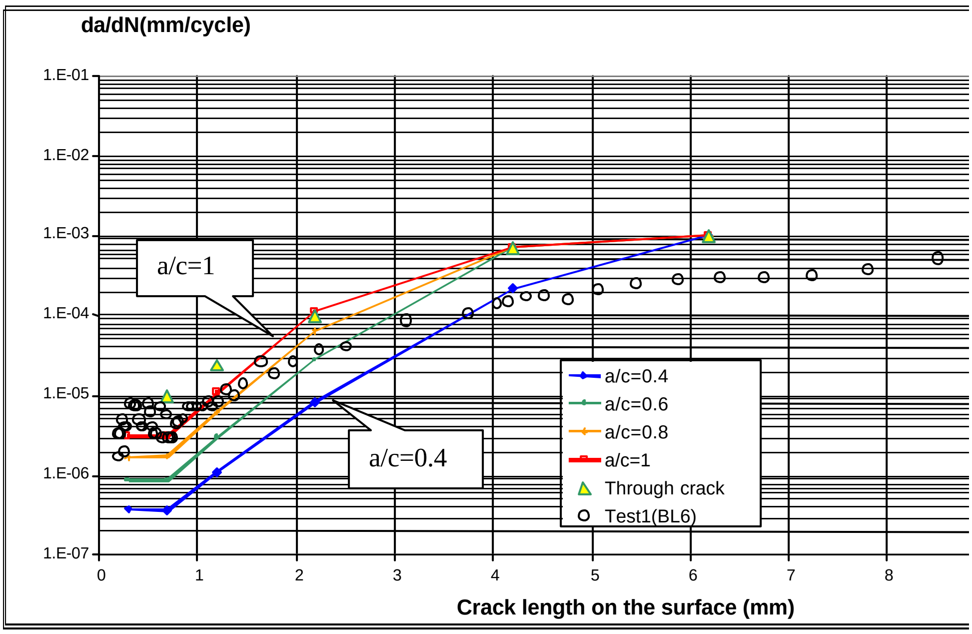


Figure 8 : 2024 interference fit hole – Crack growth rate vs surface crack length, test data a predictions

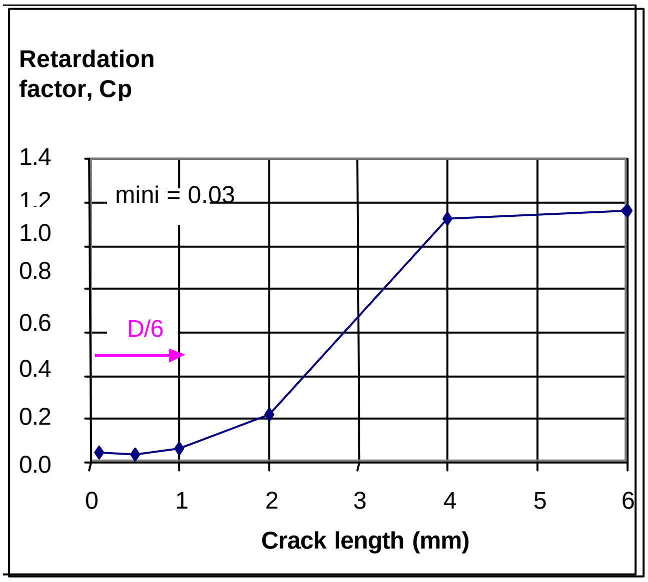


Figure 9 : Predicted retardation mastercurve on 2024 (2mm-150 Mpa) interference fit hole. Retardation ratio vs surface crack length

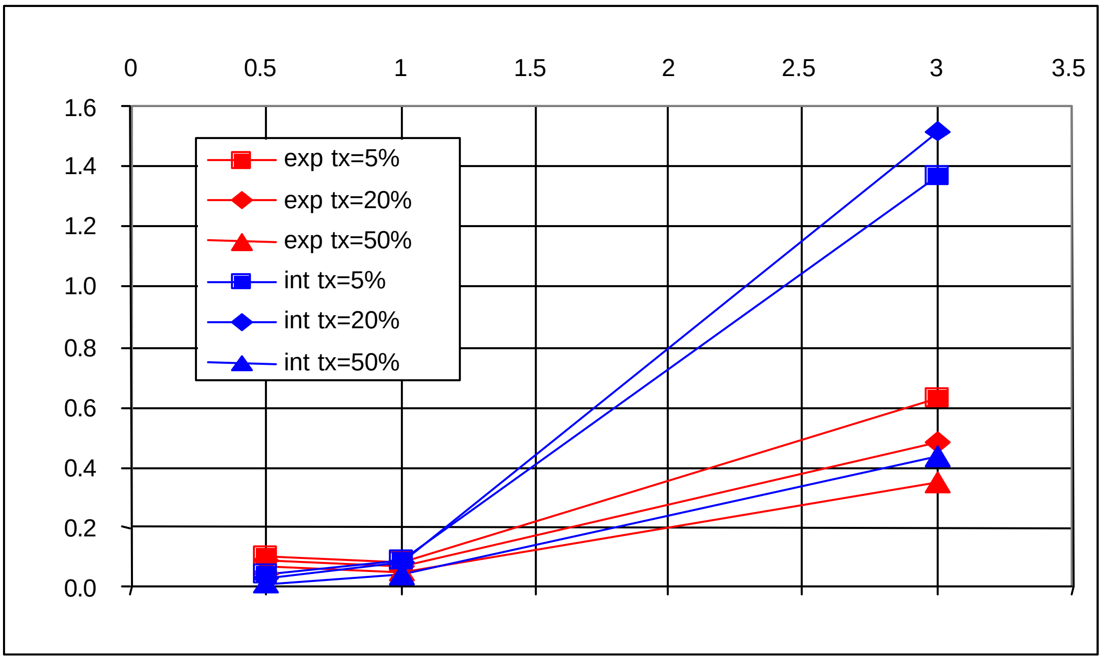


Figure 10 : Prospective calculations on 7075, cold worked hole with / without interference fitting. Load transfer coefficients of 5, 20, 50%. Retardation ratios vs surface crack length

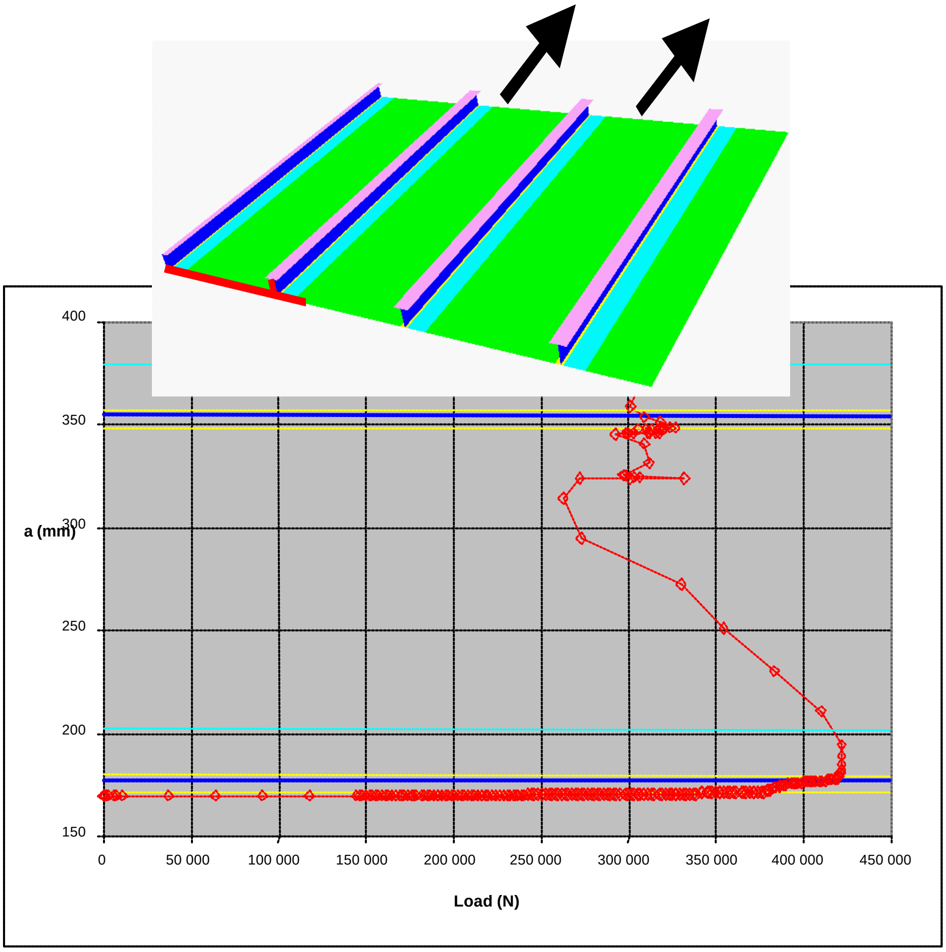


Figure 11 : Residual strength prediction

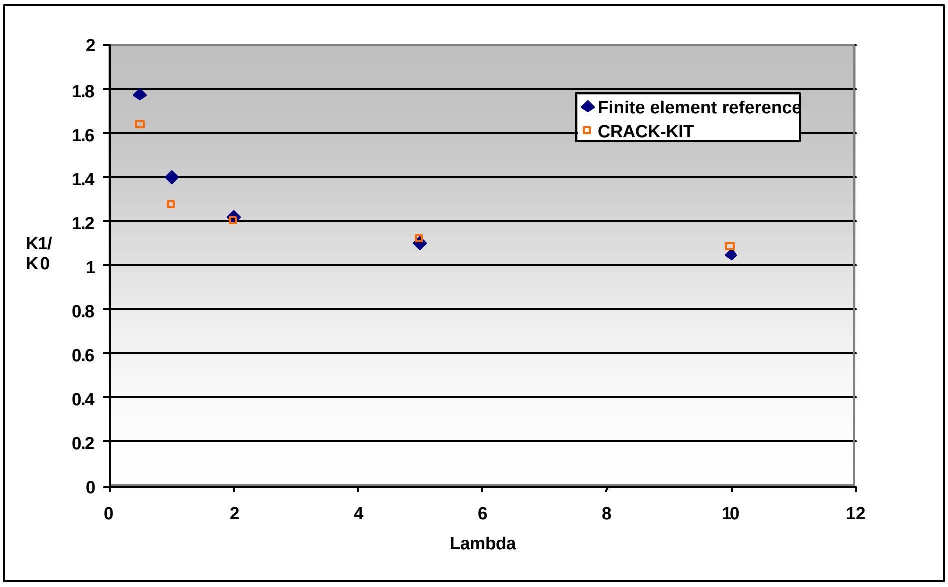


Figure 12 : Example of normalised SIF for a crack symmetric about a broken stiffener, plotted as a function of the crack size to stiffener area ratio

Two materials : nodular cast iron and C36 carbon steel.

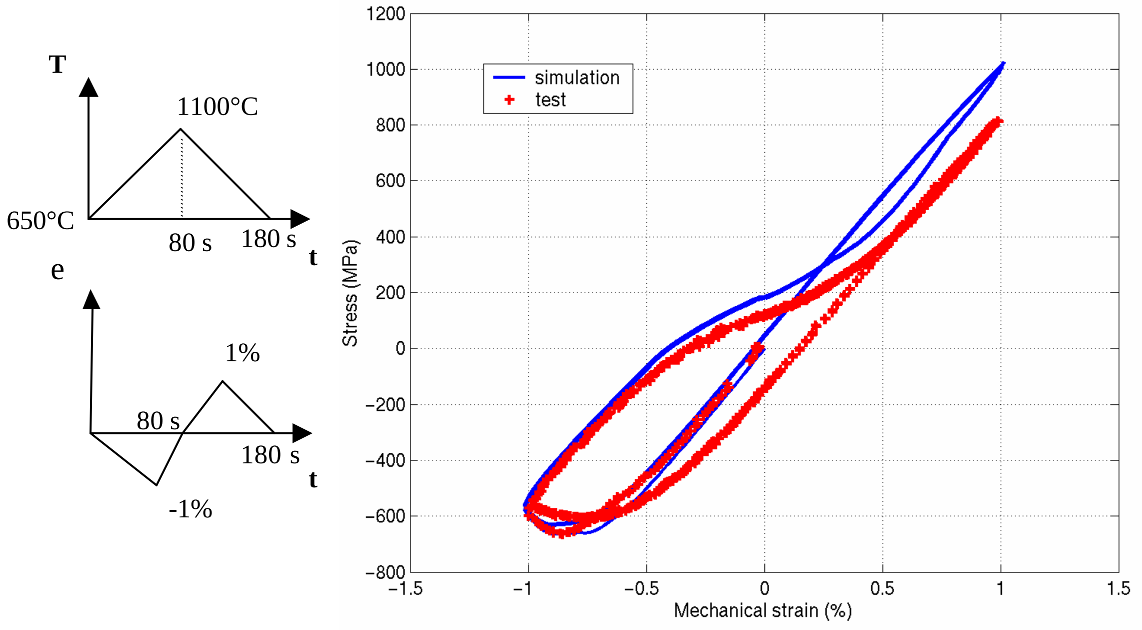


Figure 13 : Prediction for a tension test of the material stress response at the stabilised cycle

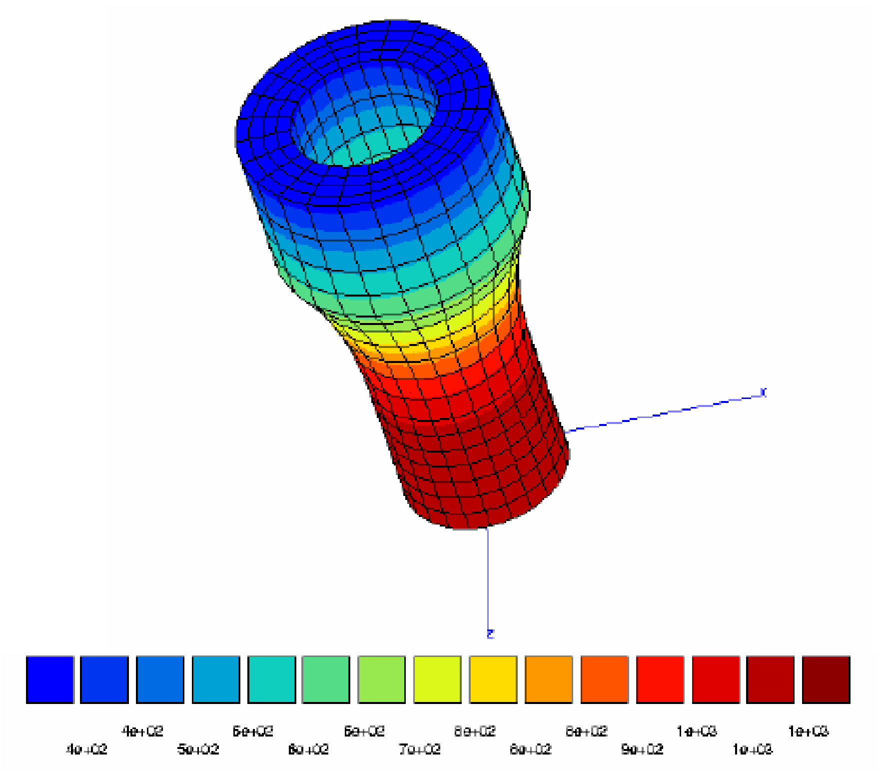


Figure 14 : Isovalues of the temperature at the maximum of the thermal cycle

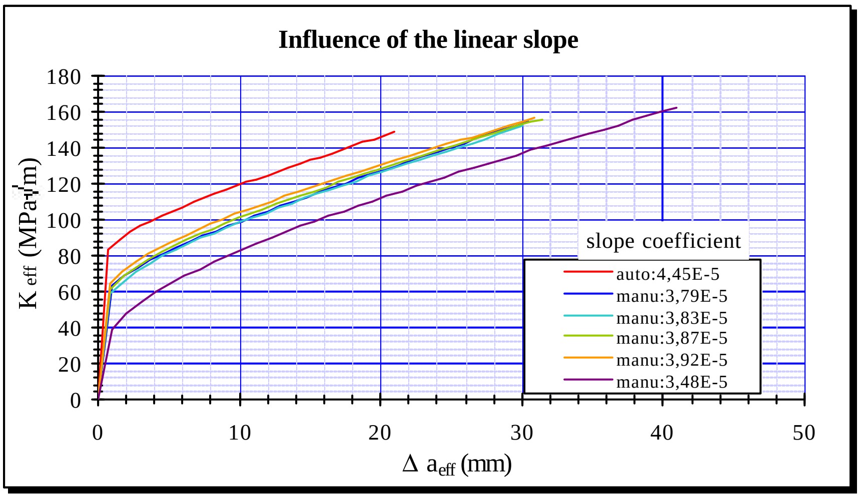


Figure 15 : Influence, on the R-curve, of the slope obtained
on the linear part of the curve displacement vs load

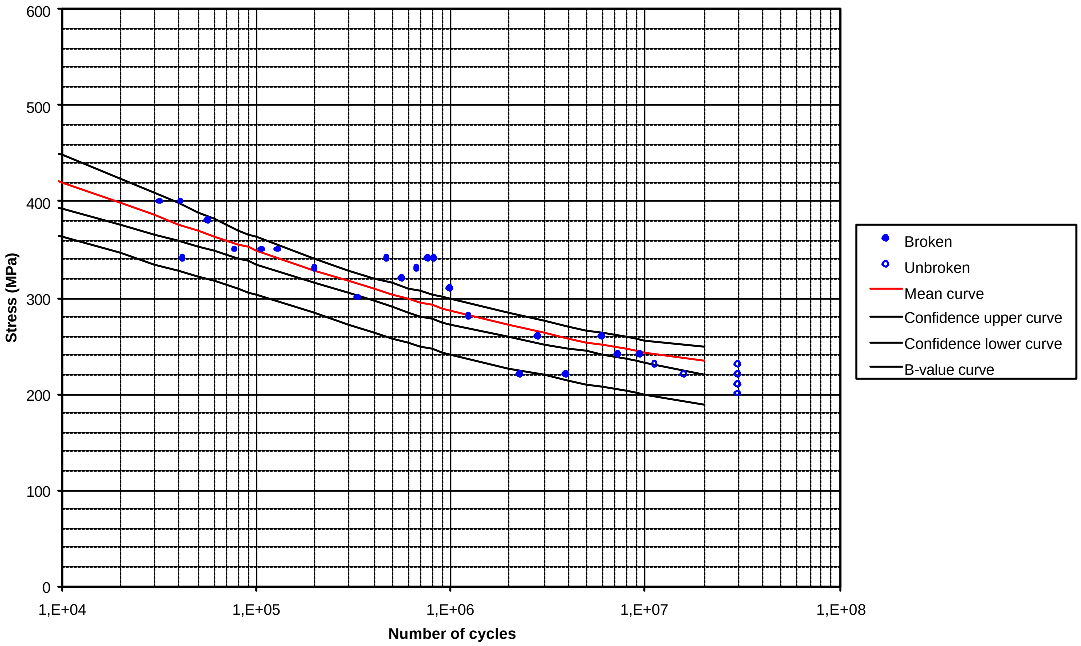


Figure 16 : Statistical reduction of fatigue test data

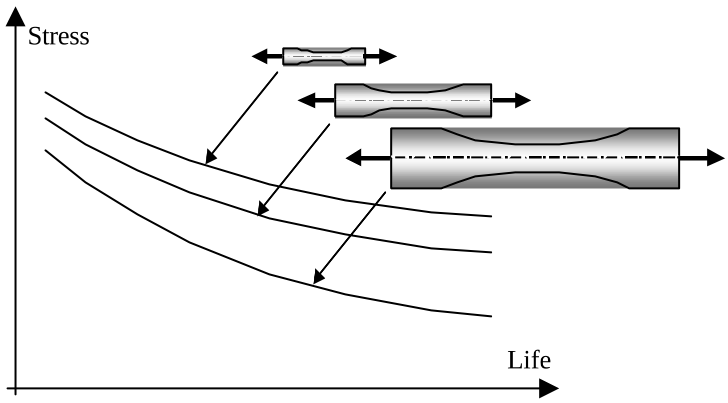


Figure 17 : Illustration of scale effects on fatigue

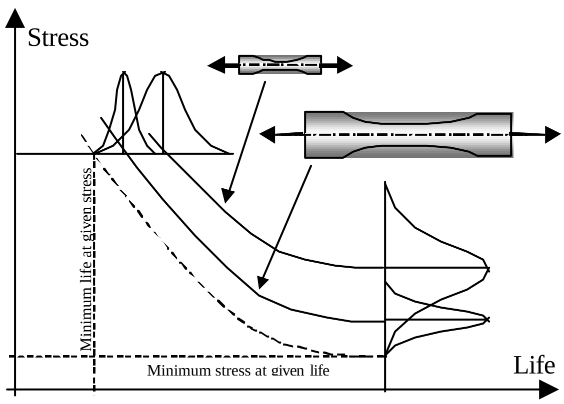
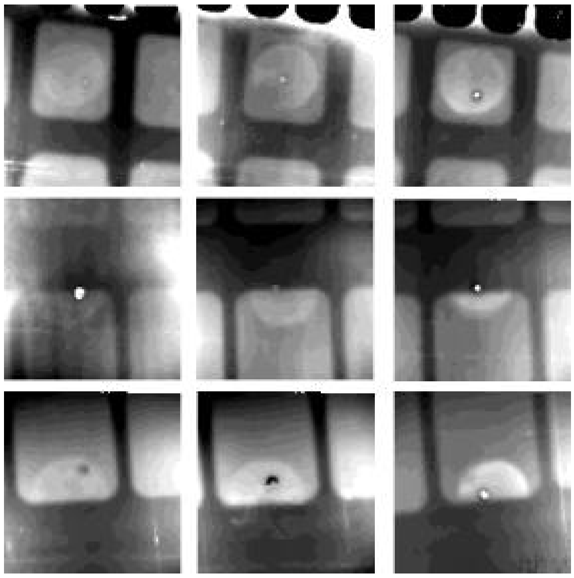


Figure 18 : Testing small specimens will exhibit more scatter than bigger ones



5% corrosion 10% 25%

Figure 19 : Flash infrared thermography of a Falcon 900 panel, showing circular “corrosion” zones

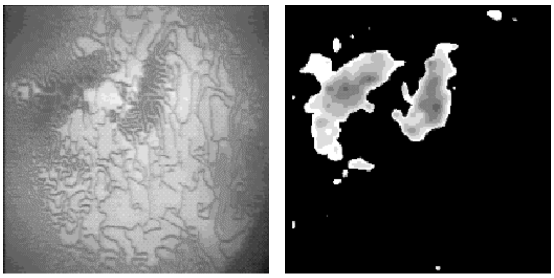
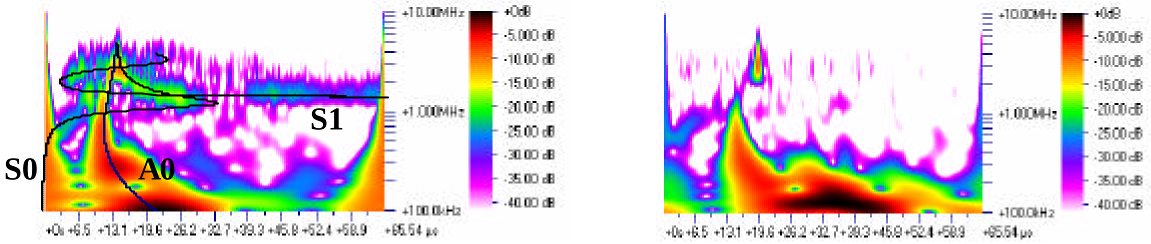


Figure 20 – Raw image (left) and after processing (right) of a corroded zone in a panel from a KC135 transport aircraft



Sound area Corroded area

Figure 21 : Wavelet analysis of the signals obtained on a sound area, comparison with theory, and disappearance of the S1 mode on a corroded area

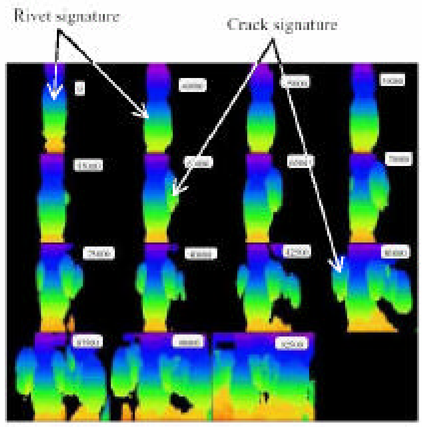


Figure 22 : crack propagation around rivet 10

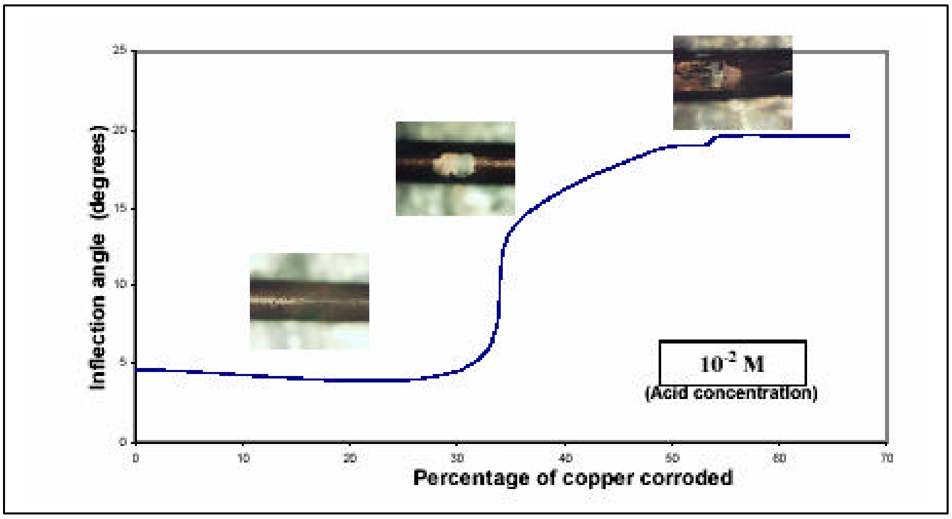
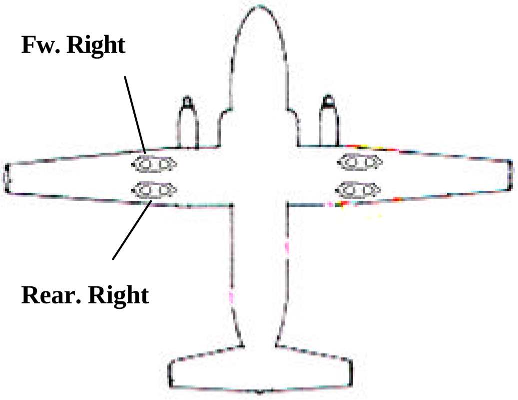


Figure 23 : Experimental result for which an accelerated chemical corrosion attack was used

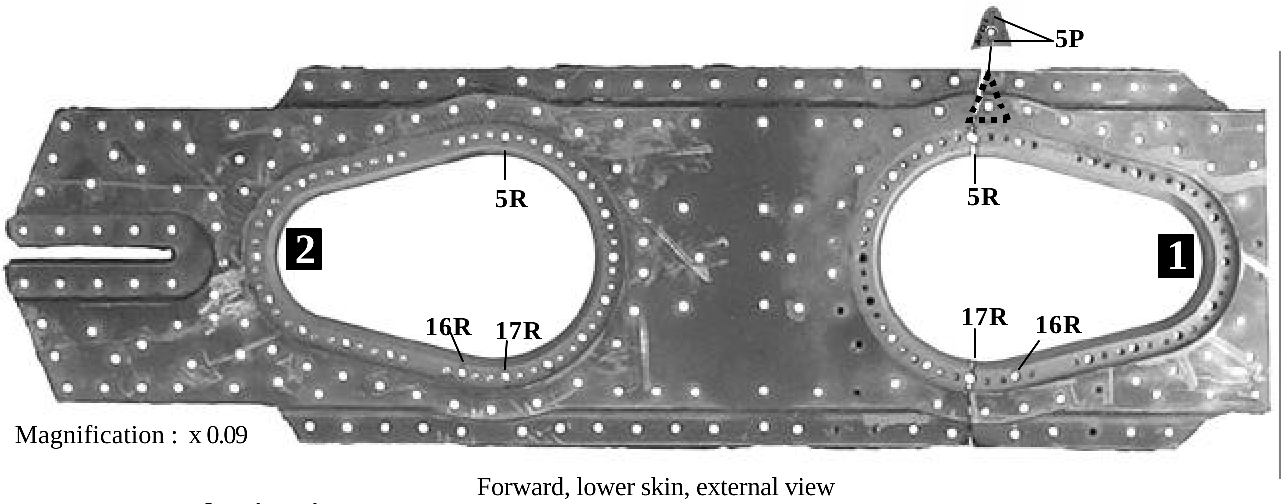
Figure 24 : The helicopter Gazelle full-scale test

Figure 25 : The Alphajet full-scale test

Figure 26 : The Transall full-scale test

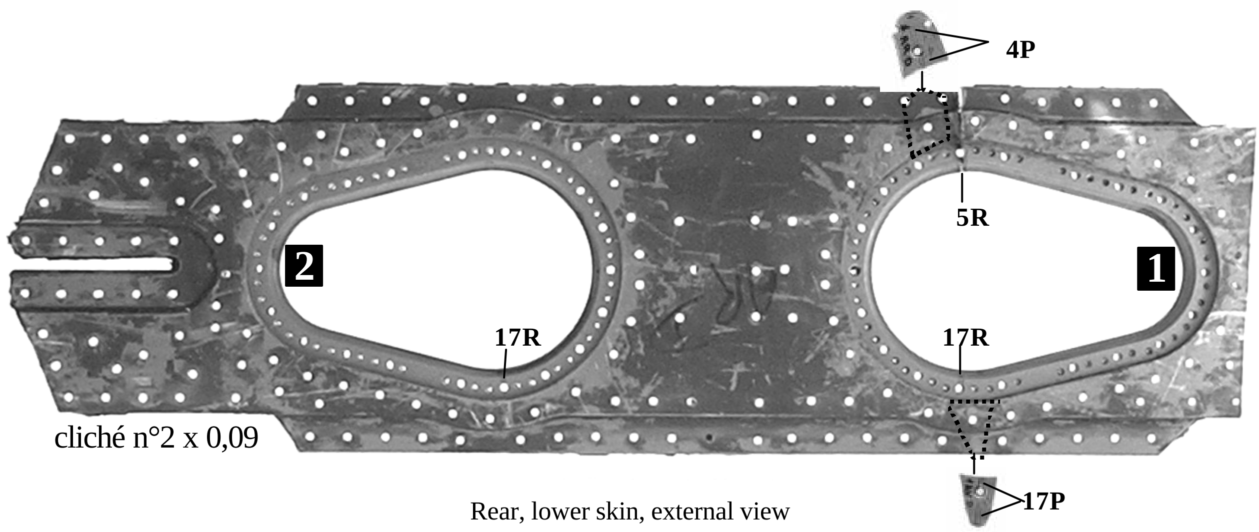


View from under the aircraft



Forward, lower skin, external view

Towards wing tip



Rear, lower skin, external view

Figure 27 : Crack initiation in fuel pump doublers

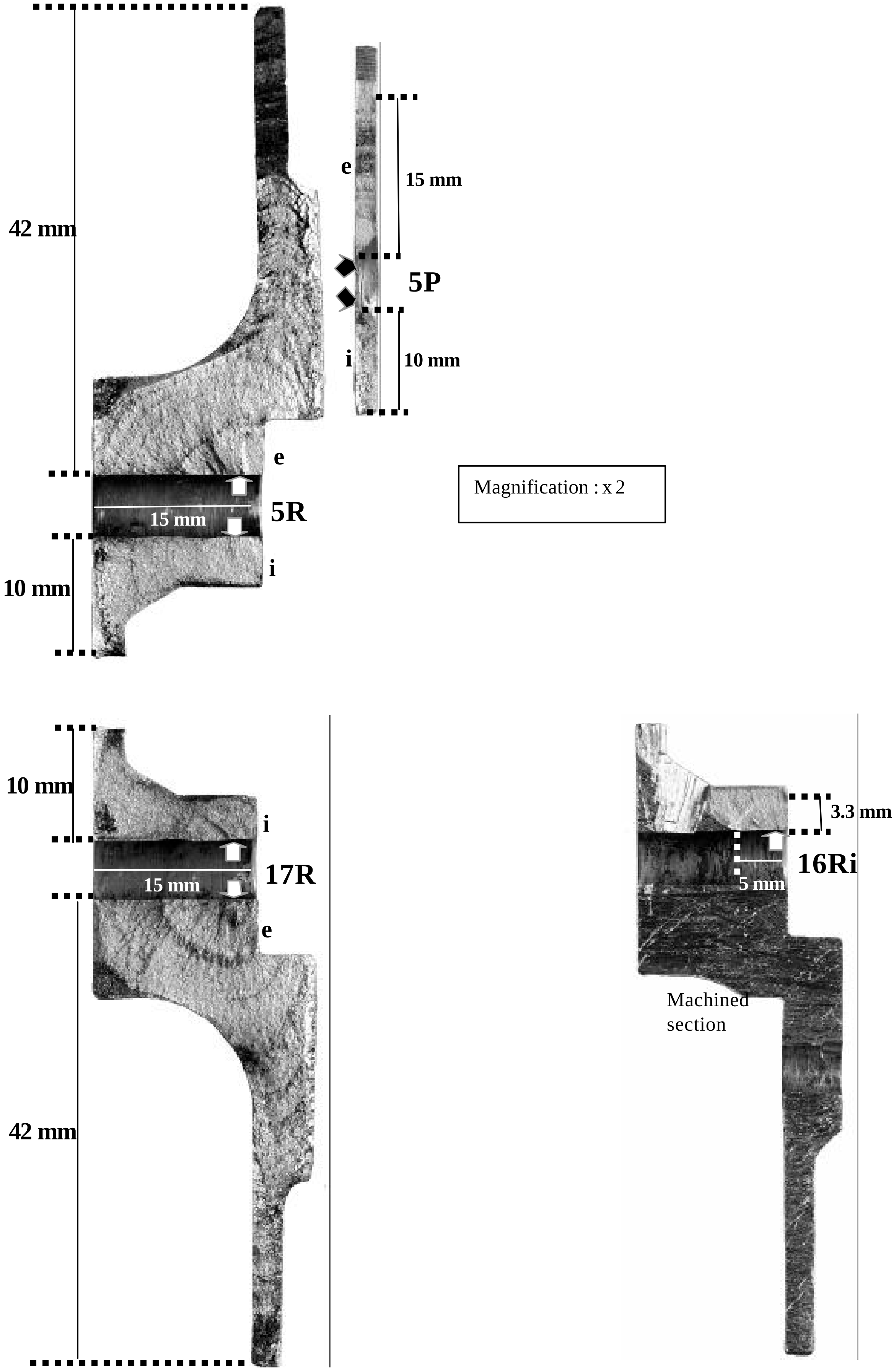
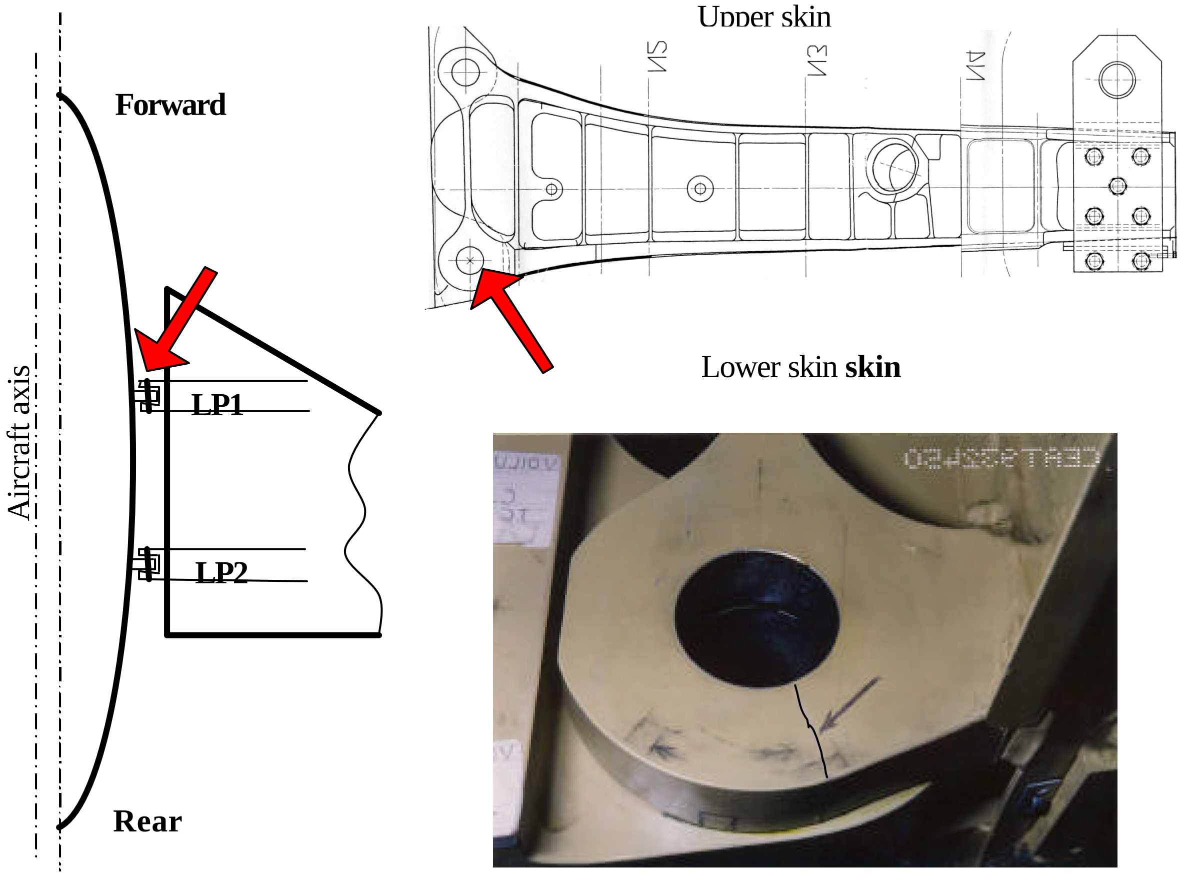
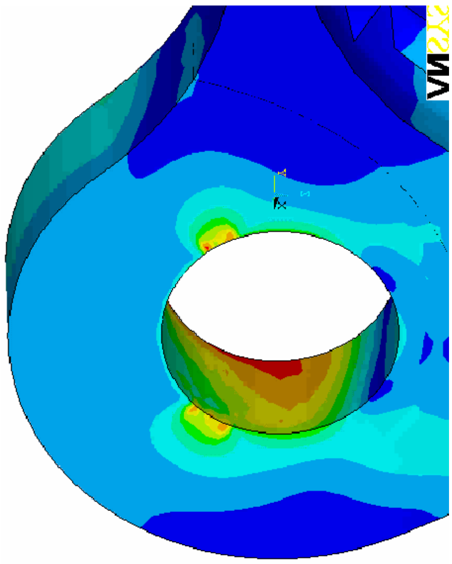


Figure 28 : Crack fractography in fuel pomp doublers



Location of the fatigue damage over the wing/fuselage junction lug

FEM Modelling of the wing spar



Initiation location
according to the fretting

Figure 29 : Mirage 2000, numerical analysis of fretting initiation in the junction lug of the wing spar

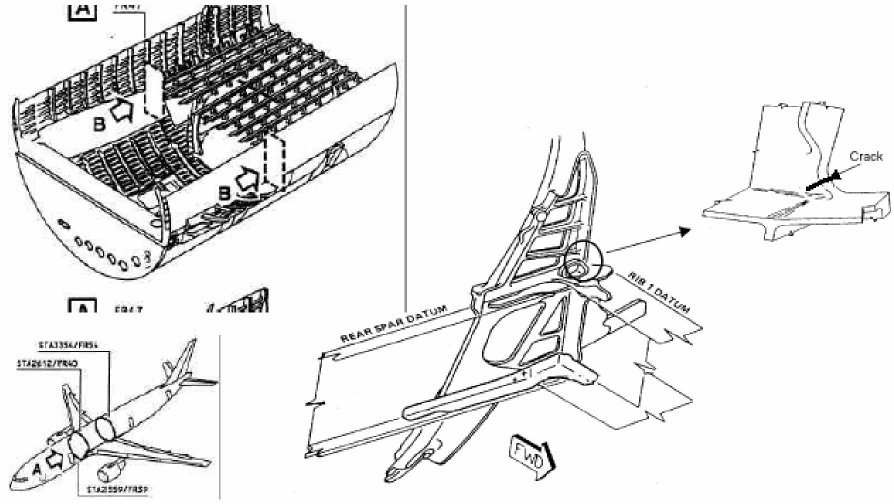


Figure 30 : Damage location

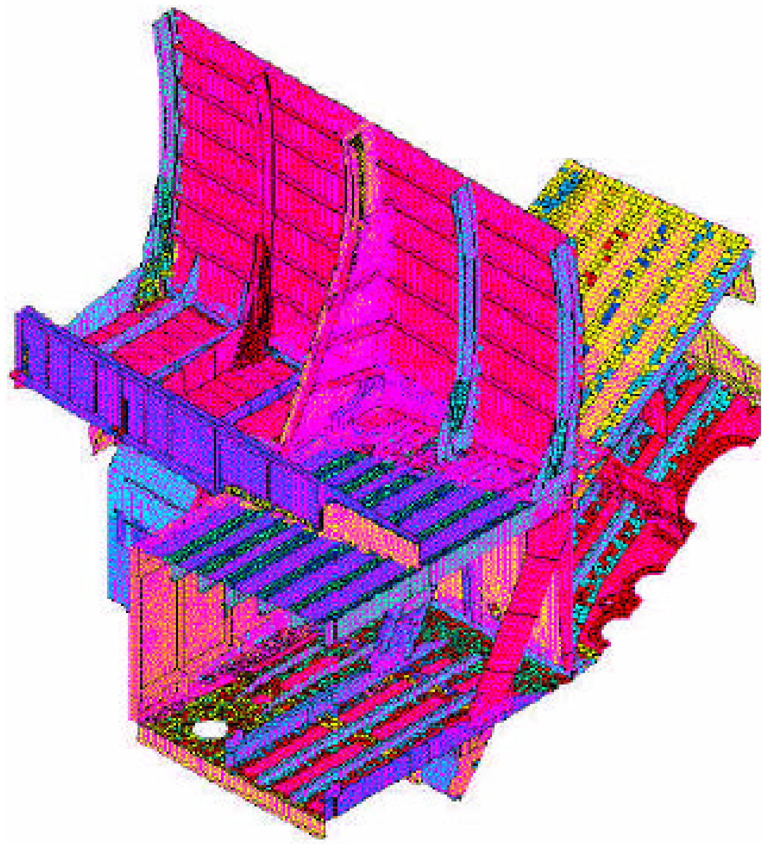


Figure 31 : Illustration of the FEM mesh to calculate Kt