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REVIEW OF AERONAUTICAL FATIGUE INVESTIGATIONS IN FRANCE DURING THE PERIOD JUNE 1999 - MAY 2001

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INTRODUCTION AND ACKNOWLEDGEMENTS

The present review, prepared for the purpose of the 27th ICAF conference to be held in Toulouse (Fr), on 25 - 27 June 2001, summarises works performed in France in the field of aeronautical fatigue, during the period May 1999 - May 2001.
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 :
Ms. Pascale Kanoute and Mr. Jean Claude Kaprez for ONERA,
MM Lionnel Le Tellier and Louis Anquez for Dassault Aviation,
MM Jean Yves Beaufils and Alain Davy for EADS Airbus SA (Toulouse Centre of Competence),
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MM Christophe Simon, Pascal Hamel, Yves Laporte and David Recorbet 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.

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6.1.1 Investigations performed at ONERA

6.1.1.1 Further validations of ONERA TMF life prediction methods

The life prediction techniques developed at ONERA and used at SNECMA for predicting crack initiation involve the combination of creep damage and fatigue « initiation » and « propagation » damages, coupled with oxidation effects. These models are applied after the full inelastic analysis of the component, using cyclic unified viscoplastic constitutive equations. They have been validated for several turbine blade materials, including single crystal superalloys, for a number of simple and complex loading conditions, including TMF conditions applied on laboratory specimens (approximately uniform temperature fields).

A specific testing methodology was recently developed, to test component like conditions. It uses tubular specimens submitted to external heating (induction) and internal cooling (air flow), allowing to develop a cyclic wall temperature gradient of about 200°C under steady conditions, on a 2 mm thickness (but an approximately uniform temperature all around the tube). These temperature fields are relevant for actual blade operative conditions (Figure 1). An additional cyclic load can be applied, which simulates centrifugal effects and allows to obtain varying lifetimes for the same temperature cycle.

Previous exploitations of this testing device has allowed to check the constitutive and damage models, and the component whole life prediction methodology, for two materials used in SNECMA turbo-engines for first stage turbine blades [1], [2] : the IN100 polycrystalline superalloy and the AM1 single crystal (coated by C1A).

A further validation has been performed recently on tubular AM1 specimens containing small cooling holes (film cooling).

The cyclic inelastic analysis of the component, including the stress localization at small holes, has been realized during a thesis at Ecole des Mines de Paris [3], using the advanced capabilities of ZéBuLoN code jointly developed by ENSMP and ONERA. Two sets of unified constitutive models giving similar results are available, either based on a slip viscoplasticity formulation [4] or an invariant approach [5].

Application of the ONERA damage models (models determined earlier [6]) as a post-treatment lead to correct predictions of the crack initiation at the critical hole, as shown on Figure 2 for one test condition. The predictions appear to slightly underestimate the life to crack initiation. Further work is under way in order to take into account, via a specific scale change, the very localised stress-strain gradients near the small holes.

REFERENCES

[1] Gallerneau, F., Nouailhas, D. and Chaboche, J.-L., (1996). *A fatigue damage model including interaction effects with oxidation and creep damages*, FATIGUE'96, Berlin .

[2] Gallerneau, F., and Chaboche, J.-L., (1999). Fatigue life prediction of single crystals for turbine blade applications, *Int. J. Damage Mechanics*, Vol. 8, pp.405-427.

[3] Cardona, J.M., (2000). Comportement et durée de vie des pièces multiperforées : application aux aubes de turbine, Thesis, School of Mines of Paris.

[4] Méric, L., Poubanne, P., and Cailletaud, G., (1991). Single crystal modeling for structural calculations : Part I - Model presentation, *Journal of Engn. Mat. and Techn*., 113, 162.

[5] Nouailhas, D., Culié, J.-P., (1991). Development and application of a model for single crystal superalloy, Fredd A.D and Walker K.P Ed., *High Temperature Constitutive Modeling – Theory and Application*, ASME, MD-26, AMD-121, New York, 151.

[6] Gallerneau, F., (1999). *Modelling of anisotropy effects of a single crystal superalloy on its fatigue-creep resistance*, ICAF’99, Seattle, July 12-16.

6.1.2 Investigations performed at EADS-CCR

These studies deal with short cracks, crack growth rates in the presence of a residual stress field, fatigue crack initiation and residual strength. The objectives are to provide aircraft and helicopter design offices with behaviour laws and predictive tools to help for assessment of inspection thresholds and damage tolerance of components.

6.1.2.1 Short cracks

The characterisation of fatigue short crack behaviour raises the issue of using the linear elastic fracture mechanics parameter K. The main reason is that local stress, plasticity which is not small compared to the crack length, and microstructural barriers exert a strong influence on the crack path and growth. Short crack growth rate tests have been run on open hole specimens in 2024 under constant amplitude loading. The short crack growth regime was characterised down to 30 microns. At high stress level, some second phase particle failures have been observed to initiate cracks ahead of the main crack leading to coalescence and acceleration effects. The graph figure 3 reports the crack growth of short cracks, R=0.1, which has been monitored on each side of the hole. Despite some scatter, the behaviour can still be formulated using a linear elastic fracture mechanics type equation, where da/dN is proportional to $\sqrt{a}^{\,n}$, n being the typical Paris' exponent as determined through standard da/dN tests.

6.1.2.2 Crack growth rates in the presence of a residual stress field

Fatigue crack growth rates at holes in the presence of a residual stress field due to cold expansion have been studied running tests and setting up a predictive method. The investigated cases are open holes on 2024 and 7075 alloys, inhabited hole in 2024 alloys. The tests data show a significant retardation effect in the first few millimeters of crack growth. The predictive method, based on finite element analysis to simulate cold expansion and to derive the elastic stress singularity to assess the crack growth rates has been validated on these cases. In addition, this approach allows to derive retardation master-curves that can be used by design offices. The main results are presented in the symposium.

6.1.2.3 Fatigue crack initiation

The implementation of the initial flaw concept on aircraft or the damage tolerance on helicopters requires fatigue data about the effect of manufacturing defects on the fatigue performance. Series of tests have been run in order to evaluate the loss of fatigue performance of coupons with initial defects such as scratches, corrosion pits, impacts. The investigated materials were aluminium alloys, steels and titanium alloys. The corrosion pits are the more severe damage since they can lead to a 50% drop in fatigue strength. Additional tests on double shear specimens made out of aluminium alloy, with two rivets, have been run. Among the investigated defects, corrosion pits, scratch in the hole, clamp loss., the combination of loss of tightening plus a scratch gave the worst performance (more than 50%) loss.

6.1.2.4 Residual strength after fatigue crack propagation

The determination of allowable fatigue cracks on stiffened panels relies on the R curve concept. The approach can be overly non conservative (more than 30%) on integrated stiffened panels. Therefore, it was decided to implement a local approach of fracture, namely the Rice and Tracey cavity growth integral to simulate the crack advance under static loading by a ductile rupture mechanism. This integral depends on the plastic deformation and the stress triaxiality ahead of the crack tip. When the integral reaches a critical value ahead of the crack tip, the crack advances. A flat stiffened panel with welded stringers has been modeled using elastoplastic finite element analysis. Load steps have been applied on the panel. The crack growth was simulated by calculating the integral ahead of the crack tip : if the critical value is reached, the nearest node is released and the integral is checked at the next node (where the crack front has gone to) to check for further release. When the crack is found to stop (the integral value is below the critical one), the load is increased. This has allowed to calculate the residual strength of the panel with an initial fatigue crack of a two stringer pitch value. The predictions of the load to failure were within 7% of the experimental value.

6.2 COMPUTATIONAL TECHNIQUES

6.2.1 Fatigue life assessment of circumferential butt-joints (EADS Airbus SA)

The objectives of these investigations are to design optimized joints against fatigue and to minimize the structural failure risks.
Introduction/context : Fuselage panels are joined either by lap-joints in the longitudinal direction or by butt-joints in the circumferential direction. The more complicated joints, which are used for areas with high fatigue stress levels (upper shells of the fuselage), are circumferential butt-joints. They are made out of the following parts :
- two stiffened curved panels (including skin, stringers and frames),
- a butt-strap that ensures the continuity of the skin,
- stringer couplings that ensure the continuity and the load transfer in the stringers,
- one frame,
- one frame/stringer stabilizer,
- fasteners between skin/strap/frame, skin/strap/stringer, stringers/couplings,….

Because of the complexity of this structural detail (several possible materials – welded stringers, possible Glare® panel/strap, thicknesses, fasteners – rivets, bolts, stacking of several parts,…), current approaches, essentially based on S-N curves obtained by test results on coupons, are not sufficient.

Field of investigation/illustration :The assessment of the fatigue life of circumferential butt-joints is based on (a) the calculation of the local stress concentration factor (called Kt) for each fastener using fully parametric finite element models and (b) the estimation of the fatigue life using the EADS Airbus SA design handbook.
Actually, the way to improve the reliability of predictions at the design stage is to better calculate Kt, using improved FEM.
These models allow to assess the stress distribution in the structure and to calculate the Kt. They take the following parameters into account :
- large displacements (non-linear computation),
- secondary bending due to the stacking of parts that induce high eccentricity,
- deformation of the fastener (modelled with beam elements).
See figure 4, illustration of the FEM mesh.
These models enable to accurately calculate Kt, which is considered as a sum of 3 components : the load transferred in the fastener, the load transferred in the plate and the bending moment.
Knowing the stress concentration factor, fatigue life can be estimated taking material fatigue properties and processes (cold-working, interference fit, CAA, fastener setting-automatic or manual,…) into account using the in-house handbook.
The two steps of the methods are validated using tests performed from the bottom to the top of the pyramid of testing : small coupons, flat stiffened panels tested in the CEAT and pressurized full-scale fatigue test (EF2).

6.2.2 Improved damage tolerance approaches for longitudinal cracks in fuselage panels (EADS Airbus SA)

The objectives of these investigations are the following :
- to reduce inspection tasks replacing visual detailed or NDT inspections by visual, or increasing inspection intervals,
- to minimize margins thanks to more reliable calculation methods (weight savings),
- to be prepared to forthcoming changes in airworthiness regulation (rotorburst, two bay crack criterion),
- to reduce costs induced by testing.

Introduction/context :The Large Damage Capability (LDC) of a fuselage panel must be maintained under control in order to reach objectives previously described and especially :
- to minimize the weight in order to increase the competitiveness,
- to comply with future regulatory requirements.

The LDC must cover both crack propagation and residual strength.
For crack growth analysis, the major difficulty is to take the so-called “bulging effect” into account. Actually, it increases dramatically the driving force curve - K=f(a) - and consequently reduces the crack propagation life (by a factor 2 on a thin sheet).
For residual strength assessment, the major problem is due to the fact that a lot of non-linear phenomena are interacting on each other (static crack extension, static frame/cleat failure, skin/cleat or cleat/frame fasteners failure).
The most reliable way to assess both crack growth and residual strength is to use non-linear Finite Elements Models and to calibrate failure criteria thanks to tests performed on curved panels (See figure 5).
Field of investigations/illustrations :The following strategy has been adopted to built reliable methods :
- development of a fully parametric non-linear (geometrical and material) FEM containing crack tips elements (to assess the stress intensity factor). Stringers and frames are modelled with beam elements except adjacent frames which are modelled with shell elements to allow an accurate prediction of the residual strength (Figure 6).
- calibration of failure criteria of fasteners, cleat and frame foots, skin thanks to tests.
- definition of an iterative method that permits to optimize a panel sized against the 2 frame bay crack criterion.

This strategy will be applied to the A380.

6.2.3 Fatigue life prediction of welded stringer runnouts (EADS Airbus SA)

The objectives of these investigations are the following :
- to orientate the design of the welded left hand lower side panel of the A318 section 14 (Figure 7),
- to build fatigue design rules for any kind of fuselage butt-joints of welded panels,
- to define welded stringer repairs able to sustain fatigue requirements (in the frame of IARCAS European project).

Introduction/context : Laser beam welding (Figure 8) is a technology now able to be implemented on future civil aircraft (A318, A380) for which the fatigue behaviour needs to be particularly investigated and assessed due to the lack of in-service experience.
In particular, the weld line run-out susceptible of fatigue cracking (essentially for stress concentration concern) must be investigated very carefully.
Field of investigations/illustrations : The fatigue behaviour of weld line run-out is not obvious because of the number of parameters that may influence the stress concentration factor Kt :

- stiffening ratio,
- angle that stops the stringer,
- weld line run-out "chaotic" shape (Figure 9).
- the presence or not of an overthickness under stringer (Figure 10), that reduces the stress level
- biaxiality,
- pillowing effect due to internal pressure,…

The method to assess the fatigue potential of a weld line run-out with or without a splicing is based on both non-linear Finite Elements Models and tests on coupons.
The first step consists in performing tests on single coupons (Figure 11) with "current" welding parameters in order to assess the Fatigue Quality Index.
The second steps consists in modelling the specimen and estimate the numerical stress concentration factor Kt. By comparison with test results, it is then possible to calibrate a so-called "test to structure factor" (Figure 12).
Finally, the real structure must be modelled (whatever the thicknesses, the angle, the loading and the coupling geometry) in order to assess the Fatigue Quality Index of this structure, taking into account the test to structure factor (with same modeling principles). See figures 13 and 14.
This method have been successfully validated by tests on several complex specimens (coupons, flat and curved panels) and with different loading conditions (uniaxial/biaxial loading, with or without pressure) :

6.2.4 Stable tearing modelling (CEAT)

A research program on modelling stable tearing in fatigue crack growth is in progress at AMRL (Australia) and CEAT.
Its objective is to investigate the conditions under which an airframe material fails by stable tearing, and to develop a predictive capability for this phenomenon under operational conditions.
This research program involves two Work Packages (WP) :
WP1 : a study of stable tearing under controlled laboratory conditions, currently in progress at CEAT, which is composed of :
- tests showing stable tearing on CCT100 aluminium alloy 7050 specimens with thickness variations : 3, 6 and 9 mm (Figure 15),
- an analysis of the influence of the precracking and loading conditions (ΔK, ΔK_{eff}, loading speed, displacement/load control…),
- the use of the test results to assess the existing models of Schijve and Forsyth,
- an investigation trying to apply the R-curve concept to the description of stable tearing,
- eventually, a fatigue test on a more representative alloy component such as a wing spar.

WP2 : Modelling of stable tearing using incremental crack growth/FE models. This Work Package 2 will use the results from WP1 to assess a Finite Element Model, already used by AMLR, to try to predict the extend of crack front shape change during tearing.
The work sharing of this programme is : WP1 : CEAT and WP2 : AMRL.

6.3 EXPERIMENTAL TECHNIQUES

6.3.1 Thermography detection of early thermal effects during fatigue tests of steel and aluminium (ONERA)

It is known, for some years, that there is a correlation between the fatigue limit of some metallic materials and the appearance of heat dissipation measurable by IR thermography. This straightforward approach unfortunately fails for aluminium alloys. ONERA objective was to analyse more deeply the temperature signal obtained during fatigue tests and to check whether there is any other damage-dependant signature.
Several fatigue tests have been performed at f = 2 Hz, both on aluminium alloy (7010) and steel (XC48, 316L), and the modulated surface temperature was simultaneously measured with a focal plane array camera [1, 2]. For each value of the stress amplitude, a series of three images corresponding to the local linear "drift" of temperature (this temperature increase is a direct consequence of cumulated heat dissipation) was calculated, the temperature amplitude at frequency f, and the one at 2f. We call them D_T , T_1, and T_2. For steel samples, three phenomena were observed when the fatigue limit was reached : the emergence of a measurable temperature drift D_T , an increase of the slope of the temperature amplitude at frequency f, T_1, and the emergence of a temperature harmonic at 2f, T_2. For 7010 alloy, only one phenomenon was clearly observed at the corresponding fatigue limit : a slope increase of the T_1 vs. σ curve.
There is thus a clear evidence, both for steel and aluminium, that the thermoelastic coefficient changes when the fatigue limit is reached, i.e. when damage and/or plasticity appears at the microscopic level, which finally leads to failure. This

thermal signature of fatigue damage is obtained in a very short time as compared to conventional fatigue tests : typically less than a couple of hours.
Figure 16 shows 7010 aluminium alloy sample results submitted to a fatigue test at frequency *f* = 2 Hz (load ratio: R_σ = -1). For each amplitude stress level, lock-in thermography provides an image of T_1, the Fourier component of temperature at frequency *f*. Plot of $T_1(\sigma)$ in fig. (a). $T_1(\sigma)$ departs from the classical linear law (Kelvin law) at about 225 MPa as shown in fig. (b) where the difference between $T_1(\sigma)$ and the linear fit of its initial part is plotted. Every point on the curve is the result of only 10 cycles. Failure occurs during the test at 400 MPa.
Figure 17 shows a summary of thermographic results with 316L and 7010 samples. Characteristic stress values as inferred from D_T (circles), from T_1 (squares), and from T_2 (diamonds). Three horizontal lines indicate the known values of the fatigue limit corresponding to the considered load ratio values R_σ = -1 and R_σ = 0. For steel, all three parameters give a good indication of the actual fatigue limit. For aluminium, only T_1 leads to a reasonably precise estimation of the fatigue limit (D_T is nearly 0, i.e. there is no measurable dissipation).

References:

KRAPEZ J.-C., PACOU D., BERTIN C., "*Application of lock-in thermography to rapid evaluation of fatigue limit in metals*", 5th Int. Workshop on Advanced Infrared Techn. and Appl., Venezia (Italy), 1999.
KRAPEZ J.-C., PACOU D., GARDETTE G., "*Lock-in thermography and fatigue limit of metals*", QIRT'2000, Quantitative Infrared Thermography, Reims (France), Jul. 18-21, 2000.

6.3.2 Biaxial test facility for engine materials (CEAT)

In the early nineties, CEAT developed a biaxial fatigue machine capable to perform tests at elevated temperature (internal pressure + axial load on thin walled tube, up to 1200°C).
This test facility is able to introduce, in the FBA specimen (Fatigue BiAxiale), tension-tension principal stresses such as in aeronautical engine rotating discs. So, a first experimental program was carried out at 650°C on the N18 alloy (a powder metallurgy nickel-base superalloy used for M88-2 discs, the Rafale's engine). The detrimental influence of biaxial loads on lifetime was shown compared to uniaxial test data.
Nevertheless, the test procedure needed to be improved, specially the finite elements analysis used to calculate the correct loading of the specimen, by taking into account :
- thermal phenomena from thermal mappings established during pressure cycling,
- the supplementary calculation of the loading ratios for tests needing pressure, and not only the calculation of the maximal levels for axial load and internal pressure,
- critical areas identification associated with uniaxial or multiaxial loading conditions, and where the stress state in the FBA specimen is defined thanks to the finite elements analysis used,
- a viscoplastic law to describe the material behaviour, that consists of a Norton law for viscous flow associated with a Chaboche model (coupling of isotropic and kinematic hardenings) for cyclic plasticity.

Then, a test campaign, including these improvements, has been carried out on a Ti 6-4 alloy at 250°C and an Inconel 718 at 650°C, where different loading conditions were applied. Results of this test campaign pointed out :
- a good accuracy between crack initiation sites observed in tests and critical areas predicted by the finite elements analysis,
- a negative effect of biaxiality that occurs between a reference uniaxial loading (pure axial load or pure internal pressure) and an equibiaxial one defined by the same amplitudes and the same maximum levels as the reference in the two directions.

This effect is summarized below, for the two characterized materials and several phase differences :

Biaxial fatigue life reduction factor			
Phase difference	**0°**	**45°**	**90°**
TA6Vpq - 250°C	1.3	1.0	1.2
Inco 718pq - 650°C	2.6	2.2	3.8

Now new widest studies will have be carried out, with the following objectives :

- biaxiality effect observation on other materials, for different fatigue lives,
- modelisation of these effects by investigating fatigue damage models.

6.4 MATERIALS AND TECHNOLOGY TESTING

6.4.1 Assessment of MMC technology for engine bladed rings (SNECMA and CEAT)

Aero-engine performances improvement requires to increase the thrust/weight ratio. This objective can be achieved either by increasing the thrust, and so the operational temperature (i.e. the turbine entry temperature), or by weight saving, and specially for rotating components.
The use of metal matrix composites (MMC), characterized by high specific mechanical properties, allows to consider the substitution of the conventional blade and disc arrangement with a bladed ring (bling). Such a solution, where a large part of the web of the disc is removed by using MMC reinforcement at the rim could result in up to a 70% weight saving.
In this context, Snecma Moteurs has already manufactured by Hot Isostatic Pressing (HIP) several small scale blings (see figure 18) with SCS-6/Ti64 titanium metal matrix composite (TiMMC), in which the Textron SiC fiber was coated by electron beam physical vapor deposition (EB-PVD) with titanium at 3M.
Then, three of these blings have been tested at the Aeronautical Tests Centre of Toulouse (CEAT) through fatigue spin testing (see figure 19) : two at room temperature and the last at 350°C. The tests consisted in a succession of sequences of 5000 cycles, at a maximal speed that corresponds to a 900 MPa peak composite stress at the first step. At the end of each sequence, the maximal speed was increased by an increment corresponding to a 100 Mpa extra stress.
The first endurance tests have shown extremely promising results. Now, tests are scheduled in order to detect the crack initiation (and its propagation) leading to the bling failure, from blings manufactured with fibres coated by an other route which aims at reducing the fabrication costs.

6.4.2 Assessment of titanium technology for landing gear applications (Messier Dowty and CEAT)

Future aircraft projects will need more and more weight saving, which constitutes a challenge, specifically for landing gears. Currently, the major landing gear components of civil aircraft (main fitting, trail arm, drag stay…) are made out of very high strength steels, which have the disadvantage of having a high density.
To take up this challenge, Messier-Dowty selected the titanium alloy Ti 10.2.3, whose specific strength and fatigue performances, higher than those of very high strength steels, promise good weight savings. However the introduction of such a new alloy in serial application requires, first, a large-scale evaluation programme.
A titanium alloy Ti 10.2.3 evaluation programme is currently in progress at Messier-Dowty and CEAT with the objective to give the technical knowledge for the design of a Ti 10.2.3 major landing gear component.
The main steps and substeps of this programme are :
- A general investigation of the material properties :
  - basic fatigue properties,
  - ultimate strength (plastic deformation) under bending and torsion loads,
  - fretting susceptibility,
  - galvanic corrosion,
  - wear protection coating efficiency.
  - evaluation of VSMPO titane alloys Ti 10.2.3 and VT22i.
- A process (die stamping) investigation :
  - adjustment of die stamping parameters for thick components.
  - validation of Messier-Dowty's design processes for titanium alloys.
- The design and manufacture of a technological demonstrator that will be tested in static and in fatigue.

Most of the tests at the coupon and technological levels are now completed. As an example, to mention one, the fretting susceptibility study was composed of 50 fretting tests on a titanium alloy Ti 10.2.3 specimens to assess the performance of technical solutions combining various surface treatments, various alloys and contact products (grease, Molykote), plus 16 fatigue tests on a specific specimen (figure 20) more representative of landing gear fretting problems to validate the technical solution.

6.4 .3 Falcon composite horizontal tail with cast titanium central box (Dassault Aviation)

The certification of a new FALCON composite horizontal tailplane, including a cast titanium central box, is achieved since May 2000 based upon JAR 25 change 14 and FAR 25 Amendment 91 requirements.
This composite horizontal tailplane is composed (figure 21) of a single upper skin, fastened to the cast titanium central box upper surface by blind rivets, and two lower skins spliced to the central box lower surface by titanium screws.

Skins are made of carbon/epoxy tapes and stiffened by co-cured hat-stiffeners.
The internal structure is composed of RTM carbon/epoxy spars and titanium ribs.
The full-scale certification test sequence, performed in a climate chamber (70°C and 85% relative humidity) on a sole test specimen including impact damages and manufacturing defects, included the following sequences :
- fatigue cycling (moisture 60%, ambient temperature) : 20000 flights with a Load Enhancement Factor of 1.17 (equivalent to more than 2 lifetimes for metallic parts),
- static tests up to limit load (moisture at equilibrium, T = 80°C),
- static tests up to ultimate load (moisture at equilibrium, T = 80°C),
- fatigue cycling (moisture 60%, ambient temperature) : 4690 flights with a Load Enhancement Factor of 1.17 (equivalent to 2 inspection intervals), after introduction of an artificial crack (3 mm saw cut) and simulation of a lug failure and a screw failure,
- residual strength tests up to limit load (moisture 100%, T = 80°C).
The casting factor of 1.5 (50% margin of safety at ultimate load) was demonstrated by analysis supported by test results validating the FE model.
The full-scale tests demonstrated the no-growth of manufacturing defects or impact damages in the composite structure and its ability to withstand ultimate loads in the presence of impact damages :
- at the maximum realistic energy of 50 J in critical areas exposed in service,
- clearly visible in critical areas protected in service.
The composite upper skin design is such that in the cast titanium central box a natural crack initiates only after extension of the saw cut up to 5 mm, two lifetimes cycling with an initiation simplified fatigue load spectrum and 3000 flights with the normal fatigue spectrum. The propagation of the crack is very slow.
The cast titanium is manufactured by PCC France company.
The process is controlled via tests and inspections : dimensions, weight, acid and penetrant fluid tolerance, Xray, tension tests on attached specimen for 100% of castings, static, fatigue and damage tolerance tests on cut up castings (#1, #25 and then 1 per set of 50).
This extensive process control should allow to apply for a reduced casting factor in accordance with ACJ 25-621, on future developments.
For new developments using the same composite design, full scale tests will be performed in ambient humidity and temperature conditions based upon experience accumulated during this programme.

6.4 .4 Falcon passenger door in aluminium casting (Dassault Aviation)

The JAA certification of a new passenger door (figure 22) in aluminium casting is in progress. The regulations applicable to this door are :
- JAR 25 change 13 and FAR 25 Amendment 69,
- Flight above 41000 ft Special Conditions.
A comparison between NPA 25D-272 (new ACJ 25-621) and the casting process control and quality plan is made to show that most of the NPA paragraphs are addressed. Then the static test may be conducted at ultimate load without taking into account the casting factor, and the additional 50% margin of safety (casting factor :1.5) is demonstrated by analysis.
The aim of the door new design is to integrate in the same casting the door substructure (vertical frames, and horizontal beams) and mechanism supports.
The door stops remain bolted, as well as rollers, on the front and rear edge frames.
The external chemically milled skin is riveted on the edge frames and beams, and on the central frame by 9 links. This reduces the number of rivets for the skin installation.
The manufacturing process is a low pressure 'HQ' sand casting realised by SFU company (Pechiney group).
The process control is based on indicators derived from each door measurement of geometrical data, radiography, material mechanical properties (attached specimens), and material chemical specification.
Moreover, some doors are cut (frequency of 1 every 50 to 100 doors), to check correlation of attached specimen test results with cut up specimens tests results for static strength and damage tolerance properties.
The certification structural tests performed on the door test article installed on a fuselage T2 section are :
- two lives time fatigue test,
- limit loads static test at 1.67 ΔP (Flight in high altitude),
- ultimate loadd test at 2.34 ΔP (Test stopped after seal extrusion and door stops failure on fuselage side, due to application of ultimate loads after 2 lifetimes). The goal was to test the door up to 2.5 ΔP.
The casting factor of 1.5 (50% margin of safety at ultimate load) is demonstrated by analysis supported by test results for the FE model validation.
- a partial damage tolerance test on a beam element, propagation for 4 inspection intervals and residual test strength (today not completed).

- fail safe tests in two configurations at 1.15 ΔP :
  - one link between central frame and skin removed,
  - the most loaded door stop removed.

In that configuration the pressurization was not possible above 0.8 ΔP, due to the seal extrusion in relation with the door distorsion. The area of the induced leakage zone is lower than the hole area allowed to comply with the 'flight in high altitude condition'. Then the seal extrusion is detrimental for the aircraft safety.
With additional material mechanical test results on specimens extracted from next door to be cut , the compliance with NPA 25D-272 shall be reached, and a casting factor of 1.0 used. This allows evolution of the ΔP applied to the door in the future.

6.5 RULE MAKING SUPPORTING ACTIVITIES

6.5.1 Initial flaw concept for threshold evaluation (EADS Airbus SA)

The objective of this study is to assess equivalent initial flaws representative of Airbus manufacturing and design practice.
Introduction /context : The "Initial Flaw" concept (see figure 23) has been the subject of considerable attention in recent years. Originally introduced by the United States Air Force, the concept is now widely used throughout the US aerospace industry in the determination of inspection thresholds for structural maintenance programmes. Initial Flaw methodologies are supported by the regulatory authorities for civil aviation (FAA and JAA), and the concept has been introduced into the airworthiness regulations as the preferred route for establishing inspection thresholds for structures on commercial transport aircraft.
However, the European aerospace industry has been slow to adopt the Initial Flaw concept, and consequently has limited experience on the applicability of the methodology. Current threshold inspection intervals are set by fatigue methods. In the absence of validated Initial Flaw distributions based on European practice, it may become necessary to accept unduly conservative assumptions in order to comply with the changes to the airworthiness regulations, with a subsequent impact on design efficiency.
Field of investigations / illustrations : EADS Airbus SA have been addressing this issue in European research projects (GARTEUR and 5th PCRD).
Three main tasks constitutes the core of their research activities :
- the assessment of appropriate manufacturing quality flaws complying with our fatigue data and methods,
- the determination of appropriate manufacturing rogue flaws representative of Airbus manufacturing practice (see figure 24 and 25),
- the development of a new approach to estimate crack growth rates in the presence of residual stresses induced by technological processes such as cold expansion and interference fit (in collaboration with EADS CCR), see figure 26.

6.6 FULL-SCALE FATIGUE TESTS

6.6.1 Fatigue tests of the Rafale landing gears (Messier Dowty and 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 going to be performed on the landing gears (static tests are completed).
Four different landing gears will be 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, will be applied.
To qualify, 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.
Test interruptions are scheduled at given intervals for NDT inspections, they will lead to subsequent analysis if needed.
Beginning of the test is scheduled in June 2001.

6.6.2 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 lifetime, extended by analysis to 10,000 FH later on. Taking into account the remaining potential of their own fleets, French and UK Forces want to prolong the utilization up to 15,000 FH. For this purpose, a second fatigue test, co-funded (25%) by the UK Forces, is in progress in CEAT (See figure 27).
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 straingages and 12 bridges are continuously acquired during the test by mean of a new CEAT device called MEFA.
For a 37,500 flight hours test target {(3 x 15,000) – 7,500}, 33,210 FH have already been simulated on May 2001.
The first phase of the test (28,125 FH simulated in training and liaison version) was achieved in December 1998, the second phase of the test (3,750FH simulated in day time tactical flight) was achieved in November 1999.
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.

Some identified damages were repaired using definite solutions already applied in service, new damages needed the desidn of specific repairs which would be validated during the test.
The third phase of the test (day and night time tactical flight with VIVIANE sight) started by the end of May 2000. 1,335 FH were simulated at the beginning of May 2001. The test has been stopped by some cracks detected on the left longitudinal beam (see figure 28). The repairs are defined and will be applied soon on the test article.

6.6.3 - Alphajet fatigue test (Dassault Aviation and 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 currently (figure 29) 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 IF experienced in service, the test airframe has accumulated 390 FI corresponding 14,500 flight hours in the FAF's current average advanced training conditions.
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.

Presently, application of flight loads on the test cell is suspended, one of the wing lower panel and the wing front spar (see damage figure 30) having to be repaired. The test is being pursued by only simulating the ground loads so as to validate an extended life for the undercarriage attachment fittings and corresponding load paths.

6.6.4 Transall structure life extension (EADS Airbus and CEAT)

Developed in the 60's, in a French-German cooperation, 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 (around 2007).

The definition and substantiation of the life extension programme 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 EADS 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 31).
After a few last adjustments, the test began in November 1999, for a duration of approximately 5 years. Up to now, 12,000 Flight Cycles have been simulated in addition to the Flight Cycles experienced in service.
First damages, observed on the lower wing panels, have already induced a specific maintenance programme 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 updated maintenance programme necessary to operate the fleet beyond 20,000 flights.

6.7 IN-SERVICE AIRCRAFT FATIGUE MONITORING

6.7.1 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 two months).

6.7.2 Evolution of the system : MICROSPEES (Dassault Aviation and 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 different 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.

6.7.3 Next 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 described in § 6.6.2. 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.

Today, the technical specifications for the calculations are written, and the computer programming is in progress. The processing of the HARPAGON records will begin at the end of 2001.

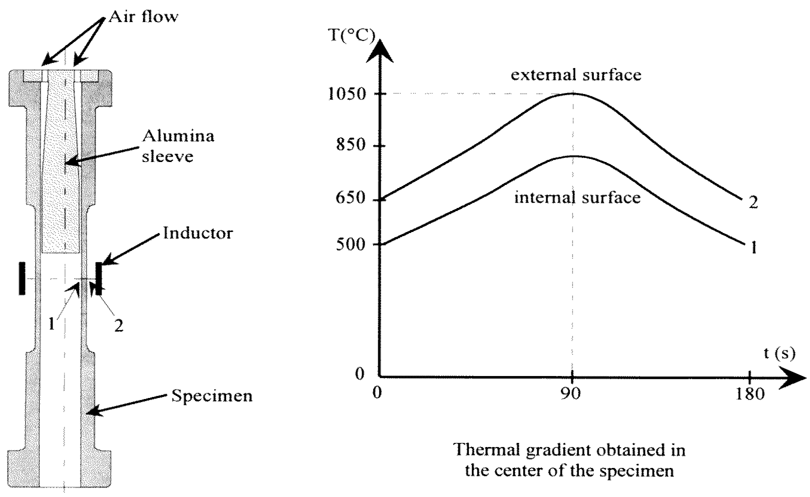

Figure 1 : Scheme of the heated/cooled specimen, and temperature distribution

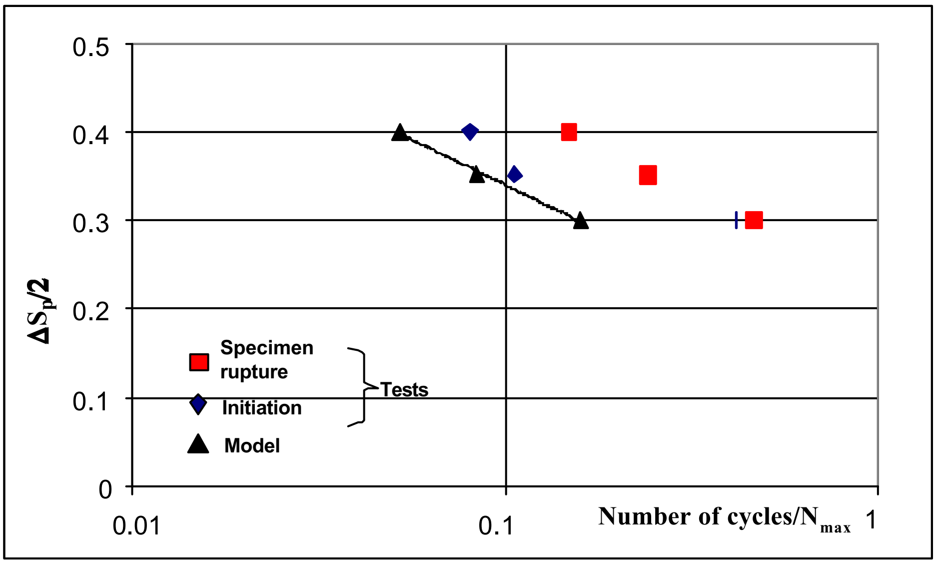

Figure 2 : Comparison of experimental and calculated lifetimes

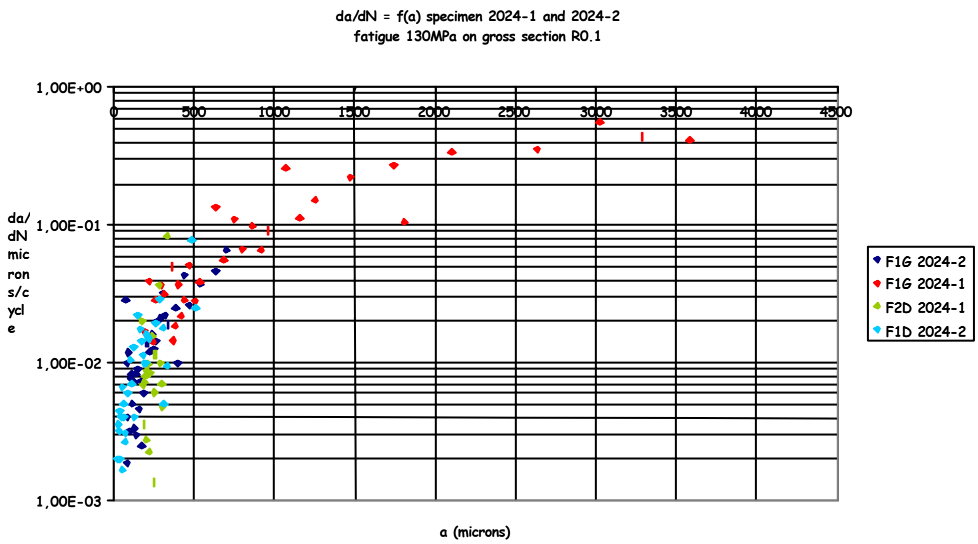

Figure 3 : Crack growth rate results in the short crack growth regime

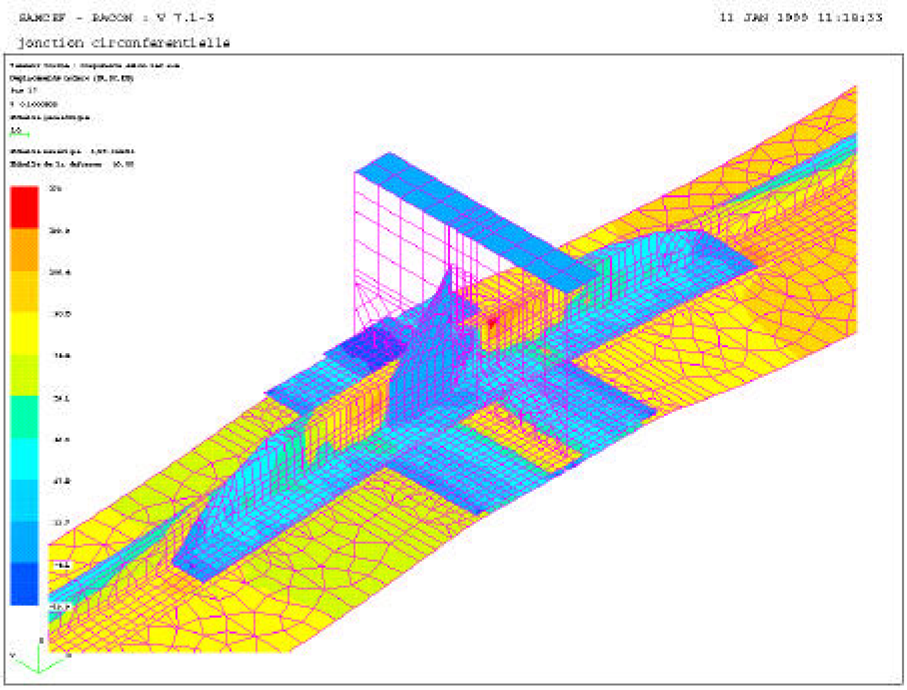

Figure 4 : Illustration of the FEM mesh to calculate Kt

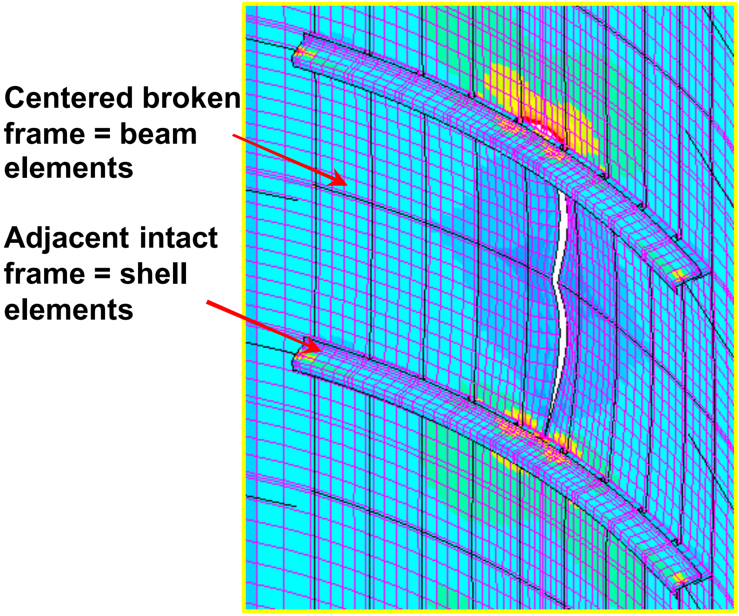

Figure 5 : Modelling fuselage large damage

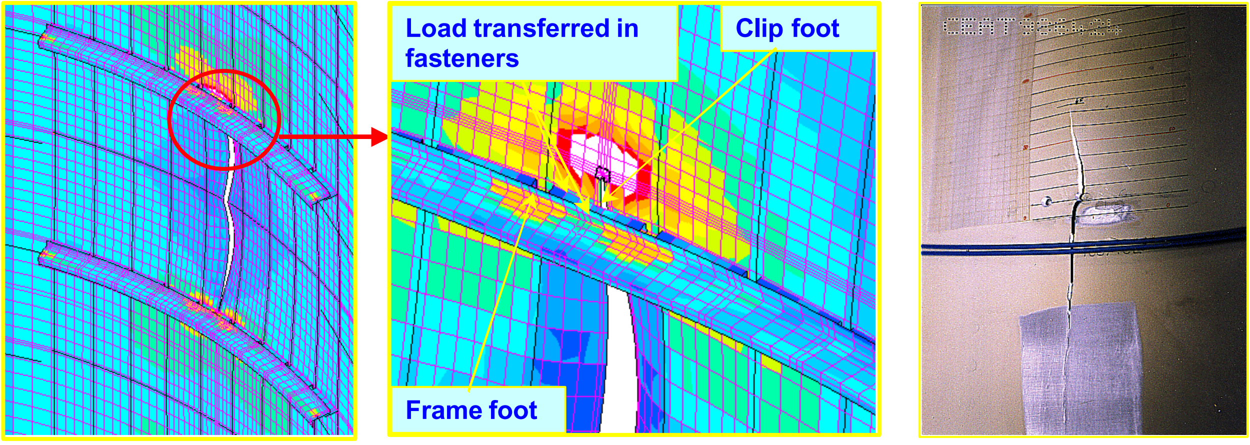

Figure 6 : Fully parametric non-linear FEM containing crack tip elements

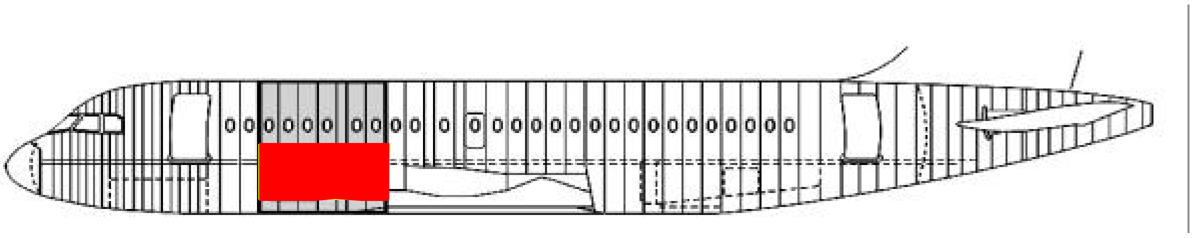

Figure 7 : Lower side panel of the A318 section 14

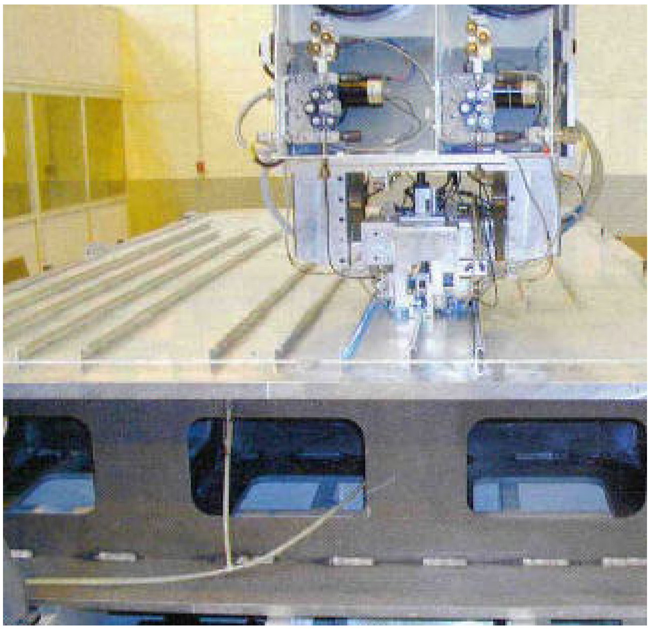

Figure 8 : Laser beam welding machine

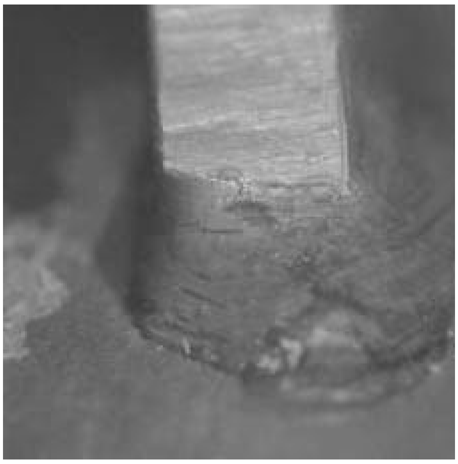

Figure 9 : Weld line run-out ‘’chaotic’’ shape

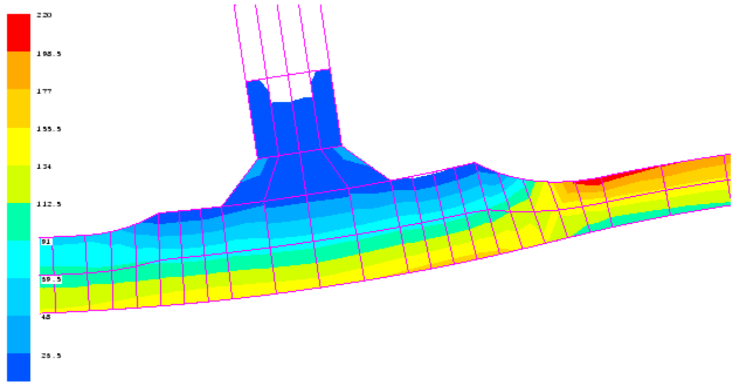

Figure 10 : Stress level reduction due to overthickness under stringer

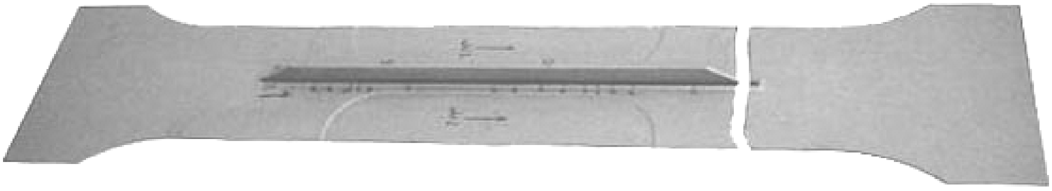

Figure 11 : Single coupon to test welded stringers

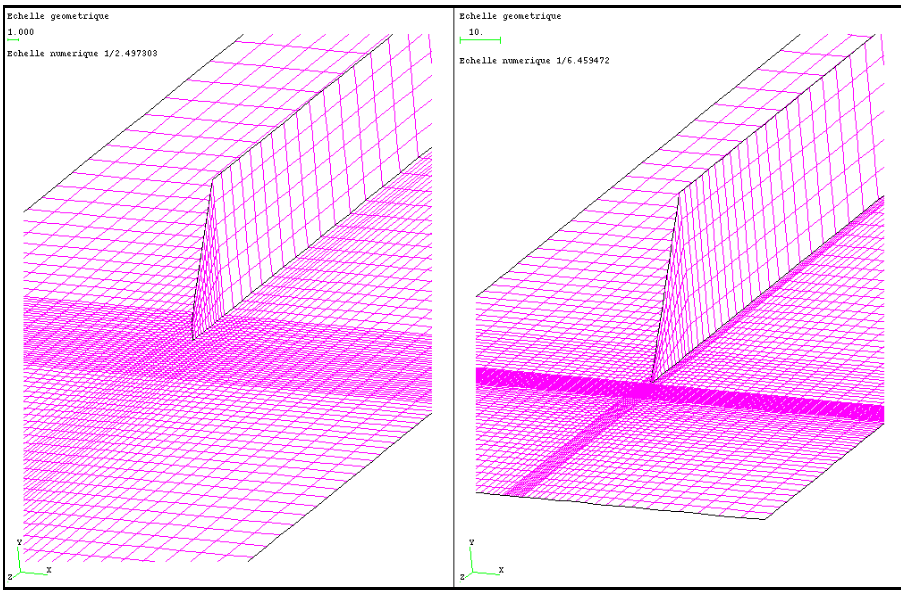

Figure 12 : Specimen modelling and Kt estimation

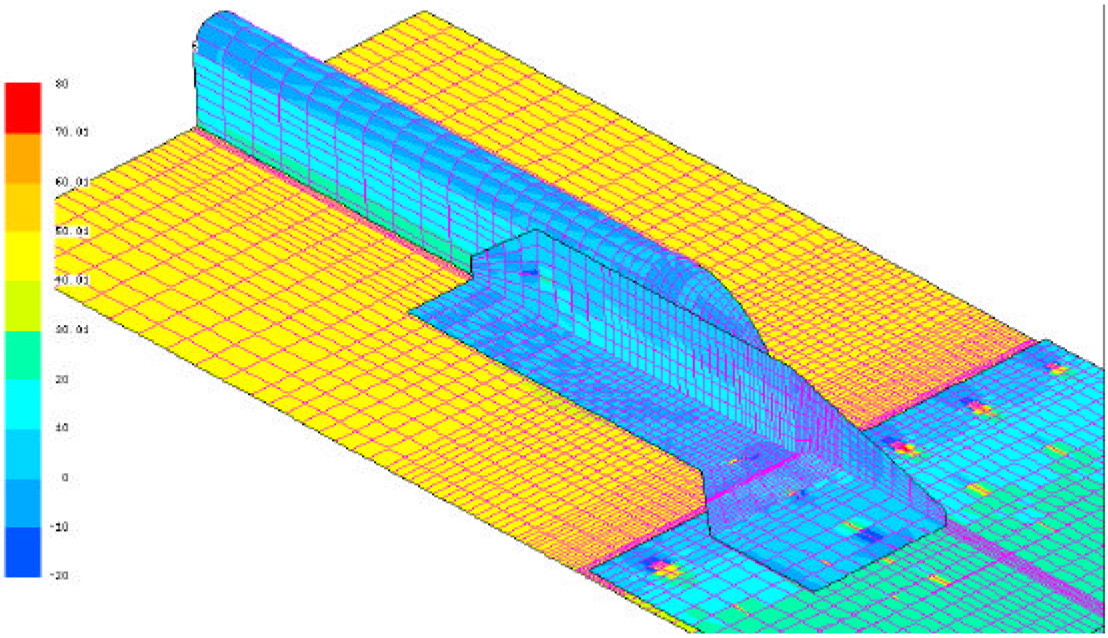

Figure 13 : Application of the method

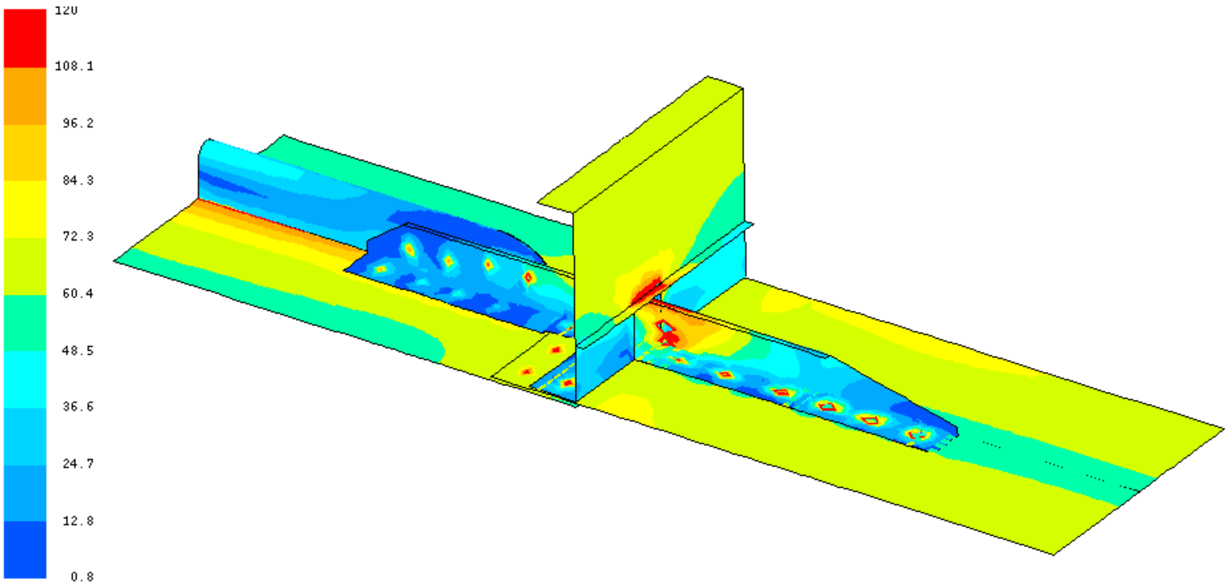

Figure 14 : Application of the method (Cont'd)

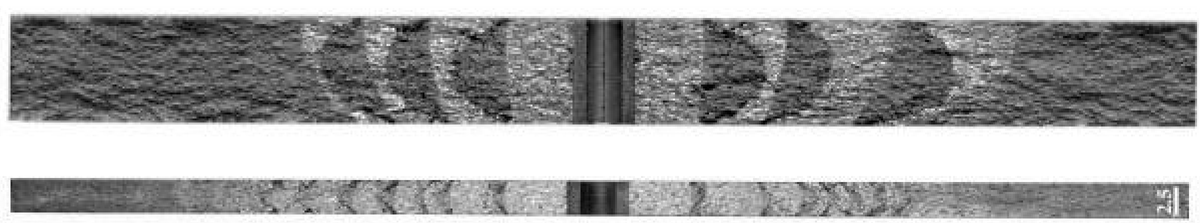

Figure 15 : stable tearing on CCT100 aluminium alloy 7050 specimen with 9 and 3 mm thickness respectively.

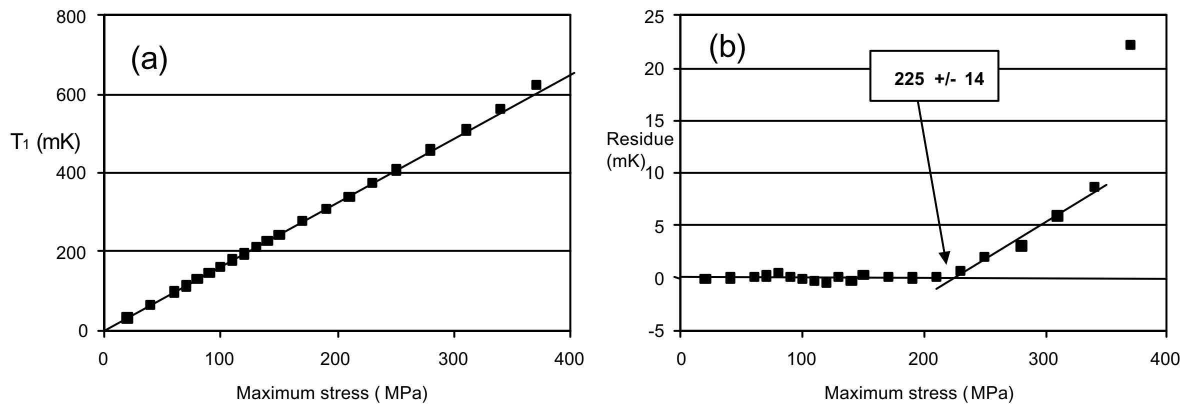

Figure 16 : Thermography results with 7010 aluminium alloy samples submitted to a fatigue test

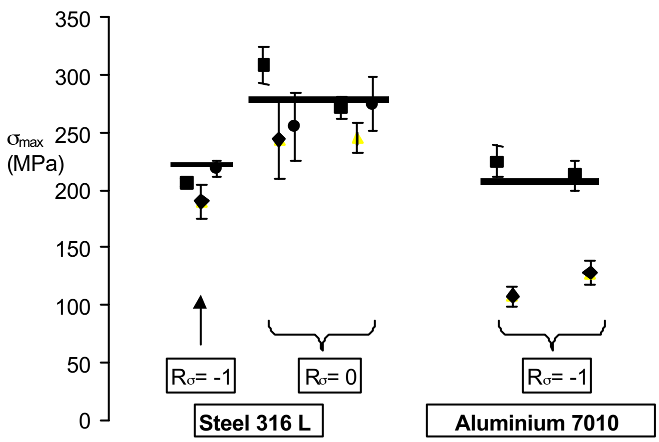

Figure 17 : Summary of thermography results with 316L and 7010 samples

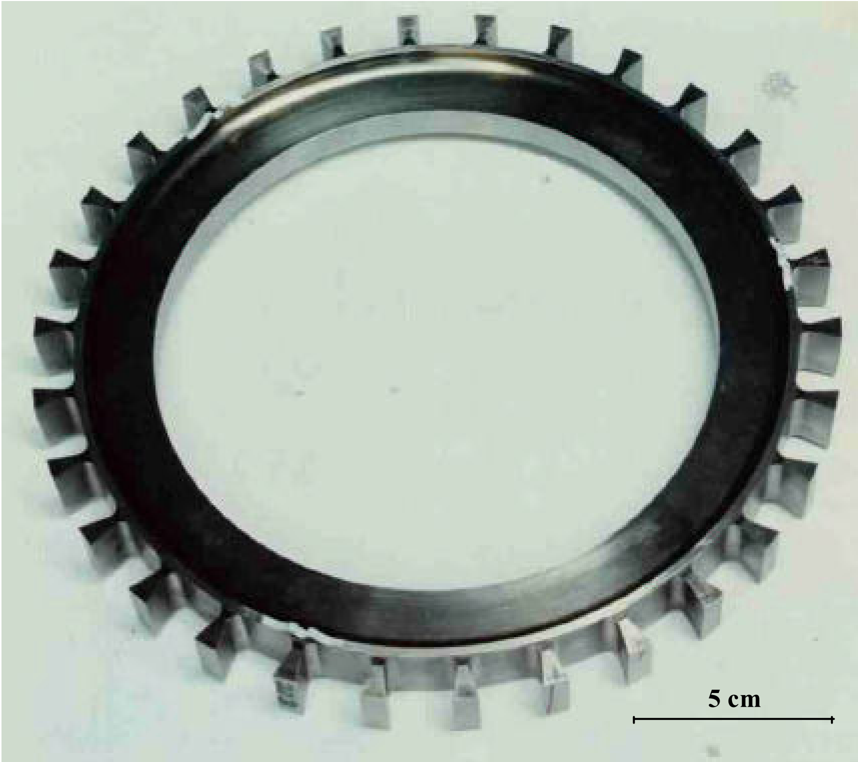

Figure 18 : Photo of a bling

Figure 19 : CEAT's spin test

Figure 20 : Landing gear representative part for fretting testing

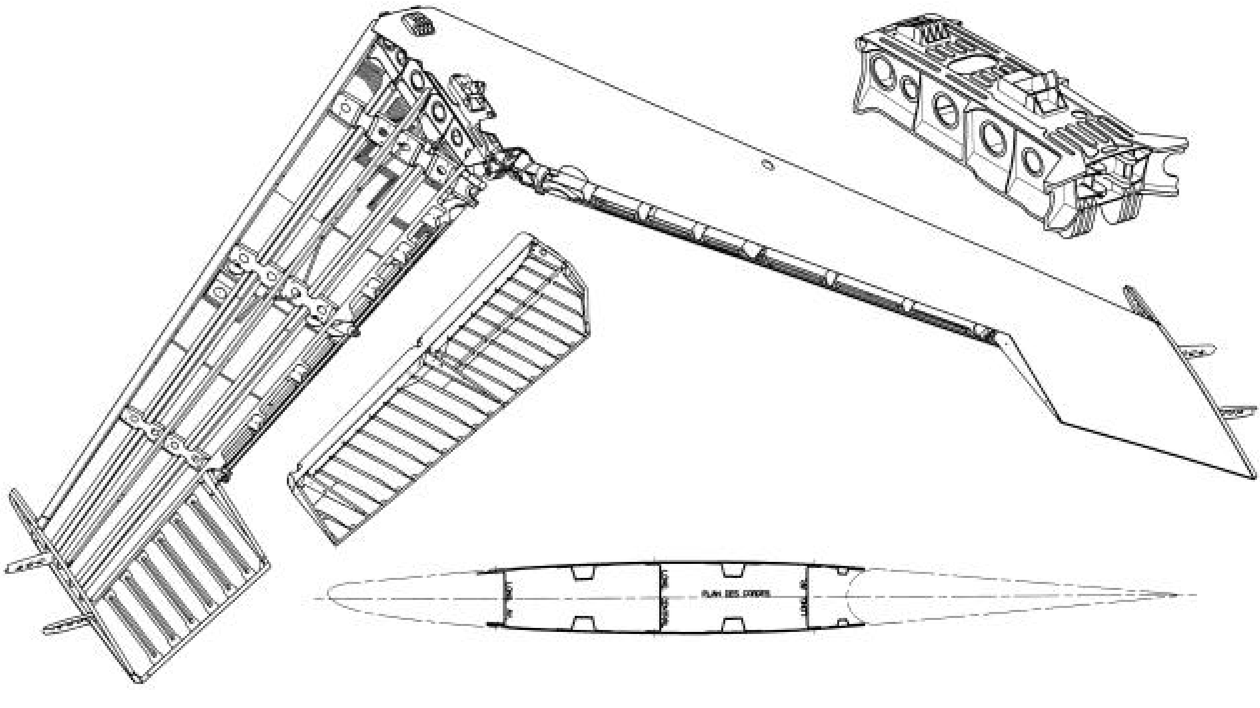

Figure 21 : Falcon carbon HTP with its cast titanium centre fitting

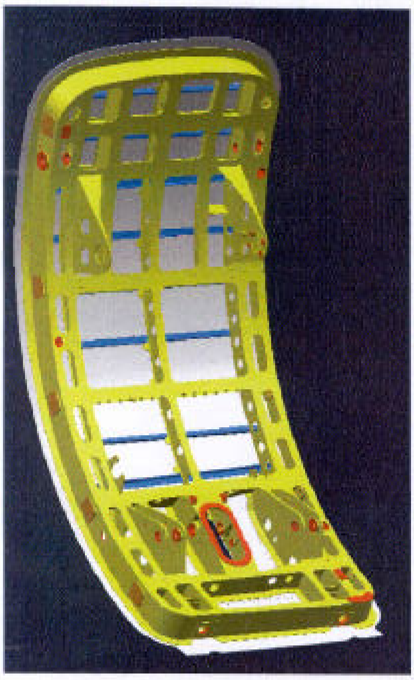

Figure 22 : passenger door in casting technology

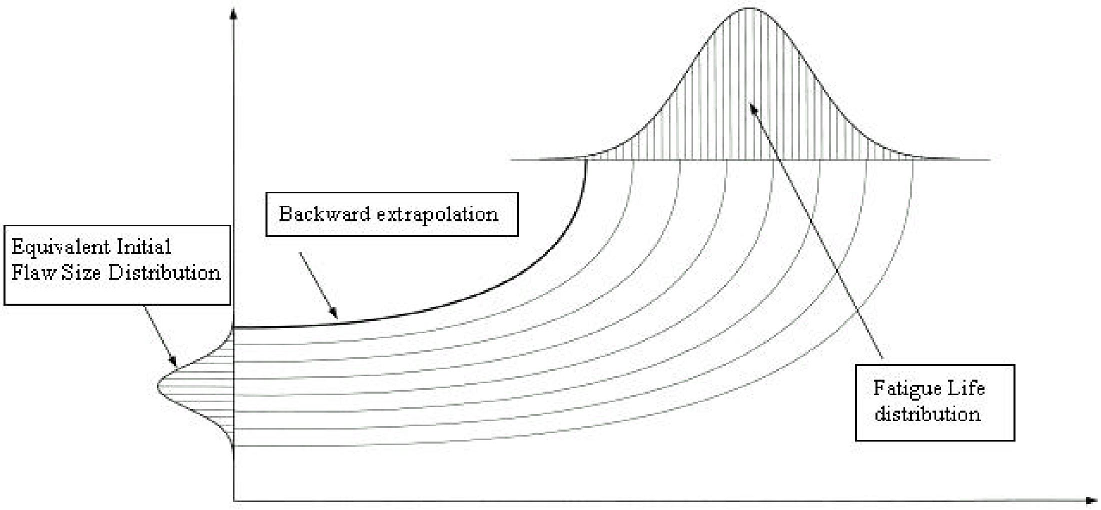

Figure 23 : Back calculation in order to generate an EIFS distribution

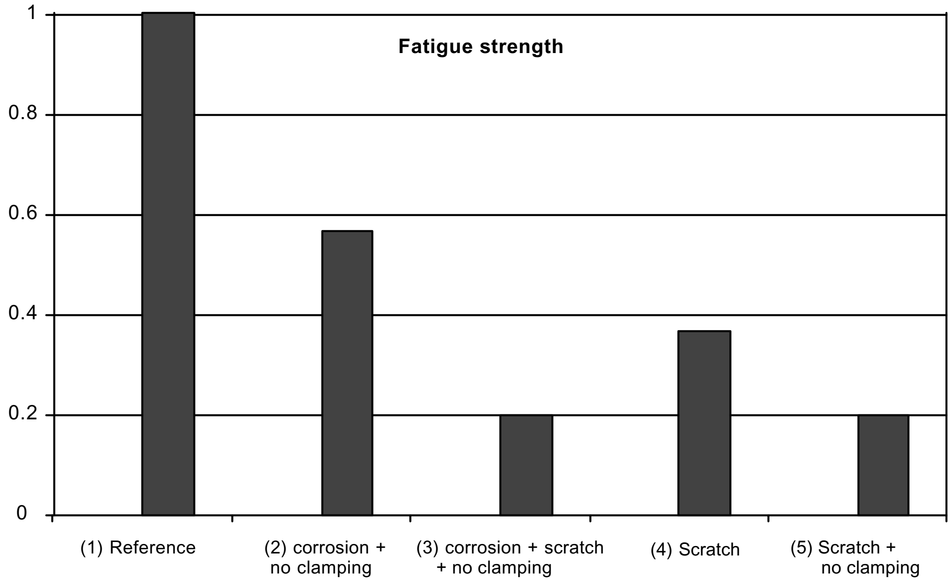

Figure 24 : Effect of combined flaws on joint specimens

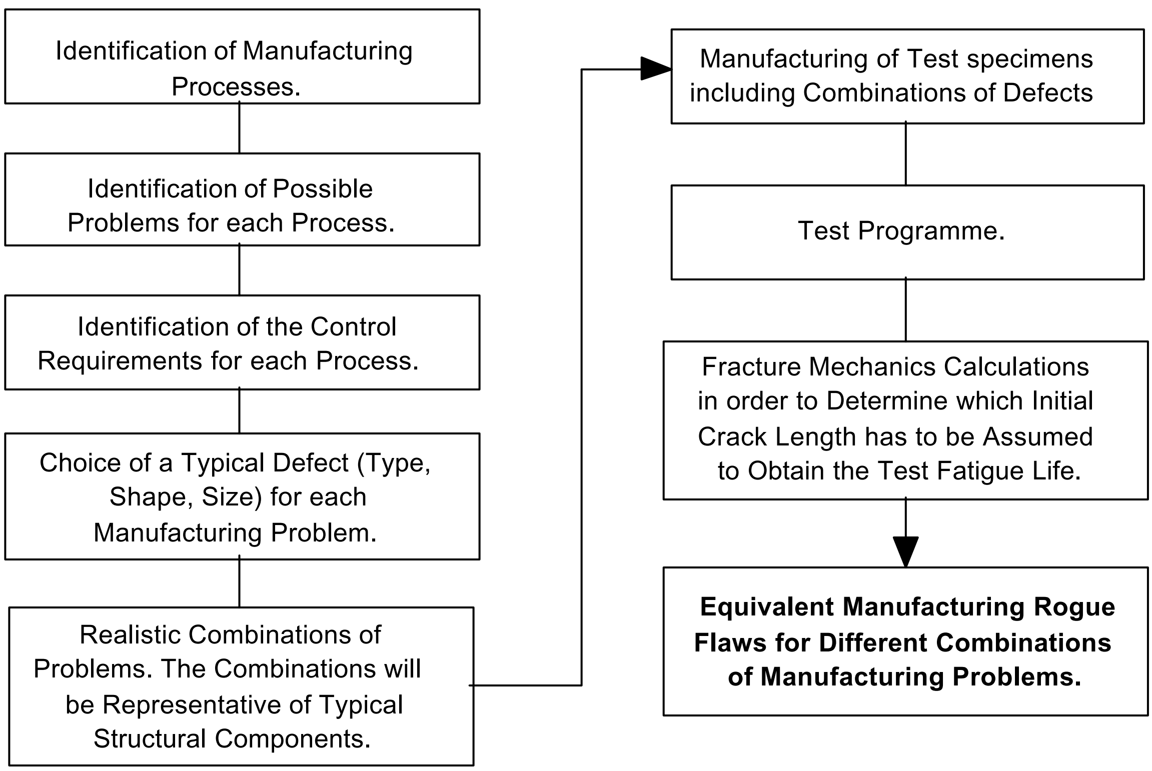

Figure 25 : Overall approach to address the issue of manufacturing rogue flaws

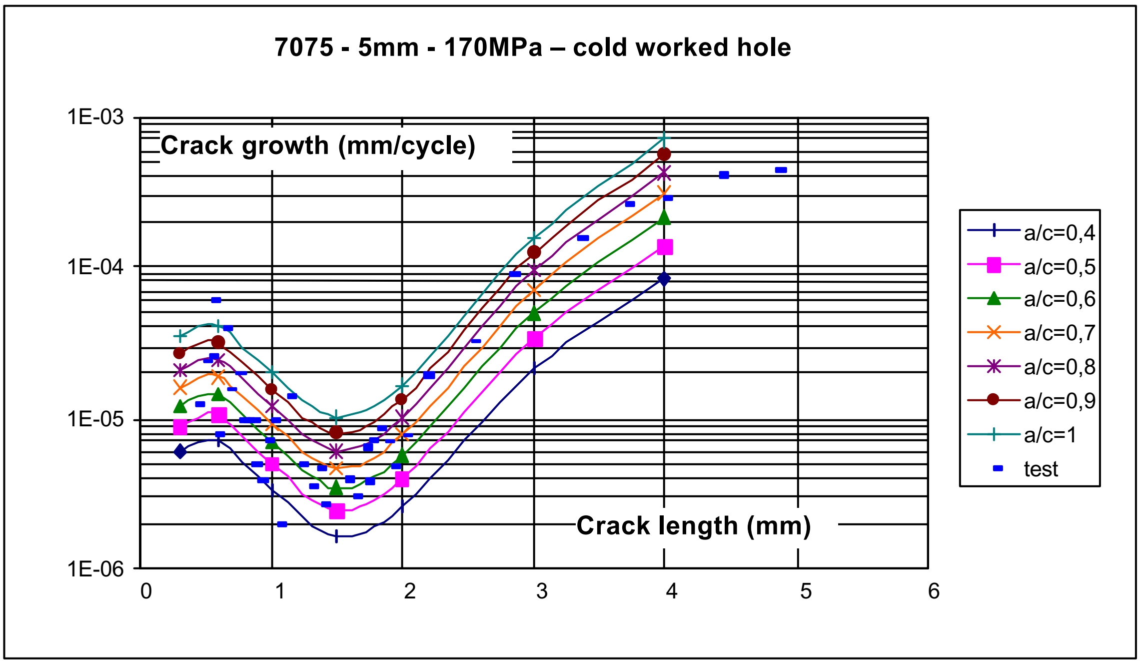

Figure 26 : Comparison test/prediction for a 4% cold worked hole

Figure 27 : Gazelle helicopter Full-scale test

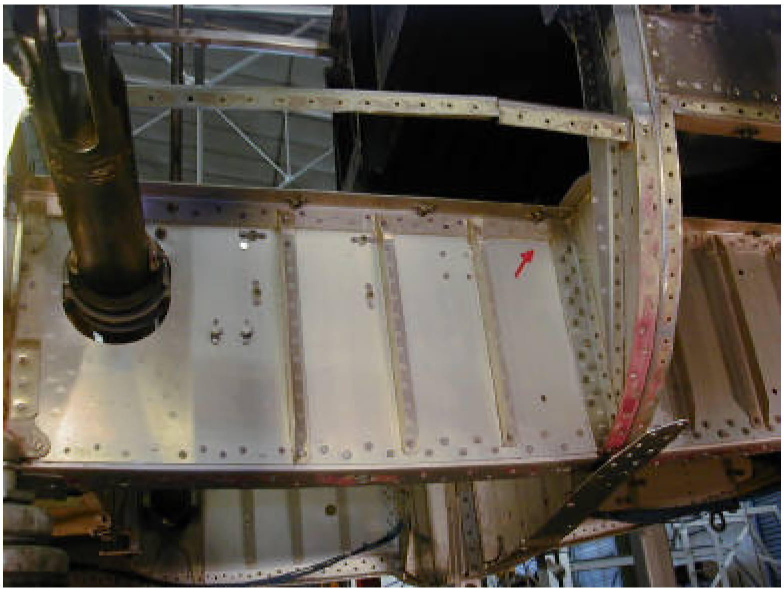

Figure 28 : Fatigue finding on the gazelle helicopter (cracks detected on the left longitudinal beam)

Figure 29 : Alphajet full-scale test

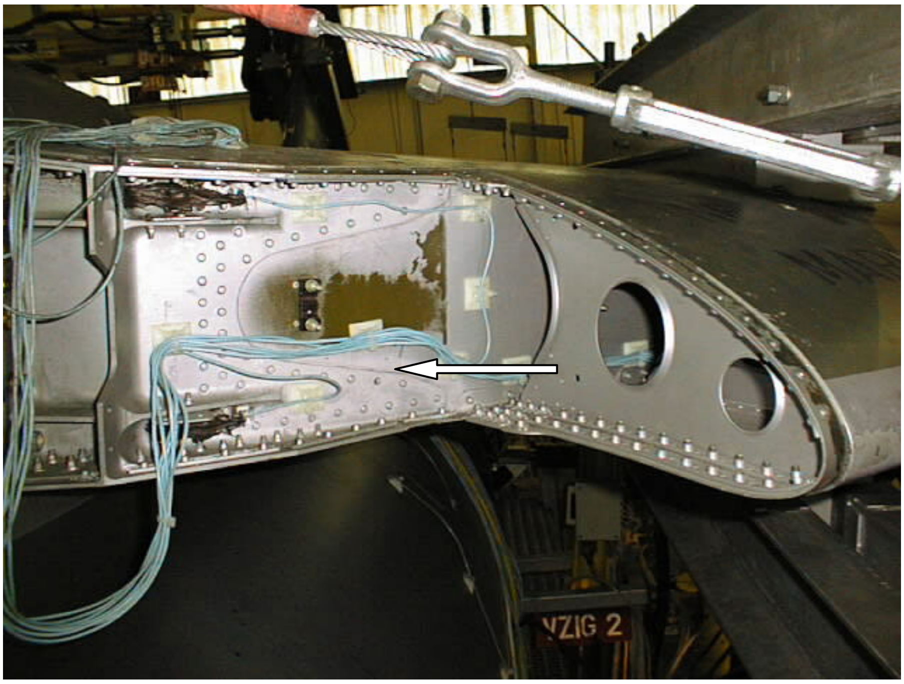

Figure 30 : Fatigue finding on the Alphajet fatigue test (front spar)

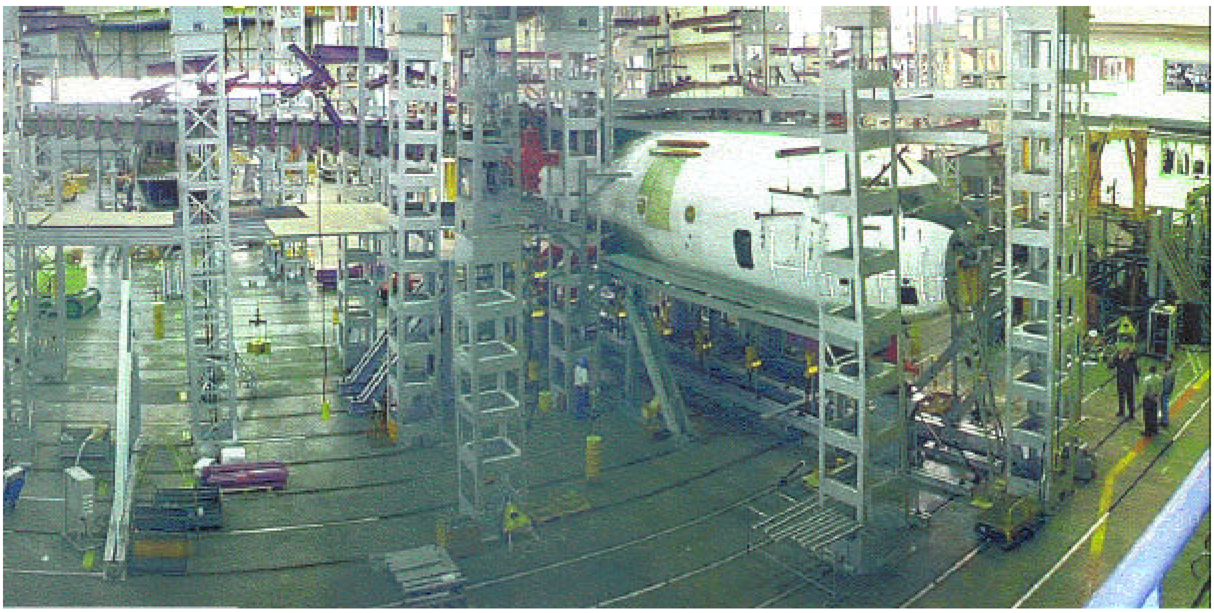

Figure 31 : The Transall Full-scale fatigue test