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Review of aeronautical fatigue investigations in France during the period May 2013- April 2015

TECHNICAL NOTE

N°15-DGATA-ST-870001-1 F-A

DGA

DIRECTION GENERALE DE L'ARMEMENT

DGA Techniques aéronautiques

DIRECTION GÉNÉRALE DE L'ARMEMENT

DGA Aeronautical Systems

Division : ST

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	NAME	Visa	Date	Title
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Abstract :

The present review, prepared for the purpose of the 34th ICAF conference to be held in Helsinki (Finland), on 1-2 June 2015, summarises works performed in France in the field of aeronautical fatigue, over the period May 2013-April 2015.

Topics are arranged from basic investigations up to full-scale fatigue tests.

References, when available, are mentioned at the end of each topic.

Correspondents who helped to collect the information needed for this review in their own organisations are :

- Bertrand Journet for Airbus Group Innovations
- Alain Santgerma and Alexis Falga for Airbus France
- Frédéric Desbordes for Dassault Aviation
- Vincent Bonnand for ONERA
- Etienne Deshaies for DGA Techniques aéronautiques.

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1. INTRODUCTION AND ACKNOWLEDGMENT

The present review, prepared for the purpose of the 34th ICAF conference to be held in Helsinki (Israel), on 1-2 June 2015, summarises works performed in France in the field of aeronautical fatigue and structure integrity, over the period May 2013-April 2015.

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- Vincent Bonnand for ONERA
- Etienne Deshaies for DGA Techniques aéronautiques.

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.

2. FATIGUE LIFE PREDICTION STUDIES AND FRACTURE MECHANICS

2.1. AN EXPERIMENTAL AND NUMERICAL INVESTIGATION OF THERMAL GRADIENT MECHANICAL FATIGUE IN THE CONTEXT OF COOLED TURBINE BLADE (ONERA)

A Thermal Gradient Mechanical Fatigue test (TGMF) facility was developed for nickel-based single crystal superalloy tubular specimens [1]. The objective was to reach a high thermal gradient in the 1mm wall thickness of the specimen to reproduce the stress state of a cooled high pressure turbine blade. Thermal gradient of the tubular hollow specimen is obtained by induction heating and a compressed air circuit for tube cooling including a thermal mass flow meter. Special attention was paid on the internal temperature measurement and its estimation. It was performed with a 3D Finite Element Method (FEM) based on a weak coupling between two ONERA codes used to calculate thermal field of the specimen. Thermal solid conduction problem is solved with the Zset suite and aero thermal turbulent calculation is conducted with CEDRE software.

The analyses were carried out in several steps increasing the complexity of thermomechanical loading, ranging from a constant thermal gradient up to a realistic engine mission, and also of the geometry of the specimens, considering a smooth specimen including or not a network of cooling holes. Depending of the studied configuration, thermal gradient is estimated 40°C/mm with a smooth surface and nearly 120°C/mm with a rough surface. Complex thermomechanical tests are simulated for validate the constitutive equations based on crystal viscoplasticity. Finally, the results given by the life prediction model, integrated stress gradient and size effect close to the holes are in good agreement with experiments, except when necking phenomena occurred because of the high applied stress.

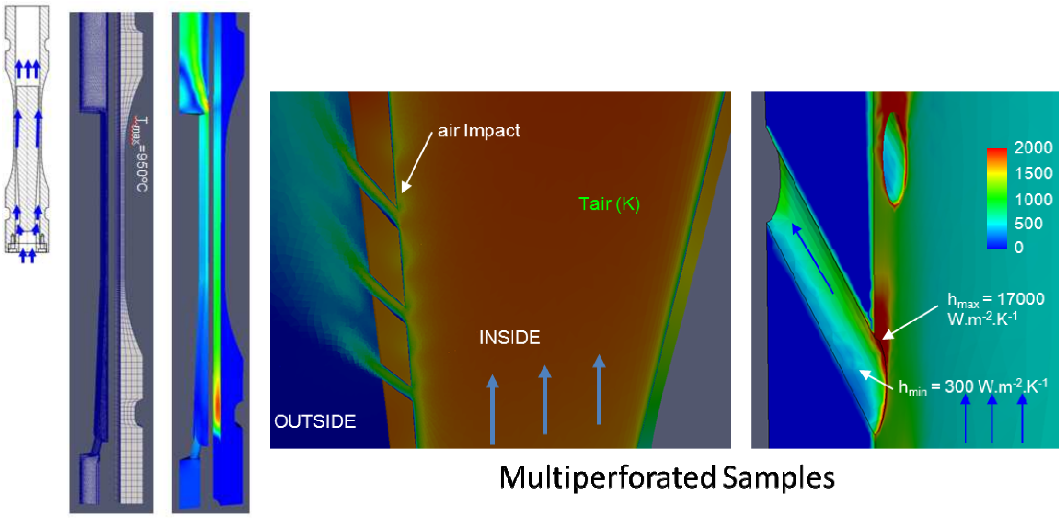


Figure 1: Thermal/Solid/Fluid coupled calculation: smooth specimen (left) and perforated specimen (right). Temperature contours and the convective heat transfer coefficient h governed by the fluid velocity

[1] R. Degeilh, Développement expérimental et modélisation d'un essai de fatigue avec gradient thermique de paroi pour application aube de turbine monocristalline, Thèse de Doctorat, ENS Cachan, 2013.

2.2. FATIGUE LIFE MODELLING OF A POWDER METALLURGY DISK SUPERALLOY WITH MICROSTRUCTURE EFFECTS (ONERA)

Polycrystalline γ/γ' nickel-based wrought superalloys are commonly used for the last high-pressure compressor (HPC) stages and for the low and high-pressure turbine (LPT and HPT) disks. Generally, lifetime assessment tools for such metallic structures involves a calculation of the structure by the finite element method or simplified approaches to determine the mechanical fields at any point in the component. Then the number of cycles to failure is computed using a fatigue damage model. These two steps are critical and require a high degree of consistency with each other. In order to propose a detailed description of the fatigue behaviour of a polycrystalline superalloy γ/γ', a precipitation model [1] was improved to predict the evolution of the size and the volume fraction of γ' secondary and tertiary precipitates as a function of the thermal history. The model was calibrated on four γ' particle distributions obtained through various in coarse grain size superalloy N18. A specific viscoplastic behaviour model is finally proposed for which the plastic threshold depends on the γ' precipitate distributions. The proposed model [2] provided a good description of the cyclic behaviour of the obtained different γ' particle distributions as shown in figure 1. Analysis of fracture surfaces of the performed fatigue tests indicated, as expected for the coarse grain N18, that cracks initiate at large crystallographic facets and not at pores or ceramic inclusions. Moreover, the effect of the microstructure on the mean stress appears to play a role in the fatigue life. The SWT fatigue criterion is finally extended and allows computing the number of cycles to failure in the disk (figure 2) from the strain and stress amplitudes and the mean stress at the stabilised cycle depending on the applied cooling paths and/or aging treatments. This proposed lifetime assessment tool enables at the end the optimisation of heat treatment on disk preform to improve the fatigue life of a disk.

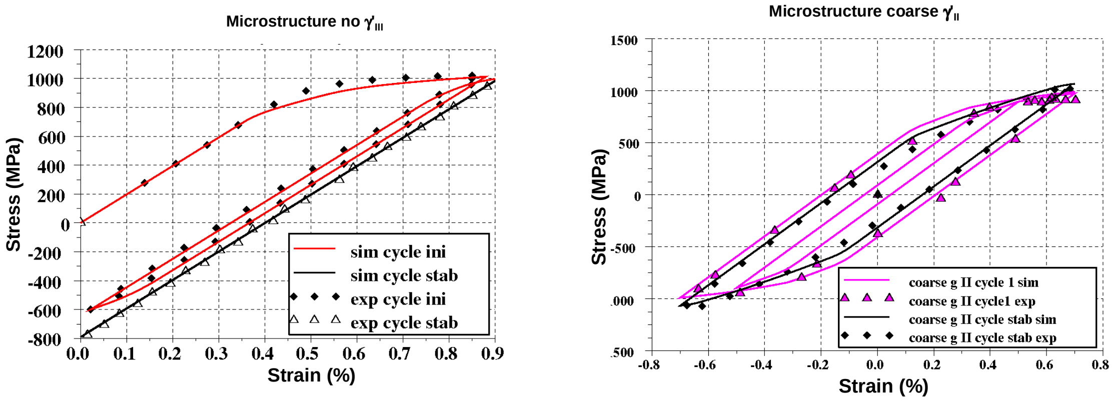
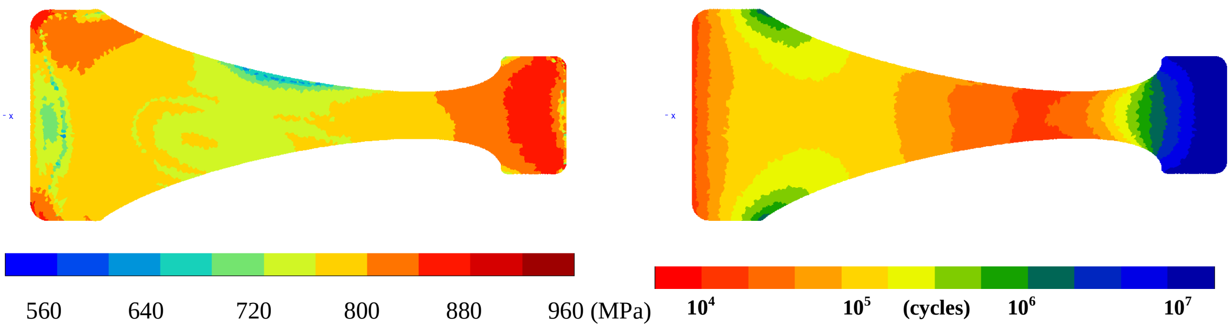


Figure 1: Influence of the precipitate distribution on the cyclic behaviour of coarse grain size superalloy N18 - Comparison of the model and the experimental response (on the first and stabilised cycle)



a) Plastic threshold stress in a disk after cooling

b) Fatigue life (90s triangular cycle between 50 and 27500 rpm)

Figure 2: Maps of the different steps of disk calculation from the heat treatment to fatigue life at 450°C (half cross-section, the rotation axis on the left)

[1] J. Park, D. Nelson - Evaluation of an energy-based approach and a critical plane approach for predicting constant amplitude multiaxial fatigue life, Int. J. Fatigue 22, pp. 23 – 39, 2000.

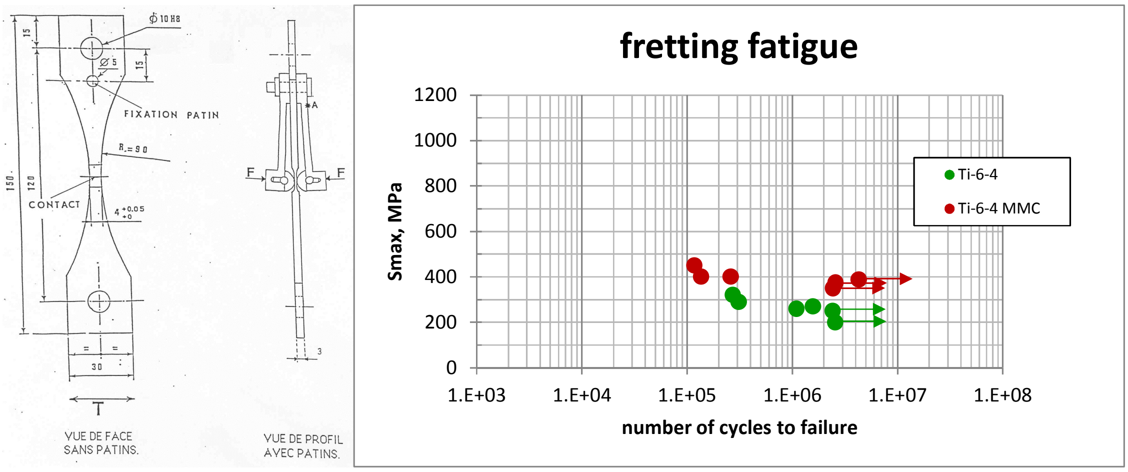
[2] G. Boittin - Expérimentation numérique pour l'aide à la spécification de la microstructure et des propriétés mécaniques d'un superalliage base Ni pour des applications moteurs, PhD Ecole Nationale Supérieure des Mines de Paris, 2011

2.3. FATIGUE WITH RESIDUAL STRESSES (AIRBUS GROUP INNOVATIONS)

The activity has just started. It deals with fatigue testing of technological coupons made out of a 7XXX aluminium alloy. Some of the coupons are pre-strained in order to simulate residual stresses. The purpose is to run representative tests of a new application case of Airbus Commercial Airplane where concentrated plastic strain might occur under some severe service loads and thus leaves tensile residual stresses. Technological coupons geometries have been defined with a stress concentration area. Finite element calculations have been run in order to simulate the pre-straining of the coupons in order to set-up a representative profile of residual stresses in the stress concentration area. Spectrum fatigue tests are under way. Crack initiation is monitored using a remote controlled camera.

2.4. FRETTING FATIGUE (AIRBUS GROUP INNOVATIONS)

In the frame of French public funded project COMETTi (*) and partnering with Airbus Helicopters, a titanium Ti-6-4 matrix composite with titanium carbide particles has been developed for rotor application. The fretting fatigue behavior of the MMC has been evaluated using a laboratory test set-up. A flat fatigue specimen is clamped at its center by two fretting pads. The application of the fatigue load on the fatigue specimen induces fretting wear between the pads and the center of the specimen where failure ultimately takes place. The fretting fatigue performance of the MMC is just above that of a standard plain Ti-6-4 alloy. Modelling work to predict the fatigue lives is under way.



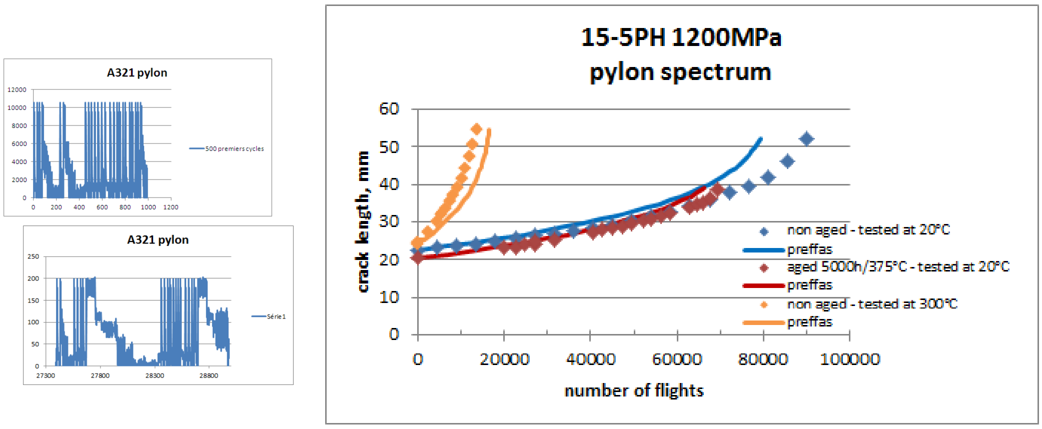
Fretting fatigue specimen set-up and test results.

(*) the funding of the French National Research Agency is acknowledged. Partners in COMETTi are: Airbus Group Innovations (coordinator), Airbus Helicopters, SMS Materials and Structures Science department of Saint-Etienne School of Mines, LTDS Tribology and Dynamics of Systems department of Ecole Central de Lyon, LMI Multimaterials and Interfaces department of University of Lyon 1, Mecachrome, HBP.

2.5. FATIGUE CRACK GROWTH UNDER SERVICE APPLICATION (AIRBUS GROUP INNOVATIONS)

In the frame of French public funded project PREVISIA (**), a martensitic stainless steel 15-5PH is evaluated in damage tolerance for pylon application. The main issue deals with ageing due to long exposure to temperatures of 300 to 400°C. The fatigue crack growth (FCG) rate at 300°C of the non-aged steel is increased compared to that of ambient temperature. Ageing 5000 hours at 375°C does not alter the FCG rates at ambient temperature, except for the end of the curve because of a lower fracture toughness of the aged alloy. During ageing, spinodal decomposition of the steel has been monitored and observed to be closely related to the evolution of the mechanical properties.

The PREFFAS methodology has been applied either on the non-aged material tested at room temperature and 300°C and on the aged material tested at RT. The figure shows the experimental behaviour under pylon flight spectrum loading in each case and the predictions of crack growth made with PREFFAS model. The tested coupon is a CT specimen.



Fatigue crack growth on 15-5PH under pylon flight spectrum loading and PREFFAS calculations.
Non-aged material tested at RT and 300°C, 5000h/300°C aged material tested at RT.

(**) the funding of the French National Research Agency is acknowledged. Partners in PREVISIA are: Airbus Group Innovations (coordinator), Airbus Commercial Airplanes, SIMaP Materials and Processing Science and Engineering department of the National Polytechnic Institute of Grenoble, CIRIMAT Materials Research and Engineering department of the National Polytechnic Institute of Toulouse, PPRIME Materials Physics and Mechanics department of ENSMA, LMT Mechanics and Technology laboratory of Ecole Normale Supérieure of Cachan, Aubert & Duval.

This application case is part of next topic.

2.6. FATIGUE CRACK GROWTH PREDICTION USING PREFFAS MODEL (AIRBUS GROUP INNOVATIONS)

The certification of structures within the damage tolerance philosophy needs predictions of fatigue crack growth under spectrum loading. For this purpose, Airbus Group has developed a model called PREFFAS. Since its first development in the early eighties within the former Aerospatiale organization, the model has always been tested and implemented to deal with various load spectra or fracture mechanics features. It is the purpose of this paper to present some of the related case studies that led to develop additional schemes or to adapt the methodology according to the service application. The paper first recalls the basics of the model and the methodology to apply in order to determine the model parameters. Then several cases studies dealing with flight spectrum including compressive loads or residual stresses, or dealing with isothermal temperature superimposed to the load spectrum will be presented. The model is based on the crack closure concept to account for the fatigue crack growth behavior and for the load spectrum effects. With compressive loads, the crack retardation effect is reduced. The analysis has been reworked dealing directly with stresses in the closure computation. Residual stresses due to welding may affect the fatigue crack growth behavior. In that case the load spectrum has been reworked with equivalent remote residual stress to have the model implement its original scheme. With temperature and/or ageing effect, the methodology to identify the model parameters is adapted to the case (PREVISIA example as above). The concerned materials in the presented case studies are aluminum alloys and steels.

One of the cases deals with representative structural application such as stiffened panels made out of aluminum alloys and welded stringers. After validation of the modelling scheme, the model is used to compare and assess, in a prospective manner, the damage tolerance behavior of different panel technologies (with welded or integral stringers).

As conclusion, the work performed recalls the features of the model together with the applications cases and its limitations. The idea to try and apply the original scheme as much as possible has allowed not to increase the number of modelling parameters. The presented works have been undertaken by Airbus Group Innovations through internal or public funded projects.

2.7. RANDOM VIBRATIONS FATIGUE ANALYSIS AND TESTS (DGA AERONAUTICAL SYSTEMS)

DGA is in charge of qualification aspects and continued airworthiness for military aircrafts. In this context, DGA have to perform tests for the qualification of equipment in vibrations environment. But tests are expensive and can take a long time to do, relatively to operational issues. The objective of this study is not to replace all the tests by calculations – many data can only be obtained by test for now, damping for example – but to be able to anticipate a test in order to reduce hazard of non-predicted failures or to extend a structural qualification from a vibrations environment substantiated by test to another one lightly different. Moreover, some subjects are raised about crack initiations which cannot be explained by “classical” fatigue models, especially for rotorcrafts, because of the type of excitation: vibrations. So, with this study, DGA could also improve its knowledge about vibration fatigue in order to explain and to bring solutions to technical facts relative to this particular subject.

Indeed, vibrations environments are characterized by low stress levels, high frequencies (many fatigue cycles) and sometimes random spectra. These characteristics lead to some analysis difficulties: to get an initiation with low stress levels and to be able to count cycles for random spectra. The aim of this study was to improve DGA knowledge about random vibrations fatigue by evaluating some specific models from literature. They were assessed by two different approaches: numerical and experimental on coupons. The objective was to answer to some questions such as: is the model accurate? Is it conservative? How many parameters are needed? Another part of this study was about the ability of using this model for real structure.

Damage calculations for random vibrations environment are based on “classical” damage calculations. The main difference and difficulty is the way to “count” cycles, because cycles don’t exist anymore. So, it is needed to estimate a probability density function of stress from the power spectral density of the excitation.

$$E[D] = \int \frac{S^b}{C}.E[P].f_S(S).dS$$

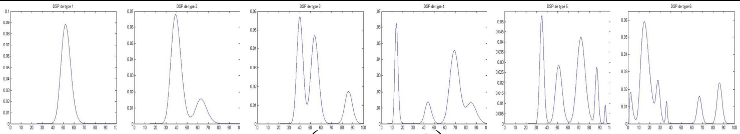
- $f_S(S)$ represents the probability density function of having extracted from the signal an elementary cycle with an amplitude S.

- $E[P]$ is the average number of peaks of the signal per unit of time.

- $\frac{S^b}{C}$ represents the damage of an S amplitude cycle when the Basquin model is used to approximate the Wöhler‘s curve.

The different models from literature give different probability density functions. The first one, the Narrow band model, is purely theoretical and well none but, as its names suggests, it isn’t adapted to wide band signal. The other models are split in two types. The first ones are based on the Narrow band model: they add a corrective factor, which is often empirical, to correct and improve the Narrow band model. The second ones are direct models.

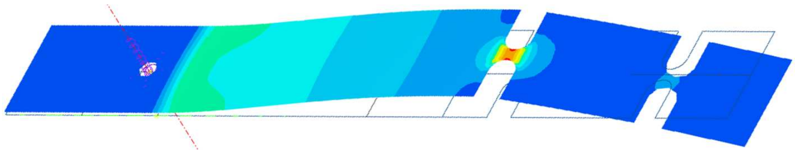
The numerical assessment of these models was made by comparison with “classical” damage calculations. More than 600 power spectral densities were generated at random on the [0 ; 100] Hz band.



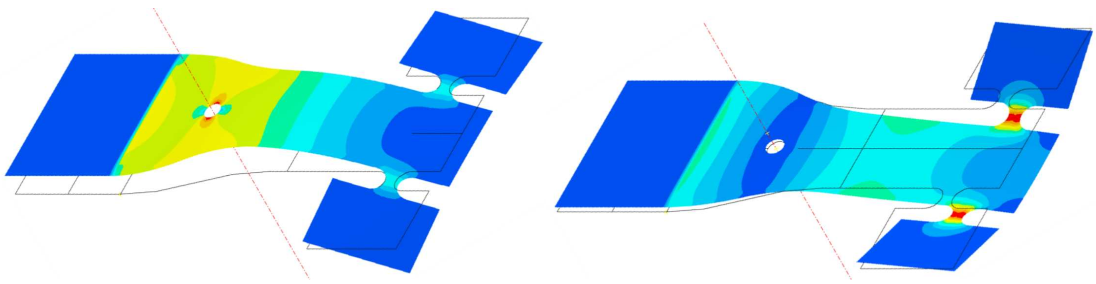
$$S(f) = \sum_{i=1}^{n} A_i \times \exp\left(-\frac{(f - f_{ci})^2}{B_i}\right) \quad with \;\; n = 1..6$$

For each one, many corresponding temporal signal were also generated then the damage was calculated for each signal by “classical” method (Rainflow counting and Miner’s rule were used). The average of all these calculations (for each power spectral density) was the reference. The damage was also calculated directly from the power spectral densities with the different previous models. A comparison of the results led to assess their accuracy, scattering and conservatism.

The aim of the second part of the study was to assess these models by confronting them to test results: some tests were made with a vibrating pot on two types of coupons. To be able to assess properly the models, the coupons had to be "wide band". Indeed, the standards for aircraft and rotorcraft, which were study in the frame of this work, present generally random vibrations on a wide band and some typical frequencies excitations corresponding to rotor or blades speed. In fact, a wide band response for "simple" coupons was quite difficult to obtain, that is why they were designed to obtain a transfer function with two natural mechanical frequencies. The first design of coupon was found in literature and has only one critical point. The second one was "home-made" by DGA: the design was optimized to obtain two different critical points, two resonance frequencies and a fixed predicted test duration.



First design of coupon

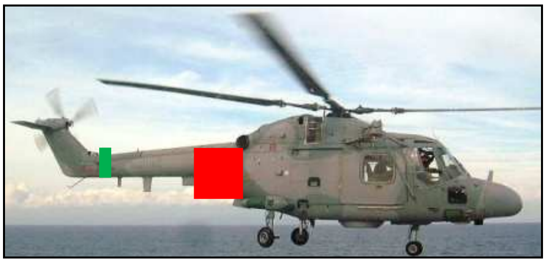


Second design of coupon

This optimization and the random excitation definition were made thanks to finite elements modelling (FEM) of the coupons and damage calculations with the different previous models. Finally, the first coupons tests enabled to adjust the complete model (FEM and damage calculation) and the second ones enabled to compare the models and to assess the possibility to make this type of damage prediction.

2.8. DAMAGE TOLERANCE APPROACH ON RECURRENT DAMAGES OF LYNX HELICOPTERS (DGA AERONAUTICAL SYSTEMS)

The Lynx fleet of French Navy is ageing and shows poor availability. This is partly due to the findings of recurrent damage on some structural parts of the airframe. Two different zones can especially be pointed out because combining high damage rates and long repair times: the first one is the frame which is at the junction between tail boom and fuselage. The existing inspection program is not based on a damage tolerance approach and the study could maximize the inspection interval. The second zone is the lateral panels of the helicopter. Currently, Lynx helicopters are not allowed to flight with damages on panels.



Zoning of recurrent damages observed on Lynx helicopters

The establishment of a damage tolerance approach on these specific areas could extend the fleet availability. A study was then launched which aims at defining:

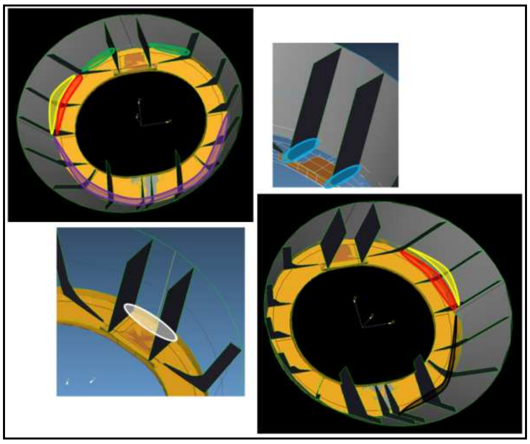
- By zone, an acceptable size of damage which enables to keep the aircraft in service

- An inspection program to be triggered in case of damage.

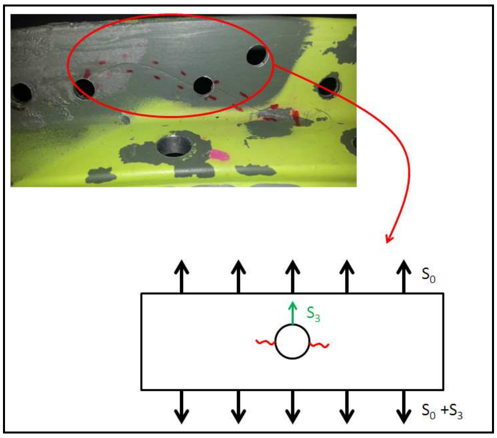
This enables, in some cases, to postpone the repair of detected damages until next adapted scheduled maintenance. In addition, a criterion which authorizes a single ferry flight is also defined.

In both cases, to be able to assess residual strength of the considered damages, flight and ground loads of affected zones have to be assessed. For this purpose, two flight tests campaigns were conducted on the Naval Air Base of Hyères (France). The tested airframe was equipped with strain gauges and numerous manoeuvers and landing were performed. Then, limit loads for each studied area was derived from these flight tests. Finite Elements Models were also used in order to extend the results of the measurement to damaged zones. Finally, a limit load and ultimate load have been identified for each zone without altering flight safety.

The methodology to determine acceptable damage sizes at the frame joining the tail section with the remaining Lynx helicopter (frame 3353A) airframe was the following. Fatigue cracks locations on the frame (Aluminium 2014-T6) were identified both from maintenance reports and from most loaded area of finite element models. Due to complex loading, flight tests required 36 strain gauges, placed all around the frame. The most loading flight manoeuvres were confirmed: yawing manoeuvres and deck-landings. Seven damages locations were studied. Finite elements model enables to obtain local stresses.



a) The seven studied areas

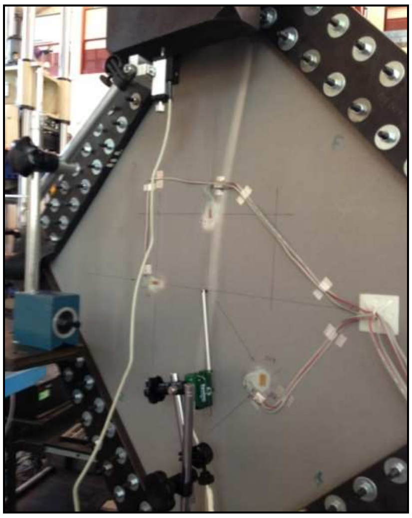


b) Simplified model of crack propagation

These results were introduced in Afgrow software simplified model (double through crack) in order to estimate the critical crack size for each zone of the field. Then crack propagation calculation with relevant safety factors gives the inspection interval for each studied zone, which depends on the damage detected size.

Lateral panels are made of sandwich structure of Aluminium 2024 skin and Nomex honeycomb with thin skins. Two kinds of damages were observed: disbonds and indents damages. Currently, no criterion allows Lynx helicopters to flight with these damages. For this zone, around 30 strain gauges were placed on the lateral panels of the tested Lynx helicopter: inner and outer sides. Shear and compression stress were studied. Thanks to flight tests, limit load of each studied lateral panel was determined.

As no robust method exists to determine residual stress which those kinds of damaged structure, test campaigns were launch to assess allowable values. The design of the specimens depends on the tests: a 400*400 mm compression specimens and 700*700 mm shear specimens were used. Five sizes of damage were investigated, from 50 mm to 150 mm diameter in static but also in fatigue to verify the no-growth.



a) Shear mechanical test

b) Compression mechanical test

Although mainly based on test results, finite elements models complete the investigation on lateral panels to better understand failure modes and scattering on results. First results give a good behaviour of damaged panels. Whatever kinds of damage, all tested specimens have a failure after ultimate load. The failure mode is a kind of general buckling: crimping mode. Moreover, no-growth of damage is observed on fatigue specimens. For each zone, an operationally acceptable damage size is defined (depending on kind of damages).

Damage tolerance approach could be applied to both zones. Once accepted by airworthiness authorities, it will enable consequent improve in fleet availability.

2.9. RAFALE FIGHTER - FATIGUE AND DAMAGE TOLERANCE TESTS (DASSAULT AVIATION & DGA AERONAUTICAL SYSTEMS)

In order to improve aircraft durability and to prepare life extension programme, a study has been launched to assess specific design features, locations or loading conditions. Elementary coupons and technological specimens will be tested at the DGA Aeronautical Systems premises to enhance the accuracy of fatigue calculations for particular cases encountered on Rafale fighter :

• Effect of the thickness plate on the Damage Tolerance : typical parts are main frames. Residual strength capability versus local thickness is assessed.

• Propagation under complex spectrum : typical areas are fuselage section areas loaded in flight but also during deck-landing and catapulting.

• Crack initiation with spectrum including high level compression cycles : the compression cycles are mainly due to deck-landing and catapulting load cases

• Crack initiation on dissymmetrical junctions : typical joints between thin skin and thick frame flange.

• Crack initiation of Titane bolts : topic proposed following some previous test findings.

3. DEMONSTRATION OF COMPLIANCE TO AIRWORTHINESS REGULATION

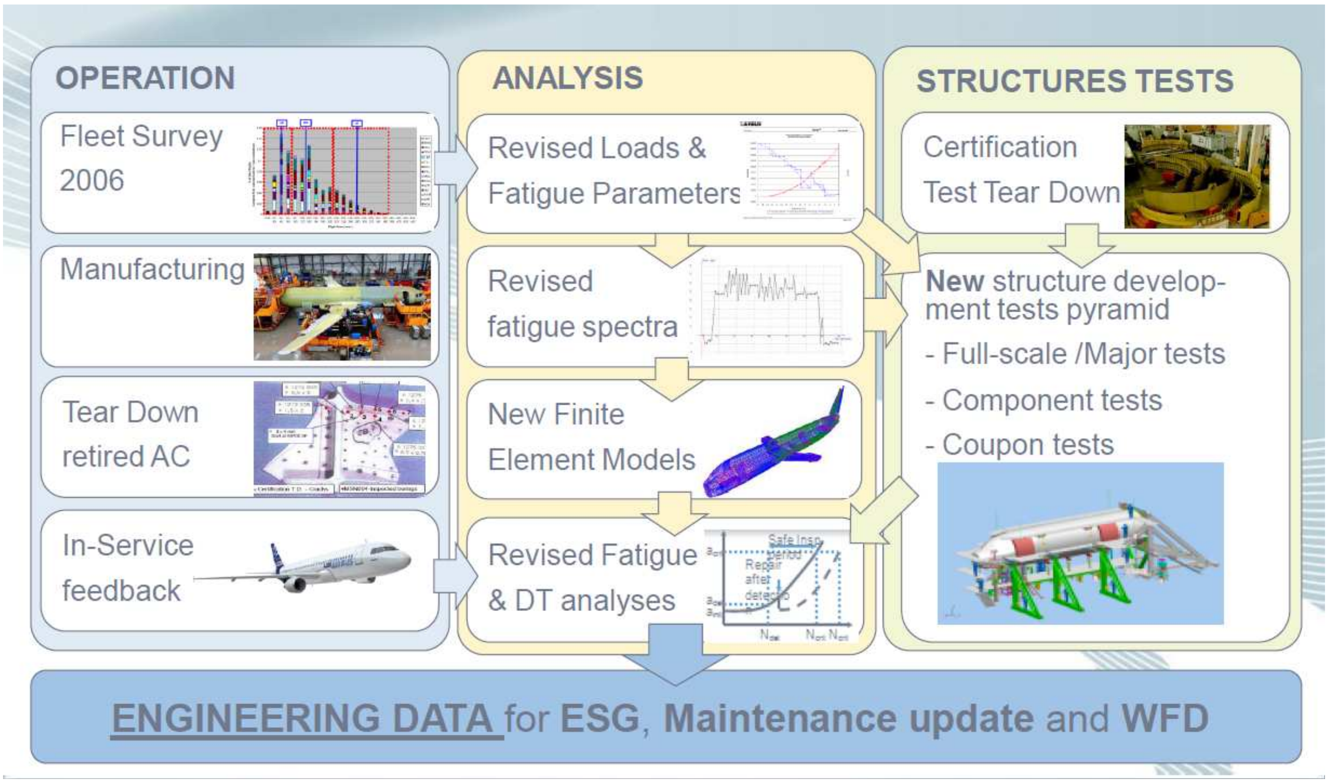
3.1. STATUS ON FAR 26 COMPLIANCE (AIRBUS)

The FAR §26.21 introduced the requirement to establish a Limit Of Validity (LOV) for each A/C models, and to demonstrate that Widespread Fatigue Damage (WFD) will be precluded up to the LOV.

Three AIRBUS families had to show compliance with FAR §26.21 in January 2015: the A300/310, A330/340, and A320 families.

The targeted LOV have been set as a compromise between the expected fleet usage and the achievable value based on the means already available.

The A320 family took full benefit of the New Full-Scale Fatigue Tests performed (see paper in previous ICAF), whereas the original Full-Scale Fatigue tests supported the evaluation of A300/310 and A330/340. The test results were complemented and supported by theoretical analyses. In-service experience was also taken into account to establish the service actions required to preclude WFD.



Consistency in the justification approach was ensured by the application of a common WFD Stress Policy. This WFD Stress Policy was built in 2011, and deeply reviewed with FAA and EASA. It was eventually agreed between all parties end 2012 for application to AIRBUS models.

The FAR §26.21 compliance demonstration requested a huge engineering work over the last 3 years. Dedicated Plateau was put in place gathering together Core team and Fuselage stress teams to ensure full consistency in the application.

All the compliance data have been provided to the FAA and EASA on 14 January 2015, which was the compliance date. AIRBUS is now looking forward for the Approval of the revised maintenance programs. Implementation of these new maintenance programs by the airlines will start from 14 January 2016. Feedback / approval from FAA/EASA is pending for the fleet mentioned.

Next steps are now the compliance for next models with the following schedule :

- January 2016: A320 NEO and A320 Sharklet (work in progress).
- January 2019: A380 (work starts now, EF tear-down still in progress)
- After 2020: A350 (EF test in progress)

4. FULL-SCALE FATIGUE TESTS, LIFE EXTENSION, FLEET MONITORING & MANAGEMENT

4.1. FATIGUE CRACK GROWTH APPROACH FOR FLEET MONITORING (DGA AERONAUTICAL SYSTEMS)

Military aircraft are subject to unpredictable variable amplitude loadings. Fighter aircraft structures are essentially sized by operational loads, which vary widely according to the missions flown by the aircraft. For safety reasons, it is necessary to be able to estimate each aircraft's actual fatigue "consumption" relative to its potential. For this purpose, most French military aircraft are equipped with load monitoring systems (flight parameter recorder or g-counters). These systems give direct or indirect access to the service loads.

The data is processed for each aircraft throughout its lifetime. The cumulative fatigue damage is calculated at various points of the structure pointed out as being critical, notably during full-scale fatigue tests. This information enables the Armed Forces to optimize the fleet management in terms of structure potential. Also, for fleet life-extension purposes and usage predictions, the use of more precise damage and crack growth prediction models is a major concern.

However, the fatigue consumption is up to now controlled by initiation models while during the fatigue life test and the operational life later, fatigue cracks are usually detected. The initiation models are yet insufficient to cover the entire lifetime.

Studies are conducted concerning the use of both initiation and propagation models to control the fatigue consumption. For the aircraft equipped with g-counters with no temporal data, only initiation models can be used to control the crack propagation. The prediction qualities of this method are analyzed. Furthermore, propagation models are investigated for the fleet providing temporal data.

Experiments are carried out on standardized compact tensile specimens CT40 B18 made of 2024A-T351 aluminum alloy.

Complex spectrum are applied to test the models. These kinds of spectrum are the well-known FALSTAFF SC, miniTWIST SC, or more specific ALPHAJET inspired from the full-scale fatigue test or a combination of g-acceleration inspired from real M2000 flights.

For each load listed above, three results are obtained:

- The damage calculation at a critical point for each sequence;
- A theoretical crack propagation curve calculated with the PREFASS model;
- A real crack propagation curve given by the test on the CT sample.

The damage calculation is the basis of the creation of the fatigue index (FI) which is used to control the fatigue life. FI is calculated as follow:

$$FI = \frac{E \times 100}{Safety\_Factor}$$

Where E is the classical cumulative damage value calculated at a critical point. The value is found with a rainflow cycle counting multiplied by an elementary damage matrix obtained from Wohler's curve.

Table 2 : PREFASS parameters for 2024A T351, 18 mm thickness.

a	b	C_{Eff} (mm / cycle)	m
0.398	0.602	1.46E-7	3.41

In practice, the crack propagation is mastered by the use of a propagation index (PI), which represents the severity of the aircraft use. A reference load is defined and crack propagation is calculated with the propagation model. Then, when the temporal data are available, the crack propagation is calculated with the "real" load and compared to the reference curve. PI is calculated as follow :

$$PI = \frac{100 \times \Delta N \times Safety\_Factor}{\Delta N_{adm}}$$

Where

- ΔN_{adm} is the critical number of flights corresponding to the crack critical size minus the detectable crack size;

- ΔN is the current number of flights read on the reference curve so that $a\_current = a\_referent$

- In this study, the reference curve is the full-scale fatigue test Alphajet.

The treatment of these values enables to draw the graphs below:

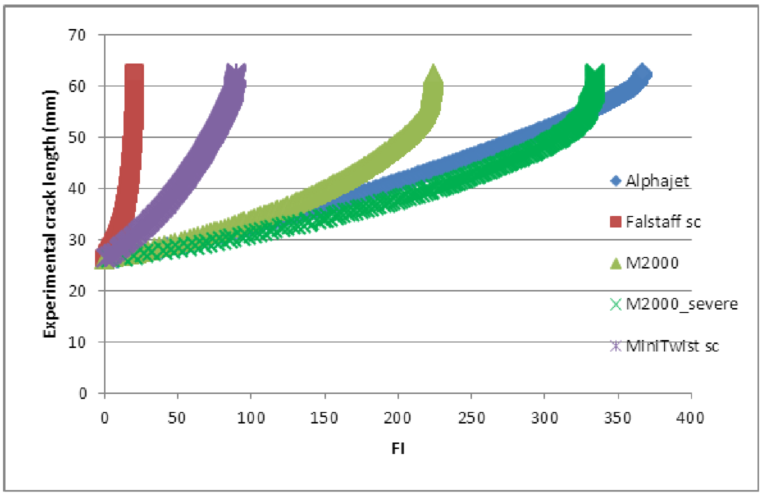


Figure 1: Experimental crack length versus FI

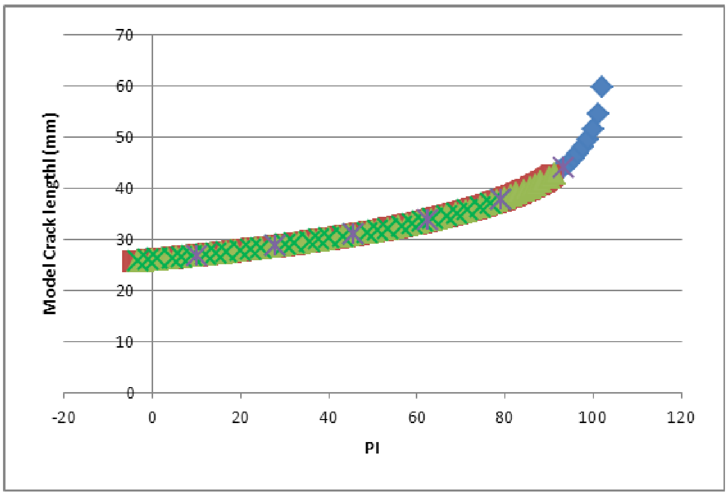


Figure 2: Model crack length versus PI

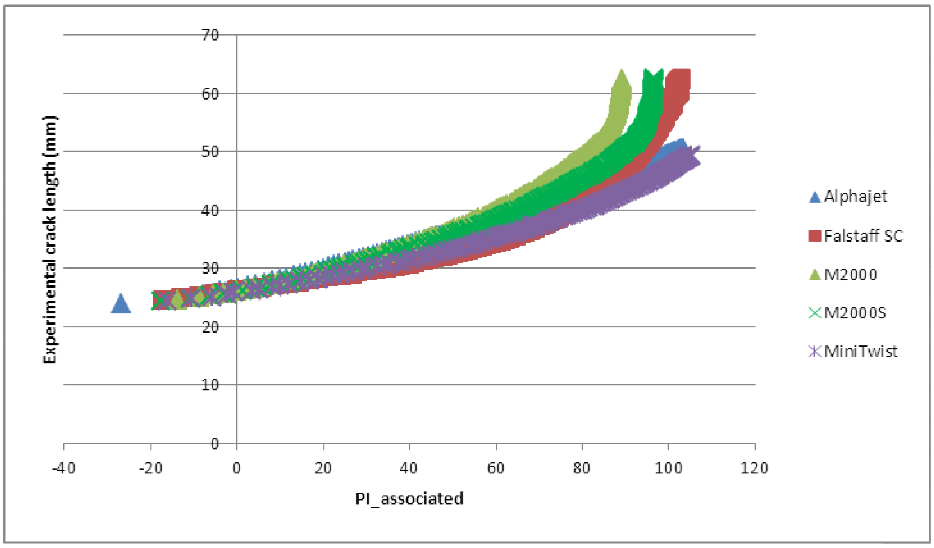


Figure 3: Experimental crack length versus associated PI

The use of a fatigue index is essential to follow the structure damaging of a military aircraft by analyzing the real loading spectrum, but it is shown at figure 1 that it is not necessary relevant to predict the propagation time and to define frequency of inspections.

The figure 2 shows the way the propagation index is defined leads a unique curve: all the curves are projected on the reference curve Alphajet.

As a consequence, by defining an associated propagation index so that flight numbers simulated with the reference curves are the same as the ones simulated with the real complex spectrum, the figure 3 shows it is possible to create an indicator on which it is possible to build a maintenance scheme. The impact of the reference curve and the propagation model predictions on the quality of this index still have to be analyzed.

The issue still exists concerning the fleet with no temporal data. Therefore, the main goal is to build an equivalent IP to control the crack propagation process. The next step will be to propose something for the fleet with no temporal data, but only g-counter and regular computations. To build an equivalent PI, and thus monitor the crack propagation process, the approach will be to make regular computation with partial exceedance numbers. The right period will be determined as a compromise between not too long period to have temporal data but long enough to minimize the influence of extrapolation and spectrum closure.

4.2. THE A400M USAGE MONITORING FUNCTION (AIRBUS)

The A400M is a versatile military transport aircraft, which is expected to cover very different types of usages during its service life. It has been designed, certified, and a maintenance programme has been established using a mix of different types of fatigue missions. Once the aircraft will be in service, a Usage Monitoring Function will be used as a mean among others to estimate the fatigue damage accumulated by each aircraft according to its actual usage.

This Usage Monitoring Function includes 3 complementary capabilities: Indirect Measurement Function, is based on the recording of parameters by the Systems of the Aircraft, which are then used to rebuild loads and stress histories at typical structure locations and to estimate fatigue and crack growth damage. The Statistical Usage provides statistics on fatigue driving parameters derived from the actual usage of each individual aircraft. Direct Measurement Function, installed only on some aircraft, includes strain sensors allowing Operational Loads monitoring and cross-checking of estimated loads and stress histories.

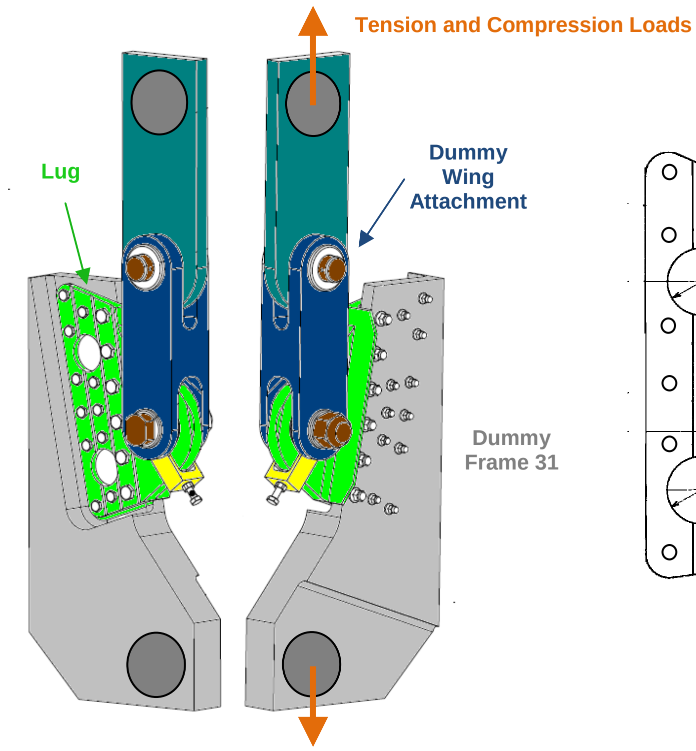
Dedicated method and algorithm have been developed to estimate damage as part of Indirect Measurement Function. Validation of output results has been performed comparing estimated loads and stress histories with actual measurements Flight Tests and Full-Scale test.

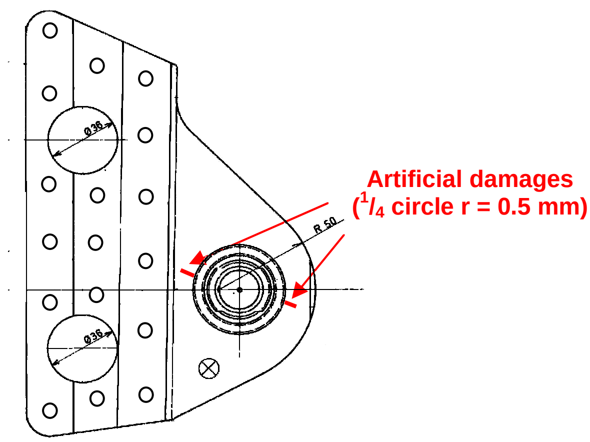
The Usage Monitoring Function will be used in the frame of A400M Continued Airworthiness, as a mean among others to help ensuring structure integrity all along aircraft life thanks to inspection periodicities appropriate for the actual usage of each aircraft and to help the Air Forces to manage their fleet cost effectively by judicious use of operations.

4.3. MIRAGE 2000 FIGHTER DAMAGE TOLERANCE TEST (DASSAULT AVIATION)

Following an in-service event, a fatigue test on the fuselage lug of the Wing to Fuselage Secondary Attachment has been carried out in the Dassault Aviation Test Laboratory.

Artificial damages have been created at the most critical locations of the lug. These damages were representative of pitting corrosion induced damage.





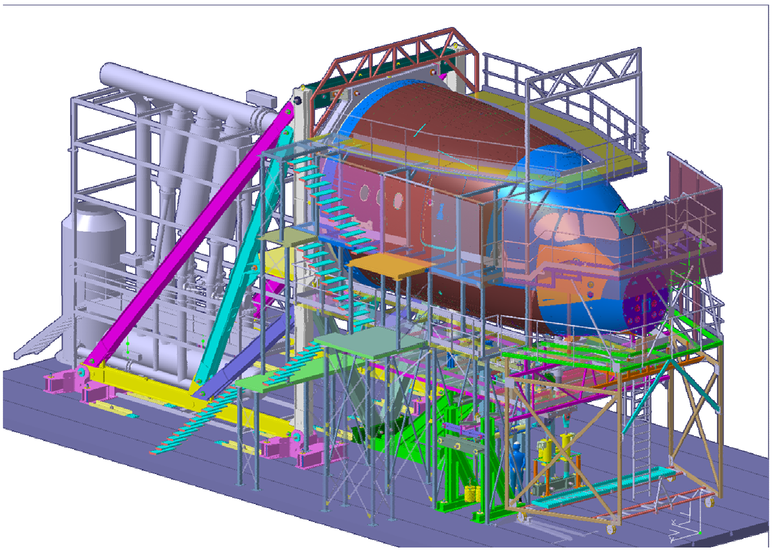
During the test, these damages have been followed up by NDT (US and Eddy Current): more than 10 fatigue lives of 7500 FH have been applied, without any propagation.

Very good Damage Tolerance of this structural element is thus demonstrated as no inspection is needed in addition to the ones required in the framework of the scheduled maintenance.

4.4. AIRBUS FULL-SCALE FATIGUE TESTING (AIRBUS AND DGA AERONAUTICAL SYSTEMS)

A350 fatigue test consists in 3 different test airframes called EF1 (forward section), EF2 (mid section) and EF3 (aft section). DGA Aeronautical Systems was selected by Airbus to perform the EF1fatigue test. Test started in September 2013 for 3DSG.

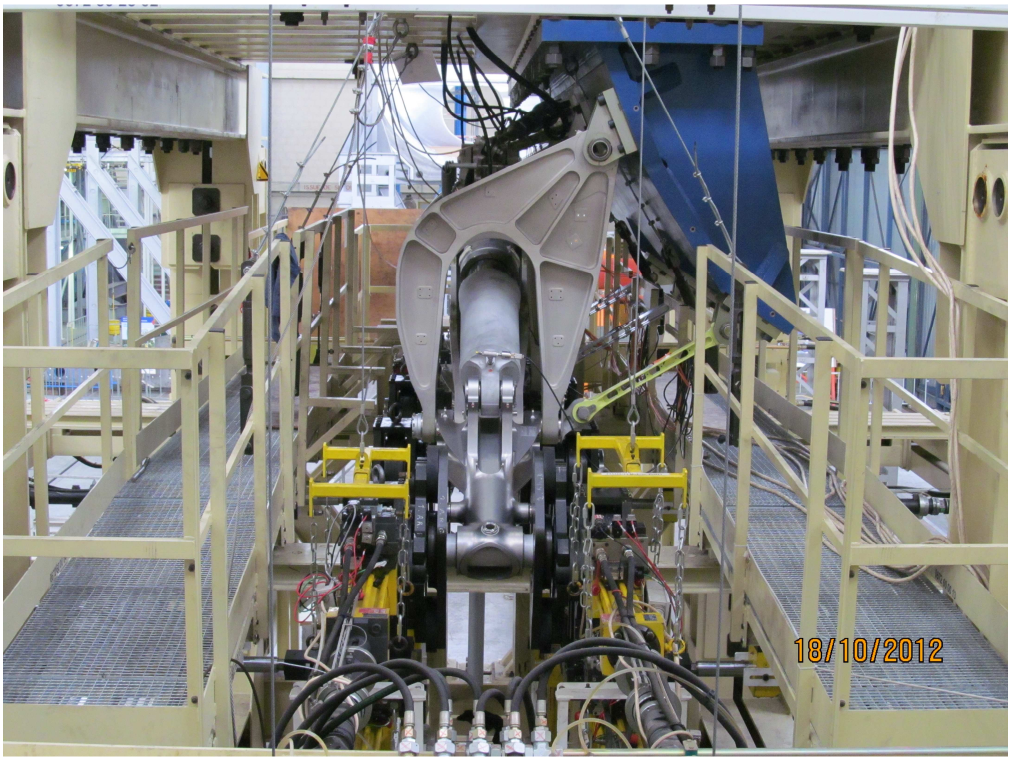
As an hybrid structure, thermal tests was performed to assess thermal stresses in the areas of composite-aluminium joints.



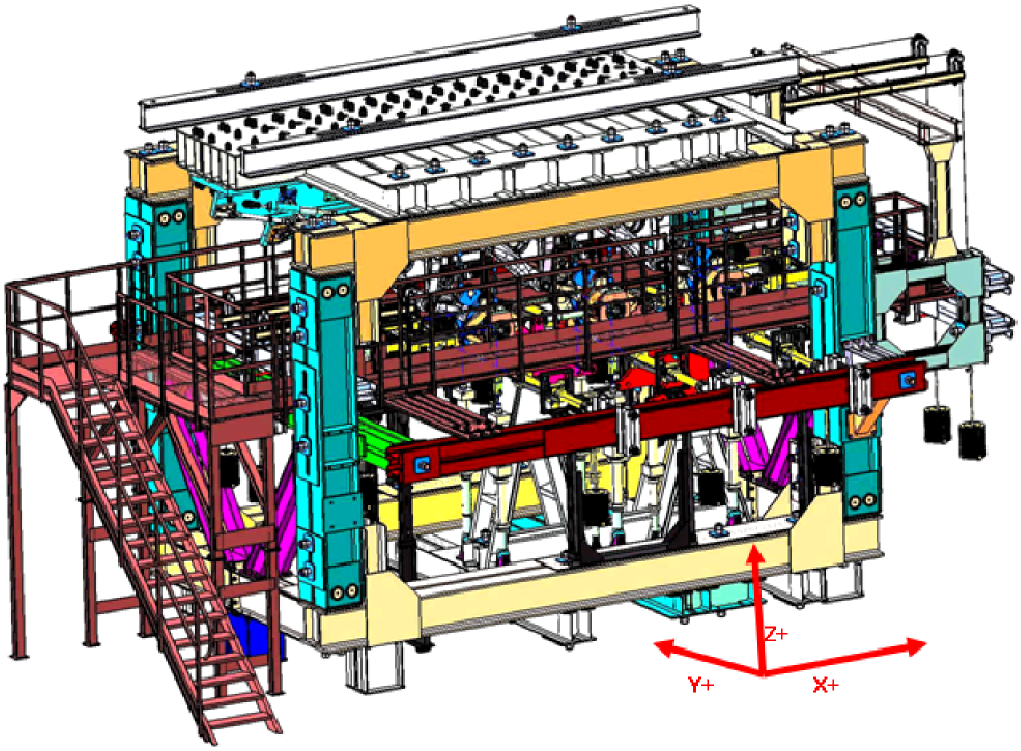
Fatigue test is still running.

4.5. A400M MAIN LANDING GEAR FATIGUE TEST (MESSIER BUGATTI DOWTY AND DGA AERONAUTICAL SYSTEMS)

DGA TA is currently performing the A400M Main Landing Gear Fatigue Test on a specimen representative of the series production.



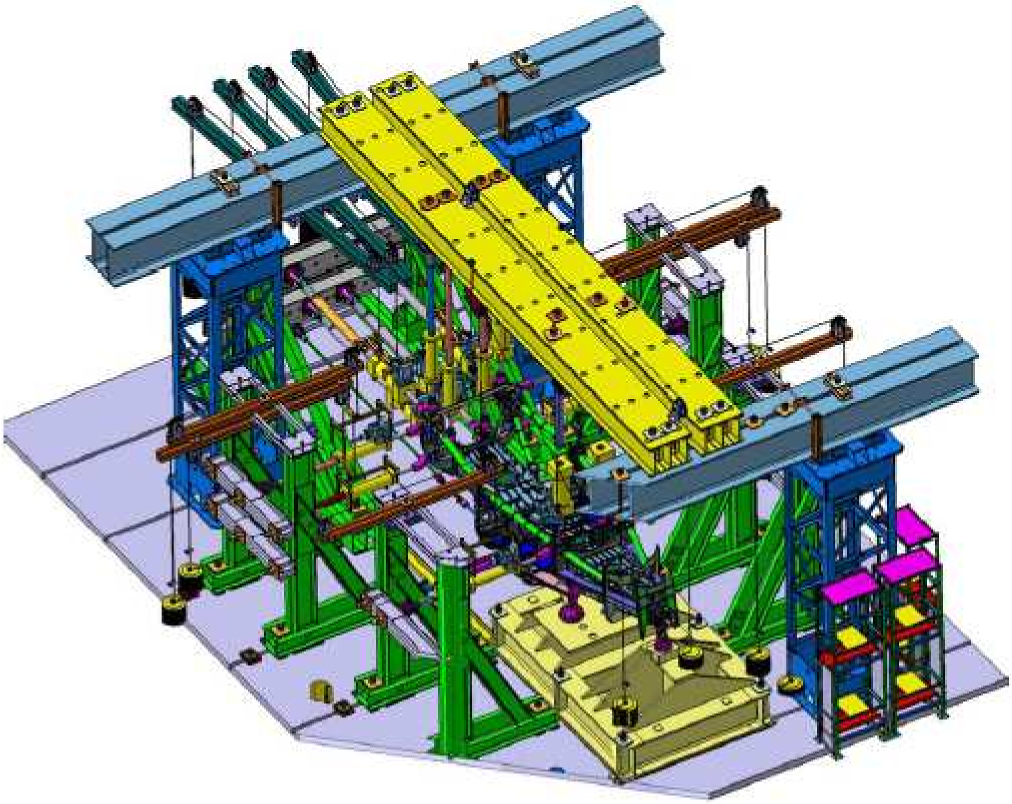
The aim is to simulate 50,000 cycles (SF) to validate 10,000 Flight Cycles (FC) which is the design goal. Load spectrum includes the application of combination of flights grouped in blocks of 200 flights. Load spectrum at the end is in line with the typical aircraft usage (mission mix) defined by a breakdown between four basic missions. Typical ground loads, kneeling actuator loads and extraction retraction loads are simulated.



The loading is achieved using twenty four independent load channels (eight per leg), each fitted with spherical bearings at both ends. All load transducers are located as near as possible of the load introduction point to have a load value recorded by the transducer very close to the real load introduced into the specimen.

4.6. A400M PYLON FATIGUE TEST (AIRBUS AND DGA AERONAUTICAL SYSTEMS)

In the frame of A400M certification, DGA TA is performing the fatigue test of the A400M pylon. This fatigue test re-uses a test rig which was the one developed and used for the A400 pylon static test. The test installation includes 16 hydraulic jacks. Specimen behaviour is followed by 26 displacement transducers and by a maximum of 450 strain gages.



Design goal is 10,000 Flight Cycles with a scatter factor of 3, leading to simulate 30,000 cycles (SF). Load spectrum includes blocks of flights built on 120 different loading cases.

The test programme is divided into several phases including a fatigue phase of 2 DSG, a damage tolerance phase of 1 DSG, a fail-safe phase (of 5 inspection intervals duration) and residual static tests.

Current phase is the fail-safe phase.

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