A.I.F.A. - ITALIAN ASSOCIATION FOR FATIGUE IN AERONAUTICS

DEPARTMENT OF AEROSPACE ENGINEERING - UNIVERSITY OF PISA

Review of aeronautical fatigue investigations carried out in Italy during the period April 2009 - March 2011

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This document summarizes the main research activities carried out in Italy about aeronautical fatigue in the period April 2009 – March 2011. The main topics covered are: load monitoring, fatigue of metallic structures, damage in composites, full scale testing.

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CONTENTS

1. INTRODUCTION

This paper summarises aeronautical fatigue investigations which have been carried out in Italy during the period April 2009 to March 2011. The different contributions have been arranged according to the topics, which are loading analysis, fatigue and fracture mechanics of metallic materials, fatigue behaviour of composites and full scale component testing. A list of references, related to the various items, is presented at the end of the document.

The review is based on the activities carried out within the various organisations belonging to A.I.F.A., the Italian Association for Fatigue in Aeronautics. The author gratefully acknowledges the fundamental contribution, which has made this review possible, given by several A.I.F.A. members, who are the representatives of Universities and Industries in A.I.F.A.

2. MEASUREMENT AND ANALYSIS OF OPERATIONAL LOADS

2.1 - AM-X life monitoring (Alenia Aeronautica)

The life monitoring program for the AM-X aircraft is an activity that has been in progress for a long time and regularly information has been given in the various National Reviews; it is based on classic mechanical g-meter readings and on information about configurations and mission profiles. With respect to the situation described in the last Review, where the data base included 169,000 flight hours, corresponding to 166,000 flights, the statistical population has increased, reaching 174,000 monitored flight hours (corresponding to 170,000 flights). The usage severity is measured by means of the Load Severity Index (L.S.I.), that is the ratio between the damage cumulated in a flight hour and the damage of an average hour of the reference spectrum. The analysis of the data, reported in Fig. 1, is an update of the similar analysis carried out two years ago, and shows that the Load Severity Index (L.S.I.) average value (among all the fleet) results just above 1. Hence the fatigue life consumption has to be considered substantially in line with the design assumptions. Looking at the flight distribution of the Strike aircraft (Fig. 2), it can be seen that a rather wide interval exists, from a minimum of 300 flights to a maximum of about 2000 flights.

2.2 - Life monitoring of the TORNADO fleet (Alenia Aeronautica)

The Tornado has entered into service with the Italian Air Force in 1980; since then, the fatigue life monitoring of the IAF fleet has been performed by Alenia Aeronautica by means of its in-house developed computer program that utilizes g-meter readings together with configuration/masses control. In total, 245,000 flights hours (corresponding to 193,000 flights) have been monitored so far. Among the 5 locations monitored (4 real, such as slat track, front spar, lower panel, frame X 8000 plus a dummy element) by means of this Interim Monitoring System, the lower wing panel remains the most fatigue affected. However, the Load Severity Index, even with a small increase, is largely below design values (Fig. 3).

Also on the basis of the low fatigue consumption rate, it was possible to extend (formally in February 2009), the aircraft life from 4000 up to 6000 Flight Hours. Anyhow, the individual tracking will be maintained, in order to assure a correct management of the fleet maintenance actions and to identify any possible anomalous fatigue consumption. At the same time, a re-organisation of the huge amount of qualification data cumulated during past years is almost concluded, together with other PANAVIA Partners, keeping all the modifications introduced into account. The re-assessment of the entire structure against fatigue, performed for the Italian Air Force, originated the strategy and the revision of maintenance manuals to extend the Italian aircraft life to 6000 Flight Hours. The avionics updating program on the aircraft, which will be maintained into service till 2025, is now in progress.

A new investigation has started on the Tornado usage, focused on the actual altitude mission profiles. The analysis is based upon representative samples of actual flights data, to be compared to pressure spectra tested (and qualified) on the Major Airframe Fatigue Test. The aim of the investigation is a tailored evaluation of the usage of structural items subjected to combined cabin differential pressure and maneuver loads.

2.3 - EF Typhoon life monitoring (Alenia Aeronautica)

Since 2003 to now a total of 50 EF Typhoon aircraft (39 single seater and 11 twin seater) have been delivered to the Italian Air Force; at the end of 2005 the aircraft officially entered into IAF service, operational in the role of patrol and interception. Up to December 2010, the Italian fleet had performed a total of 16203 flights (corresponding to 21230 Flight Hours).

Alenia Aeronautica provides support to IAF for the Typhoon fleet fatigue and usage monitoring by means of the analysis of the data collected by the Structural Health Monitoring system (SHM). As described in the last National Review, the SHM system provides the Fatigue Index calculation for 10 structural significant locations (the percentage of the design fatigue life that has been spent), Auxiliary Data complementary to flight data (g, roll rate, Mach, weight, altitude, etc) and the Event Monitor (that points out any significant structural event compared to pertinent envelopes).

At present, the usage shows a spectrum similar in shape to the design spectrum, but substantially less severe. This trend finds a confirmation in the Fatigue Indexes calculations, that are below design too (Fig. 4).

2.4 - Development of a Structural Health Monitoring System for the M346 trainer (Alenia Aermacchi)

The M346 aircraft is an Advanced/Lead-In Fighter Trainer, developed by Alenia Aermacchi. In the last two years, significant progresses have been made towards the definition of a HUMS, that is a Data Acquisition and Processing System designed to support the aircraft maintenance throughout the service life.

The system will be installed on each aircraft and is composed by an on-board segment and an on-ground segment.

The Main Functionalities are:

a. Onboard recording of aircraft data;
b. System monitoring;
c. Parameters-based structural monitoring (Structural HUMS, S-HUMS)

The last task is the basis for the development of a S-HUMS, i.e. a Structural Health and Usage Monitoring System; its main functionalities will be:

1. L/ESS Statistical Data at A/C and Fleet Level
    - Evaluation of actual spectra of parameters such as Nz, Ny, vz, etc. for comparison with Design Spectra;
    - Statistical evaluation of actual aircraft usage vs. of design usage (mission mix);
    - Statistical evaluation of actual mission severity vs. design mission;
2. IAT Residual life for each monitored structural assembly
    - Damage calculation for each monitored section and consequent spent and residual life evaluation;
    - Identification of most fatigued section for each assembly and, among these, for each A/C;
3. IAT Structural Inspections
    - Progressive crack growth calculation with actual stress spectra;
    - Tailoring of inspection intervals based on actual spectra.

The S-HUMS keeps the aircraft configuration into account and processes the loads and parameters time histories, so obtaining information on the flown spectrum and monitoring the structural fatigue. A number of Control Points have been selected, i.e. the critical section of the monitored components; for each of them, two evaluations are performed: a safe life analysis and a damage tolerance analysis. The first one is a standard fatigue life assessment, based on the use of S-N curves and Miner's rule. The residual life of a sub-assembly is the one of its most fatigue damaged section, and the residual life of the aircraft is the one of its most fatigue damaged sub-assembly. Moreover, for those components where an inspection was specified, a crack growth analysis is performed to tune the inspection interval keeping the real flown spectrum into consideration, with the objective of avoiding useless periods of stopped aircraft if the flown spectrum is less severe than the design one, and vice versa risks for the safety in the opposite case; the crack growth analysis is based on a standard 1.27 mm initial defect, and is performed using the Nasgro 4.13 code.

As far as the spectra analyses are concerned, the S-HUMS is capable of monitoring the spectrum for each of the 13 missions currently considered; for each aircraft, the exceedance/occurrence spectra of each mission are updated, as well as the mission mix. The system can obviously produce the statistical analysis of the complete fleet spectra.

2.5 - Health and Usage Monitoring System for AW101 helicopter (AgustaWestland)

In the previous review, a research program was mentioned for the exploitation of AgustaWestland Enhanced Structural Usage Monitoring (ESUM) and Transmission Usage Monitoring (TUM) to improve the evaluation of fatigue loading spectra. These usage monitoring systems are installed on AW101 variants, AW139 and NH90. Now, a first contract has been assigned for the AW101 usage data analysis to assess Structural Usage Monitoring and Transmission Power Spectrum.

The main tasks of the system are:

- Determine the true Usage Spectrum and the Transmission Power Spectrum from recorded data;
- Verify the design assumptions and point out major deviations versus true recordings;
- Perform a sensitivity study to checked potential improvements or reductions of design lives.

A comprehensive exercise has been made on a preliminary set of recorded data from Danish Royal Air Force AW101 helicopters to validate usage assumptions made in the qualification. This exercise is based on the identification of the time spent by the helicopter in a specific flight condition.

The design spectrum is subdivided into the following major flight conditions:

- ✓ Ground conditions (taxiing, …)
- ✓ Take-off and landing (flare, …)
- ✓ Hover (IGE, OGE, spot turns, …)
- ✓ Climb and descent (transition to Vy, climb TOP or MCP, …)
- ✓ Autorotation and related power-off manoeuvres
- ✓ Level flight (banked turns, pull-ups, control reversals, various flight speeds, …)

Further split is done in specific flight regimes, as quoted in brackets, and subconditions per:

- Weight
- Longitudinal balance
- Altitude

A dedicated software, ESUM, runs on an on-board computer and can recognize each flight condition with the related sub-conditions and associated weight, altitude and longitudinal balance using the following data:

- ✓ Weight
- ✓ Longitudinal balance
- ✓ Altitude
- ✓ Speed
- ✓ Load factor
- ✓ Bank angle
- ✓ Longitudinal acceleration
- ✓ Climb and descent rate
- ✓ ACSR status

Plots of the distribution of various quantities, such as weight (Fig. 5), centre of gravity position, altitude (Fig. 6), etc, have been produced, as well as Main Rotor Torque (Fig. 7), bank angle in a steady roll manoeuvre and the level flight speed (Fig. 8); this allows the identification of the most significant flight conditions and their relative occurrence frequency. The comparison of the flight conditions frequencies with the design assumptions can give a simple indication of the usage severity.

A HUMS system has been developed, following similar principles, for the AW139 helicopter, used by many civil operators engaged in the off-shore sector, characterized by a high usage rate, with the purpose of triggering a Condition Based Maintenance programme. It includes:

- a Transmission Vibration Monitoring (TVM) unit, based on 11 accelerometers and 1 speed sensor;
- a Rotor Track and Balance (RTB) system, based on 4 accelerometers, 2 magnetic sensors and 1 tracking device;
- a Usage Monitoring (UM) system, subdivided into Structural (airframe and dynamic components) and Transmission (drive system) Usage Monitoring.

A paper has been presented at the NATO-RTO AVT-172 workshop on “Implementation of Condition Based Maintenance” held in Bucharest (Romania), October 2010, [1].

2.6 - Individual Aircraft Tracking Program of the C-27 J aircraft (Alenia Aeronautica)

Fatigue life monitoring allows the operator to know the current status of its aircraft (in terms of Fatigue Residual Life and Crack Growth Residual Life) and consequently to plan and optimise maintenance and usage of each aircraft inside the fleet. The C-27 J main Customers, to which the IATP program is applied, are:

Italian Air Force	12 Aircraft
Hellenic Air Force	12 Aircraft
Lithuanian Air Force	3 Aircraft
Bulgarian Air Force	3 Aircraft
Romanian Air Force	7 Aircraft
Moroccan Air Force	4 Aircraft

About 25,000 Flight Hours have been achieved by the C-27 J fleet and a continuous monitoring by IATP program has been performed in order to track their *Structural Integrity* status. Information about the definition of the program have been given in a past National Review; it is worthwhile quoting here that it is based on fatigue consumption and on crack growth evaluation. The design life is 14,000 flights or 28,000 flight hours.

3. METALS

3.1 - Fatigue behaviour of notched and un-notched materials

3.1.1 - Fatigue behaviour of double lap riveted joints assembled with and without interfay sealant (Univ. Pisa)

A number of fatigue tests were carried out on sealed and un-sealed riveted specimens in 7075-T73 aluminium alloy. The sealant was a bi-component manganese-dioxide cured, polysulfide compound, typically used to seal fuselages and integral fuel tanks. The specimen (see Fig. 9) was a "double shear" joint, assembled by only one Hi-Lok rivet, installed with clearance fit.

Sealed specimens were pre-assembled by inserting a temporary spring fastener in a pilot hole. The holes were reamed to the final diameter after the sealant curing and then the Hi-Loks were installed; this procedure is commonly used in aerospace structures assembly.

The fatigue resistance of sealed specimens was very low with respect to those of dry assembled specimens (see Fig. 10). The lubricant effect of the sealing layer, highlighted by the hysteresis cycles measured in sealed and un-sealed specimens, justified the results obtained.

As a matter of fact, all the fatigue tests were carried out in laboratory air, being the use of sealant commonly finalised to prevent water and dust ingestion in riveted joints: different results could thus be obtained by performing fatigue tests in corrosive environment. The corrosive aspect can be in any case considered of secondary importance when the use of sealants is aimed at preventing air or liquid leakages.

The results of additional tests, performed on specimens assembled by applying different clamping forces during the sealant curing time, indicated an independence of the fatigue resistance on this parameter.

Furthermore, different failure modes were observed in sealed and un-sealed specimens: fatigue cracks nucleated outside the hole at a location in the shadow of the rivet in the un-sealed joints, while fatigue cracks nucleated at both sides of the holes in sealed joints (see Fig. 11).

The first failure mode is consistent with high clamping force exerted by the rivet, while the second failure mode is typically observed with low clamping force, or in alternative, with low friction coefficient between the mating surfaces.

Finite element calculations, performed by assuming several values of the friction coefficient between the mating surfaces, confirmed the different location of the fatigue critical areas in sealed and un-sealed specimens. In particular, as shown in Fig. 12, the stress distribution in the specimen along the loading direction, reported for two cases of friction coefficient (f=0 and f=0.5), highlighted how an increase of the friction coefficient results in a movement of the most fatigue critical zones from the lateral sides of the rivet hole to regions located far from the hole.

In the graph of Fig. 13 the longitudinal stress is plotted versus the distance from the hole edge, up to the free edge of the specimen, for increasing values of the friction coefficient. The stress at the hole edge tends to decrease very rapidly from f=0 to f=0.5, while a further increase of the friction coefficient causes a slower decrease of the stress at hole edge.

More details on this research can be found in [2]. Future activities will be dedicated to the same topic in joints assembled by more than one rivet; different scenarios could be observed in this case.

3.1.2 - Developments of Laser Shock Peening technology for fatigue life improvement of metallic components (Univ. Bologna)

Laser shock peening (LSP) is an innovative surface treatment technique, which has been successfully applied to improve fatigue performances of metallic components. The key beneficial characteristic is the establishment of compressive residual stresses under the treated surface of metallic materials, mechanically produced by high magnitude shock waves induced by a high-energy laser pulse. Compared with the traditional mechanical methods (e.g. shot peening process) to introduce residual stresses, LSP can produce high magnitude compressive residual stresses several times deeper than traditional technologies. The residual stresses have significant improvements on the fatigue behaviour, i.e. a longer nucleation and initial propagation periods.

Experimental tests were made on thin (2.3 mm) and thick (20 mm) aluminium specimens and results of these research activities are reported here.

1) Experimental tests on thin specimens

The specimens used were dog-bone specimens, obtained from 2.3 mm thick sheet of Al 7075-T73 (Fig. 14).
In LSP treatment, the characteristics of the introduced residual stresses depend on the laser parameters. It is important to define:

- Laser power density - compressive stresses increase with the increase of laser power;
- Laser wavelength - different peak pressures are developed at different wavelengths;

- Peen size - superficial residual stresses increase with the size of the impact;
- Pulse Duration - reducing the pulse duration reduces the depth of the compressive residual stresses through the thickness;
- Overlapping of laser peens - increasing the number of layers increases the value of compressive residual stresses;
- Coating – the treatment can be performed with or without the use of a protective coating.

The laser used for the research activity on thin specimens (thanks to collaboration with Centro Laser of the Polytechnic University of Madrid), has the characteristics summarised in Table I.

Since the laser available has a relatively small output energy, the best setup is with small peen diameter (1.5 mm) and consequently a high overlapping rate (900 shots/cm^2). This fact leads to the decision to work in a direct ablation mode, since the great overlapping would increase both the costs and the time of the procedure if it were necessary to apply the protective painting after every shot.

The hole drilling technique was used for the residual stresses measurement, Fig. 15.
When treating the specimen on one side only, its bending was clearly visible (Fig. 16a): this is obviously unacceptable. However, the specimens were treated on both sides in order to ensure the symmetry of the stress field. So, after treating on both sides, the specimen re-established its original shape (Fig. 16b).

Fatigue tests were carried out under sinusoidal loading; the load conditions (S_{max} 160 MPa and R=0.1) were chosen on the basis of previous works in order to be able to make a direct comparison of different technologies. However, the results relative to LSP technology have turned to be very disappointing, performing three times worse than the baseline (no enhancement treatment, basic open hole) in the terms of fatigue lives of tested specimens, Table II.

Obviously, the focus of this investigation has moved towards understanding the causes of this result. The sequence of operations used for the specimens preparation: open hole and then LSP treatment might have caused peaks in tensile stresses at the edge of the hole. Laser and plasma interaction in the presence of a sharp edge could have caused an effect of plasma lens, in which the laser beam passes through the created plasma and is being focused on the inner side of the hole. Future investigations will be directed towards understanding the importance of the sequence of operations (first LSP treatment and then open hole drilling).

2) Experimental tests on thick specimens

The specimen used for this activity has the geometry illustrated in Fig. 17. This experimental activity was done entirely at the University of Bologna, using the laser with the properties given in Table III.

Three point bending fatigue tests have been carried out: the results are given in Fig. 18. In this case, the expected increase of fatigue live was encountered (3x fatigue live increase in respect to the base materials) since the tensile residual stresses that balance the surface compressive residual stresses are in the thickness of the specimen. In addition, the effect of the sharp edge of the open hole was not present.

The following conclusions can be drawn:
- LSP needs to be optimised, particularly when applied to low thickness components;
- In the investigated configuration of the thin specimen, LSP turned to have detrimental effects on the treated specimen; the tensile residual stresses are the cause of shorter fatigue lives.
- In the investigated configuration of the thick specimen, the fatigue lives after LSP treatment have increased three times with respect to the base material.

3.1.3 - Fatigue of gears (AgustaWestland)

A relevant cooperation with Milan Polytechnic – Dept. Mechanical Engng. started more than 6 years ago to improve the evaluation of root bending fatigue of AgustaWestland gears; some details have already been given in the last National Reviews. The endurance limit of some specific steels has been studied in depth, with significant observations that have important consequences on the time required for the tests (up to 100 million cycles).

An Integrated Project, partly funded by the EU, is in progress on the fatigue behaviour of gearboxes and transmission systems; more details of the program and its major outcome will be described in a paper, [3], to be presented at the 26th ICAF Symposium in Montreal.

Another important activity is in progress to improve AgustaWestland fatigue analysis of gears and is based on an integration of the fatigue analysis methodologies for transmissions design. A common exercise will be performed on Lynx transmission components; four complex components have been selected: MGB casing, MGB conformal pinion, TGB output shaft and Tail Take-off bevel gear. The same operating spectrum and state will be used to carry out the fatigue evaluation with different codes.

3.2 - Crack propagation and fracture mechanics

3.2.1 - Damage Tolerance evaluation of flat panels stiffened by bonded pads (Univ. Pisa)

Within the framework of a collaboration with Piaggio Aero Industries (PAI), an experimental program has been performed at the Department of Aerospace Engineering of the University of Pisa, aimed at assessing the fatigue crack growth and the residual strength of a number of selected configurations of stiffened panels.

Two couples of flat panels were tested, external dimensions 1470 x 1200 mm, stiffened by means of multilayered bonded pads., The panels of the first couple had six pads and the initial through crack was in the skin, in the middle of the central bay; the panels of the second couple had seven pads and an initial through crack in the central pad (and in the skin underneath). The panels within each couple differed for the thickness of the layers in the pads, while the skin was in all cases 1 mm thick; all the panels were made of 2024-T3.

A particular problem for all the panels of this activity was the design of the gripping system and the interface with the machine, as a consequence of the small height-to-width ratio of the panels. The risk in these situations is to have a non-uniform stress distribution in the skin, with the central portion loaded more severely. a dedicated solution for the gripping plates was designed, with a slot in the central part, in order to reduce locally the stiffness of the plate, and consequently also the amount of load introduced in the central part of the panel. Finite Element analyses were carried out to establish the most appropriate extension and position of the slot to allow the obtainment of a stabilized stress distribution in the central region of the panel, which is the location of interest.

Table IV gives some details about the panel configurations.

In the case of Panel type 3, a complete cut was introduced in the pad so that the initial damage was a through crack in the skin with a completely broken pad. For the fatigue crack growth test, all the panels have been subjected to a constant amplitude cyclic load, characterized by a maximum average stress level of 90 MPa and a stress ratio R = 0.1.

After completion of the fatigue crack growth test, a residual strength test was performed for each panel.

The results of the crack propagation tests on the four panels are summarized in Fig. 19.

Analyzing the results in figure 19, the following observations can be made:

- The panels with seven pads and an initial crack in correspondence with the central pad showed the highest crack propagation rate.
- The crack propagation life of panel type 3 is much shorter than the one of panel 4 due to the fact that the central pad was completely cut prior to test (as a remedy to a manufacturing defect).
- Six pads panels benefit from a considerable crack retardation due to the effects of unbroken pads that limit the central bay.
- Thicker pad layers slightly decrease the crack growth rate.

The results of the residual strength tests on the four panels are summarized in Fig. 20 in terms of mean gross stress vs. half crack length increase.

The half crack length increase has been measured from digital frames, captured by a video camera, using a graduated scale traced on the test articles for this purpose. The mean gross stress has been derived dividing the measured load by the cross sectional area of the whole panel (undamaged skin and multilayered pads); the gross stress has been used because the tests articles had different numbers of pads with different thicknesses (and hence different cross sectional areas with the same skin).

Data in Fig. 20 shows that the four test articles underwent a stable crack propagation phase of different duration; the failure stress was always higher than twice the maximum load considered for the fatigue crack propagation.

3.2.2 - Development of a Flaw Tolerance helicopter fatigue design methodology (AgustaWestland)

This topic was also included in previous editions of the Italian National Review, because AgustaWestland has a long term research program in progress on the applicability of the Flaw Tolerance EASA CS 29 requirements. AgustaWestland preferred method for compliance with Flaw Tolerance requirements for dynamic components is the adoption of the “no damage growth” concept.

The typical flaw size assumed is a corner or a semi-circular crack of radius r = 0.38 mm, for parts exposed to accidental damage in flight, and a smaller flaw of radius r = 0.25 mm for parts protected in flight, after maintenance inspections. As far as the no-growth concept is concerned, it is applied through the use of the Kitagawa-Takahashi diagram. For this purpose, a Flaw Tolerance Data Base (FTDB) is being progressively extended, within a collaboration with Milan Polytechnic - Dept. Mechanical Engng.; it is a large activity of characterization of materials and also of processes used to improve the fatigue behaviour of typical components. In the last Review some data about the influence of shot peening was given; recently, also a portable ultrasonic shot-peening treatment (Stressonic ®) has been evaluated.

3.2.3 - Damage tolerance of adhesive bonded stiffened panels (Univ. Bologna)

Adhesively bonded solutions are very attractive for primary aircraft structures, since they provide significant benefits concerning the damage tolerance (DT) proprieties in comparison with the more conventional riveted and monolithic structures. The stiff adhesive joints permits indeed a high load transfer between the cracked and the intact elements of a built-up bonded structure, while maintaining the differential structural behaviour with fail-safe features provided by the multi-load paths. Nevertheless, all the DT potential benefits that the adhesive bonded structures could provide are not yet fully exploited. This is also due to the difficulties in predicting the behaviour of cracked bonded structures as a consequence of the complexity of the interacting phenomena involved during the crack propagation. The load redistribution between all the elements of the bonded structures can indeed determine delaminations at the interface of the bonded elements and promote fatigue damage and failures in such elements, which obviously affect the crack propagation itself.

Bonded stiffened panels are widely employed in aircraft areas where the DT is the most important design criterion such as the crown of pressurized fuselages and the low part of the wings. Figure 21 shows the general crack growth behaviour of a skin crack propagating through a bonded stiffened panel over a broken stringer. Up to now, as a consequence of the aforementioned limits in predicting the DT performances of bonded structures, the allowable stress level definition is based only on the value ΔN_A; the long crack propagation period through the skin underneath the bonded stringer (N_B - N_A in Figure 1), normally is not considered, as highlighted by H.J. Schmidt [4]. Significant benefits, in terms of structural weights, reliability and maintenance costs, could be achieved by taking into account also the crack propagation period underneath the intact bonded stringer. Therefore, a deep investigation of the phenomena involved during this crack propagation phase and of the mechanisms that can induce the stringer failure, while the skin crack is still hidden under the stringer, is fundamental.

To this end, with the background of well-known literature on the displacement compatibility method (C.C. Poe and T. Swift), an analytical tool (LEAF) was developed at Bologna University - Forlì Campus for the prediction of the DT proprieties of bonded stiffened panels.

The estimated stress field around the advancing skin crack tip and loads carried by the stiffeners were validated by the results of extensive test campaigns performed with specimens representative of typical fuselage concepts. The analytical predictions, in conjunction with the experimental results, provided a deep insight into the investigated crack propagation through the skin underneath the stringers and the mechanisms which drive those stiffeners failure.

Experiments and analyses confirm that bonded stringers show their most effective crack retarder effect when the crack tip propagates through the skin underneath them; this is due to the crack bridging mechanism. The stiffer is the stringer and the more intense is the crack restrain contribution which, however, is exerted as long as the stringer is intact (Figure 22/a). The failure of the stringer, which can occur when the skin crack is still in the stringer covered area, is driven by the stiffness characteristics of the stringer as well, in accordance with a low-cycle-fatigue mechanism. The thinner is the stringer and the higher are the maximum stresses acting in it (Figure 22/b). Such stresses approach the ultimate tensile strength (UTS) of the stringer material and can drive the fast failure of thin stringers. On the contrary, thick stringers fail typically with stresses well below the UTS value, after a long period of crack propagation under the stringer, as a consequence of a more conventional fatigue mechanism.

In conclusions, bonded stringers are confirmed to behave as effective crack retarders, mostly when the crack grows underneath the stringer foot. In order to exploit all the benefits that the bonded stringers can provide, high-static strength thin stringers and fatigue insensitive thick stringers are to be preferred. The aforementioned conclusions can be generalized in order to design all kind of stiffeners adhesively bonded on panels, such as thin doublers or crack retarders. More details on this research will be presented in a paper at the ICAF Symposium in Montreal, [5].

3.2.4 - Damage tolerance of adhesive bonded fuselage panels (Piaggio Aero Industries)

Piaggio Aero Industries is developing a new aircraft, of the business jet class, characterized by higher performances and more payload than the turbo-propeller three lifting surfaces P180. One of the major issues for this class of airplanes is to have the largest possible internal space, for the same external dimensions; the fulfilment of this objective requires a careful design of the fuselage lay-out, for which the Piaggio designers are considering a stringer-less configuration, characterized by a few longitudinal reinforcements and many circumferential straps, with a low pitch, to guarantee adequate resistance to compression loads. This arrangement allows a reduced structural thickness and a larger internal volume for the same external dimensions.

An additional challenge for this aircraft, whose temporary name is P1XX, is that it will be certified according to the EASA CS25 regulation, which means in compliance with the Damage Tolerance requirements.

Therefore, a number of activities have been planned and performed in order to obtain basic experimental data on the fracture mechanics behaviour of aluminium skins reinforced by bonded straps, useful to calibrate analysis tools. Two major activities will be shortly described in the following:

- Fatigue crack growth tests on flat panels with different crack stoppers features;
- Fatigue crack growth test on a dedicated fuselage barrel with different upper skins.

1) Fatigue crack growth tests on flat panels with different crack stoppers features

Four flat panels, made of 2024-T3, 1 mm thick, have been tested; the dimensions were 1470 in width x1200 mm in height and the initial damage was a central through crack of 14 mm total length. For all panels, a cyclic loading characterized by a maximum stress of 90 MPa was applied, with a stress ratio R of 0.1. Four configurations have been tested:

a) a simple flat panel, unstiffned, for reference;
b) a chemical milled panel with integral pads, 2.5 mm total thickness;
c) a flat panel with bonded pads, 2.5 mm total thickness;
d) a flat panel with bonded pads, 2.0 mm total thickness.

The chemical milled panel showed a crack growth life only a few cycles longer than the reference panel; anyhow, it provided a test case useful to validate an analysis procedure based on the use of FE models for calculating the stress intensity factor and the AFGROW code for crack growth prediction. A good agreement between test results and prediction was obtained.

The bonded panels had 7 pads (i.e. the initial damage interested also a central pad), each of them was 50 mm wide, while the spacing was 210 mm; two options were studied: a 1.0 mm thick pad or a 1.5 mm pad. In both cases, the crack reached a 4 bays length without failure and with a very long life, as shown in Fig. 23. The results show the superior crack arrest capability of the bonded pad solution, with a strong crack growth rate decrease when the tip passes under the pad.

An extensive damage tolerance test program is in progress, with the objective of optimizing pitch and thickness of bonded pads. At the same time, also different pre-bonding treatment and bonding products are assessed.

2) Fatigue crack growth test on a dedicated fuselage barrel with different upper skins

A special test rig has been designed and manufactured, to allow the test of curved panels representative of fuselage crown panel structural solutions under pressurization cyclic loading, see Fig. 24. The barrel structure is representative of a small portion of P1XX central fuselage.

In order to analyze the damage tolerance characteristics against a circumferential crack in a stringer-less architecture, three types of crown panels (in all cases, the skin thickness was 1 mm) have been analyzed/tested. A short description of each test will be given in the following.

During the tests, the barrel has been loaded with the following fatigue loads:

a) Operative pressurization test: p = 9.6 psi for the first two tests and p = 10.6 psi for the third test;

b) Bending due to flight loads, by means of an applied force of 90000 N × 2 (the load to achieve the bending fatigue stress on the crown panel; see Fig. 25).

In the area of interest, the ratio of longitudinal to hoop stress is a bit higher than 1.0. A FE model has been developed to analyze the behavior of the fuselage barrel, see Fig. 26. The fatigue target stress σ_L = 92 MPa is achieved between Frame 5 and Frame 6. A total of 44 rosettes and 47 strain gauges were attached to the external and internal surfaces of the fuselage panels (the crown was monitored with 19 rosettes and 35 s.g.).

The combined loads were applied to the barrel according to the Fig. 27. The total time for a cycle was 46 sec.

Numerical simulation was carried out by means of global and local FEM analysis in order to evaluate the stress intensity factors at different crack lengths.

The three configurations already tested were:

a) Plain Panel (i.e. without pad)

The only crack arrest features are the two riveted longerons and the two riveted butt splices. This structure is penalized because crack arrest features are absent and residual strength calculation is required with the item totally broken.

As initial artificial damage, the LH upper longeron has been cut and an initial crack of a0 = 30.5 mm on both sides of the rivet hole (D = 4 mm) was introduced in the skin with an electro-erosion tool.

The test served as a global check of the test set-up and the results were used to calibrate the analysis methodology.

b) Panel with bonded pads

Bonded pads are placed at the same location of integral pads; they act as crack stopper features.

The bonded crown panel has been made with 4 bonded layers. The adhesive system was Cytec FM-73. The initial damage was similar to panel 1 (broken upper LH longeron plus a 30.5 mm crack in the skin).

The crack growth showed a substantial improvement over the un-reinforced skin.

c) Panel with chemical milled integral pads

Integral pads (thickness 1.5 mm, pitch 210 mm) were obtained in the skin of this panel by means of chemical milling and were placed in the same location of the pads of the bonded panel; such longitudinal integral pads provide a crack growth retardation feature. The initial damage was similar to test 1 and test 2.

A fourth test, on a crown panel with bonded pads in optimized position, is scheduled to start in May 2011. This last panel has been defined on the basis of the previous activities and has a lower skin thickness (0.8 mm) and a pad thickness of 1.5 mm. In order to accommodate larger windows, the maximum longitudinal pad pitch has been increased to 285 mm.

In order to analyze all the damage scenarios possible in bonded structures, also bond line delamination, adherent damage growth and damage interaction will be considered/addressed in the future activities. A collaboration with Delft University of Technology is under definition.

4. COMPOSITES AND FIBER METAL LAMINATES

4.1 - A novel cohesive zone modelling approach to evaluate Damage Tolerance of composite laminates (Milan Polytechnic)

A numerical approach for modelling the development of interlaminar damage in different conditions has been developed and assessed. This approach is based on cohesive laws [6-8] to model both the fracture propagation in pre-damaged specimens as well as the onset and subsequent evolution of interlaminar fracture in originally undamaged composites laminates. A peculiar aspect is that a preliminary identification of the layers where interlaminar cracks will propagate is not required because, thanks to the characteristics of this modelling technique, models including all the interlaminar layers can be quite easily managed. In fact, such approach is based on an alternative modelling technique of laminates, with respect to the conventional finite element schemes that employ cohesive elements [9]. This technique can exploit the use of different cohesive models presented in the literature [10] to describe a composite laminate as a collection of bi-dimensional sub-laminates, mutually connected by interface zones represented by means of traditional brick elements. In a first version of the approach, such elements are characterised by a null in-plane response and by a non-linear out-of-plane behaviour, which models the onset and propagation of the interlaminar damage. The stiffness of cohesive elements is modelled on a physical basis, by using the real elastic moduli of the material. Thanks to the finite thickness of the elements and to the low stiffness levels, the resulting finite element scheme can be conveniently solved by means of an explicit integration approach adopted to model both static and dynamic problems [9].

The numerical-experimental correlations of the responses obtained in the simulations of DCB and ENF tests show the capability of the proposed numerical approach to follow both stable and unstable crack propagation, respectively [11]. The numerical technique has been subsequently applied to model the interlaminar fracture onset and propagation within an undamaged curved laminate in a quasi-static bending condition (Fig. 28). The comparison between numerical and experimental damage patterns points out both the capability of this technique to identify the different locations of the interlaminar damage onset, in the absence of pre-damage zones in the laminate, and its capability to follow the propagation of multiple interlaminar cracks [11, 12].

The interlaminar constitutive model has been completed by the implementation of an in-plane constitutive law. Such law has been developed on the basis of a biphasic approach in order to model the intralaminar damage mechanisms, which play a fundamental role in the onset of delamination in particular load conditions such as low energy impacts on composite laminates.

On the basis of the original hybrid scheme of the numerical approach, a biphasic decomposition of the material, at three dimensional level, has permitted to implement within the interlaminar solid elements the constitutive model of an idealised matrix phase and within the shell elements the constitutive model of an idealised fibre phase. The interaction between intra- and inter-laminar damage mechanisms has been performed thanks to the presence of all the damage variables in the interlaminar solid elements. These variables represent the degradation of the stiffness properties of the idealised matrix phase. The hybrid coupled approach has been adopted to simulate low energy impact tests on composite laminates and subsequent compression after impact tests [12].

All the numerical results have been compared with the experimental evidences. Encouraging results have been obtained in the simulation of low energy impacts (Fig. 29) with the correct evaluation of many of the physical phenomena that affect such a type of load condition. Good results have been obtained in the numerical simulation of compression after impact tests (Fig. 30), both in terms of qualitative identification of the interlaminar damage mode propagation during the compressive action and quantitatively by means of the correct identification of the compressive residual strength level [12].

In the light of all these results and on the basis of the correlation level obtained both in dynamic and static load conditions, the hybrid coupled approach can be considered a valuable numerical technique capable to provide a significant help in the development of damage tolerant composite elements.

4.2 - Interlaminar Fracture Mechanics characterization of composites (Univ. Pisa)

A collaboration has started between AgustaWestland and the Department of Aerospace Engineering of the University of Pisa with the objective of characterizing a few composite material systems as far as their fatigue and delamination resistance is concerned. The activity has started with two systems: a fabric 5H 8552S/AGP280 and a UniDirectional 8552/AS4.

For each of them, static tests to evaluate the mode I, mode II and mixed mode I+II (with two different values of the mode partition ratio) G_c have been planned. Three batches of specimens have been manufactured for mode I (DCB coupons) and mode II tests (ENF coupons), to cover adequately the scatter in the properties.

Moreover, also fatigue curves, again on three batches for the pure mode I and II conditions, are in progress to evaluate the number of cycles for delamination onset. A criterion of 5% stiffness decrease has been selected for these tests, which are quite complex and delicate, due to the very small differences in forces that must be appreciated. For the mode II fatigue tests, a pre-crack has been introduced, to avoid over conservative results.

5. INTERNATIONAL AND NATIONAL RESEARCH PROGRAMS

5.1 – Clean Sky Joint Technology Initiative - European Technology Platform (Alenia Aeronautica)

The purpose of this research, partly funded by the European Union within the 7th Framework Program, is to demonstrate the validity and the degree of maturity of the aeronautical technologies developed with the goal of reducing the pollution due to the air traffic.

The project is based on six platforms (in brackets the leaders):

- ✓ Green Regional Aircraft (Alenia Aeronautica);
- ✓ Smart Fixed Wing Aircraft (Airbus & SAAB);
- ✓ Green Rotorcraft (Agusta Westland & Eurocopter);
- ✓ Sustainable and Green Engine (Rolls-Royce & Safran);
- ✓ Systems for Green Operations (Thales & Liebherr);
- ✓ Eco-Design (Dassault & Fraunhofer Institut).

The objective of the Green Regional Aircraft platform is to develop a new generation Regional Aircraft that includes enhanced technologies for :

- ✓ More efficient aerodynamic configuration
- ✓ Weight reduction (minimum 9% less)
- ✓ New generation engine (10-20% fuel reduction)

In addition to the development of the new technologies for the achievement of the above mentioned objectives, the project comprises the validation of a demonstrator up to flight test; an existing platform will be modified to receive panels and components where the developed technologies will be implemented and integrated. The objective of the Flight Test will be to obtain an in-flight validation for the advanced structural technologies that require data acquired in a real operating environment. Strains measured in the full scale test and in-flight will be compared.

5.2 - MAAXIMUS Research (Alenia Aeronautica)

The MAAXIMUS acronym stands for **M**ore **A**ffordable **A**ircraft Structure Lifecycle through e**X**tended, **I**ntegrated, & **M**ature n**U**merical **S**izing and this project, partly funded by the European Union within the 7th Framework Program, consists of a key enabler for drastic changes in the development of a mature virtual product sizing technology.

The objectives are quite ambitious and can be summarized in the following points:

- Structure **development cost** reduction of 5-10% (more mature technologies, less unexpected failures, lower number of tests, …);
- Structure **development lead time** reduction of 10-20% (reduced time to introduce new technologies, less test, lean & efficient processes);
- Structure **direct operation cost** reduction of 5% (structure weight reduction: lighter materials, reduced conservatism, optimised designs, etc);
- Structural **maintenance cost** reduction of 5-10% (reduction of unexpected maintenance, extended life cycle, delayed planned maintenance, focused/anticipated maintenance).

Alenia Aeronautica's contribution to the research is focused on:

- ✓ Automation in the assembly: monolithic structures

- ✓ Virtual structural sizing
- ✓ Repair numerical simulation
- ✓ Virtual testing techniques

In this project, “virtual testing techniques” refers to all the disciplines, included fatigue design. The activities are focused also on smart materials (for durability assessment) and Health Monitoring, that have important implications for the fatigue management.

An important activity has been dedicated to an analytical/experimental activity, aimed at minimizing the use of anti-peeling fasteners to stop crack/delamination growth by using Eisenmann study. The damage arrest is achieved by reducing the stress intensity at the crack/delamination tip due to the composite materials capability to be tailored in stiffness, thickness, local laminate lay-up and toughness.

The analytical phase was completed; the test activity, to validate the theory, is in progress.

5.3 - HECTOR Research (AgustaWestland)

The HECTOR project (HElicopter fuselage Crack moniToring and prognosis through on-board sensOR network) is an EU funded research programme, within the framework of the 7FP, with partners from Italy, Greece, Poland, Slovakia and Norway. The main purpose of the project is to propose modelling techniques for the analysis of multiple sensors in producing a unified and comprehensive method for identification, monitoring and prognosis of potential damages (like cracks) in the fuselage of a helicopter.

This goal will be obtained through the combined use of advanced models for the assessment of the state of stress in the system and the identification of likely areas of nucleation and growth of cracks, and the use of a network of on-board sensors of last generation (Comparative Vacuum Monitoring, Optical Fibre Sensors, Crack Propagation Gauges, etc). A sketch with the logical flow of the actions is shown in Fig. 31.

The final research aim is to obtain a reliable method to assess the damage accumulated in the fuselage by means of an on-line and advanced prognostic models that allows the definition in real time of a plan for periodic and special inspections.

The results could be applied on a model of military helicopter. These issues are extremely important in military scenarios where hostile environment and accidental load play an important role. In addition, SHM can improve the life of ageing airframe without reduce safety.

The purpose of HECTOR is the definition of a reliable method, useful for helicopters, that permits, on the basis of a complete knowledge of the stress state on the fuselage (advanced FE model) and the data collected by a low power sensor network positioned in the most critical areas, to perform the Structural Health Monitoring and the on-line evaluation of the damage mechanism (cracks) and evolution based on advanced damage criteria.

This package could be installed on new helicopters to define a more reliable, efficient and economic program of inspections, or on ageing airframes to perform a life extension of the machines.

Critical points are the definition of an integrated network of sensors of different types (crack gages, comparative vacuum monitoring, fibre sensors, eddy current, etc.) chosen on the basis of the facility of positioning and fuselage dismantling and considering their reliability (online or off-line mode) and the construction of an advanced communication system, choosing the most reliable and adapt between wireless and traditional one.

Safety, Reliability and availability, Performance and Reduction of maintenance costs are keywords.

A SHM integrated design can lead to a more optimized structure with the possibility to increase the payload and the performance while keeping provide the required strength. Moreover, a SHM is a powerful tool for the ground but also air crew, monitoring the health condition in order to make a correct real time prognosis for the achievement of the mission task.

5.4 - OAST-OASB Fatigue Test Program (Alenia Aeronautica)

In the last National Review, a large experimental fatigue test campaign was mentioned, planned by Alenia to substitute the traditional but low environmental friendly OAC (Chromic acid oxidation) process with the more ecological ones OAST (sulphur-tartaric acid) or OASB (sulphur-boric acid).

The activity on OASB anodization has been completed, showing a reduction in fatigue life of the order of 2-3% with respect of the bare material, as in the expectations. Now, two facilities (Casoria and Nola) have dedicated treatment lines; the evaluation of the OAST treatment is planned in the next year.

6. COMPONENT AND FULL-SCALE TESTING

6.1 - Development fatigue test of M-346 components (Alenia Aermacchi)

A development fatigue test has been planned on the Carbon Fibre cabin floor, namely in the area of the joint with the fuselage skin (see Fig. 32). The aim of the test was to provide evidence of the capability of the structure to sustain fatigue loading in presence of defects and to show also the no-growth of such defects. The test set-up was such that the T-section (see Fig. 33) is maintained on only one side of the floor panel, and cut away on the other side (in order to introduce more easily the proper constraints and loads). The T-section is connected to a back-up structure, that simulates the fuselage skin and frames, in a very similar way to the real aircraft. Two dummy elements simulate the spars of the NLG attachment. The pressurization loads are applied to the floor structure by means of 5 pads, loaded perpendicularly to the floor, while the loads in the plane of the floor are introduced in the test article by means of a whiffle-tree system connected to 8 sandwiches.
The test is articulated in various phases:

1. Conditioning at 75°C and 85% RH;

Application of loads: only those deriving from the cabin pressurization (they are predominant with respect to those due to manoeuvers), that means definition of a flight-by-flight sequence of the pressurization events, representing 500 FH, applied to reach 2 lifetimes;

3. Residual Strength static test at UL with a further load enhancement factor equal to 1.15 as “Environmental Knock Down Factor”;
4. Ultrasonic NDI during and at the end of the fatigue test, and after the residual static test.

The start of the test is planned in April 2011.

6.2 – M-346 Certification Full Scale and Component fatigue tests (Alenia Aermacchi)

Some information on these tests was already given in the last National Review; in the last two years progresses have been made in the definition of the load spectra (by means of an extensive load survey activity performed on prototypes 1 and 2) and in the preparation of the tests. The Full Scale Fatigue Test (FSFT), that is going to start by the end of March 2011 (see Fig. 34), will comprise the fuselage and the wings, including the leading edge flaps (i.e. nose droops) but not the trailing edge flap and the aileron, that will be separately tested, as well as the vertical and horizontal tails. The loads introduced on the FSFT by these components will be applied using suitable dummies/tools or directly by actuators acting on their aircraft structure attachments.

The non-stationary (buffet) load spectra will be applied basically with two different approaches:

- Quasi-static approach, introducing loads by means of actuators; the dynamic contribution is superimposed to the static one. This procedure will be followed for the FSFT and, partially, for the horizontal tail.
- “Pure” dynamic approach, exciting the structure normal modes involved in the buffet phenomenon by means of shakers, appropriately applied to the item. This procedure will be followed for the vertical tail and, partially, for the horizontal tail.

Dedicated test rigs are under development for the performance of the Vertical Tail test and the Horizontal tail test. In both cases, manoeuvres and gust loads will be applied to the structure by means of actuators, while buffet loads will be applied with two approaches: in the HT case, a quasi-static part of the loads will be applied by means of actuators, while the dynamic part will be applied by means of shakers (some details about the management of buffet loads were given in the previous National Review), while for the VT buffet load will be introduced only by means of shakers. A number of “control points” have been selected in the Durability assessment, and the test load conditions have been validated by means of the equivalent damage approach. The flight-by-flight load sequence will be applied two times for the durability test and one more lifetime for damage tolerance demonstration.

In the case of the HT test, particular attention has been paid to the simulation of the buffet phenomenon in the test:

- a modal reduction has been carried out (from 7 used in the analysis to 2 included in the test; progressive elimination of modes has been carried out in order to assess their influence on the final fatigue damage);

- appropriate selection of the points of shaker positioning, for a better simulation of the two modes identified;

- identification of the mission segments where the “static” fatigue load induces a non-negligible damage, in comparison with the damage due to buffet, with the objective of defining an equivalent time-history to be applied by means for actuators, simulating both contributions.

6.3 - EF Typhoon (Alenia Aeronautica)

The Production Major Airframe Fatigue Test (PMAFT) is ongoing within BaeSystems facilities at Brough (UK). At the moment 9000 Flight Hours have been simulated (with design spectrum). Alenia Aeronautica is involved in this full scale test for the components under its responsibility and for the definition of buffet load spectra. The methodology followed by Alenia Aeronautica in defining the fatigue damage associated to buffet conditions was described in [13].

7. AIRCRAFT FATIGUE SUBSTANTIATION

7.1 – C-27 J program (Alenia Aeronautica)

The C-27 J aircraft is a derivative of the Alenia G222/C27A aircraft modified to meet expanded or more stringent system-level requirements as determined primarily by market assessments and certification requirements. The main modifications, with respect to the G222, are related to the new engine installation (new engine nacelle design) and to the new landing gear design.

The certification basis is EASA CS 25, for European Customers, and FAR 25 amendment 87 for USA Operators.

JCA (Joint Cargo Aircraft) is the name of the US version of the C-27 J. The Fatigue and Damage Tolerance Design and Certification (according to FAR 25 regulation) have been completed. The first aircraft has been delivered to the US Customer on October 2008, and up to now 21 aeroplanes orders are closed by the customer.

Information about the Individual Aircraft Tracking Program has been already given in section 2.6 of this Review.

7.1.1 - High cycle aerodynamic fatigue on wing leading edges

An investigation on the fatigue behaviour under high cycle aerodynamic fatigue loads has been performed on a wing item, by means of the Time History Derivation technique already presented at ICAF 2007 [13]. The aim of the investigation was to analyze the reasons of L.E. fatigue cracks experienced in service, due to vibrations induced by propeller aerodynamic turbulent wake, and to evaluate the relevant design modifications. The problem was originated by the change of the propeller; the new one has 6 blades and an interaction exists with a natural frequency of the structure.

A dynamic model was developed in order to simulate the dynamic response of the interested wing panels. To tune up the dynamic FE model, an aerodynamic excitation test was performed by means of a dedicated ground test. The wing panels response due to in-flight dynamic loads has been analyzed and the relevant stress Power Spectral Density graphs produced (see Fig. 35), from which a time-history has been deduced, following the procedure already outlined in [13].

Comparative fatigue calculations between the original and the modified structural arrangements were performed, in order to assess the efficiency of the new design. Unfortunately, the analyzed situation required the performance of calculations in a region where the S/N curves are not well defined, for the very high number of cycles; moreover, the calculation did not explain the existence of those cracks. It was therefore necessary to use a slightly unusual approach, in order to assess the potential improvements in fatigue associated with a change in the design: a power law (characterized by a typical value) was used to evaluate the relative damage of the cycles of such low amplitudes, in combination with a factor on the stress.

The method is very pragmatic and has low scientific bases, but has the merit of allowing the tuning of calculations with experimental evidence and of allowing the evaluation of the improvements introduced in the structural design. A more scientific test campaign should be necessary for a confirmation of such evaluations.

7.2 - ATR 42- ATR 72 Ageing program (Alenia Aeronautica)

The ATR aircraft family was designed for a DSG (Design Service Goal) of 70,000 Flights.

The ATR Ageing Structures Program has the goal to extend the Airframe Service Life from the original DSG to 105,000 flights. The ATR Fleet is composed by more than 1000 aeroplanes, delivered since 1985, and has reached about 20 millions Flight Hours. The current life of the Fleet Leader Aircraft (a -72 aircraft) has overcome 60,000 Flights (corresponding to about 50,000 Flight Hours). All the activities necessary for the *Life Extension* have been competed and presented to the Airworthiness Authority: a description of the activities planned was presented at the ICAF Symposium in Naples, [14].

The Official Certification of the Extended Life is expected before the end of 2011. In the meantime, a new version of the aircraft is under development.

ATR 72 - TMPA (Turkish Maritime Patrol Aircraft) Program

The Structural Design, Analysis and Certification were carried out for a Maritime Patrol version of the ATR 72, requested by the Turkish Air Force. New Damage Tolerance Analysis is in progress according to the dedicated Maritime Patrol mission profiles and mixing, as defined by the Customer.

7.3 - Boeing B 787 – Fuselage Sections 44-46 and Horizontal Tail Damage Tolerance design (Alenia Aeronautica)

Information about the level of involvement of Alenia Aeronautica in this program was already given in the last Review. Here, it is important to recall that, together with the design, Alenia is in charge of the Fatigue/Damage Tolerance analysis and certification of the components under its responsibility. For the fuselage sections, the design and manufacturing are Alenia's responsibility, while the testing is planned in Boeing facilities. As far as the horizontal tail is concerned, Alenia is in charge for the performance of its Static and Fatigue Test.

The tests on the Horizontal Tail of the basic version of the aircraft (the -8), were described in the previous National Review; the experimental activities have been successfully concluded both for the composite test article and for the metallic one.

The Fatigue and Damage Tolerance design of B.787-9, a stretched version, is in progress, with the same work sharing among the various partners involved in the program.

7.4 - Development of the RRJ Superjet 100 (Alenia Aeronautica)

At the end of the year 2006, the cooperation with the Russian Industry SCAC (Sukhoi Civil Aircraft Corporation) was launched in Alenia, for a common activity related to the Design, Analysis and Manufacturing of the RRJJ Superjet 100.

Alenia has performed the Fatigue Design of the rear fuselage, the rear pressure bulkhead and both the empennages. The prototype was rolled out from Sukhoi Komsomolsk-on Amur factory on 26 September 2007. The first flight was successfully accomplished last May 19, 2008.

During 2010 Alenia, in the Turin facilities, has successfully performed the Certification Tests for Community Noise and acquired data for Sonic Fatigue clearances.

To date Alenia, by a structural team in Moscow, is supporting the Full Scale Fatigue Test which is in progress. The aircraft will be certified according to EASA CS25 regulation. So far, the equivalence of two years of usage have been simulated, so that the clearance has been reached for the flight.

7.5 - Development of the Bombardier C Series (Alenia Aeronautica)

This is a new project, or better a restart of a suspended previous project, for a narrow body, medium range twin engine aircraft. An artist's impression is shown in Fig. 36. Together with the Design, Alenia is in charge of the Fatigue/Damage Tolerance Analysis and Certification of Horizontal and Vertical Tail Structures. Most of such structures will be manufactured in composite materials.

In addition Alenia is in charge for Static and Fatigue Test development for: material, structural details and component characterization. A very large experimental activity is planned, to qualify material and processes, according to the "pyramid of test" concept. The performance of Full Scale Tests is under Bombardier responsibility.

7.6 - AW 139 Helicopter (AgustaWestland)

The civil certification of the AW139 helicopter has further been developed with:

- 6800 Kg ETOW (extended take off weight);
- Goodrich dual rescue hoist.

Moreover, in February 2010 the AW 139 FIPS (Full Ice Protection System) was certified by EASA; it is the first civil helicopter of relatively small size (6 tons) to have this kit. From the structural point of view, the major consequences of FIPS are that the MR blades are equipped with a de-ice system to remove periodically ice accumulated on the blades themselves and the TR Blades have an anti-ice system preventing ice formation by blade heating. Some fatigue related issues are relevant to FIPS:

- assessment of fatigue strength of heated composite blades, that required the performance of dedicated tests;
- impact damage due to ice shedding; this was evaluated by means of susceptibility tests plus flight trials evidences and proper trajectories
- change in fatigue loading spectra due to higher drag (for the MR and the airframe, due to ice accretion), greater power required and higher TR trust to balance the increased MR torque. This was the challenging task, because of the target of minimum penalty.

Icing occurs in cloud and therefore turbulence is superimposed to the flight condition. Moreover, ice accretion change loads and power required during any 'nominally stabilized' flight condition. A direct correlation of flight data in ice and out of ice condition showed a relevant load amplification and therefore a high fatigue life penalty.

Considering that the pilot maintains the aircraft within torque range and usually operates in IFR conditions with a more gentle maneuvering due to the adverse environment, a compensating effect should occur.

The fatigue evaluation was therefore managed using several continuous flight recordings of all critical parameters (about 120) asking the pilot to fly representative maneuvering during icing condition. Rainflow analysis was carried out for all data during the FIPS flight phases (about 40 h) and the damage rate was computed and correlated to ‘normal’ environment, proving no penalty.

7.7 - AW149 development (AgustaWestland)

AW149 is a new model in the development phase; the first prototype flew in 2010 and the second prototype is ready to fly and activities for achieving the Flight Readiness Review have been successfully concluded.

The fatigue clearance was supported by tests and analysis; the fatigue tracking of the first prototype is in progress.

7.8 - M-346 trainer (Alenia Aermacchi)

M-346 is the Alenia Aermacchi Advanced/Lead-In Fighter Trainer, designed to allow the accomplishment of all the needs in Phase III and Phase IV of a military pilot training. The M-346 Durability and Damage Tolerance assessment proceeded in the last two years covering the activities related to development and qualification. Information has already been given in sections 6.1 and 6.2 of this Review about a development fatigue test planned on the CFRP cabin floor joint to the fuselage skin and on the preparation activities for the Full Scale Fatigue Test on the series configuration.

The design spectrum has been updated on the basis of the results of the load survey activity performed on prototypes 1 and 2. The analytical verification of the airframe has been iterated with the updated data and the Full Scale Fatigue Test has been designed accordingly. Fatigue and Damage Tolerance criteria are applied following the military international regulations, such as JSSG-2006.

As far as the analytical methodologies development is concerned, with particular reference to buffet and flight-by-flight spectra generation, this activity was already described in detail in the last National Review. The load spectra have been updated, considering the results of the load survey on prototypes.

For what concerns the Durability assessment, it is divided into two types of analyses: the first one aims at demonstrating that no crack will initiate in the design life (with a scatter factor of 4) and the second one is based on a crack growth analysis, with the objective of demonstrating that a standard quality initial flaw of 0.254 mm will not grow to critical dimensions (or cause functional impairment) in 2 design lifetimes.

The Damage Tolerance approach defines the threshold inspection, as the time required by a “rogue” flaw of 1.27 mm to propagate to criticality, divided by 2.

Extensive analytical and experimental work has been performed, and already presented in previous National Reviews.

7.9 - UAS Technology (Alenia Aeronautica)

In the last years Alenia Aeronautica was strongly interested in the UAS (Unmanned Aerial System) technology development. The main UAS’s programs in which Alenia was involved are:

- ✓ SKY – X for basic technology demonstration;
- ✓ SKY – Y for MALE (Medium Altitude Long Endurance) technology;
- ✓ Molynx – Civil UAV for Control and Surveillance;
- ✓ Neuron – European UAS technology demonstrator.

The Sky-Y MALE Technology Demonstrator is a dedicated platform for validating several key enabling technologies, for the surveillance in either military and civil operational scenarios. The demonstrator had its first flight in June 2007; later its FCS was modified and a new version, completely developed by Alenia, was validated in flight in 2010.

Within the international Neuron program (a stealth UAV, see Fig. 37), Alenia has the primary responsibility for:

- ✓ Sonic Fatigue Design;
- ✓ Glass Window Design and manufacturing under Damage Tolerance and Residual Strength requirements.

The Neuron weapon bay door, during opening, is subjected to vibration phenomena and so a fatigue analysis has been carried out; the mechanisms for opening and supporting the door are metallic parts, and the analysis performed assured an available life in the present situation and evaluated the life in presence of improvements in stress levels and design (Kt reduction).

The NATO RTO AVT-174 Working Group has the task of preparing a report with "Guidelines on Structural Design and Qualification for Unmanned Military Air Vehicles". Alenia Aeronautica takes part in this WG and its contribution is focused on the preparation of the chapter on Durability and Damage Tolerance evaluation and on their relevant Structural Qualification aspects. Special attention is paid to evaluate the possibility of reducing the level of effort required, particularly the testing requirements. The issue of the final report is foreseen for Fall 2011.

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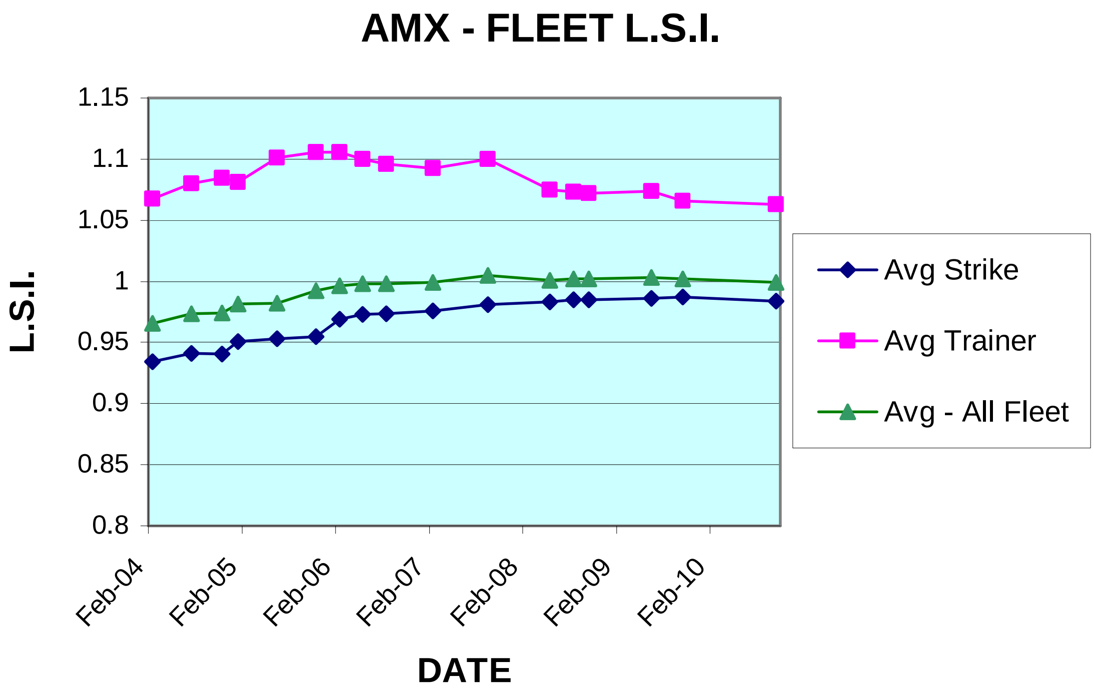


Fig. 1 – Load Severity Index distribution for the AM-X fleet.

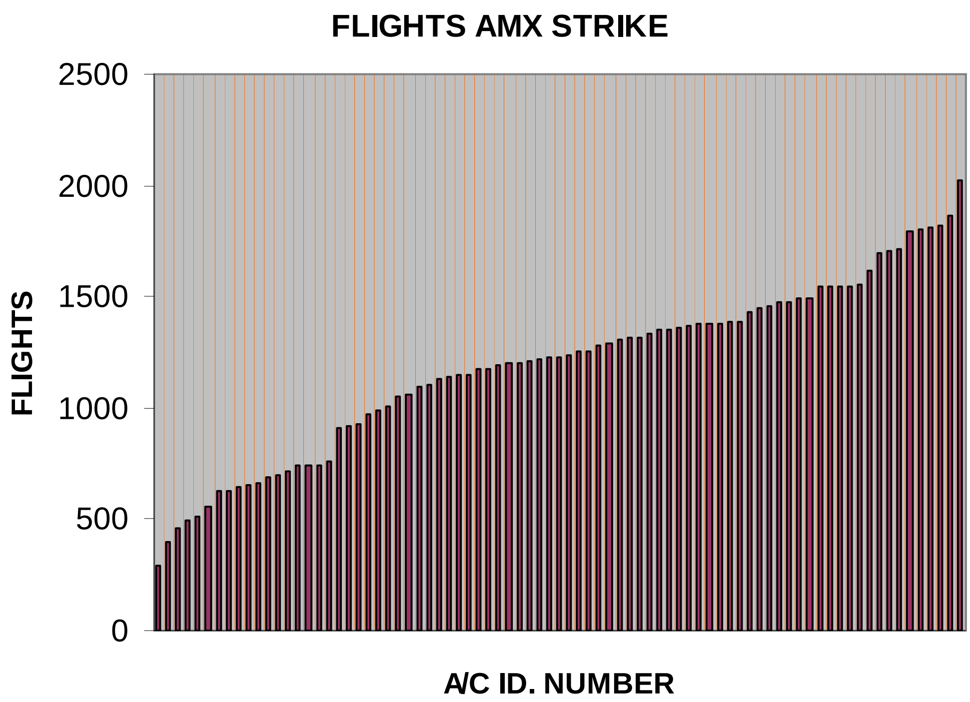


Fig. 2 – Distribution of Flight Hours in the AM-X Strike fleet

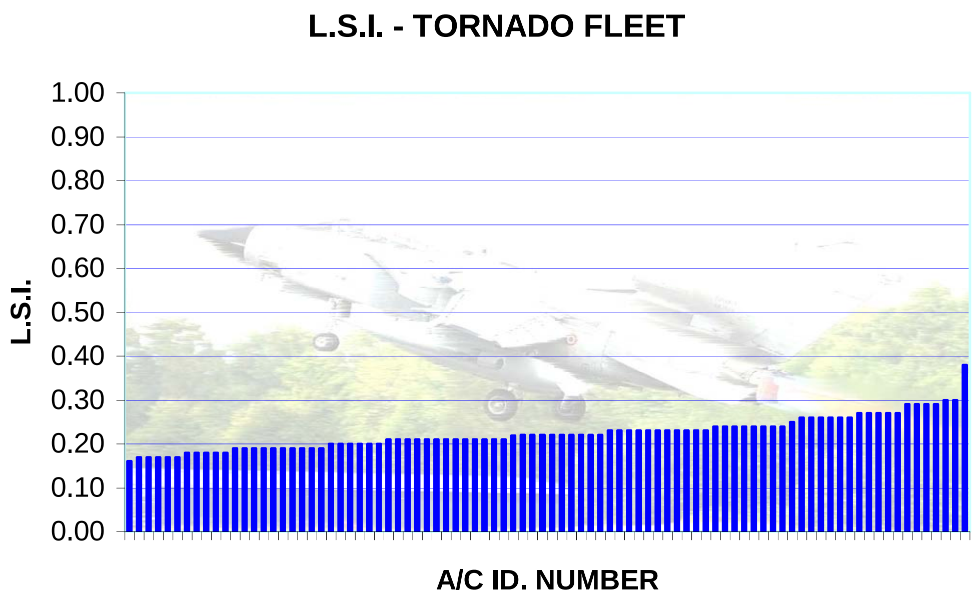


Fig. 3 - Load Severity Index trend for the aircraft of the I.A.F. Tornado fleet.

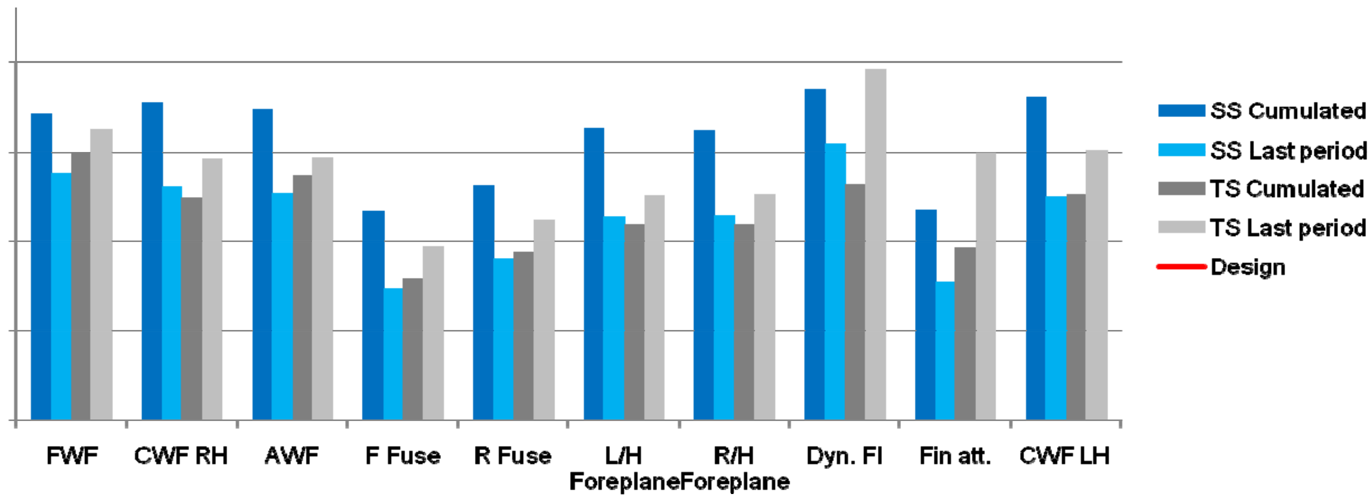


Fig. 4 - Fatigue Indexes values for IAF Typhoon aircraft.

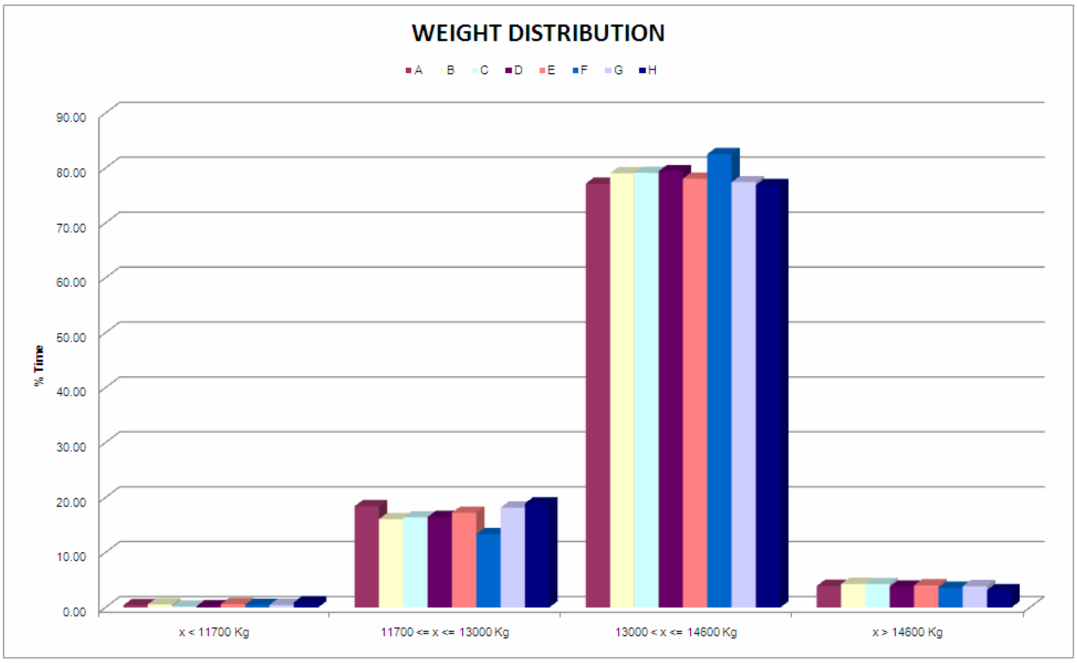


Fig. 5 - Weight distribution of the monitored machines of the Danish AW101 fleet.

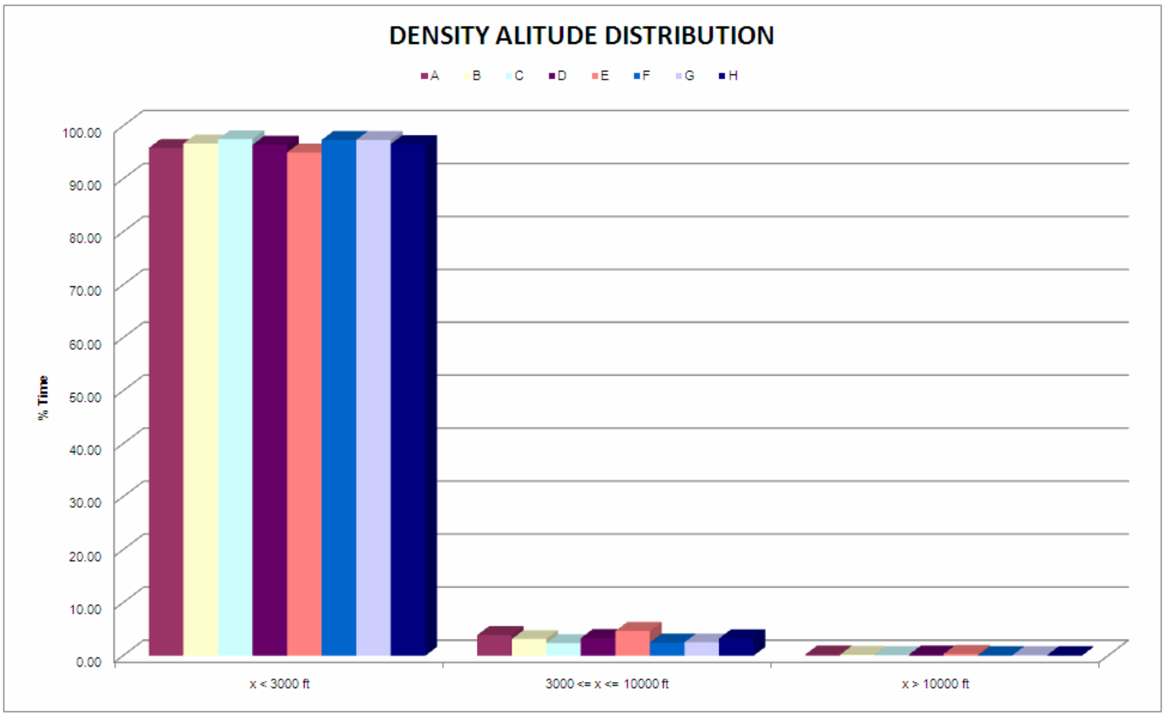


Fig. 6 - Altitude distribution of the monitored machines of the Danish AW101 fleet.

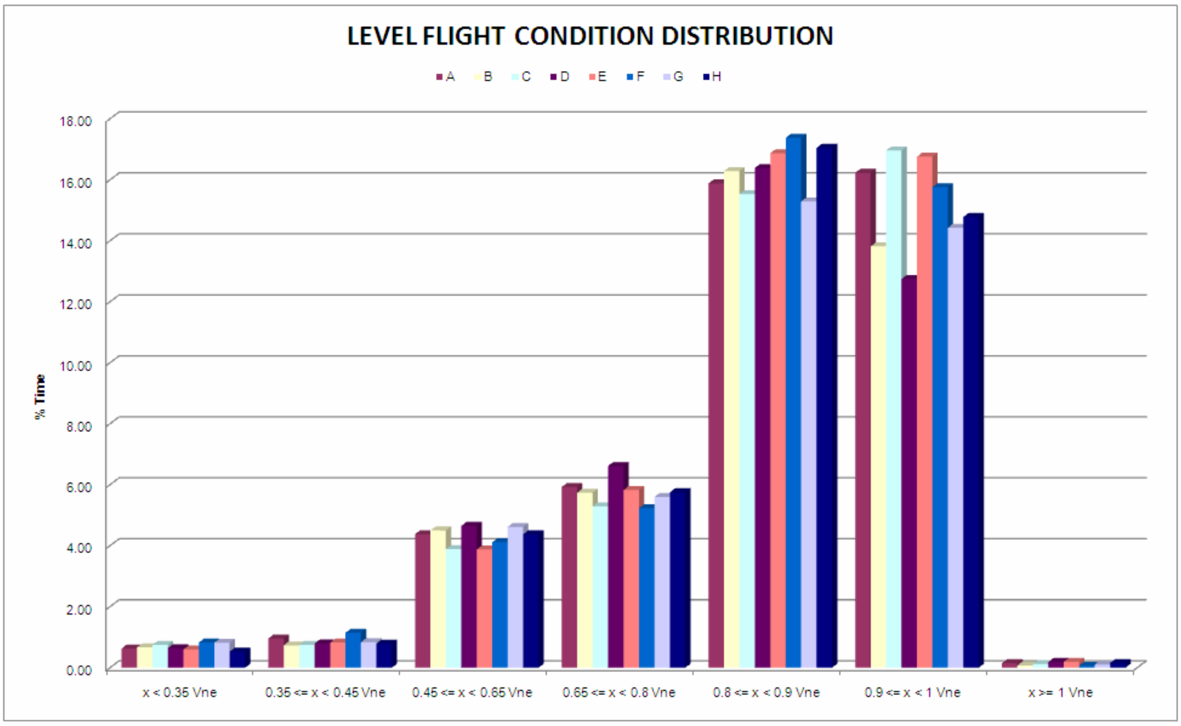


Fig. 7 - Distribution of level flight speed in the monitored machines of the Danish AW101 fleet.

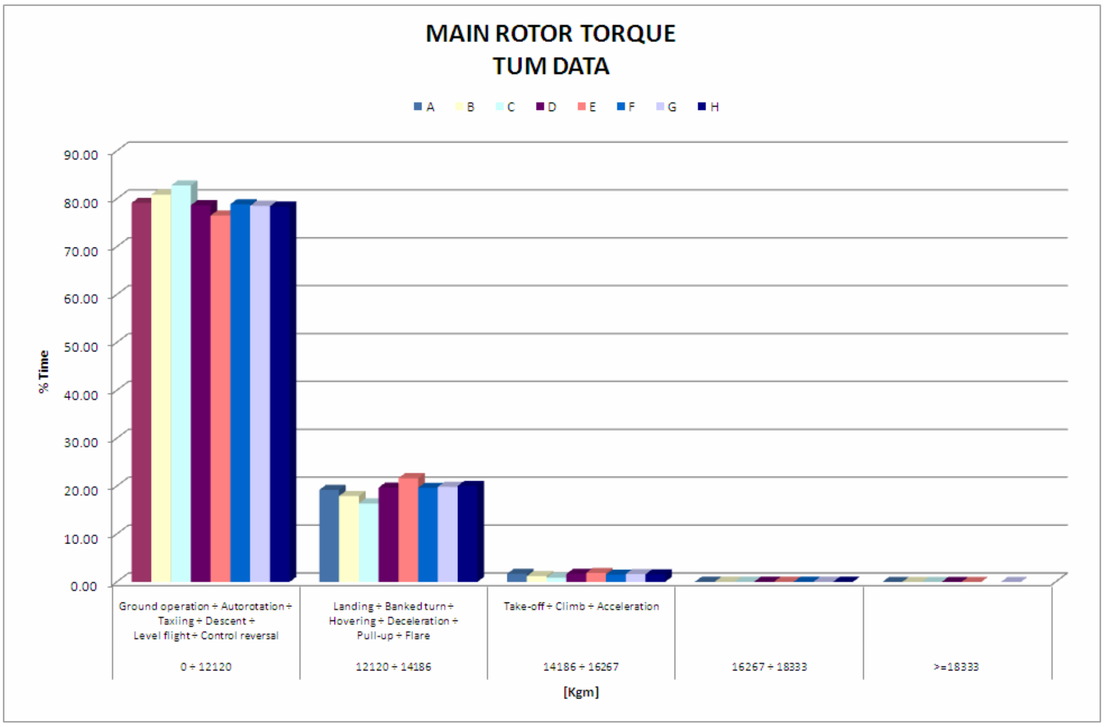


Fig. 8 - Main Rotor Torque distribution in the AW101 monitored machines of the Danish fleet.

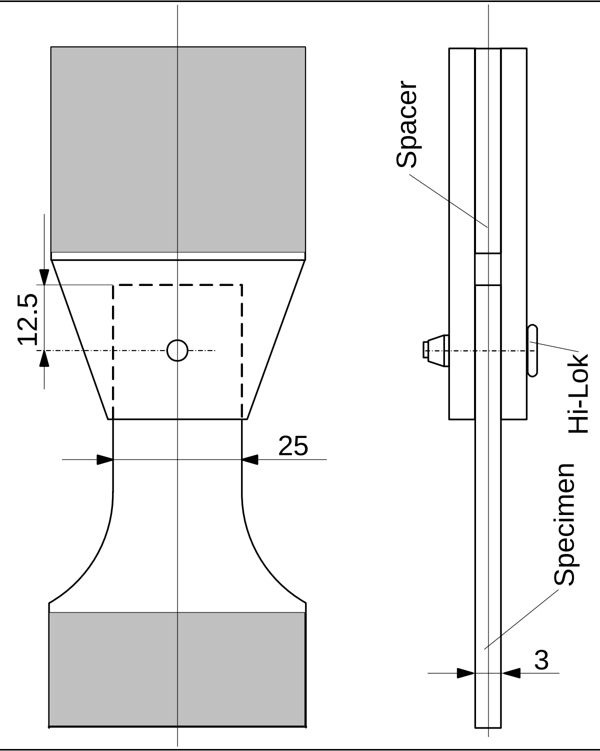


Fig. 9 – Specimen used for assessing the sealant influence on fatigue behaviour.

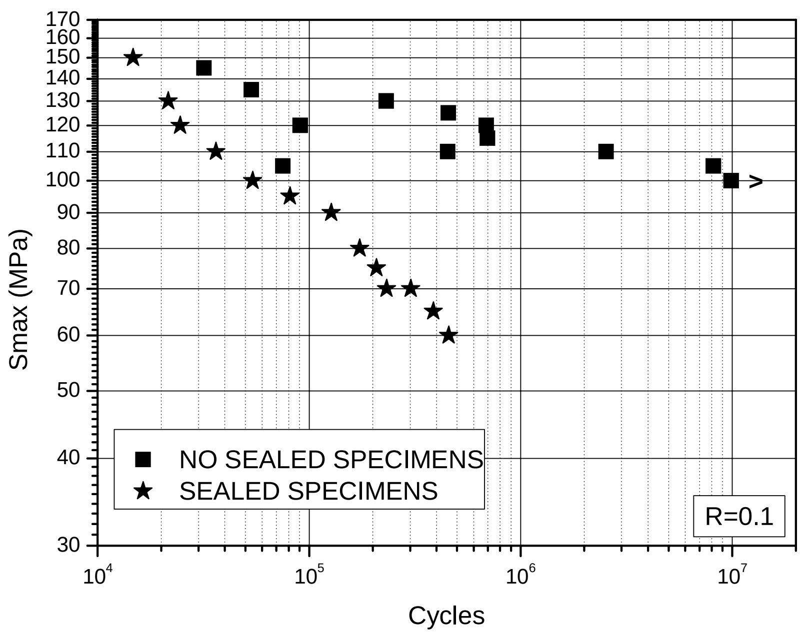


Fig. 10 – Fatigue test results of sealed and un-sealed specimens.

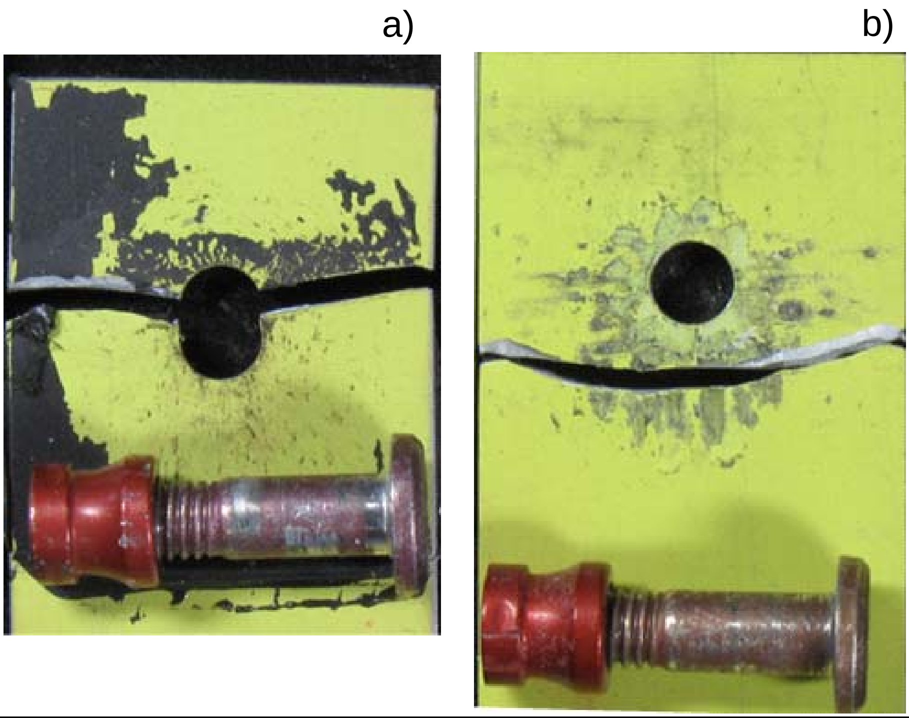


Fig. 11 –Typical failure modes of sealed (left) and un-sealed specimens (right).

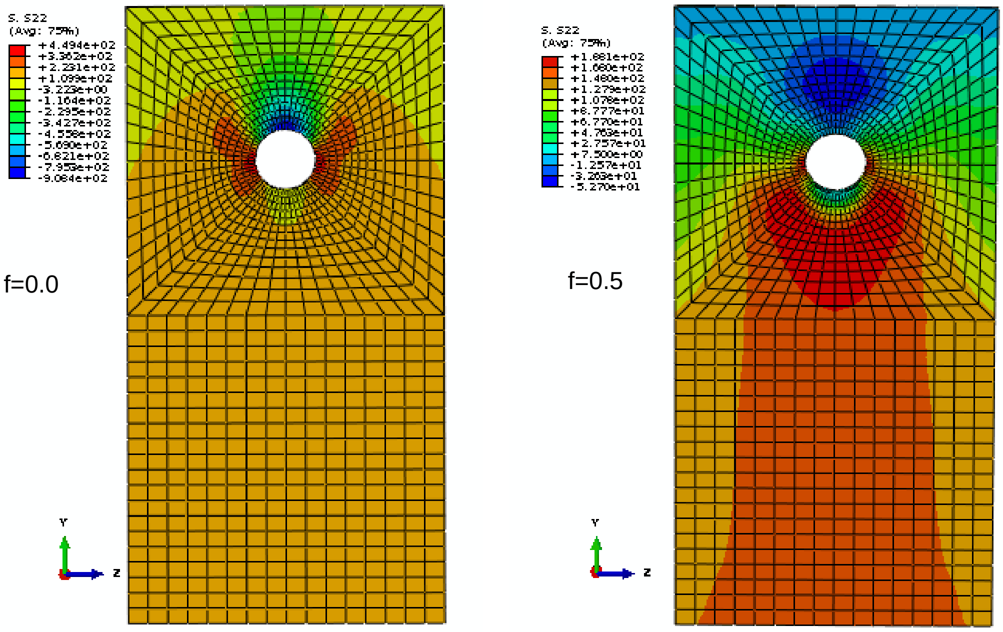


Fig. 12 – Modifications in the longitudinal stress state for different values of the friction coefficient, f=0.0 (left) and f=0.5 (right). Applied stress S=150 MPa.

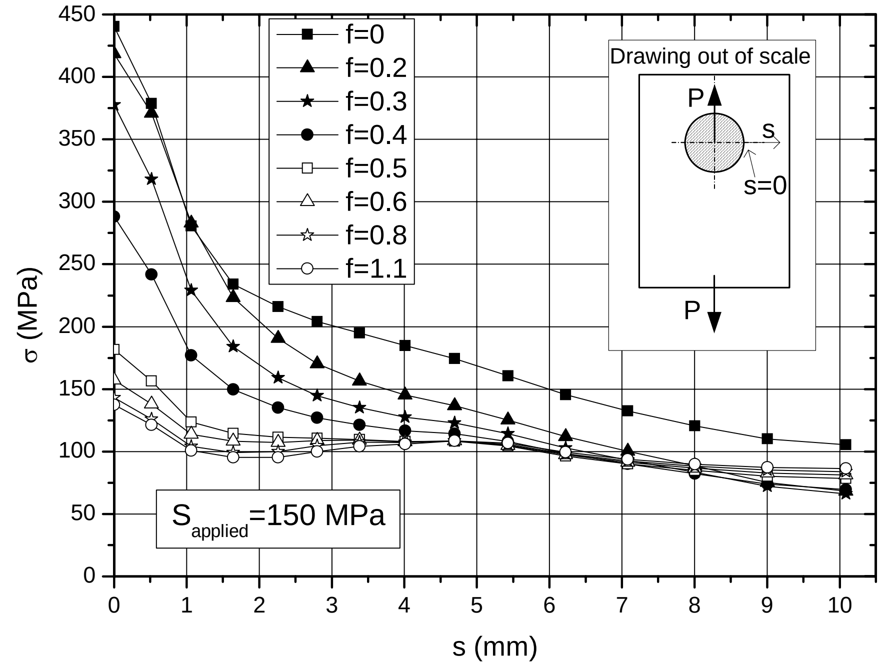


Fig. 13 – Longitudinal stress along a transverse path for different values of the friction coefficient.

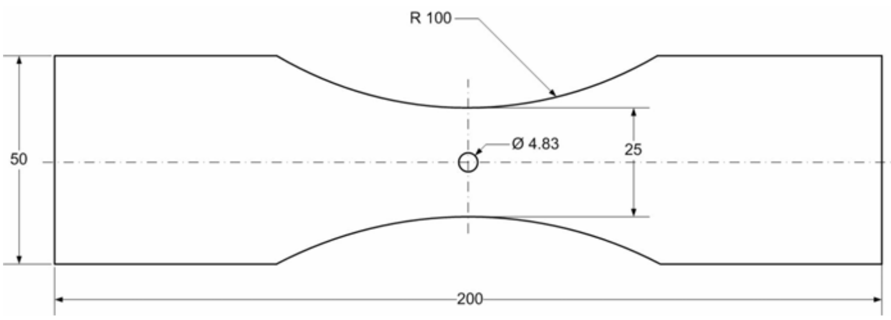


Fig. 14 - Open hole specimen in thin sheet (Laser Shock Peening)

Laser type [-]	Wavelength [nm]	Output energy [J]	Pulse duration [ns]	Laser frequency [Hz]
Nd-YAG	1064	2.8 (10% loss)	9	10

Table I - Laser properties used for LSP treatment of thin specimens (Laser Centre, UPM Madrid).

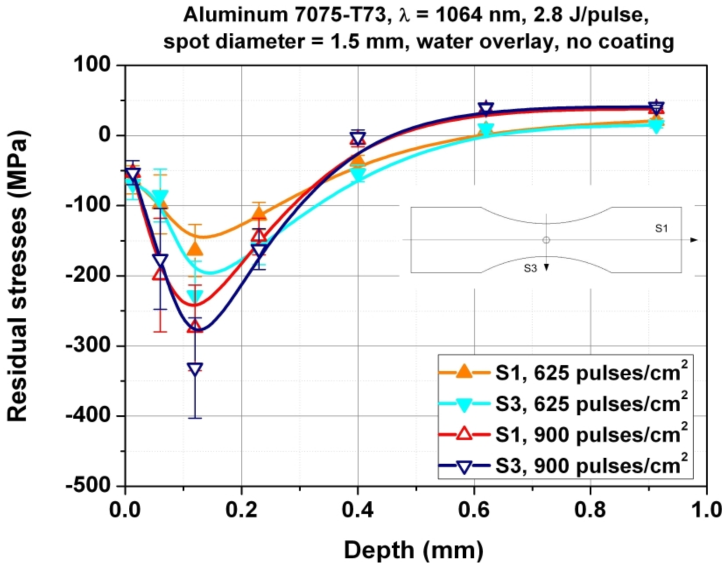


Fig. 15 - Residual stresses introduced in the treated specimen measured using hole drilling method.

a) One side treatment

b) Two side treatment

Fig. 16 - Deformation of the LSP treated specimen

Treatment	Fatigue life (cycles)	Difference
Baseline	10^5	1x
LSP as treated	$3.47x10^4$	0.3x
LSP polished hole	$3.45\ x10^4$	0.3x
LSP polished treated zone	$3.21\ x10^4$	0.3x
Cold working	$6.5x10^5$	6.5x
Stress Wave	$8.5x10^6$	85x

Table II - Fatigue lives with different open-hole treatments, at S_{max} 160 MPa and R=0.1.

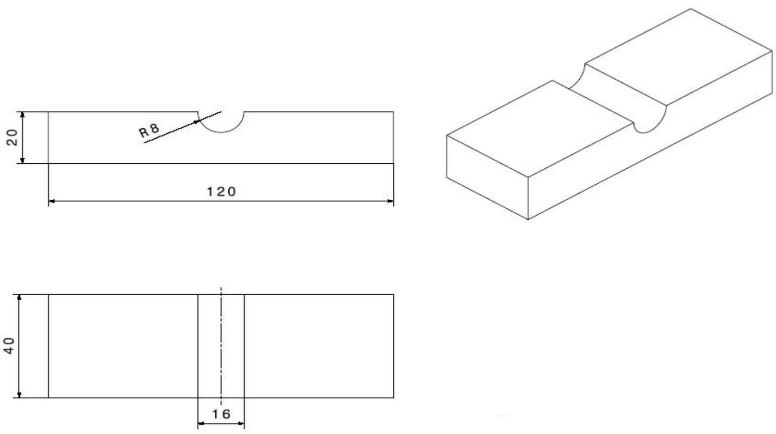


Figure 17 - Geometry of the thick specimen used to assess Laser Shock Peening effects.

Laser type [-]	Wavelength [nm]	Output energy [J]	Pulse duration [ns]	Laser frequency [Hz]
Nd-YAG	1064	1.8	9	10

Table III - Laser properties used for LSP treatment of thick specimens (University of Bologna)

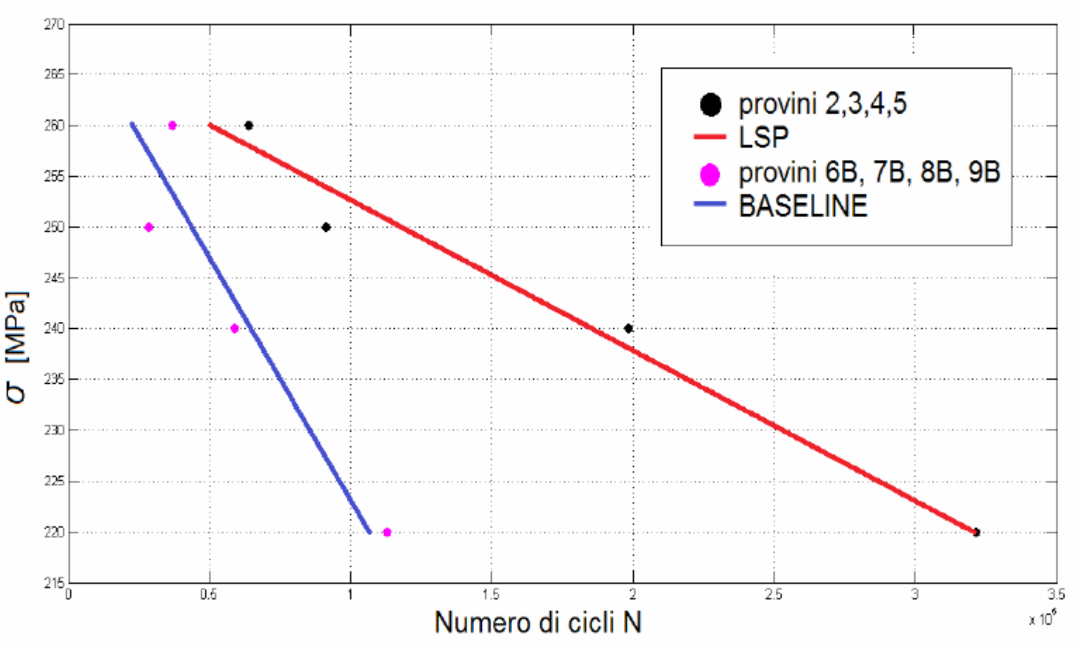
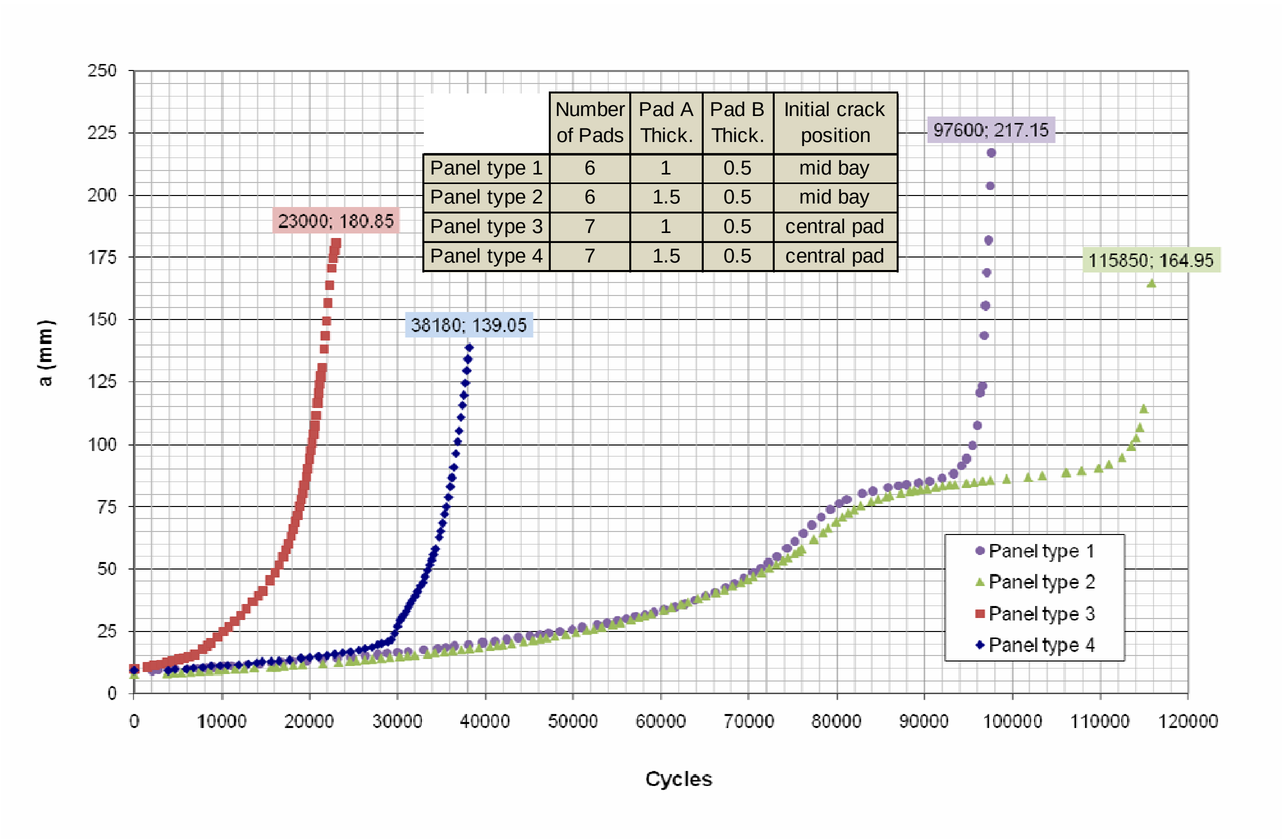


Figure 18 - Fatigue lives of thick specimens treated with LSP.

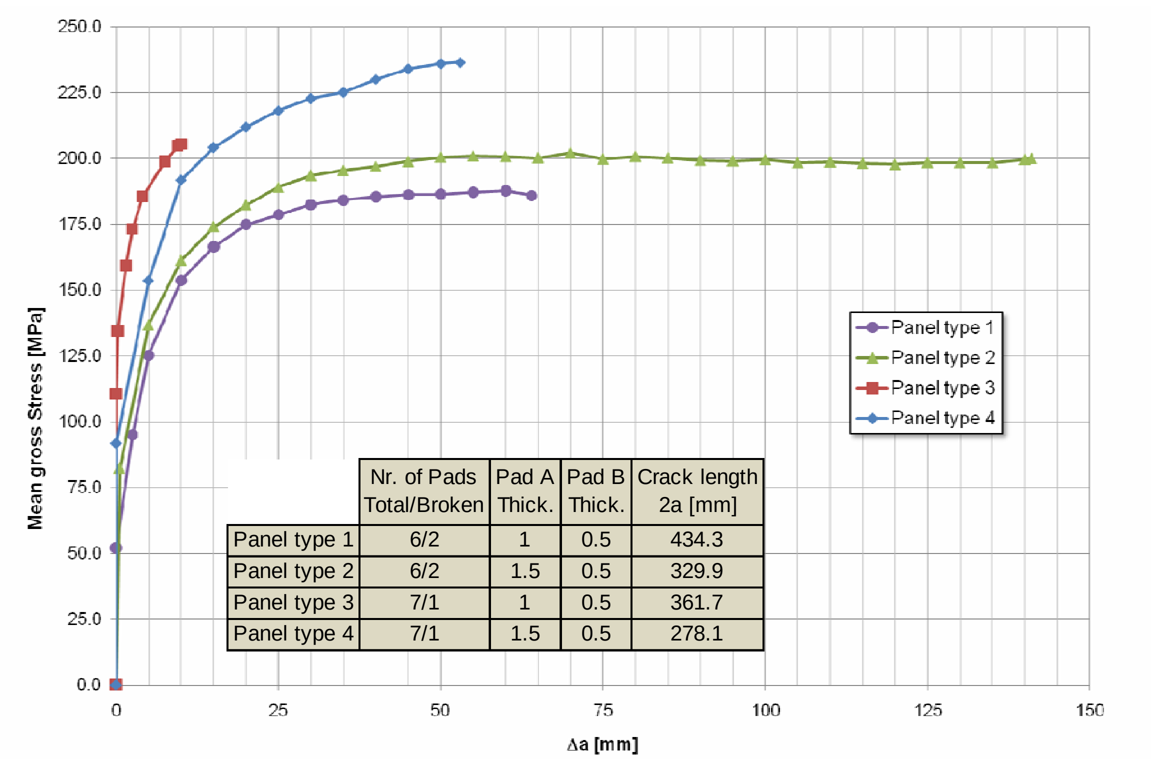
Panel Type	N. of bays	N. of bonded pads @ 210 mm pitch	Skin thickness mm	Pad layer A thickness mm	Pad layer B thickness mm
1	7	6	1	1	0.5
2	7	6	1	1.5	0.5
3	6	7	1	1	0.5
4	6	7	1	1.5	0.5

Table IV - Test articles relevant data.



	Number of Pads	Pad A Thick.	Pad B Thick.	Initial crack position
Panel type 1	6	1	0.5	mid bay
Panel type 2	6	1.5	0.5	mid bay
Panel type 3	7	1	0.5	central pad
Panel type 4	7	1.5	0.5	central pad

Fig. 19 – Summary of crack propagation results: a vs N curves



	Nr. of Pads Total/Broken	Pad A Thick.	Pad B Thick.	Crack length 2a [mm]
Panel type 1	6/2	1	0.5	434.3
Panel type 2	6/2	1.5	0.5	329.9
Panel type 3	7/1	1	0.5	361.7
Panel type 4	7/1	1.5	0.5	278.1

Fig. 20 – Summary of residual strength test results: mean gross stress vs. Δa curves

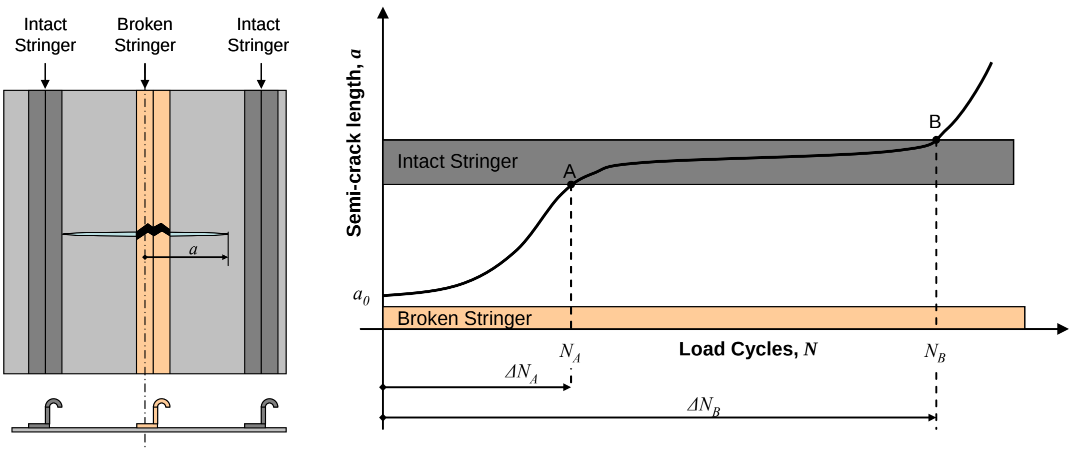


Fig. 21 - Fatigue Crack Propagation (FCP) through bonded stiffened panels.

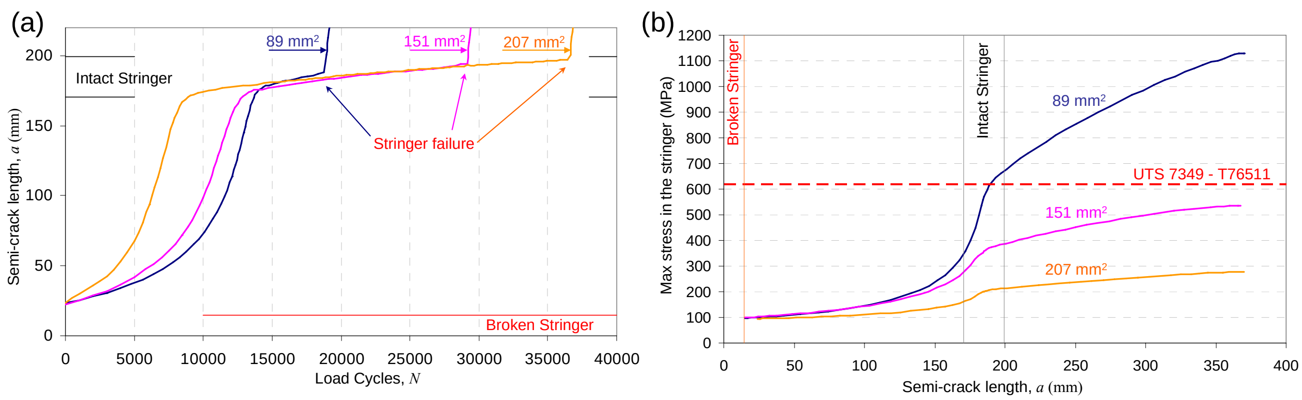


Fig. 22 - FCP in bonded stiffened panels over a broken stringer: (a) experimental FCP curves; (b) analytically predicted maximum stresses in the stringer adjacent to the broken one. The panels have 7349-T76511, “z-shaped” stringers of, respectively, 89, 151 and 207 mm2 cross-section.

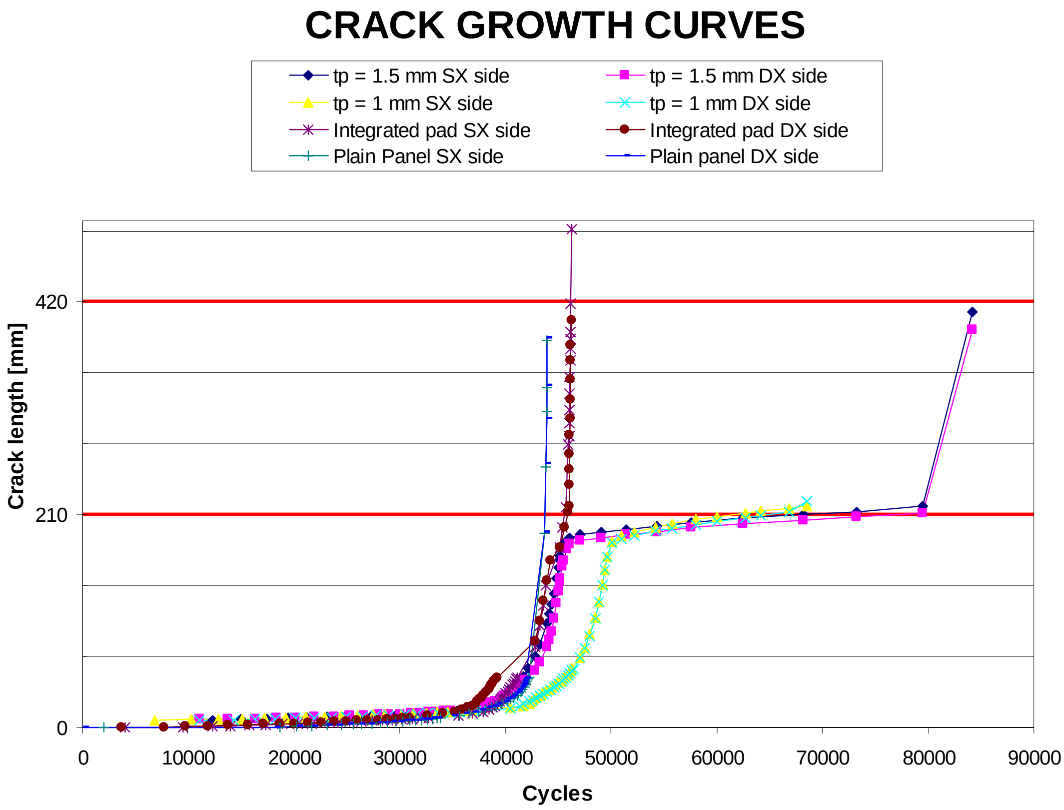


Fig. 23 - Comparison of various crack growth curves obtained.

Fig. 24 - Photo of the barrel test set-up.

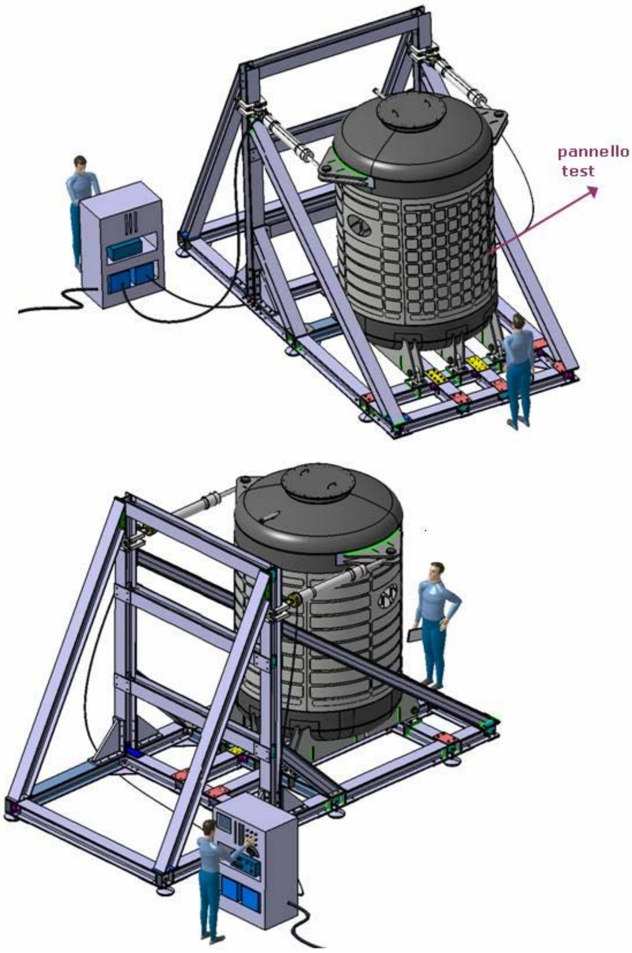
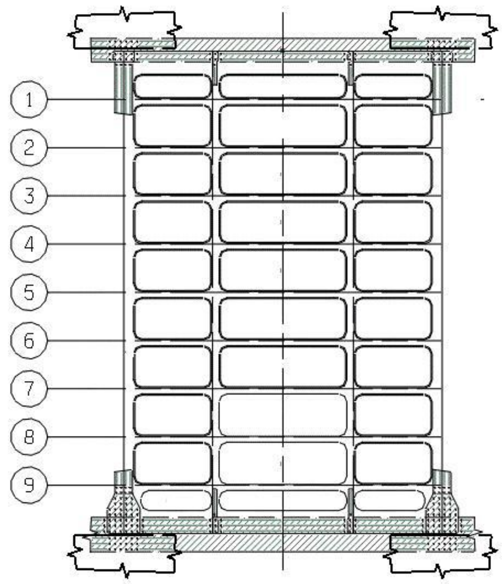


Fig. 25 - Barrel test set-up: sketch of the constraints and load application system.



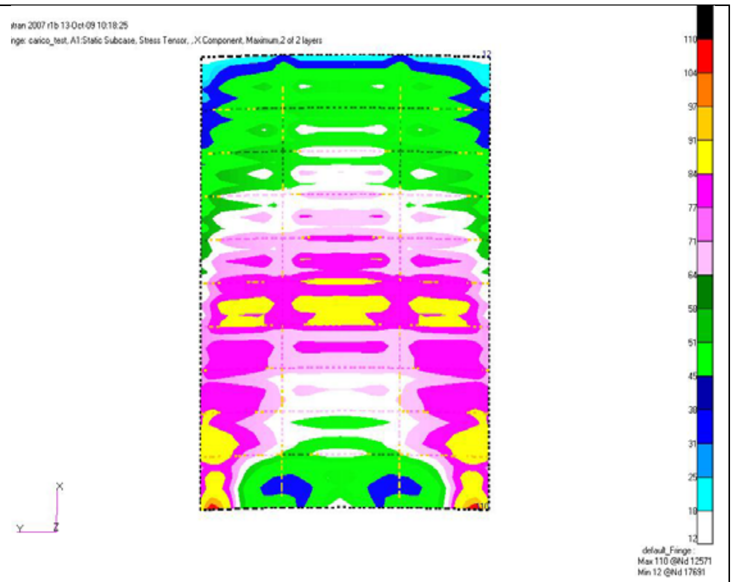


Fig. 26a - Sketch of the crown panel tested in the barrel rig.	**Fig. 26b - FE model analysis: stress distribution in the skin.**

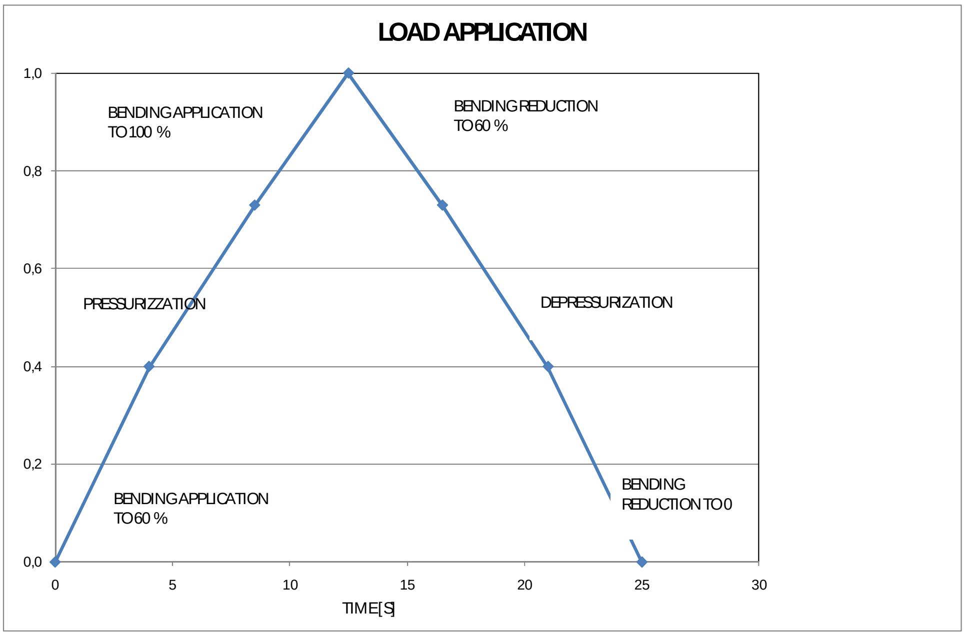


Fig. 27 - Sequence of combined load application in the barrel test.

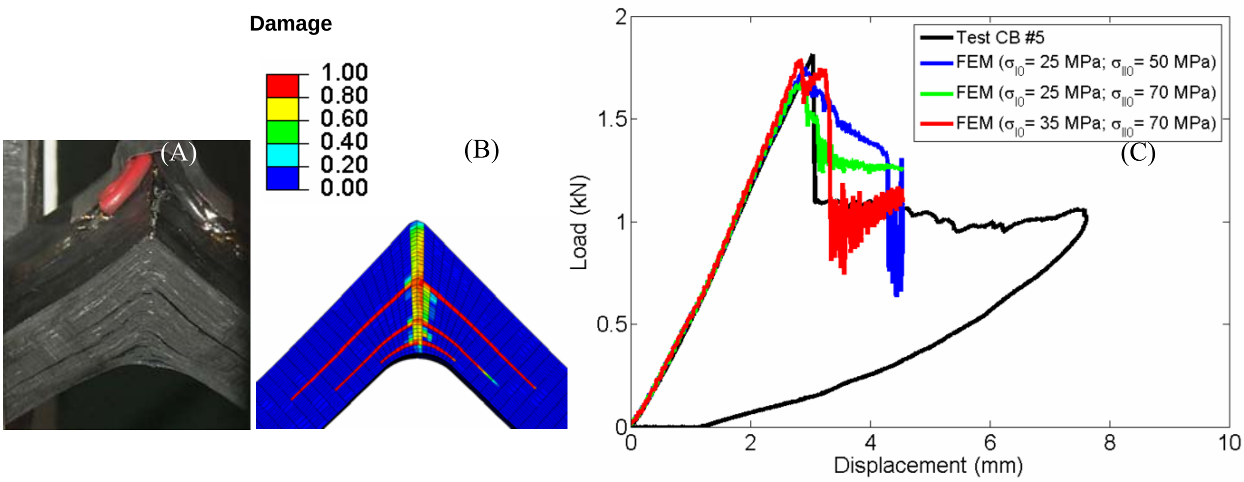


Fig. 28 – Four point bending test on curved laminate: (A) experimental evidences, (B) interlaminar damage in the numerical model and (C) load vs. displacement numerical-experimental correlation

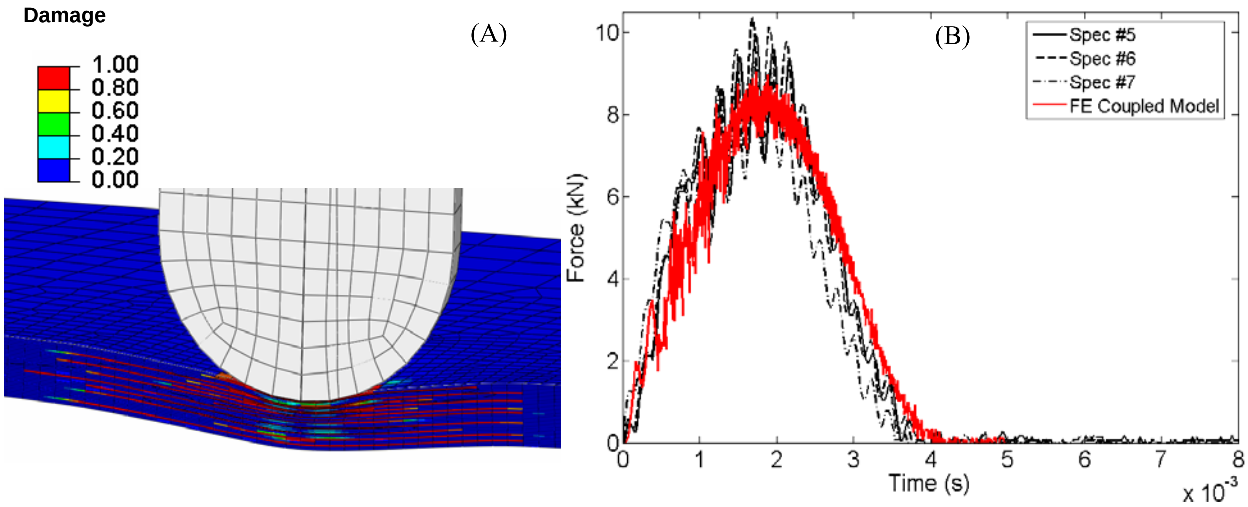


Fig. 29 – Low energy impact test simulation: (A) contour of interlaminar damage and (B) numerical-experimental correlation of the force vs. time history response

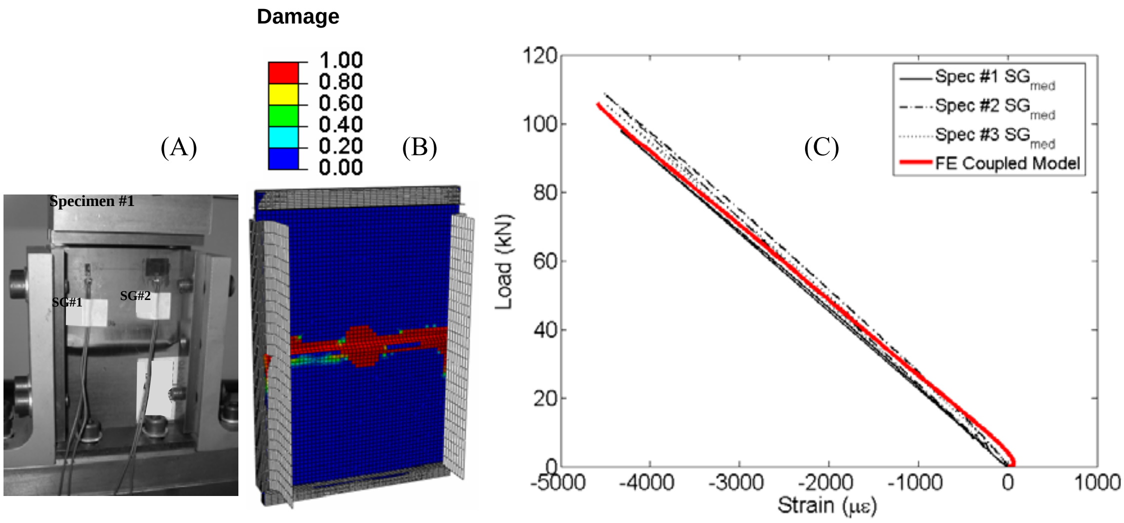


Fig. 30 – Compression after impact test: (A) experimental evidences, (B) final failure mode in the FE Coupled Model and (C) numerical-experimental correlation of the load vs. strain response

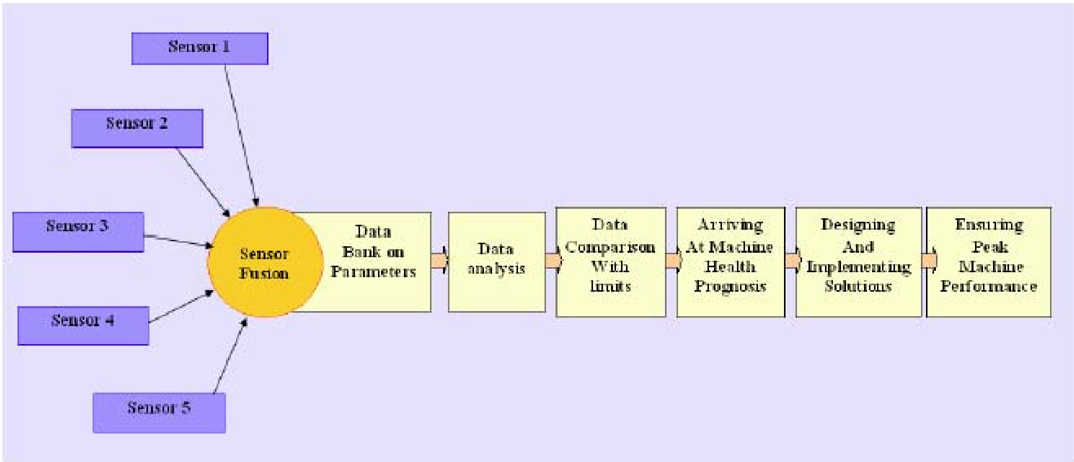


Fig. 31 - Sketch of the logic flow of actions for the damage detection and prognosis in helicopter fuselage.

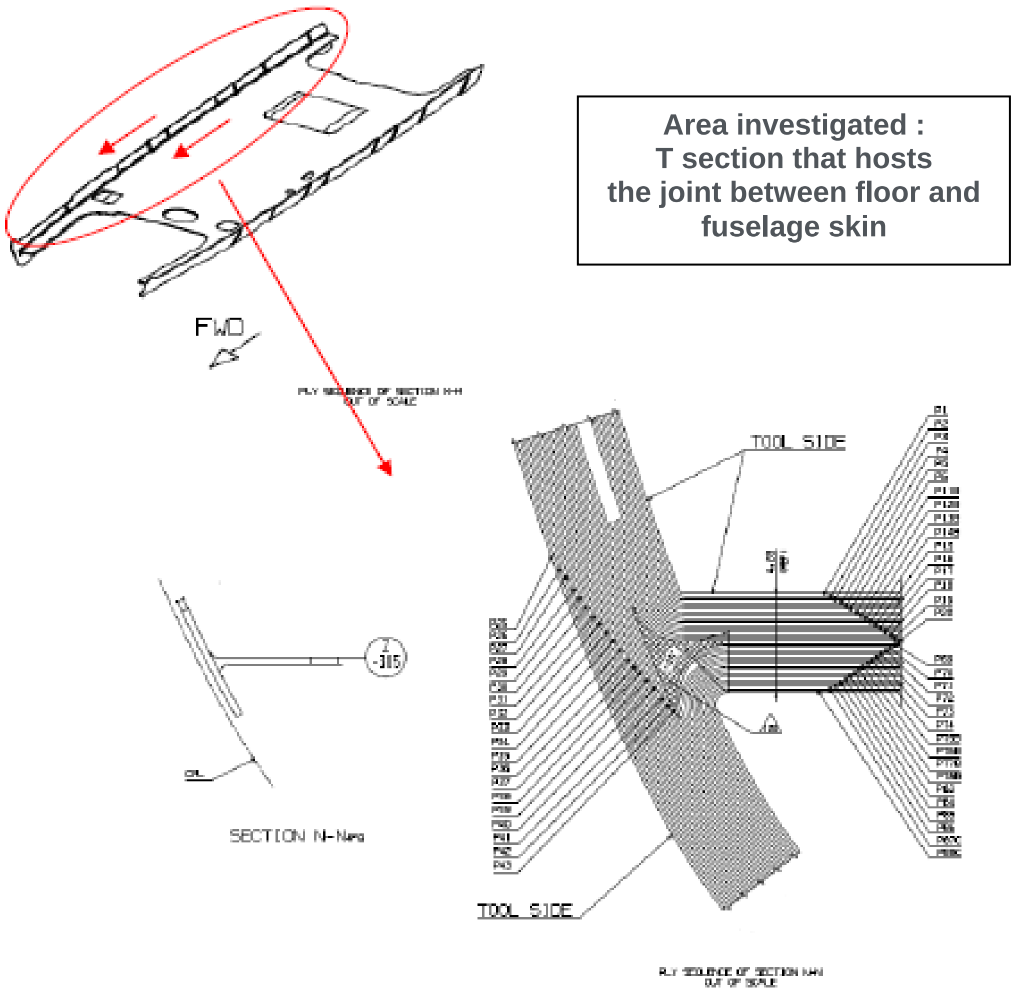


Fig. 32 - M346 CFRP cabin floor to fuselage joint.

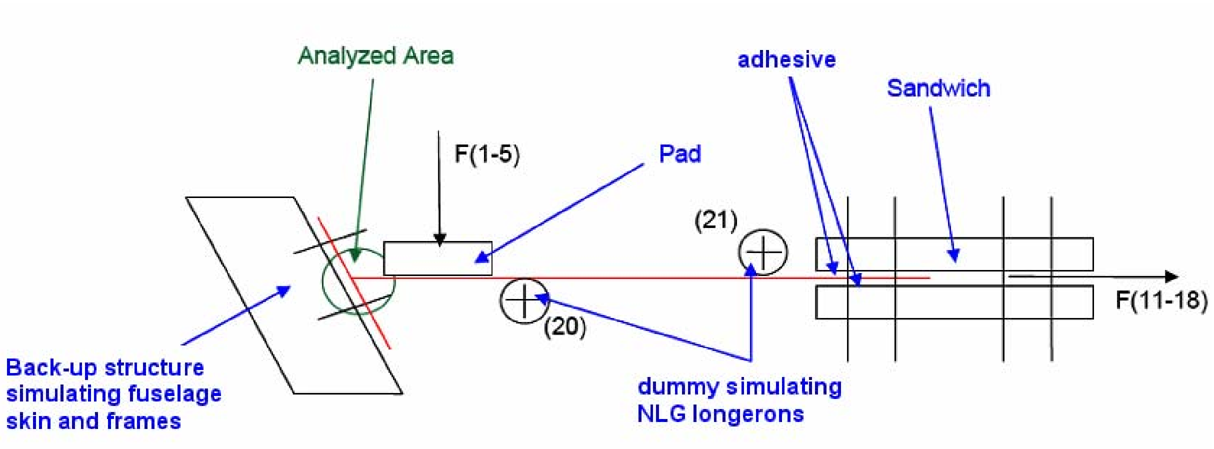


Fig. 33 - Sketch of the set-up for the M346 CFRP cabin floor to fuselage joint fatigue test.

Fig. 34 - Preparation of the M346 Full Scale Fatigue Test

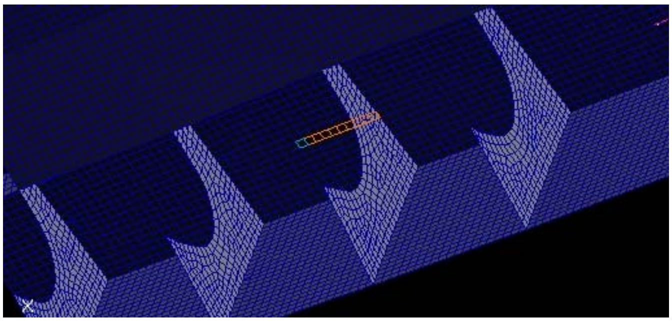
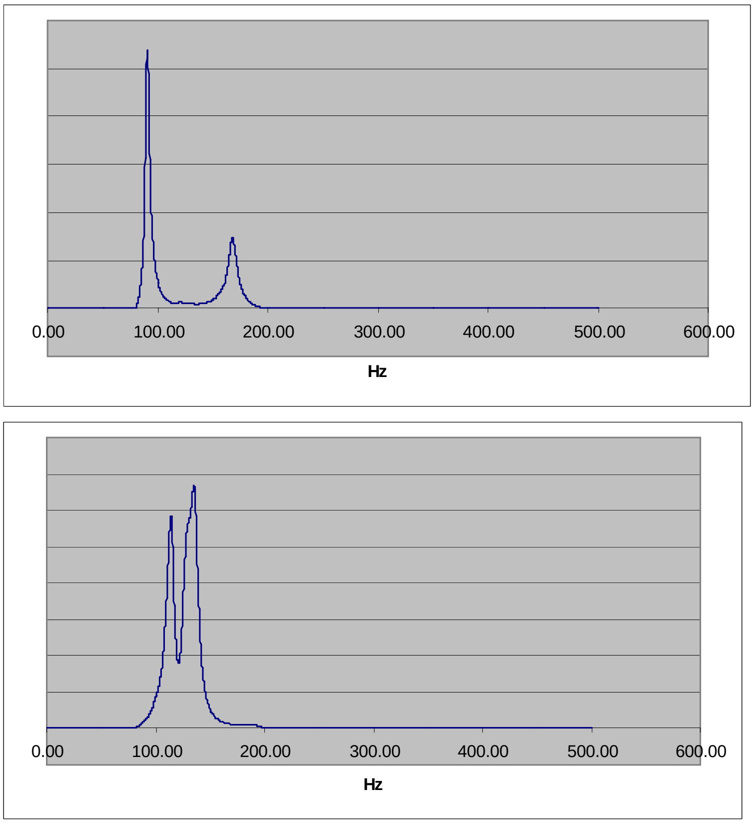
	
Dynamic FE model of LE structure	PSD of original (up) and modified (down) structure

Fig. 35 - C27J LE Prototype Wing development fatigue test – crack at spar root section

Fig. 36 - Bombardier CSeries aircraft: artist's impression.

Fig. 37 - NEURON, an international UAV program.