Review of Aeronautical Fatigue Investigations in Switzerland

June 2001 – April 2003

KOR-TTA2003-0061

M. Guillaume
RUAG Aerospace
CH- 6032 Emmen

SUMMARY

The Swiss review summarizes fatigue work in Switzerland. It includes main contributions from the RUAG Aerospace (RA) and Pilatus Aircraft Ltd. This document later forms a chapter of the ICAF conference minutes published by the conference host nation. The format of the review reflects ICAF requirements.

Prepared for the presentation at the 28th ICAF Conference
Lucerne, 05 and 06 May 2003

4.1 INTRODUCTION

The present review gives a brief summary of the work performed in Switzerland in the field of aeronautical fatigue, during the period from June 2001 till April 2003. The various contributions to this review come from the following sources:

- RUAG Aerospace; Fatigue Engineering, and Full Scale Fatigue Test Project Team
- Pilatus Aircrfat Ltd; Structural Engineering
- CFS Engineering
- Gottier Engineering Consultant
- Swiss Federal Laboratories for Materials Testing and Research,
- Sulzer Innotec

A part of the preparation of this review is sponsored by the Swiss Defence and Procurement Agency (DPA). This fond is gratefully acknowledged.

4.2 MIRAGE III RS (M. Guillaume*)

* RUAG Aerospace

The Mirage III RS (Swiss recoinassant version) will be retired at the end of 2003. In the year 2001 Dassault Aviation released a new service bulletin for inspection of rib No. 4 using eddy current. The engineering basis were established during the Swiss Mirage III Swiss Full Scale Fatigue Test which took place at Emmen from 1976 till 1986. All 16 Mirage III RS have a slightly modified rib 4 which should eliminate the occurrence of cracks in the lower flansch. This modification was introduced during wing spar refurbishment program, which took place between 1988 and 1992. The Mirage III RS have on average accumulated more than 3100 FH per airplane. The fleet leader has accumulated more than 3300 FH. Between 1600 FH and 2000 FH all the Mirage III RS were upgraded with a small Canard to improve the performance. After 1200 FH of canard usage some canards showed debonding of the honeycomb structure at the outboard leading edge. During the last two years no fatigue problems were reported during maintenance. During the year 2003 only 4 Mirage III RS will be in service. A special paint for the last missions will ensure the final attraction of this very nice plane, see figure 4.2_1.

Figure 4.2_1: Mirage III RS, current configuration of the Swiss Air Force

4.3 ACTIVITIES FOR THE SWISS HAWK Mk 66 FLEET (S. Büsser*)

* RUAG Aerospace

The Hawk Mk66 fleet consist of 19 aircraft, used by the Swiss Air Force as jet trainer. Several locations are known as fatigue critical such as tailplane, center and rear fuselage. The structure of the Hawk Mk66 fleet is inspected periodically, but in none of these locations any fatigue related damage could be found.

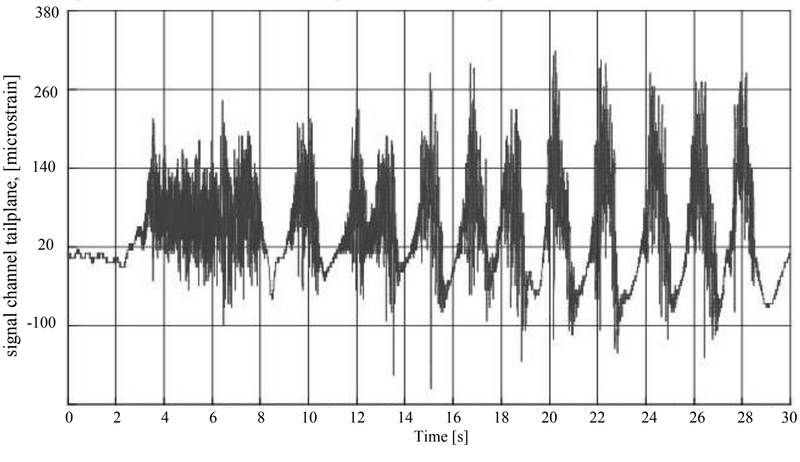
Revision of Fatigue Index Calculations (FIC)

At the introduction of the Hawk Mk66 in 1989 every aircraft was equipped with an Electronic Structural Data Acquisition system (ESDA) to monitor fleet usage. During each flight the normal acceleration, the fuel level and the strains at the critical locations of fin and tailplane are measured. With the collected flight data a FIC is performed for the components fuselage, wing, fin and tailplane using a Miner's damage summation coupled with BAe fitted S-N data.
During last year the FIC for the wing has been revised. The influence of the aircraft weight and the several wing stores has been verified. Also the process to determine the stress at the critical location of the wing out of the flight data was updated.

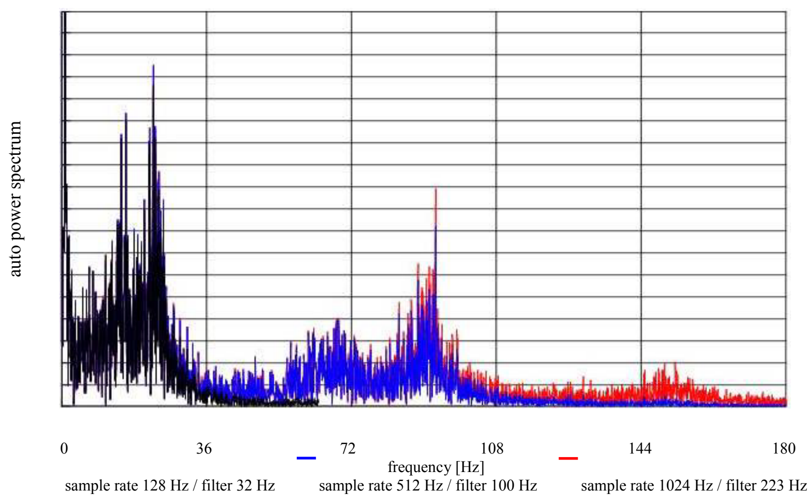
Empennage vibration studies

Vibration load due to buffeting do occur at the a the empennage of the Hawk Mk66. With the ESDA – System the strain at the critical location of the tailplane top skin is measured with a sample rate of 128 Hz and filtered to 32 Hz. It is obvious that this measurement setup is not sufficient to properly detect structural vibrations with frequencies up to 90 Hz (see plot 1 and 2). For this reason a special Operational Loads Monitoring (OLM) program was established, measuring the tailplane and the fin strain signals simultaneously with three different sample rates.

plot 1: manoeuvre turn stall, tailplane channel sampled with 1024 Hz



plot 2: manoeuvre turn stall, comparison of the frequency spectra for tailplane channel



Several flight manoeuvres which produce heavy empennage vibrations have been studied in detail and the influence of increasing the sample rate on the measured signal and on the calculated fatigue index for the tailplane was analysed. It was found that the increase of sample rate leads to an increase of usage spectrum severity particularly for the tailplane channel. The fin channel is not that much affected.

The main task of this signal analysis was to determine the remaining safe life of the Hawk Mk 66 empennage for Swiss usage and to develop a procedure to calculate a fatigue index out of the flight data sampled with 128 Hz which accounts for the "missed" flight data.

4.4 THE F-5E/F SWIS TIGER FLEET (A. Uebersax*, M. Geering*, A. Oswald*)

* RUAG Aerospace

Crack retardation testing for damage tolerance analysis update

19% of the Swiss Air Force F-5E fleet and 100% of the F-5F fleet are equipped with an electronic structural data acquisition system (ESDA). The collected data was combined with mission and configuration distributions to generate Swiss specific usage spectra for the F-5E and F-5F. For all locations assessed by a Swiss specific damage tolerance analysis (DTA) these spectra are used.
In the SWISCRAK program crack growth retardation effects may be accounted for by using the modified Willenborg retardation model. The Willenborg model utilizes an effective stress concept to reduce the applied stresses and hence the crack tip stress intensity factor. The Willenborg model has been modified by Gallagher and Stalnaker to incorporate an overload shut-off ratio S_{ol} and a threshold stress intensity K_{th}. In the process of updating the DTA with newer spectra, crack retardation testing had to be performed in order to determine the overload shut-off ratio S_{ol}.

Dorsal longeron: In addition to the analytically defined stress sequence based on the measured N_z sequence, a stress sequence was established directly based on strain gauge measures. As it was decided to use this strain gage sequence to define inspection intervals, the corresponding retardation parameter had to be determined.
The crack retardation tests were performed in combination with the tests to verify the benefit of the introduced life improvement (standoff relocation) on the dorsal longeron, see [1]. The corresponding overload shut-off ratio S_{ol} changed from 2.4 with the previous sequence to 2.9 with the new strain gauge sequence.

Upper longeron at splice F.S.284 aft end: At location F.S. 284 the longerons are spliced together with a tension bolt and a barrel nut contained in the aft longeron P/N 14-11304, s. Fig.1. Being a single load path structure, a DTA was required for this location. The fracture critical location in the longeron P/N 14-11304 is mainly loaded by fuselage bending moment, but is also influenced by the pre-load of the splice bolt. The geometry of the test specimen was chosen to represent the aft part of the splice. The peak stress location was determined in [2] to be at the mid point of the member in the barrel nut hole. The splice bolt was torqued to the value specified for the aircraft in order to allow representative pre-load stresses, see Fig. 2. Retardation parameters were found to be 2.2 for the F-5E sequence and 2.1 for the F-5F sequence.

Wing lower skin: The wing lower skin is machined from 7075-T7351 plate and represents the vital part of the wing as it runs trough from one wingtip to the other. The tests to determine the shut-off ratio S_{ol} for the wing lower skin are also carried out for both, the F-5E and F-5F sequence. The test specimens represent a non coldworked hole in the wing lower skin at 33% MAC / W.S.0.0. Final results are not yet available as testing is still in progress.

Investigation of in-service incidents

Forward attachment bolt failure of wing tip missile launcher: A new forward attachment bolt (EWB 26-10H12) failed only 45 flight hours after installation, s. Fig.4. The bolt is made of H-11 steel (consumable electrode vacuum melted) 5.0Cr – 1.3Mo-0.50V (0.38-0.43C), heat treated to 260 ksi and vacuum cadmium plated. This failure is the second of this type. In the first bolt, the crack initiated in the bolt thread, whereas in the present case, the crack initiated in the transition radius between the bolt head and the bolt shaft. Investigation results of the first failure were already presented in the last review [1].

The design of the connection between the wing tip rib lug and the launcher is such that flight and landing loads cause a bending moment in the bolt, s. Fig. 3. The bolt is mounted with a specified torque value. The resulting pre-load ensures contact between the lug and the launcher throughout the flight. The shaft and the contact surfaces of the bolt head were treated with dry lubricant, which is not called out in the specification. This resulted in a higher pre-load of the of the bolt, increasing the potential for stress corrosion cracking. Some additional effects were noted during the failure analysis: The cadmium plating of the bolt was mechanically damaged by a small scratch and chemically degraded by alcoholic solvents of the dry lubrication spray (molybdenum sulfides). Surface corrosion was induced by the presence of water and the contaminants sodium and chlorine, which were found on the crack surface. Hydrogen was formed by the corrosion process resulting from the lack of air and helped to induce stress corrosion cracking. Only about 2% of the fracture surface, s. Fig 4, was identified to be inter-crystalline corrosion. The rest of the fracture surface was caused by brittle failure due to an overload.

Actions taken include the replacement of the bolts by the following procedure: Bolts and contact surfaces must not be lubricated, bolts are cleaned from contaminants prior installation and the tolerance range of the prescribed torque value was reduced and fixed at the minimum value. Additionally, the bolts will be periodically replaced in the future.

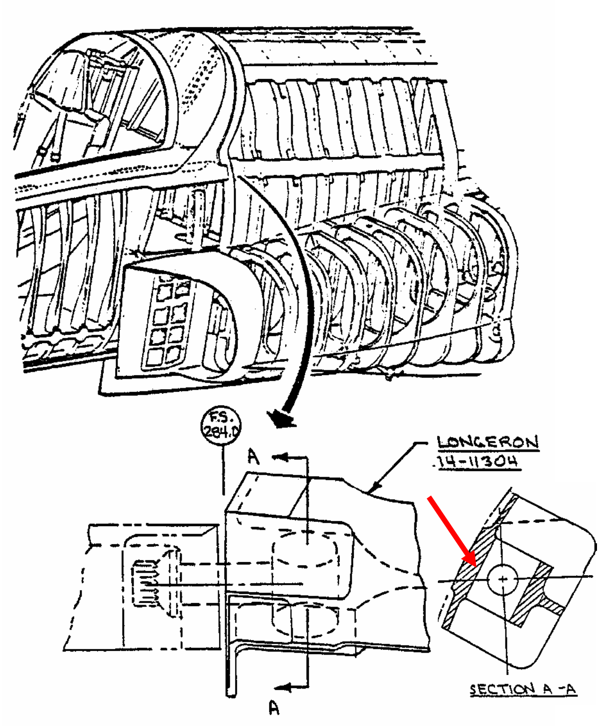


Figure 1 F.S.284 longeron splice of the F-5E. Arrow indicates the peak stress location.

Figure 2 Crack retardation test setup for the aft location of F.S.284 splice with the test specimen at the bottom

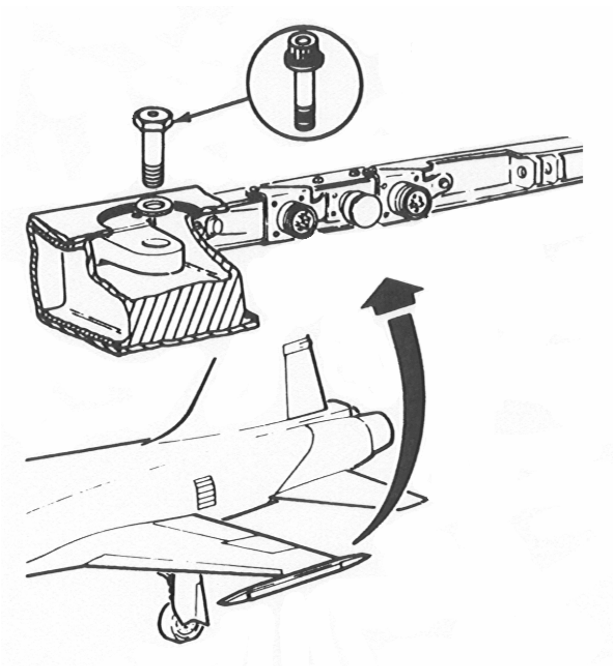


Figure 3 Location of the forward attachment bolt of the wing tip missile launcher.

Figure 4 Failed forward attachment bolt (2nd case). Arrow indicates the corroded crack origin.

4.5 DEVELOPMENT OF A PROCEDURE TO CALCULATE AIRCRAFT BALANCED LOAD DISTRIBUTIONS FOR F/A-18 SERVICE FLIGHTS (S. Oesch*, J-P. Weiss+)

* RUAG Aerospace, + Consultant Engineer

Background of spectrum generation

During the past years Switzerland made in-depth structural integrity analysis with the Swiss specific F/A-18 design spectrum. Detailed studies were performed before the production of the aircraft, leading to several structural changes. The design spectrum is applied as well to the ongoing Full Scale Fatigue Test.
Today the flying fleet shows a usage severity of stabilized characteristics. The amount of accumulated service flights is representative enough, to be in a position to define a Swiss usage spectra. The need came up to develop a tool to generate spectra at any location in the structure of the aircraft. The maneuver severity of real aircraft usage has to be known in detail, in order to calculate life estimates for locations of interest.
The development of a usage spectrum for service flights was tasked to RUAG. An engineering effort lead to a process called 'Symmetric Loads Prediction' (SLP). Today SLP allows to generate the sequences of fuselage symmetric balanced load distributions for F/A-18 service flights.

Description of the development

The development of SLP included three major steps:

1. It was necessary to establish a procedure to identify flight maneuvers. For every flight the F/A-18 fatigue tracking system (Boeing SAFE software used for data extracting from flying airplanes)and software provide several usage data files of different format. The data consists of flight parameters and strain gage readings in a real time sequence. Those files have to be combined to one file. The data must be smoothed to avoid unrealistic spikes, and to remove higher frequency buffeting effects. Further data pre-processing is done for the parameters available. For every flight a smoothed file is created, that is used as input for the loads prediction.

2. To the identified maneuvers, loads must be assigned. The loads prediction is the core part of SLP. It will be described in the next subchapter. It is important to note that SLP additionally assures the overall balancing of the aircraft.

3. The third step is the easiest: It's the generation of the load sequence for the aircraft at specific stations. This consists of putting the loads generated under step 2 in the order according to step 1.

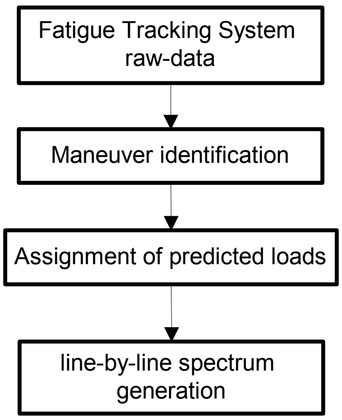


Figure 4.5_1: Overview of SLP procedure

A simplified overview of the SLP procedure is given in Figure 4.5_1. The process is widely automated for application today.

Derivation of Symmetric Loads Prediction

The main engineering task was to develop the loads prediction for SLP. For every load to be predicted a formula had to be defined, that is based on the input parameters. The quality of the prediction has to be reasonably good, which required a substantial engineering effort.

The base of the derivation are the loads data sources. All loads data available with at least a certain credibility were used in the best possible way. Sources are:

- Loads data from RUAG CFD – program
- various sources of Boeing load trends
- extract of a flight test database

Loads values were listed against all available flight parameters. A clear assignment of load versus parameter is necessary. This allowed to start to search for the best fit formula to determine a load as a function of the parameters. It was an optimization task to find the maximum correlation using different methods and different sets of parameters as input. Methods used for the loads regression was the Lagrange isoparametric formulation, and least square fits.

The following example shows the solution for the Vertical Tail Bending Moment. The most influencing parameters were found to be the parameters Cn and M.

Cvtbm = f(Cn, M) ; where
M = Mach number
Cn = Normal force coefficient
Cvtbm = Vertical Tail Bending Moment coefficient

Using the Lagrange method the derived loads formula can be expressed as a function of the following:

Cvtbm = f (Cn, M, Cn*M, Cn^2, M^2, Cn^2*M^2, Cn^2*M, Cn*M^2, Cn^3, M^3, Cn^4*M, Cn^3*M, Cn*M^3, Cn^3*M^2, Cn^2*M^3, Cn^4)

A matrix calculation allowed to determine the necessary coefficients. The final formula can be illustrated graphically. The equation for the Vertical Tail Bending Moment is shown in Figure 4.5_2.

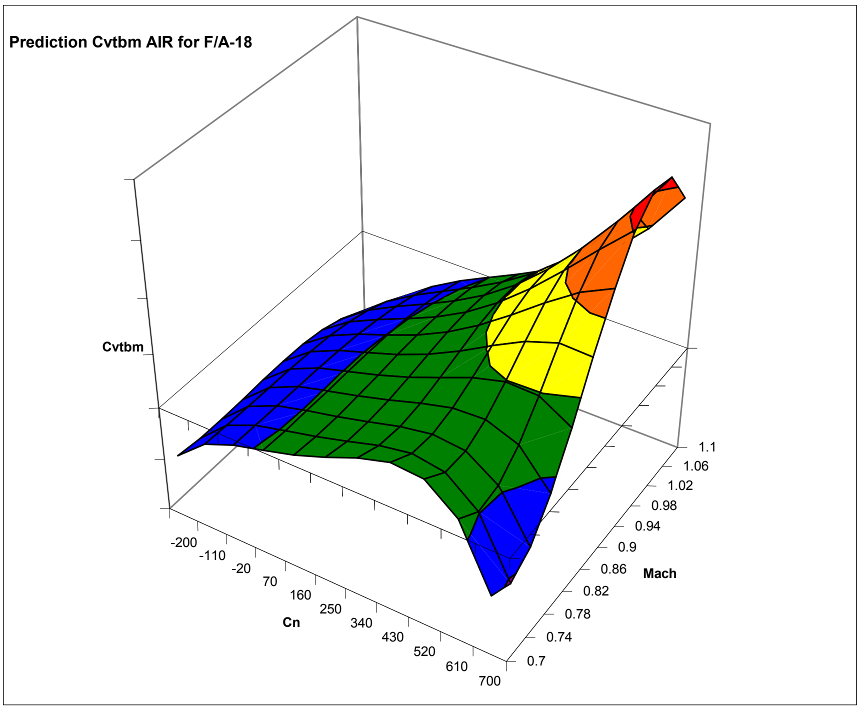


Figure 4.5_2: Loads prediction as a function of two parameters

All the loads formula can be derived as described above. Whenever more than one solution was found, arguments of quality, higher stability or more reliable data source were used to pick the best equation.

After the definition of all needed SLP – loads formula, they were hardcoded into the SLP-Software. SLP is taking care of transforming the input file of flight parameters and strain gage readings into component loads. The balancing of the aircraft is fully automated in SLP.

Input data to SLP are strain gage readings and flight parameters such as angle of attack, Mach number or control surface deflections.
The program delivers component load-distributions along the fuselage. An example is given in figure 3 below.

Results

The SLP Software is written in MS Excel. The results are given as tabular outputs and a choice of graphical illustrations is automatically generated. An example for a single maneuver is shown in figure 4.5_3.

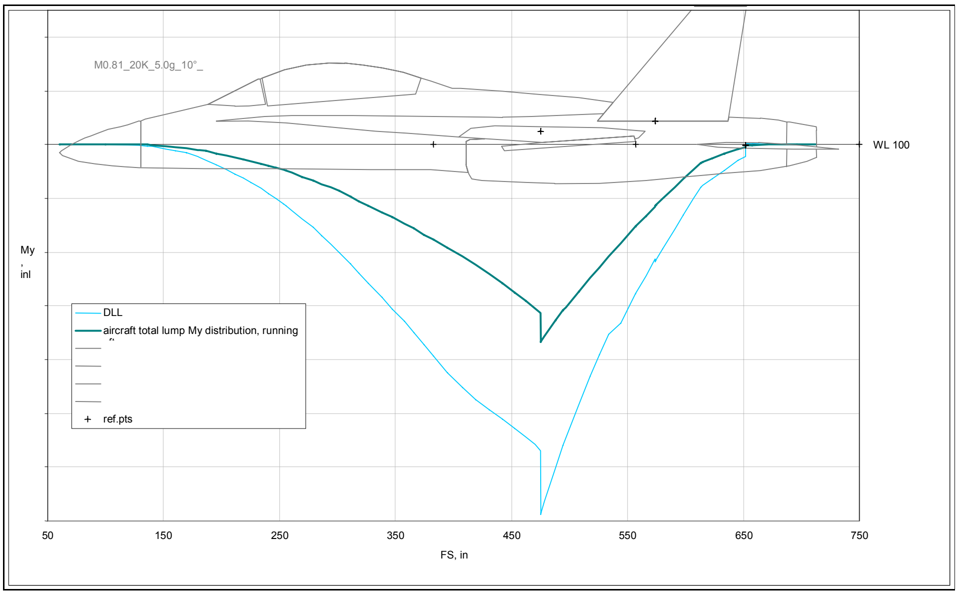


Figure 4.5_3: Net Bending Moment distribution along the fuselage for a 5g Load Case

Current restrictions

Derivation of the fuselage symmetric vertical Force (Fz) and Pitch Moment (My) loading sequence was addressed only. This loading was considered in the first instance since it can be considered to represent the ‘core’ flight loading from which asymmetric or dynamic departures are made. Also, it alone represents the loading that imparts proportionally the greatest structural fatigue damage. For each the Vertical Tail, Horizontal Tail and the Wing Root three attachment loads are given: Shear, Bending Moment and Torque. Asymmetric maneuvers are identified, but substituted with a symmetric load.
The process currently available is valid for the fuselage and the wing attachment loads.

Go ahead

The tool currently is in a validation and testing phase. First flights are analyzed. A Swiss usage spectrum of typical severity is planned to be generated.
Methods are evaluated to transform loads into stress. This is required for those locations necessitating stress sequences for the life calculation.
In future, when considered necessary, the possibility exists to add capabilities to the tool. Additional capabilities could be to address drag effects, to add wing components, to handle additional aircraft configurations or to address asymmetric effects.

4.6 SWISS F/A-18 STRESS SEQUENCE GENERATION USING SLP DATA (B. Bucher*)

* RUAG Aerospace

Introduction

Using the SLP procedure, section loads data sequences (Shear and Bending Moment) along the fuselage are now available at every fuselage frame station. In addition, all the component interface loads are known from the wings, vertical tails and horizontal tails. All the loads belong each to a line from the SAFE sequence of flight parameters.
In order to determine local stresses sequences analytically, the loads have to be distributed to a structural analysis model, such as the FE3 Fe model, representing the flying Swiss F/A-18.

Known Methods for Stress Generations

Currently we know a couple of different methods to derive local element stress based on externally applied loads, panel loads or applied loads at reference locations:

- Boeing LOADS 98 procedure generating and using influence coefficients based on Boeing component FE models loaded by design loads
- NGC procedure using SPECGEN based on CTR/AFT fuselage ½ FE model, given MES and unit stress output from a set of panel and distributed loads
- In house developed tool GENERATE based on Full F/A-18 FE2 FE model, FSFT configuration using unit stresses from 74 unit jack loads applied and 2011 unique jack loads cases.

Without going too much into details about the methods mentioned above we will address now some possible approaches to come up with elements stress sequences using SLP.

Outline of a Method for Stress Sequence Generation Using SLP

In common of every method is the use of internal loads FE models. For the SLP procedure we will use a complete FE model of the Swiss F/A-18 D structure. Since loads are defined at loads reference locations those loads have to be spread again into the structure. As a first approach we will use Nastran RBE3 Elements, which don't add any additional stiffness to the structure but distribute a local load or moment onto a set of grid points as an equivalent load set. This way – at the loads reference locations - a 100% loads match is guaranteed. At other locations the level of load match depends on a sensible definition of the RBE3 elements. The highest accuracy can be expected - fortunately - in the CTR fuselage, the inner wing and wing root area. This is due to the fact, that those portions of the structure are predominantly affected by the through loads and much less by the locally applied loads.

In addition a more closely look has to be taken at the internal tank (due to tank overpressure) and duct pressure loads. Those loads may depend on the specific type of flight maneuver. An investigation has to be performed to know how those local internal loads relate to the flight parameters known from SAFE. During previous FEM activities for the generation of FE3 for the Swiss F/A-18 FSFT those internal pressure loads have actually been developed for a set of 30 balanced fatigue load cases. The nz levels for those load cases range from –2.25 and +4.5 to +8.25. This data will provide valuable information to improve the internal loads match and therefore the local stress response.

4.7 THE SWISS FULL SCALE FATIGUE TEST PROGRAM (M. Guillaume*, I. Pfiffner*)

* RUAG Aerospace

Introduction

During summer of 1998, the Swiss Defense Procurement Agency (DPA) commissioned the Swiss Aircraft and Systems Enterprise Corporation (SF) today called RUAG Aerospace to perform a full scale fatigue test (FSFT) on the Swiss F/A-18 a/c. In order to perform this test in the most economical way with state of the art test set up and equipment, SF decided to team up with and use the knowledge of IABG. The IABG Company, a test and analysis center in Germany works as subcontractor on the test concept. The details of the development of the test concept are documented in reference [3].

The goals to be achieved by the Swiss F/A-18 FSFT are as follows:

- A basis of a safe operation of the Swiss F/A-18 fleet combined with a most economical maintenance program shall be established: the tota1 fatigue life of critical components will be determined.
- In addition, inspection intervals established by analysis and inspection methods will be validated.
- Furthermore repair designs which occurred during the FSFT will be verified.

Test Set-Up and Equipment

During July 2001 the wings including the control surfaces and the Leading Edge Extension (LEX) were paded to apply the loads through whiffle trees using adhesive bonding technology. Each wing (figure 4.7_1) has 219 pads of the size of 10x10 cm^2. The distribution of pads is optimized to simulate the correct loading at the wing root, and the wing fold. Also the hinge moments of each control surfaces were matched by choosing a representative center of pressure. Each LEX is paded with 42 pads to mach the shear and pitching moment of the LEX component loading.

Figure 4.7_1: Pad application on the inner wing torsion box bonded by IABG

To introduce the high loads into the center fuselage which affects critical locations on the dorsal deck and the bulkheads tension/compression shear pads were used. The procedure for the pads bonding on the fuselage was quite different then for the wing. A special constrained lever was adjusted to connect the pads. A strap plate connect the pads to the actuator. The pad layout was designed to have dominant shear loading along the bondline between pad and fuselage surface. The most critical pad configuration was qualified by a real test spectrum running a component test on a servo controlled MTS test machine.

In August 2001 a modular strong steel floor was installed in the test hall in Emmen. This strong floor has to sustain the whole test-set-up with the high loads during fatigue cycling.
An important step was the review of the whole test-set-up with all the details. The test-set-up is designed on CAD version CATIA 5 (figure 4.7_2). To simplify the detail design and reduce the risk of assembly problems of the rig and the whiffle trees.

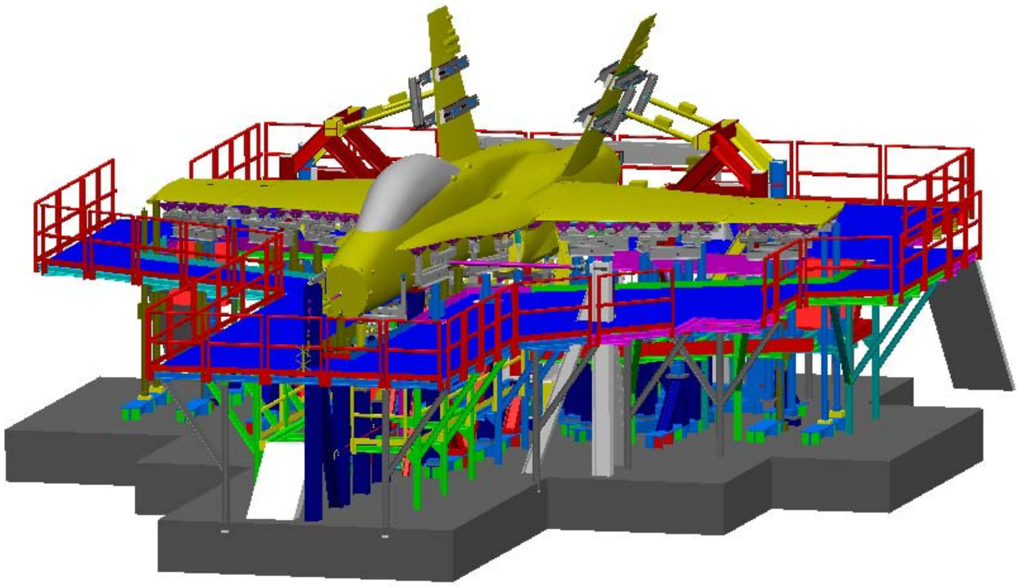


Figure 4.7_2: Final test set-up designed on CATIA version 5.

During the period December 2001 to February 2002 the loading rig including three platforms were installed on the structural floor.
During the same time 68 actuators (push/pull type) were installed and the hydraulics tubes through the whole rig to each actuator were fixed.
The hydraulic system was ready approximately end of April 2002. In Mai 2002 the test article with the inner wings, the dummies, and the fittings was installed to the loading rig. It was fixed on the 6 restrained struts, see figure 4.7_3. Additional components had to be assembled for example outer wings with flaps, inner wings with flaps, horizontal dummies and the loading components for the vertical tails.

Table with overview of the fixed reactions:

Fixed Reaction	**Location**	**Restraint Mode**
RX1	Engines (x = flight direction)	F_x
RY1	Forward Fuselage (y = left hand side direction)	F_y, M_z
RY2	Rear Fuselage (y = left hand side direction)	F_y, M_z
RZ1	Forward Fuselage NLG (z = vertical direction)	F_z, M_x, M_y
RZ2	Rear Fuselage MLG (z = vertical direction)	F_z, M_x, M_y
RZ3	Rear Fuselage MLG (z = vertical direction)	F_z, M_x, M_y

The installation of the test article in the loading rig was a major milestone during the project, see figure 4.7_4. The predicted weight based on component measurements was within 0.5% compared with the load cell measurements on the three vertical struts.

Figure 4.7_4: Lifting the test article into the rig.

One of the major differences compared with other tests are the tension compression whiffle trees. There dimensions are big and also there weight. So special consideration is required to install these components.
The heavy whiffle trees for tension and compression loading were installed mainly on the wing areas and the fuselage zones, see figure 4.7_3. All the whiffle trees on the wing consist of two levels and have proper degree of freedom for the functionality.

Figure 4.7_3: Wing whiffle trees to simulate distributed loads and simple vertical tail load introduction system.

The control and monitoring system (figure 4.7_4) was delivered from FCS Test Systems in The Netherlands.
The systems consist of one host computer and two monitoring computers.

One real time front end (RTFE) with processor boards and I/O cards and SmartTest controller units are placed in the main cabinet, close to the test rig. The SmartTest Control Unit (SCU) is a digital controller unit.

Using SmartTest Manager, this computer generates the command profiles for each SmartTest Control Units and send these profiles via the ethernet systems to the real time front end.
The monitoring computer uses the SmartTest Explorer to display the commands and feedback signals of all channels as well as the data acquisition boards within the control systems.
The connection between the SCU's and RTFE is made using a real-time fiber-optic communication link.
The interface between the control & monitoring (C&M) system and the data acquisition (DAQ) system are ethernet connections and CAN bus interface.
At the moment there are two operating modes installed: Static Test Run for strain survey and Dynamic Test Run for the fatigue cycling, respectively.

The Static Test Manager (function of control system for strain survey) used for strain survey mode allow us any existing load case of the fatigue test program to be run in predefined steps or in increment mode. The Dynamic Test Manager (function of control system) used for fatigue cycling supports the following actions: Set test position, start, adjusting test speed, and pause and stop the test run.

Figure 4.7_4: Racks for the control and monitoring system (middle three) and the data aquisition system (two left an rigt)

To get some experience with the control and monitoring system an advance test were build up. Therefore two small actuators were installed to a small rig, see figure 4.7_5 were used. The interface between the hydraulic manifold of each actuator and the SmartTest controllers were tested in detail.

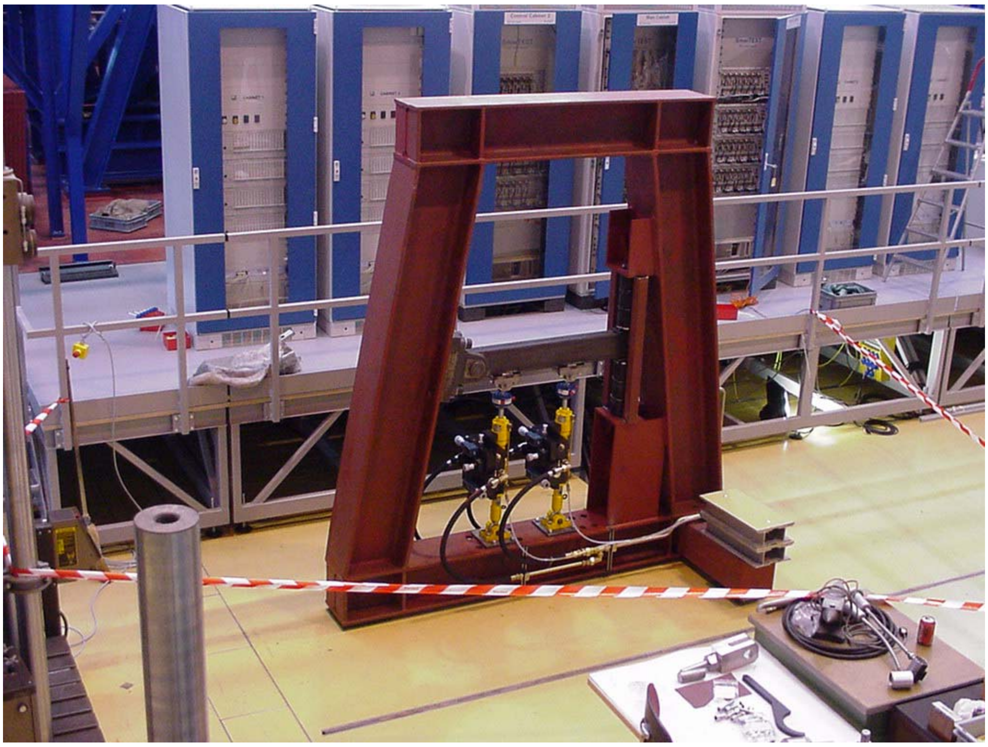


Figure 4.7_5: Small rig for tests with two small actuators connected to the control system.

In June and July 2002 the hydraulic tube system was tested with full pressure. Leakage checks were done and all connections were checked through out the hydraulic system.
In August 2002 all 68 actuators were connected to the whiffle trees, dummies and fittings.

The hydraulic system has special requirements to ensure a good handling and a safe operation:

- Each actuator is equipped with a manifold block, which supports the servo valve, the shut-off valve, the throttle valve and a load limit device. Each actuator is switched on or off individually via the host PC of the control system. To load the FSFT article, all actuators and a main hydraulic solenoid are automatically switched on by a start-up procedure.
- The servo valve is the final control element in the closed loop of the hydraulic actuator. The load cell, which is attached to the end of the piston rod, generates two independent signals (A and B). One is used as feedback in the control loop and is compared to the command value in the control unit, the other one is used for redundancy and measurement purposes.
- To avoid the exceedance of maximum loads, all individual actuators are equipped with a load limiter, which limits the oil pressure in the two actuator chambers to the maximum tension and compression load of the actuator to be applied during testing.
- In case of an emergency shutdown (hard shutdown) the specimen could be damaged by a sudden unbalanced load case. This is prevented by means of a throttle valve available at each actuator. Simultaneously by switching off the test, the shut off valve is closed and the short circuit valve will be opened causing an immediate unloading of the actuator. The time for unloading can be controlled by adjusting the throttle valves. To guarantee a balanced and safe unloading of the specimen the throttles of all actuators are adjusted within a number of pre-tests to be performed before actual cycling loading starts.

Static and dynamic Commissioning
The static commissioning of the whole system was done in August till October 2002. To adjust the trottle valves several test had to be done to ensure the safe unloading of the test article. This is an empirical testing which took some time. During this tests the actuators were only loaded up to 30% of the maximum spectrum loading.

The static phase was completed with the strain survey measurements.
For the strain surveys which will be performed every 1000 flight hours seven load cases were selected. The following four load cases are from the test spectrum:
- 8.25g symmetric steady state pull up maneuver with clean configuration
- -2.00g symmetric steady state push down maneuver with clean configuration
- 6.25g asymmetric right roll maneuver with 5.5 rad/sec2 with clean configuration
- 6.25g asymmetric right roll maneuver with 5.5 rad/sec2 with light bombs under the wings

The remaining three load cases represent Swiss symmetric 9g design maneuvers:

- 7.96g at sea level with max wing root bending moment
- 9.0g at sea level with max forward fuselage bending moment
- 9.0g at 10'000 feet with max aft fuselage bending moment

During a strain survey all strain gauges (total of 1095) are recorded using a state of the art HbM data acquisition system. The loads are applied to a load level of 70% in steps of 10% to avoid high loading of the test article which may introduce residual stresses.
During the dynamic test phase a problem with the fail safe function of the control system occurred. The hard shut down may cause an overloading of the test structure. The control & monitoring systems had to be modified to fulfill the specified safety requirements. Further problems with the stability of the SmartTest control system delayed the commissioning.
During fatigue cycling the wing is deflected up to 51 cm at the aft spar tip, see figure 4.7_9. In fatigue cycling mode the data acquisition system can measure up to 300 channels continuously. The 6 fatigue tracking sensors common to the Swiss fleet and 16 strain gauges at critical locations analyzed in the Swiss ASIP study are recorded and reviewed continuously. First results confirm the accurate simulation of the Swiss design spectrum at the wing root, see figure 4.7_10

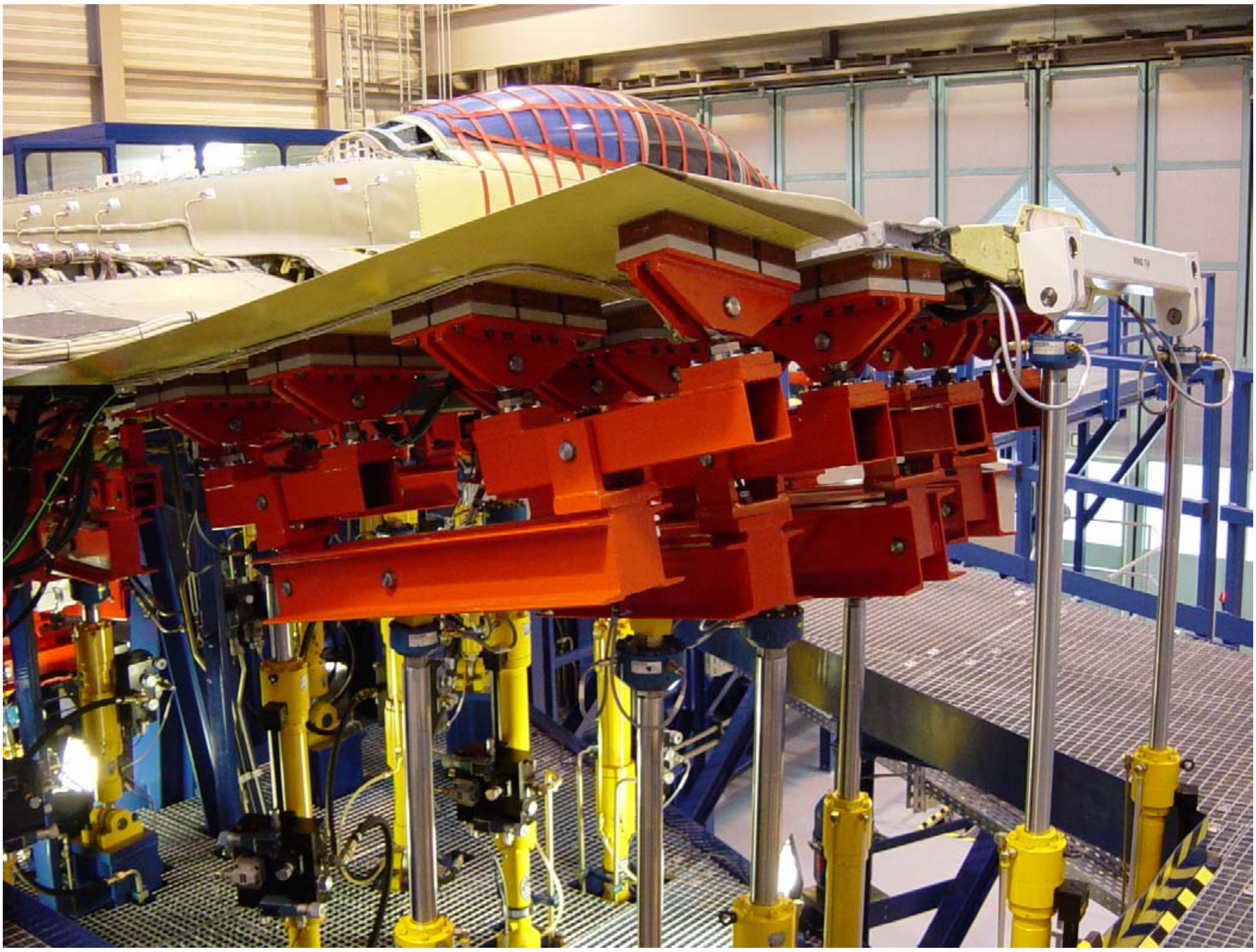


Figure 4.7_9: Maximum wing tip deflection during fatigue cycling.

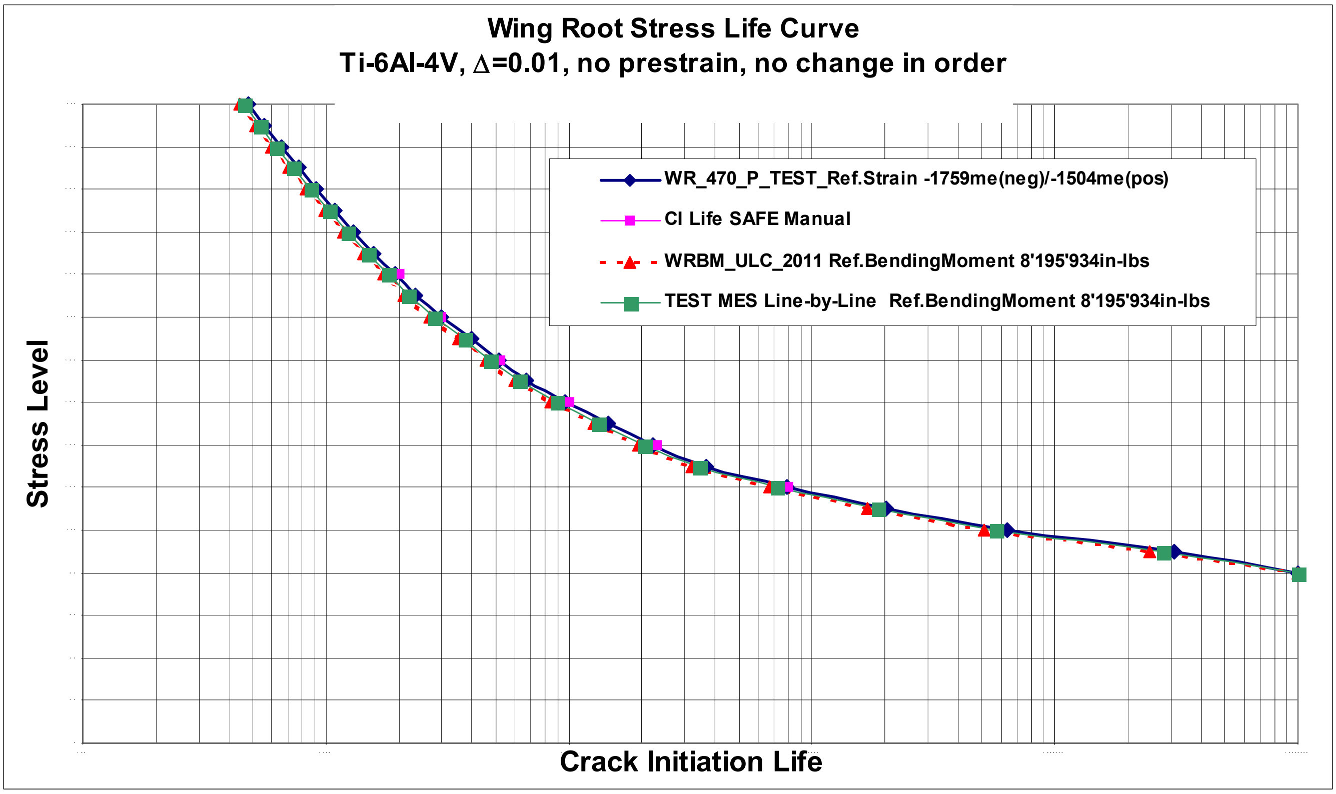


Figure 4.7_10: Crack initiation life curve for the Swiss design, for the test, and actual measured data from fatigue sensor.

In early December 2002 we were able to run the spectrum for the first time. After some software updates of the control system we were able to cycle up to 2000 SFH until early February 2003. The test speed is optimized to run at 0.23 Hz/cycle. It has to be mention that our spectrum contains only randomized maneuver loads and no taxi loads.

Within the last two years an inspection program was developed based on the results of the ASIP study. During the ASIP study for all fracture critical parts and for most maintenance critical parts a crack growth analysis was done. This information was very helpful for the inspection program.
During fatigue cycling the test article will be checked using visual inspection. Every 1000 SFH additional inspections will be done using close visual, eddy current, ultra sonic, as well as video scope inspections. To get access to critical areas in the inner wing torsion box some fasteners of the upper level were removed to do inspection using video scope. At the fracture critical locations no anomaly was observed so far. But in the inboard leading edge flap at the lug area a first crack was discovered during routine inspection at 3000 SFH. The crack was unexpected at the outboard lower lug. The Swiss have a redesign of the first inboard lug set, beef up of 50%, and all the lugs have parabolic fillets instead of circular. It was decided to continue testing because the cracks are about 0.4 inch long along the lug bore direction. The crack seems not to grow along the bore but it is growing in the lower flange of the lug along the chord direction. Further strain gauge instrumentation will be done to monitor the strains and to understand the loading actions at the lug.
The figure 4.7_11 shows the test during cycling with inspectors doing visual inspections.

Our customer, the Defence and Procurement Agency (DPA) requested to start the test no later than end of January 2003. We were able to match the customer goals within four years starting from scratch.

Figure 4.7_11: Test-set up during fatigue cycling at RUAG Aerospcae in Emmen.

4.8 OVER-/ UNDERTESTING CHECK AT CRITICAL LOCATIONS OF THE SWISS F/A-18 FSFT ARTICLE (M. Gottier+, M. Guillaume*, L. Schmid*)

* RUAG Aerospace, +Gottier Engineering Consultant

The Swiss F/A-18 Full Scale Fatigue Test (FSFT) article is loaded by a test spectrum, which has been derived from the Swiss Master Event Spectrum (MES). In addition to the test spectrum, which is a slightly simplified spectrum from the MES, which results in very similar fatigue damage at the most critical locations. However, the loads are not introduced into the test article using distributed (flight) loads, but are applied using hydraulic actuator (test) loads. Due to discrete load introduction the target load distribution cannot be simulated correctly over the entire aircraft. It is assumed that, locally, the aircraft structure can be over- or untertested. Hence, the Over-/Undertesting Study was initiated addressing the following objectives:

- Identification of overtested locations and, potentially, planning of additional inspections during test performance.
- Preparation and validation of test data processing and interpretation.

The 34 locations chosen for this study were selected based on different criteria:

- Location previously used to verify the Test MES
- Critical location in the Swiss ASIP study, i.e. low crack initiation life and/or crack growth life
- Critical location identified in the USN F/A-18 full scale fatigue test
- FEM stress comparison between the flight and the test configuration of the F/A-18 FE model for four enveloppe load cases:
  - on the FEM with the flight load distribution (FEM flight configuration) and
  on the FEM with test load distribution (FEM test configuration, where actuator loads are applied on FEM of
  - the test article).

Once these critical locations were determined two specific tools were developed to check the stresses and to determine the crack initiation life:

- A comparison tool that compares the FEM stresses for a selected (common) element of the flight and the test configuration of the F/A-18 FE model for 30 fatigue relevant load cases. The results are presented by means of a cross-plot.
- A spectrum generation tool that computes stress spectra for a selected element using the FEM output of a unit loads run (74 unit load cases). The spectra are fed into crack initiation software and the results are plotted and analysed using S-N curves.

A stress comparison is made by means of a cross plot, shown in figure 4.8_1. It reveals for this location that there is almost no difference between the stresses from the FEM test configuration and those from the FEM flight configuration.
Due to the fact, that this location is loaded by secondary bending which is not considered in the cross plot, it has been decided to check in a second step the impact of the test loads on the CI life. Figure 4.8_2 shows the KTDLS curves for the test and the flight configuration respectively. It results, that the test life is only 8000 flight hours, i.e. this location is overtested. The actual CI-life is however 8500 flight hours, when the test life is adjusted to the CI- life of 10640 flight hours from the design phase.
It can be concluded that this method to check whether a location will be over- or undertested is a very helpful to find potential crack in early stage of the test and to prevent big damages in the test article, which would lead in long test stops. However, the analyst has to be aware of the FEM's, which are used for this kind of check. Different FEM for the two load configurations may lead to big differences in stresses at the same location and as result of that in a wrong interpretation of the fatigue lives.

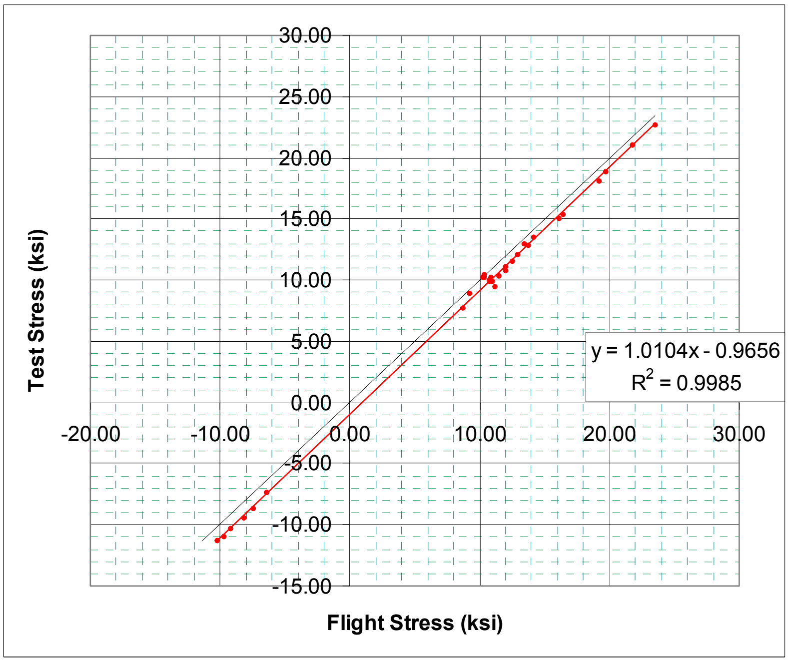


Figure 4.8_1: Crossplot for compare of flight stress vs test stress using the FE3 and the FE2 model for the 30 balanced fatigue loads which were also used for the development of the test concept.

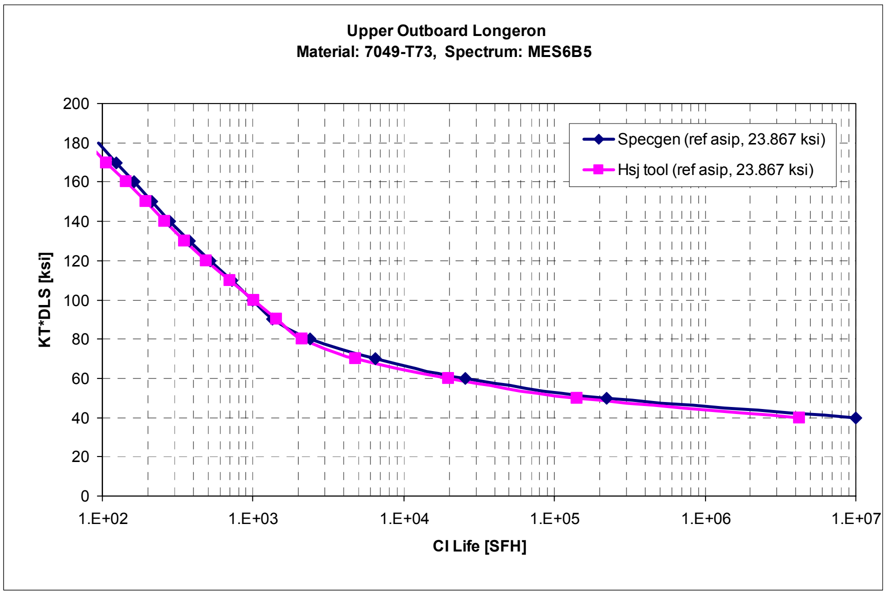


Figure 4.8_2: Comparison of crack initiation curves for ASIP data used for redesign and RUAG sepctrum development based on the FE2 model.

4.9 AERODYNAMIC CFD LOADS STUDY FOR THE SWISS F/A-18 (M. Guillaume*, A. Gehri*, M. Schleicher*, J. B. Vos+)

* RUAG Aerospace, +CFS Engineering

During the Swiss Full Scale Fatigue Test (FSFT) program actuator loads where developed based on the Boeing component loads approach derived during the ASIP study.

Boeing used for the aero loads on the fuselage only wind tunnel data from early YF-17 measurements. Data were only used for the forward fuselage. For the center and aft fuselage aero loads were based on a simple engineering judgement. In a first feasibility study RUAG Aerospace developed aero component loads used Computational Fluid Dynamics (CFD) and compared it with the Boeing data used in the ASIP study and applied in the FSFT test.
In a first step a CAD cleaning was done for the loft surface. An important and time consuming step was the generation of a topology with the associated mesh to apply a Navier Stokes algorithm with special turbulence modeling, see figure 4.7_1. For the component loads calculation a post processing procedure was created where the corresponding loft surface for the loads summation was defined for each component. Total 24 reference points were defined for aircraft components for moments and shear loads.

In the forward fuselage higher shear loads compared to the Boeing loads were found. Quit different and for us not surprising just the opposite happened in the center fuselage. The Boeing loads there are much higher. In the aft fuselage the CFD study showed at the horizontal tail area a down load whereas Boeing loads are positive.
It has to be mentioned that at the end of the fuselage some convergence problems arised which where more difficult to handle for high AOA angles.

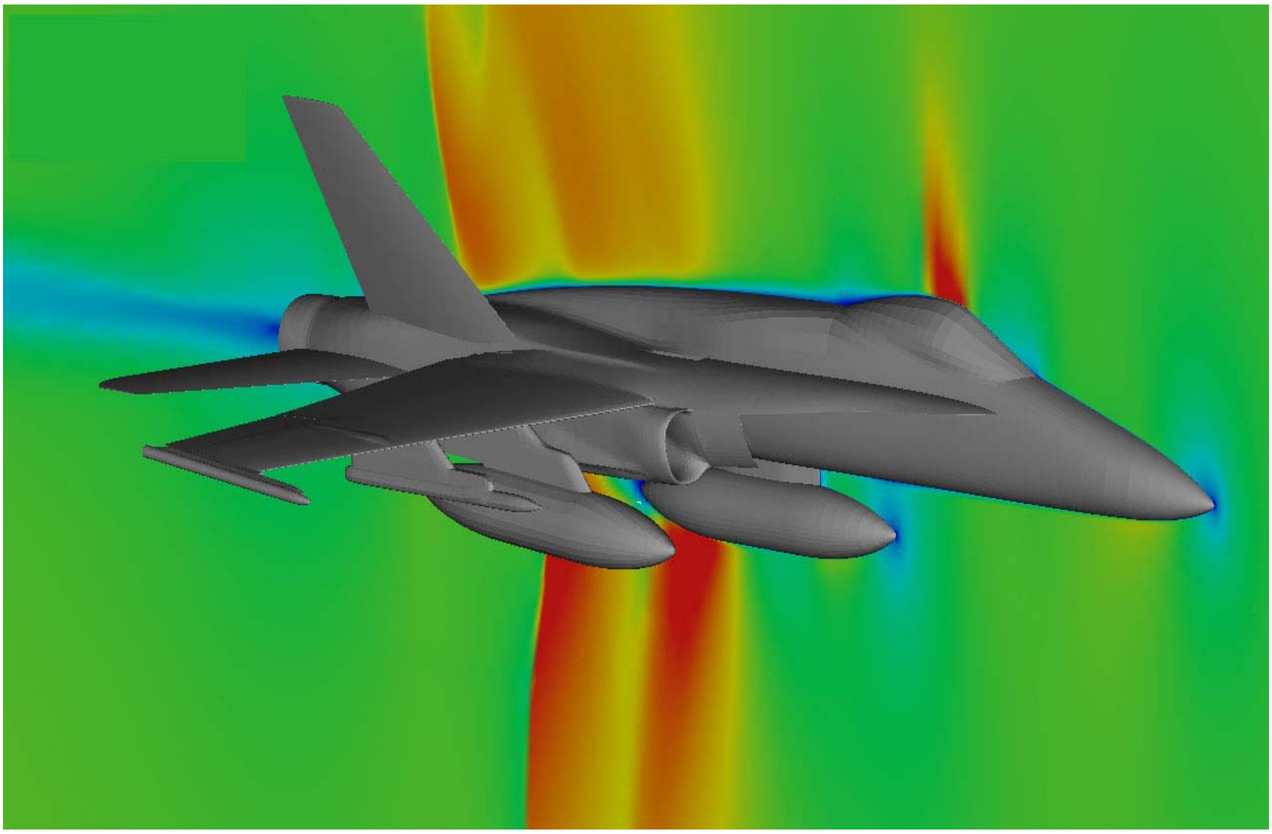


Figure 4.7_1: CFD calculation for M = 0.95 and AOA of 1° with centerline tank and two wing tanks and AMRAAM missile

5.0 DAMAGE TOLERANCE ANALYSIS OF PILATUS TRAINER AIRFRAME (C. Spedding*)

*Pilatus Aircraft Ltd

Introduction:

The Pilatus family of trainer aircraft are tandem two-seat, turboprop powered, primary military training aircraft which are certified to FAR 23. Originally, safe fatigue life of over 10,000 hours was demonstrated by analysis. Following the full-scale fatigue test (FSFT) carried out by the RAAF on a PC-9 airframe (reported at earlier ICAF conferences), during which some of the components cracked before the anticipated safe life was reached, it was decided to review the safe-life philosophy for the aircraft and justify the use of damage tolerance analysis in order to demonstrate required aircraft life.

Discussion:

The present project is to demonstrate that the cracked structure is able to withstand the critical static flight loads when cracked to at least the minimum detectable limits in all areas considered damage tolerant, and that after embodiment of Pilatus-designed modifications it will achieve full operational life.
During the FSFT, cracks were allowed to propagate to a known length before being repaired. The crack length measurements made on the FSFT are used to substantiate the damage tolerance analysis.
The load spectrum applied on the FSFT was based on known Royal Australian Air Force (RAAF) usage, which is less severe than the Pilatus design requirement, so the difference in damage levels is to be addressed. There have been subsequent increases in MTOW and MZFW whose effect is included in the project.
Subsequent to the FSFT, corrosion has been found in a number of locations on the aircraft. The efects of corrosion in combination with fatigue, and any adverse effect of modifications to the aircraft which makes them unrepresentative of the FSFT specimen, will be examined in the final phase of the project.
The intention is to demonstrate for the trainer aircraft that continued airworthiness is maintained, whether by safe life or by damage tolerance philosophy, so the required operational fatigue life is achieved while complying with FAR 23 regulations.

Pilatus is performing the project in three phases.

Phase 1 was an interim study in the second half of 2001 to allow aircraft that were approaching the certified safe life of the airframe, to continue flying.

Equivalent stress cycles (one-per-flight) were determined which represented the fatigue damage due to the stress spectra at fatigue-critical locations on the FSFT airframe. These stress cycles and other appropriate parameters were used to carry out damage tolerance analysis. The output was a preliminary series of inspection intervals which allowed sub-critical cracks to be managed safely based on FSFT spectrum usage. This enabled the continued operation of aircraft approaching their safe life until the programme in Phase 2 is complete.

Phase 2, adjusted for the difference between the FSFT spectrum and FALSTAFF, and the difference in performed during 2002/2003, repeats the analysis in more detail using the FSFT measured load spectrum MTOW / MZFW between the various marks of trainer.

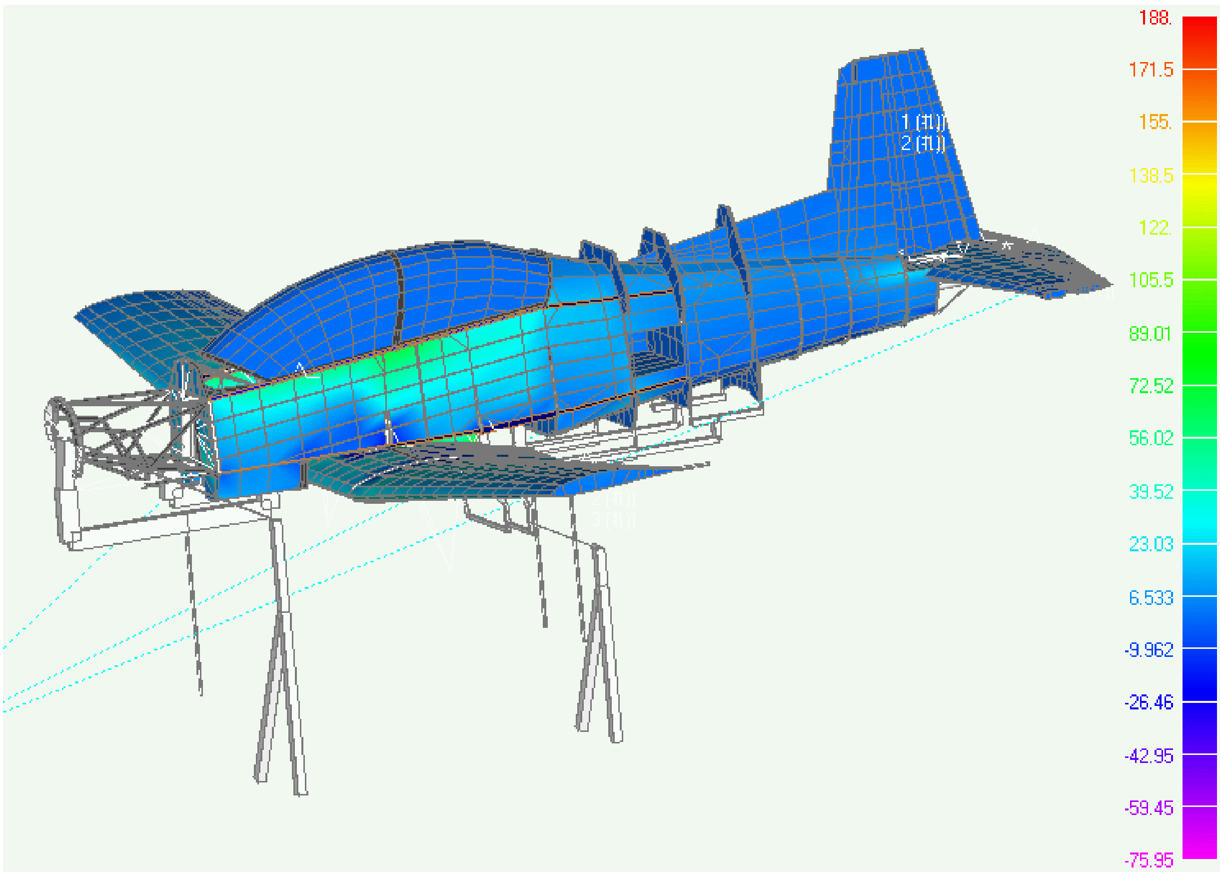
The FSFT load cases have been grouped by type of event and flight segment. The aircraft FEA model has then been modified to fully represent the FSFT configuration and validated against selected cases and the FSFT strain gauge outputs.

Each of the ten areas where significant cracking was found on the FSFT are being analysed in turn. DTA is also being used to show that if cracking is found at the initial inspection at that location, the repair embodied will be adequate for the safe operational life of 10,000 hours to be completed. This phase will be completed by the end of 2003.

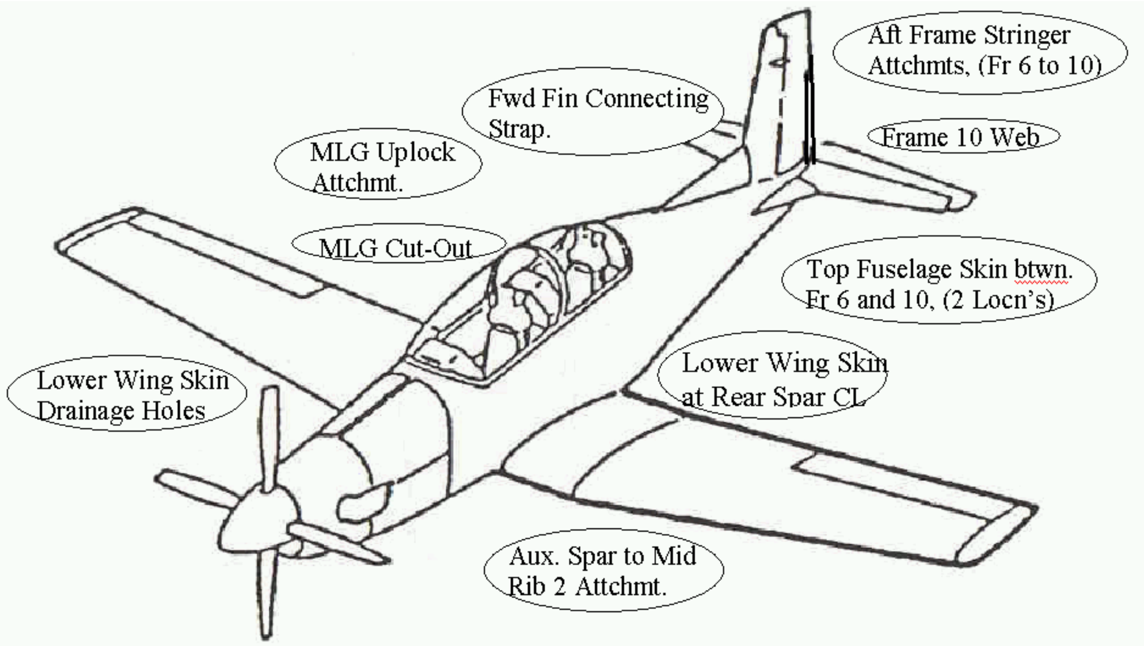
Phase 3 is planned for 2004 and will extend the study to cover the increased weight PC-9(M), operation at the FALSTAFF spectrum load levels, and any fatigue-sensitive structural variations across the fleet, which may not have been adequately represented in the FSFT specimen.

There will be a survey to seek out any detail design areas which may be more fatigue-prone than the FSFT airframe fatigue-critical locations, or prone to corrosion. This will be required to encompass all supported modification standards of all trainer fleets. Safe life or DTA parameters, as appropriate, will be calculated for all extra areas identified.
Modifications will be prepared to enable each area to complete the required operational life. Fatigue analysis (and if necessary DTA) will be performed on each repaired area, assuming damage is detected at its initial inspection, to verify this.

Pilatus PC-9(M)



FE Model representing FSFT setup



PC-9 Damage Areas

5.1 FATIGUE DESIGN PHILOSOPHIE OF THE PC-21 AIRCRAFT (M. Gottier*)

*Gottier Engineering Consultant

General description
The Pilatus PC-21 trainer is a low-wing monoplane with a pressurized stepped, tandem-seat cockpit. It's powered by a 1600 SHP Pratt & Whitney turboprop engine. The aircraft is designed to satisfy the needs of the basic and the advanced flying pilot training. In addition to the high aerodynamic performance, it is equipped with a mission computer that will provide a capability that is far beyond any current generation training aircraft.

Structural requirements
The PC-21 a/c will be certified to FAR 23. Statically it has to fulfill an operational load factor range of –4g to +8g for symmetric maneuvers, and up to +5.33g for asymmetrical maneuvers. For fatigue the a/c is designed as damage tolerant, and has to meet the requirements of FAR 23.571 through 23.573. These paragraphs are supplemented by MIL-specifications and MIL-standards, when detailed information is needed.
The required service life is 15'000 flight hours based on the PC-21 design spectrum.

Structural description
The primary structure of the aircraft is made mainly of aluminum alloy in machined and sheet form of the type AA2124-T851 and AA2024-T3/T42. There has been extensive use made of integrally machined parts to a large extent in order to reduce the parts count and tolerance problems during assembly as far as possible. Since this technology has a negative effect on inherent crack stopping compared to conventional technology with sheet metal parts, the design of the aircraft has been done very carefully with respect to fatigue. Therefore, FEM analysis of the overall structure and detail parts has been widely employed (figure 5.1_1 and 5.1_2).

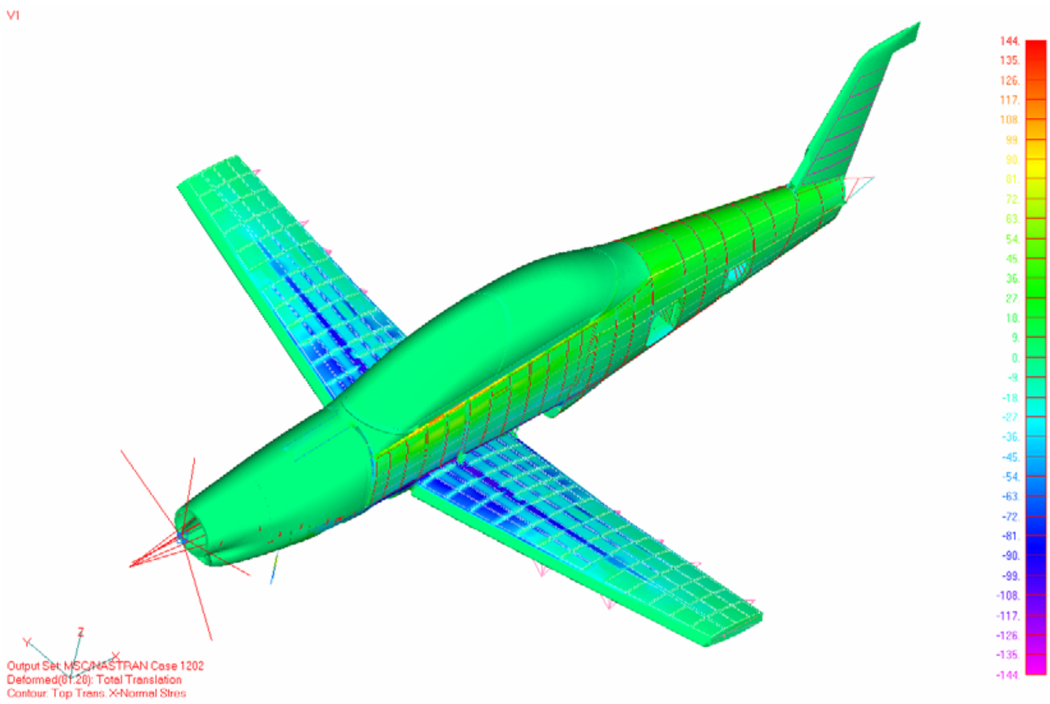


Figure 5.1_1: PC-21 FEM Model (8g steady pitching case)

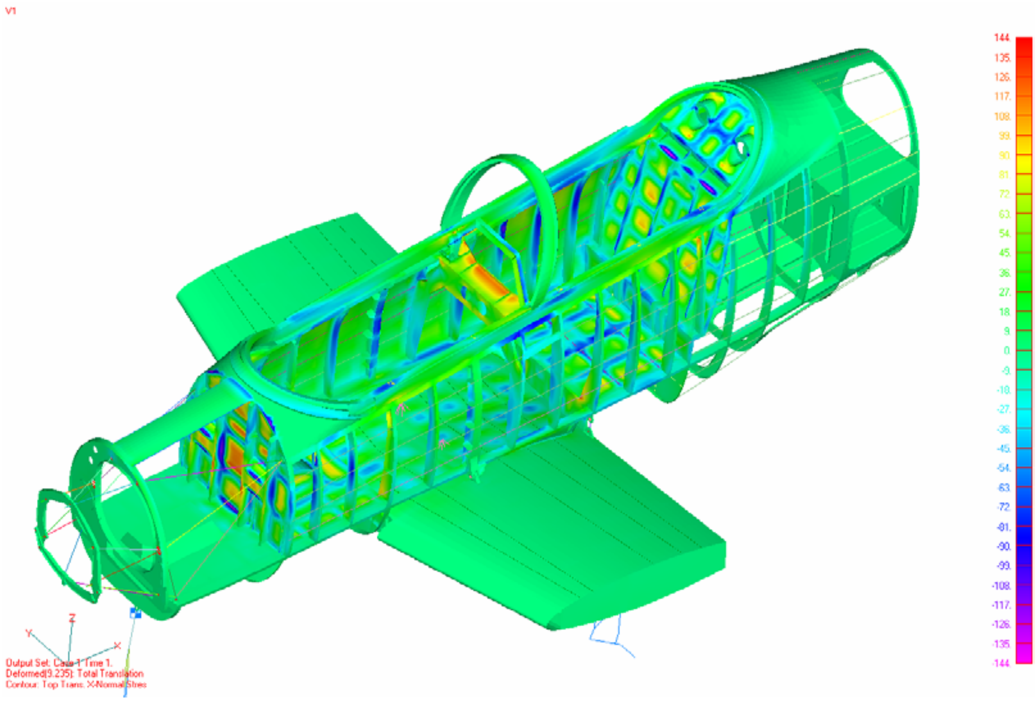


Figure 5.1_2: PC-21 nonlinear cockpit model (cockpit limit pressure case)

Spectrum

The PC-21 design spectrum was created based on the PC-21 mission specification from Pilatus Aircraft Ltd. In order to make the spectrum easy to understand, sorties, maneuvers and events have been defined. The parameters, which define each event were determined from measured data of the PC-21 "Proof-of-Concept" and information provided by flight test personal.

This PC-21 design spectrum is called the Master Design Spectrum (MDS), which is the basis to create local load or stress spectra for crack initiation or crack growth analyses. It is also the basis to define the test spectrum for the future full scale fatigue test. The n_z values of the MDS were compared to two reference spectra, i.e. the FALSTAFF spectrum and the Hawk1997 spectrum. This is a usage spectrum of the Hawk fleet of the Swiss Airforce. A comparison was made based on exceedance curves and based on crack initiation (CI) and crack growth (CG) analyses respectively. It revealed that the PC-21 n_z Design Spectrum is about as severe as the FALSTAFF spectrum, however more severe than the Hawk 1997 spectrum (see figure 5.1_3).

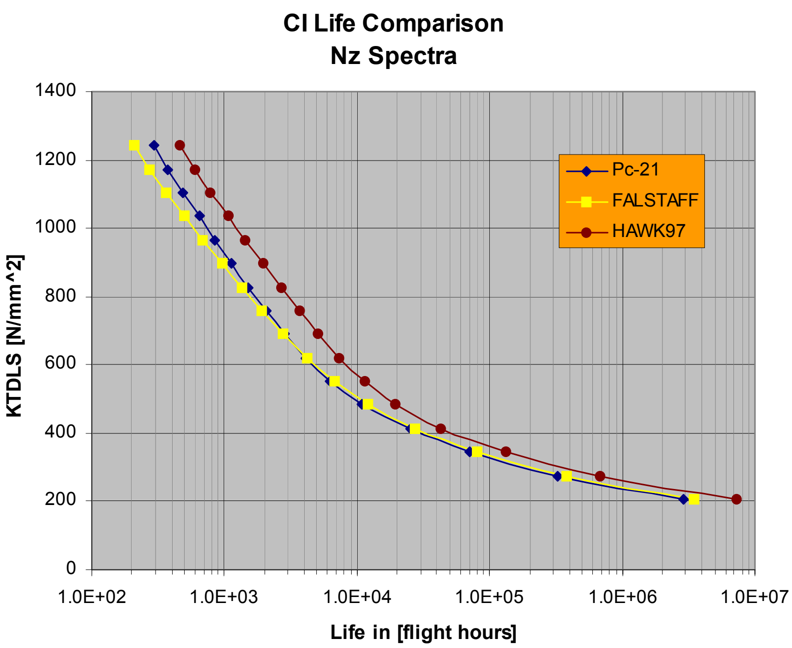


Figure 5.1_3: CI Life comparison for n_z spectra

The motivation to create a specific PC-21 MDS was, to have a realistic spectrum that is in accordance with the planned mission for the aircraft. In contrast to the FALSTAFF spectrum the PC-21 Design Spectrum contains symmetric and asymmetric maneuvers as well as different speeds and altitudes.
The PC-21 MDS consists of the following parameters:

- 3 types of mission (PT, AT, TWT),
- 22 design sorties,
- 36 unique maneuvers and
- 69101 events, consisting of a combination of flight parameters.

The MDS represents 500 hours of flying time or 246 sorties containing 530 flights.
This includes a mission mix in terms of flying time of 50 % PT (primary flying training), 40 % AT (advanced flying training) and 10 % TWT (tactical training).

Local spectra at critical locations are determined with results from the FEM shown above and the sequence load cases representing the MDS. Stresses are calculated for 55 fatigue load cases, representing maneuver conditions. Stresses due to the cabin pressure are superimposed to the stresses produced by maneuver loads. It is assumed that two pressure cycles of maximum operational cabin pressure occur per sortie.

Design Approach
The PC-21 airframe is designed to the damage tolerance and durability criteria of MIL-A-83444. This design should allow economical in-service inspections, minimize the cracking of primary structural parts and prevent the loss of the aircraft due to unstable propagation of undetected cracks and flaws.
The design of fatigue-critical parts is based on the crack initiation analysis using the Neuber-Notch approach and a scatter factor of four. In addition crack growth analyses with a scatter factor of two will be conducted for the most critical parts. A total of 43 locations are analyzed before performing the full-scale fatigue test (FSFT).

Structural testing
In order to certify the airplane to FAR 23 a FSFT has to be carried out. This FSFT also has to comply with damage tolerance requirements of MIL-STD-1530. The test will be conducted with 25 push-pull actuators and a test spectrum, which represents the PC-21 MDS. The goal of this test is to perform a durability test of 2 x 15'000 FH and a damage tolerance test of 1 x 15'000 FH. At the end of the damage tolerance test, a residual strength test will be conducted.
A detailed tear down inspection is planned after completion of the FSFT.
FSFT results will be verified by component tests for locations where the accessibility for inspections is limited.

Determination of inspection intervals
Inspection intervals for fatigue critical part will be determined based on crack growth analyses. These analyses will be adjusted to the test results from both the FSFT and component tests. This approach is similar to the approach chosen for the redesign of the Swiss F/A-18 a/c.

Monitoring System
A monitoring system will be installed in each PC-21 aircraft. The goal of this monitoring system is to allow the user to track each airplane individually and as a result of that, to manage each aircraft to the end of the planned service life with an economical maintenance effort.
The system will track the basic flight parameters as well as 6 strain gauges, located in each major structural assembly (see figure 5.1_4).

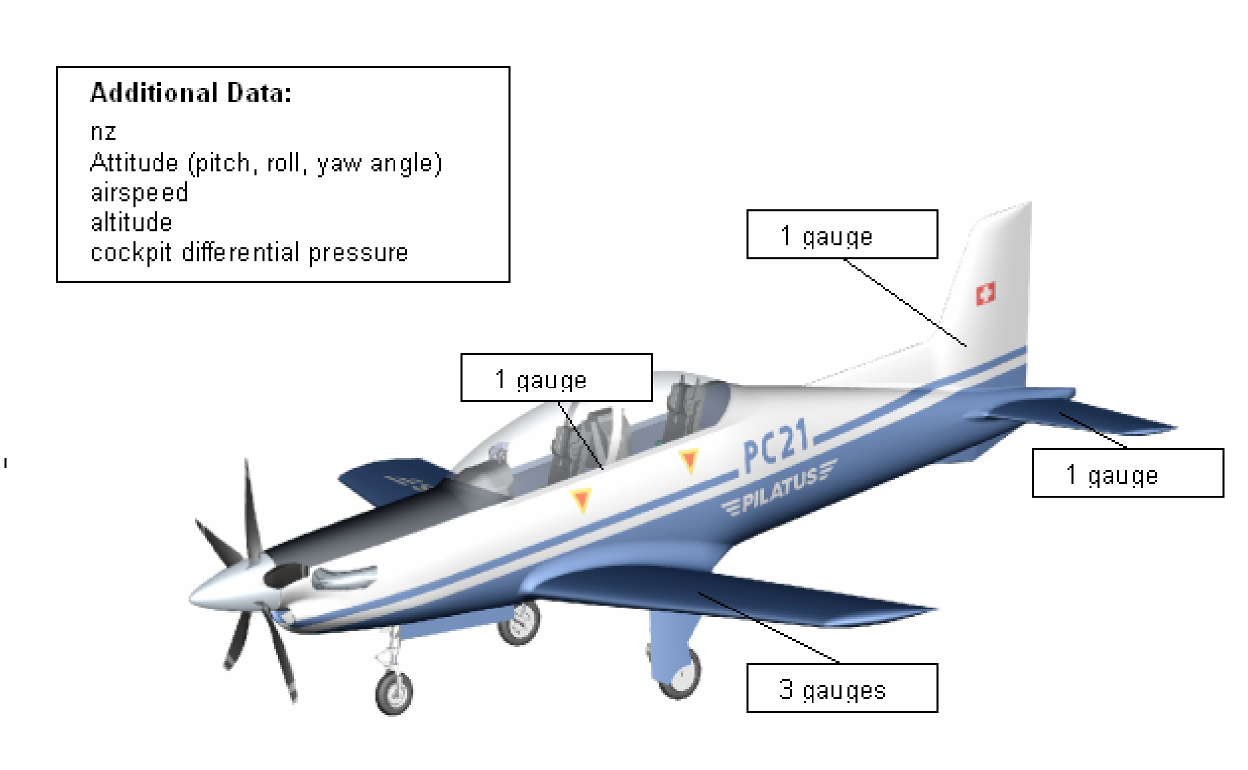


Figure 5.1_4: Gauges of the PC-21 fatigue monitoring system

5.2 PC-12 – FATIGUE ASSESSMENT OF PRODUCTION DAMAGES IN PRESSURE DOME (D. Haenni*)

* Pilatus Aircraft Ltd

Description of the Pilatus PC-12

The PC-12 is a large single engine turboprop utility aircraft, designed to perform a wide range of missions, such as transport of passengers, cargo, a combination of both, Medevac operations, etc. The Maximum Takeoff Weight of 4500 kg allows for a crew of two and up to nine passengers or for up to 1500 kg of cargo. The maximum operating altitude is 30'000 ft. The PC-12 Pratt & Whitney PT6A-67B turbine, flat rated to 1200 SHP during takeoff and 1000 SHP during cruise enables the PC-12 to cruise at 270 KTAS at high altitude. Over 400 PC-12s have been produced until spring 2003.

Figure 1: Pilatus PC-12

Certification Basis of the Pilatus PC-12

Certification basis is FAR Part 23, Normal Category. The type certificate for the 4100 kg version was obtained in 1994, the 4500 kg version followed in 1996. The PC-12 primary structure is certified as “Safe Life”. The Fatigue life was demonstrated with a Full Scale Fatigue Test where seven lives were simulated. One life equals 20’000 flying hours which is equivalent to 27’000 flights.

Introduction

A number of PC-12 pressure domes were found in the fleet with production damages, such as drill marks and production tool damage in the surface and also with dents.

Description of Pressure Dome

The PC-12 pressure dome, which is stretch formed from 1.6 mm AA 2024.T42 sheet metal and chemi-milled down in steps to 1.0 mm and 0.8 mm, separates the pressurized cabin from the unpressurized tail cone during flight (see Fig. 5.2_1). The pressure dome is an integral (one piece, monolithic) structure, which is connected to the fuselage at the interface of cabin and tail cone. The max. design differential pressure at 30'000 feet operating altitude is 5.741 psi (396 mbar).

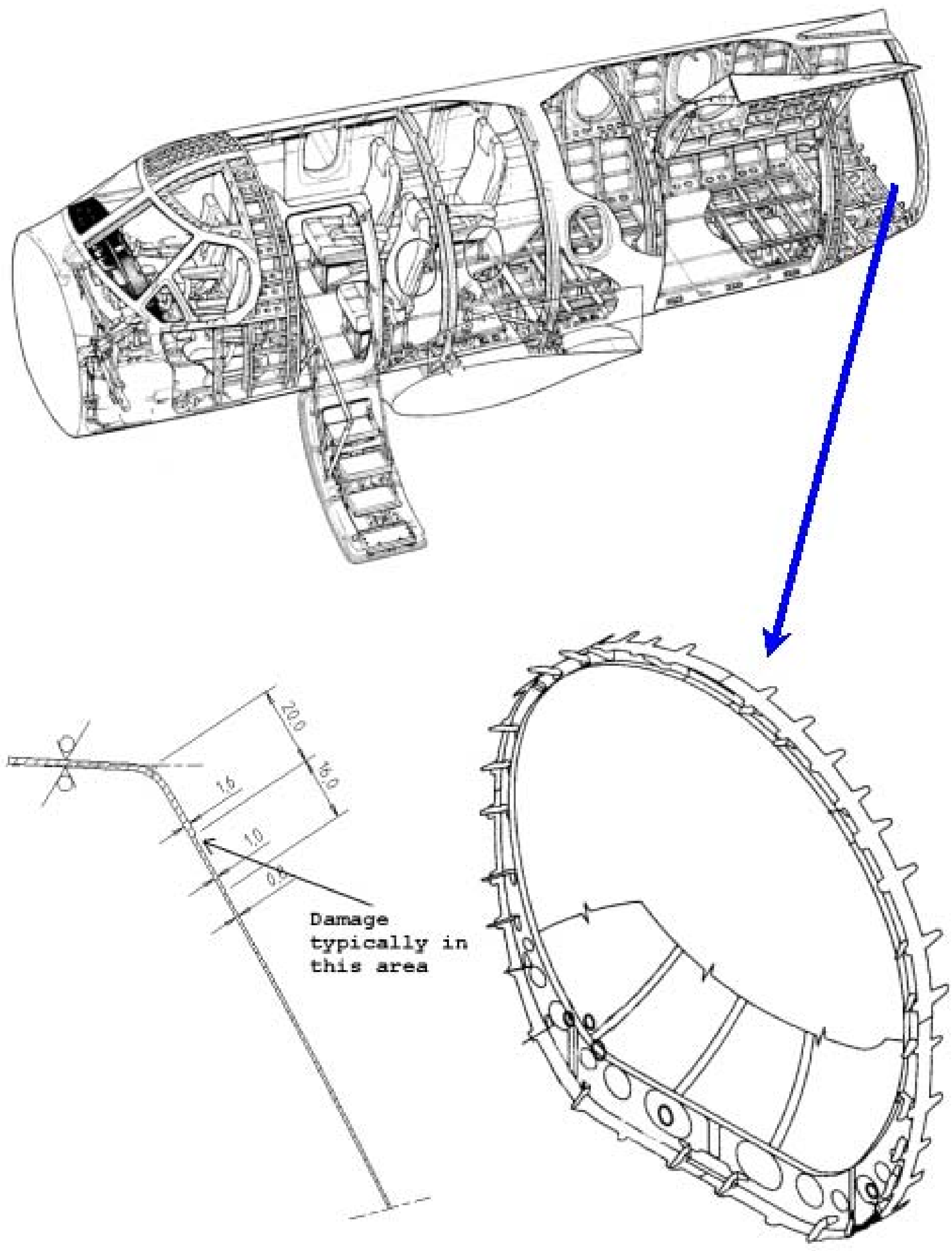


Figure 5.2_1 2: Location of pressure dome

Crack Growth / Residual Strength Analysis

A crack growth and residual strength analysis was performed (stresses derived from FE-model) with the software AFGROW. Conservatively one pressure cycle corresponding to 30'000 ft was assumed for each flight. In addition a factor of 1.33 was included in the establishment of the stresses. Initial flaw sizes of a = c = 0.3 were assumed. The crack growth life from this initial crack size to the critical crack length of 52 mm was 170'700 flights. Using a scatter factor of 8 a life of 21'377 flights resulted. At that time the PC-12 fleet leader had accumulated 12'000 flights.

Ultimate Strength Test / Residual Strength Test

An ultimate strength test (900 mbar differential pressure) was performed with a pressure dome component test article, which was induced with artificial drill marks of 0.5 mm depth. The test article successfully passed the test.

A number of residual strength tests (with 450 mbar limit differential pressure) were performed with a pressure dome component test article. Tests were performed with saw cut lengths of 22 mm, 30 mm, 40 mm, 50 mm, 70 mm, 90 mm and 120 mm. The saw cut was sealed with silicone to allow for pressurization. The saw cut did not become unstable, even with a length of 120 mm.

Pressure Dome Component Fatigue Testing

The test was set up to prove that non-conforming pressure domes (with simulated repairs) still fulfill the requirements for a “Safe Life” structure as it was demonstrated with a conforming pressure dome during the FSFT. The component fatigue test had the primary goal of quickly obtaining sufficient data to clear the non-conforming aircraft and the secondary goal of establishing absolute limits for acceptable damage.

The component test was performed by IABG (Germany) to the differential pressure of 396 mbar, corresponding to 30,000 ft, for every flight. This would ensure conservative results and simplify the test procedure. The scatter factor was set to 6. After the test there was no structural failure was detected and no cracks were found. The test article was therefore considered to have successfully passed the component fatigue test.

Criticality Assessment

The static strength of the tail cone was checked for a combination of flight loads and differential pressure loads as a result of a sudden decompression of the cabin into the tail cone following a failure of the pressure dome based on FAR 25.365.

Conclusions

The combination of analysis and testing allowed within a relative short time to evaluate the criticality of the damages and to demonstrate that the PC-12 pressure domes with blended out areas still fulfill the design requirement of a “Safe-Life” of 20’000 flying hours (27’000 flights) without additional restrictions. It also showed that the residual strength requirement of this structure is exceeded.

5.3 PC-12 – ASSESSMENT OF FATIGUE FAILURE IN THE NOSE LANDING GEAR DRAG LINK (D. Haenni*)

*Pilatus Aircraft Ltd

Dscription of Failures
Three nose landing gear drag link assemblies failed after relatively short life times. The design life of the PC-12 is 20,000 hrs or 27,000 flights whichever comes first. The failures occurred at 5950, 6170 and 5000 flights. See Figure 5.3_1.

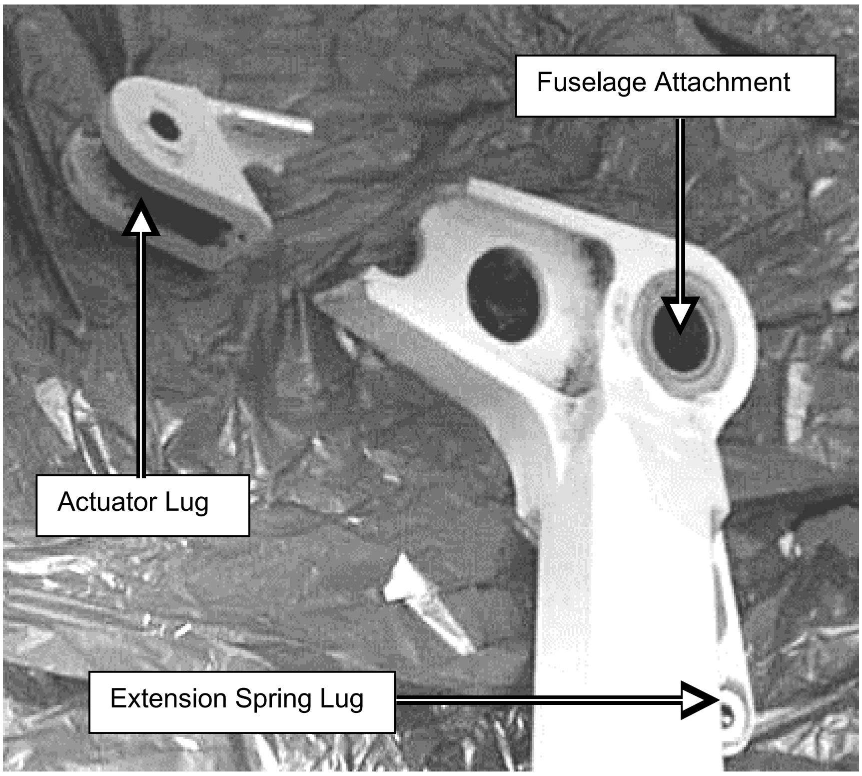


Figure 1: Drag Link after failure

Assessment
The original stress analyses underestimated the design limit load (thermal expansion of the hydraulic fluid was ignored) and incorrectly identified the critical section. As the PC-12 is certified to FAR Part 23 there was no fatigue analysis carried out for this landing gear part.

A new static finite element model calculated the stress concentration factor at the critical section. See Figure 5.3_2.

Fatigue analysis was performed for the original part, which indicated that the theoretical life was less than the design goal and critical crack length was relatively small (unlikely to be found during visual inspection).

The fatigue analysis was performed with the crack initiation program CITIME (to create KTDLS curves) and crack growth program AFGROW.

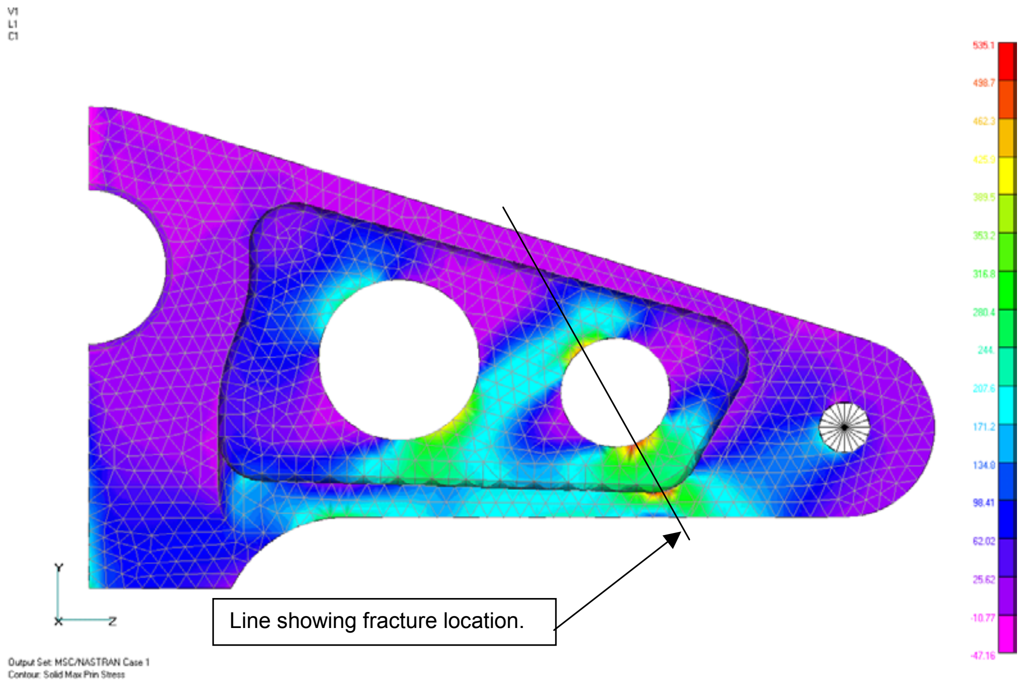


Figure 5.3_2: Finite element model of original drag link

Solution

Based on these findings the original part was modified (with the goal to eliminate the high stress concentrations and reduce general stress levels (see Fig. 5.3_3) and the fatigue and crack growth analyses were repeated. Several modification steps had to be performed until the fatigue life of the new design met the requirement.

Analysis has shown that the new part exceeds the design life of 20'000 hours using a scatter factor of 8.

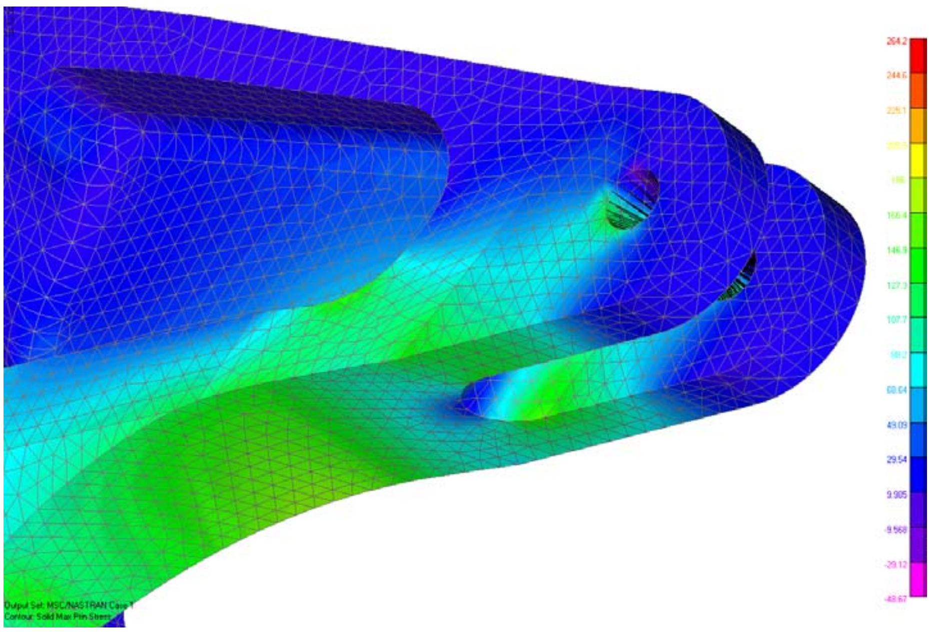


Figure 5.3_3: Finite element model of modified drag link

5.4 PC-12 – AIRCRAFT PREVIOUSLY EVALUATED WITH «SAFE-LIFE» SUBJECTED TO A WEIGHT INCREASE (D. Haenni*)

*Pilatus Aircraft Ltd

Introduction

The design fatigue life of the PC-12 with a Maximum Takeoff Weight (MTOW) of 4100 kg and the PC-12/45 with a MTOW of 4500 kg is 20’000 flying hours or 27’000 flights. To substantiate the design fatigue life a Full Scale Fatigue Test (FSFT) was performed with a representative PC-12 FSFT article, see Fig. 5.4_1. The loading spectrum used during the FSFT was based on the loads corresponding to a MTOW of 4000 kg, as the initial design target for the MTOW of the PC-12 was 4000 kg. The required testing was completed after four lives. Three additional lives were tested. The life number 7 was performed with increased loads in comparison to lives 1 to 6.

None of the cracks and failures detected during the FSFT led to a catastrophic failure of the primary structure. The fail-safe design of the cargo door was demonstrated when it continued to withstand limit load after the complete failure of a cargo door hook fitting. After the failure of the lug at fuselage frame 24 during life 7 the aircraft was continued to be tested. It withstood a residual strength test without catastrophic failure.

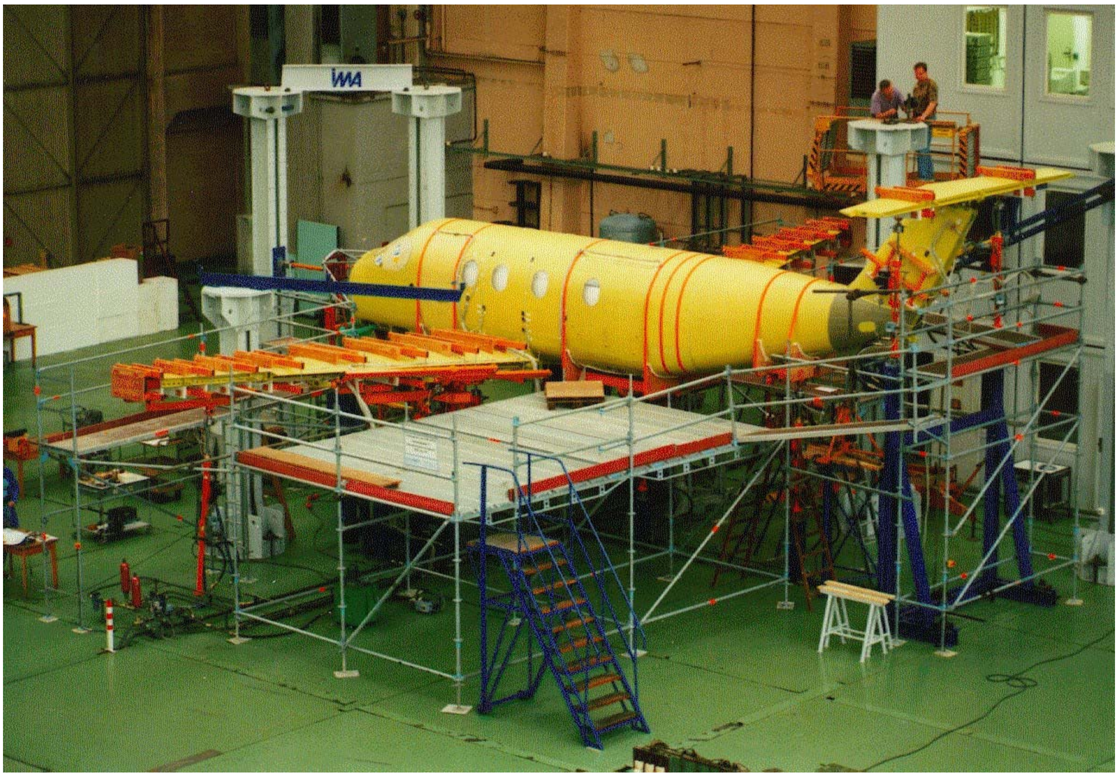


Figure 2: Pilatus PC-12 FSFT

In order to obtain the approval for 20’000 flying hours for the PC-12 and the PC-12/45, additional analytical substantiation was required.

Problem Discussion and Solution Path

In a first step structural areas, which are affected by the weight increase, were determined. A first selection of locations was made based on such criteria as fatigue test incidents, structural significance of items, highly loaded areas, locations of high stress concentration and areas of high load transfer.

A second and final selection was then made based on a prioritization analysis. This prioritization analysis took into account different factors influencing the fatigue life. Each factor was assigned with a rating value, which included also a weighting factor. The sum of all these rating values gives an indication of the criticality of the part: The higher this sum is, the more critical the part is.

Based on the above approach 6 areas with 9 specific locations were selected to be representative for the PC-12/45 configuration.

In a second step the stress spectra of the selected locations had to be derived in order to be able to perform later fatigue analyses.

The basis for these spectra was the three different test spectra applied on the PC-12 structure during the FSFT:

- L1 is the original test spectrum used to simulate the 1st life time of 20’000 hours.
- L2-6 is the slightly modified test spectrum used to simulate the 2nd – 6th life time during the next 5 x 20’000 hours.
- L7 is the modified test spectrum with 20% increment load increase used to simulate the 7th lifetime.

Stresses determined from the finite element model for configuration 1 (MTOW = 4000 kg) where then linked with the appropriate load cases in these spectra. For the configuration 2 (MTOW = 4500 kg) the stresses were again determined at the same location. These stresses were assigned to the load cases of the spectra L2-6.

The third step consisted of defining the CI lives at the selected critical locations.

The crack initiation lives (CI-lives) were calculated using the appropriate stress spectra. The CI life calculation is based on the Neuber-Notch approach and the Miner’s rule of linear damage accumulation.

The CI-life calculations were applied twice for each location: In a first run, the CI life was computed with the stress spectrum for configuration 1 and in a second run, the CI life is computed with the stress spectrum for configuration 2 (PC-12/45). The outcome of this procedure is two KTDLS curves per location (see Fig. 5.4_2).

The forth step was to determine the final fatigue life of each critical location: It was assumed that the crack initiation life is completed at the end of the seventh life in the test, i.e. at 140’000 flight hours. The Ktσ value from the test is defined by reaching the KTDLS curve of the PC-12 configuration at 140’000 flight hours. The CI-life of the PC-12/45 configuration was now determined by reacting the number of hours from the PC-12/45 (4500) curve at the same Ktσ value.

This procedure is called the pegging procedure using FSFT results.

For the FSFT the scatter factor was set to 4. For the substantiation of the PC-12/45 the same scatter factor of 4 was used.

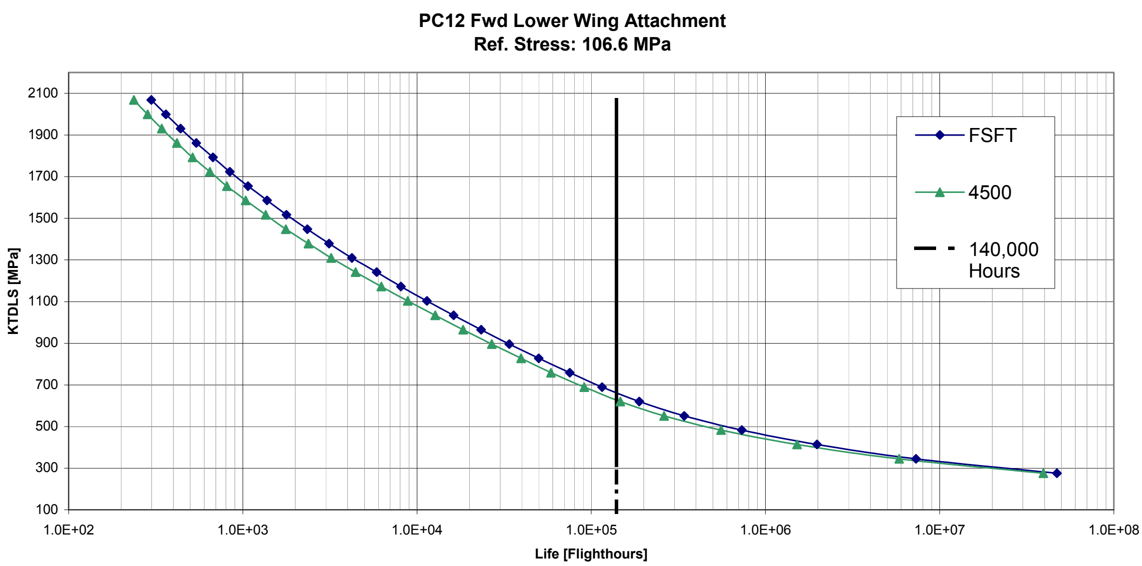


Fig. 5.4_2: KTDLS curves of PC-12 forward lower wing attachment

FSFT: 4000 kg version of the PC-12, which was the basis for the Full Scale Fatigue Test)

4500: 4500 kg version of the PC-12

Conclusion

Based on this approach it was possible to substantiate the service life of 20’000 flight hours or 27’000 flights for the PC-12/45.

5.5 FATIGUE RESISTANCE OF CARBON FIBER REINFORCED POLYMERS UP TP THE GIGACYCLE REGIME (S. Michel*, R. Kieselbach*, H-J. Martens+)

*Swiss Federal Laboratories for Materials Testing and Research, +Sulzer Innotec

Material systems for aerospace applications must withstand static as well as fatigue loads. Service load spectra show very high occurrences of very low loads, specially for control surfaces and empennage structures. Carbon fiber reinforced materials, based on epoxy-matrices are widely used in these structures. However, there is an interest to introduce carbon fiber reinforced materials with a thermoplastic matrix, for example Polyether-Ether-Keton (PEEK). The qualification of such a material requires a broad knowledge of the properties of such material systems in various environments. While static properties and fatigue resistance curves in the low cycle, medium and high cycle fatigue regime are well defined, data beyond 10^7 cycles (ultra high cycle fatigue regime) are not very well known.

In the presented work, in a first part, a testing program has been developed which allows to define the material properties beyond 10^7 cycles. Practical aspects of very time consuming testing due to the very limited possibility of a test acceleration and strong requirements for the stability of the testing set-up including the monitoring equipment were the major difficulties which had to be overcome. In a second part, a fatigue design concept was developed, which allows to predict the fatigue life of a typical structural detail under bending loads. Various failure criteria used in the static mode have been extended to fatigue loads. In a verification test program the fatigue strength of a waisted beam under bending loads have been performed and compared with the predicted fatigue life.

For the material system APC-2/AS-4 the fatigue resistance curves have been determined up to 10^9 cycles for an R-value of 0.1. Testing frequency, specimen geometry and testing equipment have been optimized to allow a testing speed, such that the self heating of the material remains low compared to a critical heat production, which would result in a thermodynamic degradation of the system. The testing equipment was a electromechanical resonance machine with a monitoring installation using a LabView software. During the test force, elongation and temperature on the specimen surface were systematically monitored. A measured reduction of the specimen stiffness correlated well with a frequency reduction of the whole system as well as with a temperature increase at the surface. However, the scatter in the data was large and a material degradation before final failure was normally not detectable for fatigue tests in the ultra high cycle regime. The specimen show in most of the cases a quasi brittle failure mode. With this set-up unidirectional (UD) and orthotropic (OT) laminates have been investigated. The fatigue resistance curves have been generated for purely tension and compression loading. For bending loads a limited amount of data is available for orthotropic (OT) and quasi-isotropic laminates (QI).

With the data generated, a fatigue limit at 10^7 cycle as commonly seen in steel materials is not found in APC-2/AS-4. The fatigue resistance in the ultra high cycle fatigue regime follows the same trend as seen in the low cycle to high cycle regime. The scatter in the data is increasing with number of cycles.

A waisted beam made from a orthotropic laminate was designed such that the various failure criteria developed predicted different failure locations. The static and fatigue strength of this beam were then determined experimentally. Based on the failure locations found in the tests, the number of appropriate failure criteria could be reduced. The predicted fatigue strength of the beam based on the remaining failure criteria correlated well with the experimentally determined fatigue strength up to 10^8 fatigue cycles. First results are shown on figure 5.5_1.

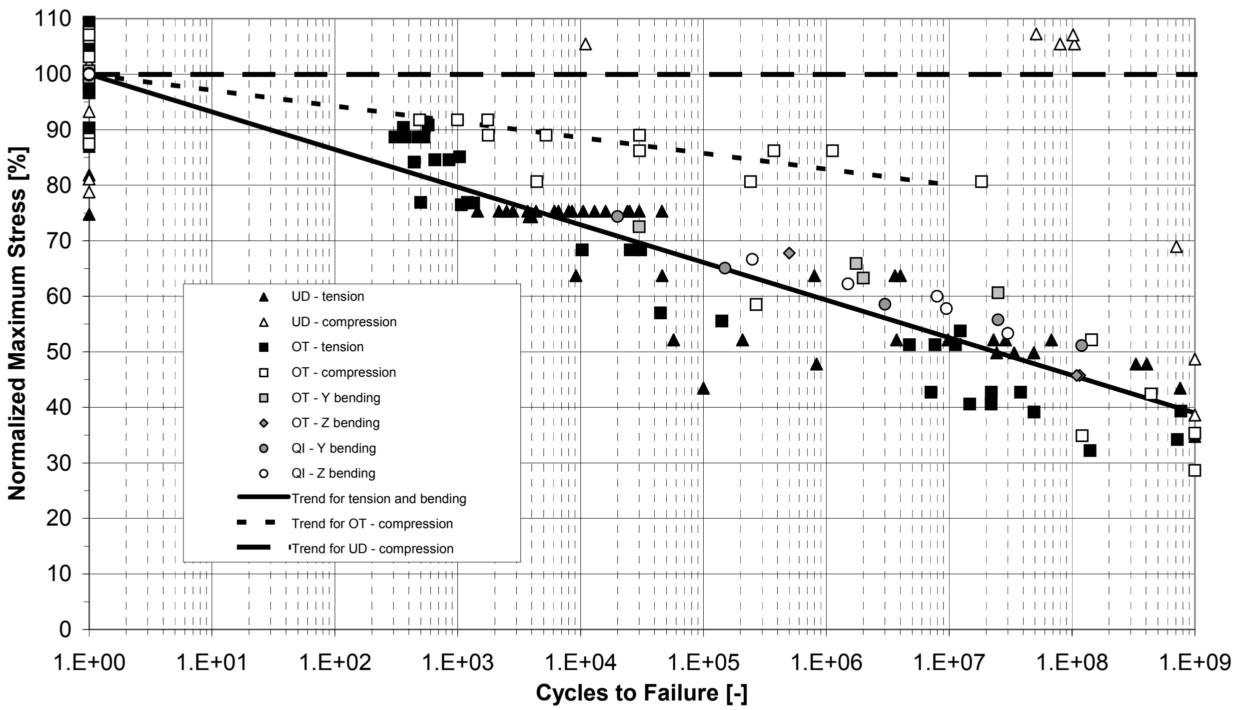


Figure 5.5_1: Normalized maximum stress versus cycles to failure data and fitted curves.

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