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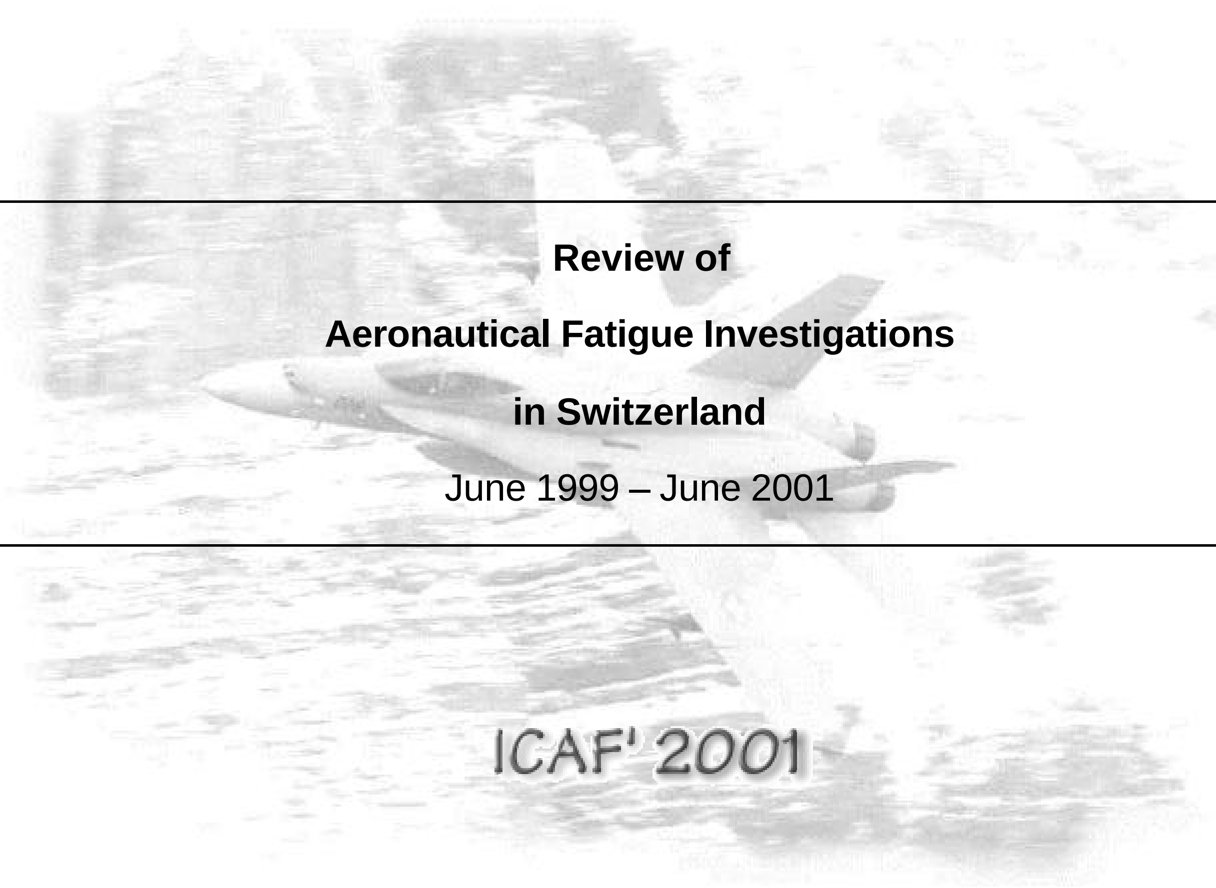

Review of

Aeronautical Fatigue Investigations

in Switzerland

June 1999 – June 2001

ICAF' 2001

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SUMMARY

The Swiss review summarizes fatigue work in Switzerland. It includes contributions from the RUAG Aerospace (RA) which is responsible for the maintenance and engineering of the Swiss Air Force. 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.

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4.1 INTRODUCTION

This review paper summarizes fatigue investigations which have been carried out in Switzerland during the last two years from 1999 till 2001.

The contributions to this review come from RUAG Aerospace (RA) and include the activities in the field of support for the Swiss Air Force to ensure the aircraft structural integrity of the fleet.

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 (M. Guillaume)

The Mirage III S (Swiss fighter version) were retired at the end of 1999. Most of the 29 airplanes were in a good structural condition. The airplanes were retired due to the introduction of the F/A-18 C/D Hornet. This was because they could not compete with the more modern F/A-18 and the availability of the spare parts for hydraulic and electronics was getting more and more difficult. On average the Mirage III S fleet reached close to 2800 FH per airplane. The Mirage III was cleared for 4000 FH based on the Swiss Full Scale Fatigue Test which took place at Emmen from 1976 till 1986, see Figure 1. Overall 20'000 SFH were simulated based on usage spectrum developed in the early seventies. Up to four main spars were broken during testing, the fleet showed severe cracking too at around 1000 FH. It was therefore decided to develop and test a refurbished main spar. Most of the fleet was later upgraded with the new main spar and by the introduction of the canard to increase the capability during maneuvers.
The remaining 16 Mirage III RS (reconnaissance version, see Figure 2) and the 4 double seaters will stay in service possibly until the end of 2005. The Mirage III RS have on average accumulated more than 3000 FH per airplane. All Mirage III RS with the new main spar have to date no cracks observed during usage. 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.

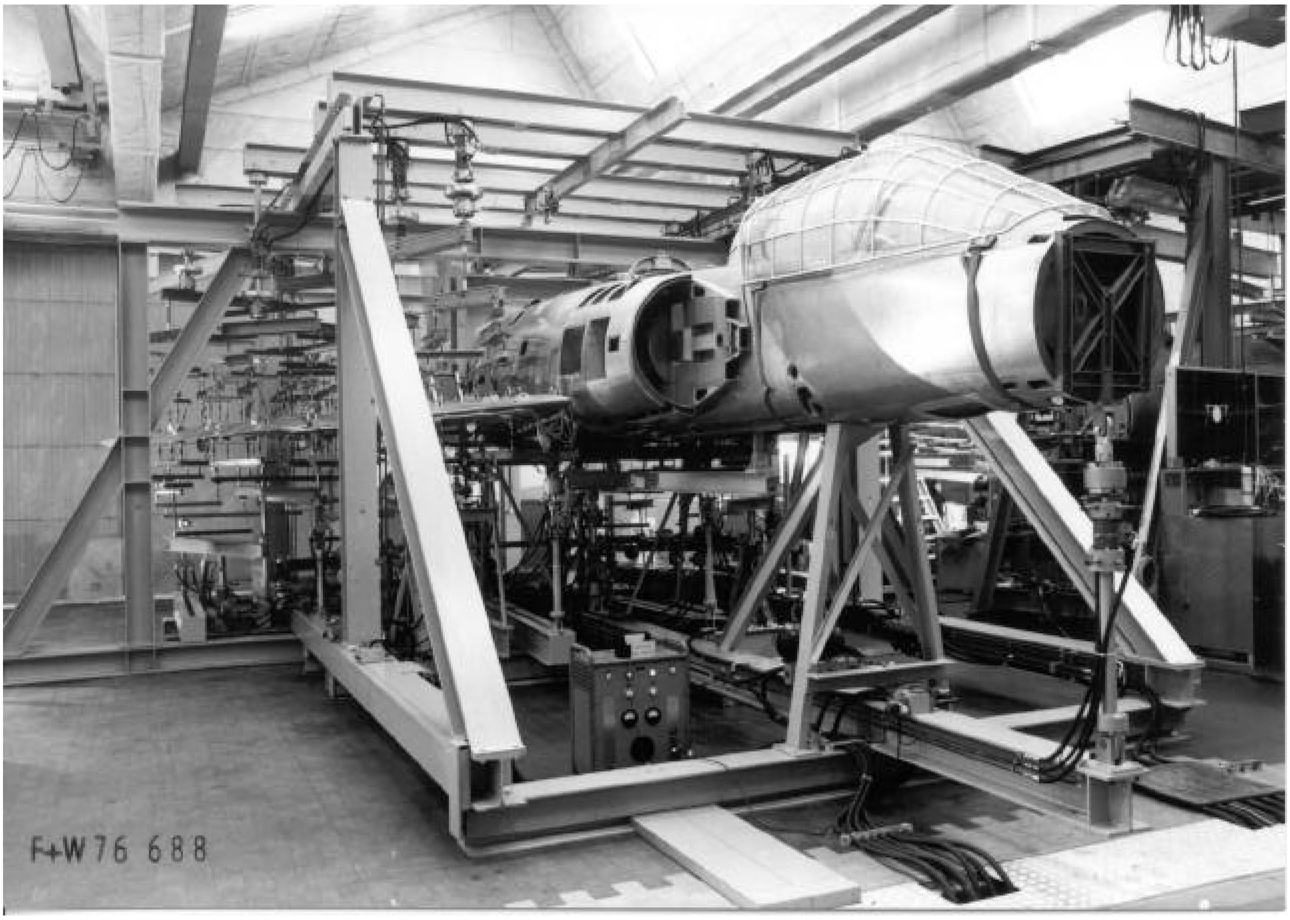

Figure 1: Mirage III full scale fatigue test at the Swiss Federal Aircraft Factory (F+W) Emmen

Figure 2: Mirage III RS, current configuration of the Swiss Air Force

4.3 THE HAWK Mk 66 FLEET (I. Kongshavn, P. Årebo)

During the period 1999-2001, a second phase in the fatigue monitoring of the Hawk Mk 66 (see Figure 3 and 4) was begun. In the first phase, fleet monitoring was performed strictly according to the safe-life philosophy. This philosophy and a description of the Fatigue Index (FI) calculation was given in the last national review summary at ICAF 1999 [1]. As the Hawk Mk 66 is now into its second decade of service, a more Swiss-specific methodology has been developed to better monitor fleet usage. In addition to FI monitoring of every aircraft, a mini-Operational Loads Program (OLM) was initiated to more closely monitor loading at critical locations in the center fuselage and tailplane.

4.3.1 Center Fuselage Modifications and Loads Monitoring

In order to extend the fatigue life of the center fuselage, a dual datum relief valve (DDRV) was installed after roughly 1000 FH. This valve limits pressurization loads during refueling, which are known to significantly contribute to fatigue damage in the center fuselage. In addition to the valve, strain gauges were installed to monitor loading near critical locations in frames 14 and 15. It is hoped that with the installation of the DDRV, a replacement of Frames 14 and 15 and multiple site damage in the tank bag side walls will be avoided during the remaining service life of the fleet.

4.3.2 Tailplane and Rear Fuselage

In the last year, a mini-OLM program for the tailplane was initiated. Tailplane vibrations between 20 Hz and 90 Hz are known to affect the Hawk [2]. Six strain gauges in the tailplane will be sampled at 512 Hz, and filtered to 100 Hz, to monitor fleet usage. This will provide both the data not currently captured by the Electronic Structural Data Acquisition system (which samples at 128 Hz and is filtered to 32 Hz), and will also provide data at six new locations. These six channels will be monitored over the next 2 years to record typical fleet usage.

Two years ago, a Swiss-specific tailplane inspection program was put into place. No significant fatigue damage has yet been found in either the tailplane, rear fuselage or center fuselage locations. However, based on other user experience, damage at these locations is to be expected. The data from the loads programs will be used to optimize the inspection planning and estimate the remaining life of these components.

Figure 3: Swiss Hawk Mk 66 used as jet trainer for Swiss Air Force

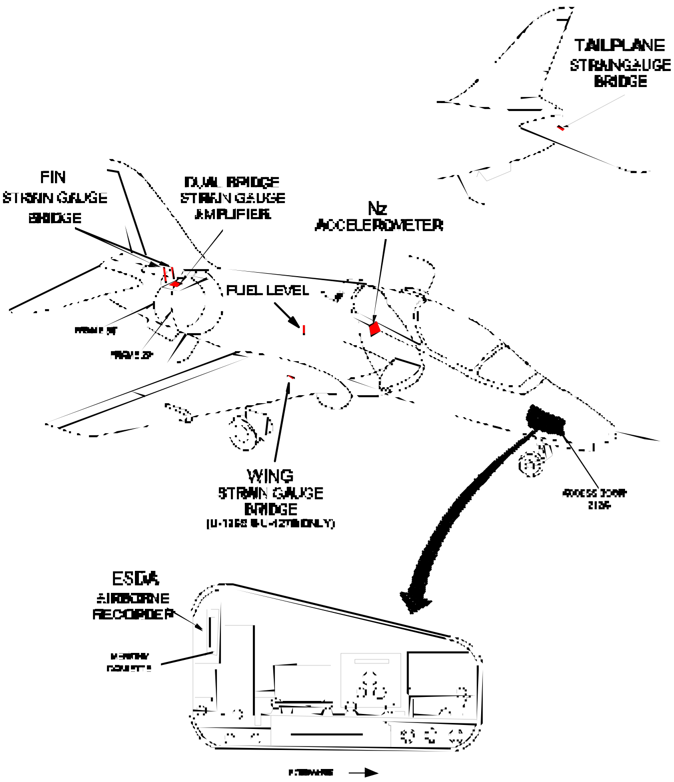

Figure 4: On board monitoring equipment of Swiss Hawk Mk 66 (ESDA = Electronic Structural Data Acquisition System)

4.4 THE F-5E/F TIGER FLEET (M. Nievergelt, A. Uebersax)

With the introduction of the F/A-18C/D Hornets in the Swiss Air Force, the F-5E/F fleet was reduced by 16 aircraft. There are currently 85 aircraft in the F-5E/F fleet, which is still the largest fleet of the Swiss Air Force (see Figure 5).

The aircraft are tracked by a system, which monitors all of the non-destructive testing (NDT) inspections which is stored on a databse. An individual component tracking has been introduced for the wing, as wings are swapped between different aircraft in order to repair them with a maximum efficiency and a minimum impact on fleet readiness.

Figure 5: Swiss F-5E aircraft are also used by the Swiss Air Force aerobatic team, the Patrouille de Suisse

4.4.1 Repair and life improvement programs

Wing refurbishment: The wing repair program is required to maintain fleet readiness in the second decade of service of the F-5E/F in Switzerland. The program includes the installation of new 15% inboard spars and new 66% spars. The spars are both manufactured and installed by RUAG Aerospace.

Temporary repair of 15% inboard spars: When cracks in the spars were detected in 1998, spare parts had drastic lead times, which would have resulted in an unacceptable fleet shortage. Therefore, it was decided to develop a temporary repair solution until part replacement [1]. The repair, consisting of a tapered steel strap, has proved to be reliable. New spars are now replacing these temporary repairs.

Lower longeron supports: The installation of Swiss redesigned lower longeron supports [1] has been introduced preventatively during the main overhauls.

Dorsal longeron standoff relocation: The electrical wiring is fixed by standoffs on the dorsal longeron. The standoffs were originally located above a fatigue-critical countersunk fastener hole. The large field-detectable crack size of this configuration leads to short inspection intervals. To minimize the inspection effort and the risk of crack initiation, the standoffs are now relocated on the dorsal deck. The initial countersunk hole is drilled out to a cylindrical hole, which is additionally coldworked and filled by an interference fit fastener [3]. This life improvement is carried out in the middle of the planned service life of the F-5E/F fleet in Switzerland.

4.4.2 Investigation of in-service incidents

Forward attachment bolt failure of wing tip missile launcher: An F-5F aircraft returned from a mission with a broken forward attachment bolt (EWB 26-10H12) of the wing tip missile launcher (LAU 100), see Figure 6. The vertically mounted bolt assures the connection between the wing tip rib and the missile launcher. The launcher was still in place, as the bolt was prevented from falling off by the cover cap. 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.

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. The bolt is mounted with a specified torque. The resulting pre-load guarantees contact between the lug and the launcher throughout flight.

The crack initiated in the forward-outboard direction (60° from the flight direction) in the 2^{nd} thread engaged in the nut. Only about 2% of the fracture surface, see Figure 7, was identified to be inter-crystalline corrosion. The rest of the section underwent a brittle failure due to an overload.

The result of the investigation showed that the failure mechanism was hydrogen induced stress corrosion cracking. The hydrogen was formed by a corrosion process resulting from the lack of air within the sealed thread. This same failure mechanism was encountered in the failure of a centerline pylon attachment bolt on the F-5E/F of the Swiss Air Force.

The two bolts are currently being replaced throughout the fleet and will be checked within a 300 FH interval. A reliable NDT procedure is being developed.

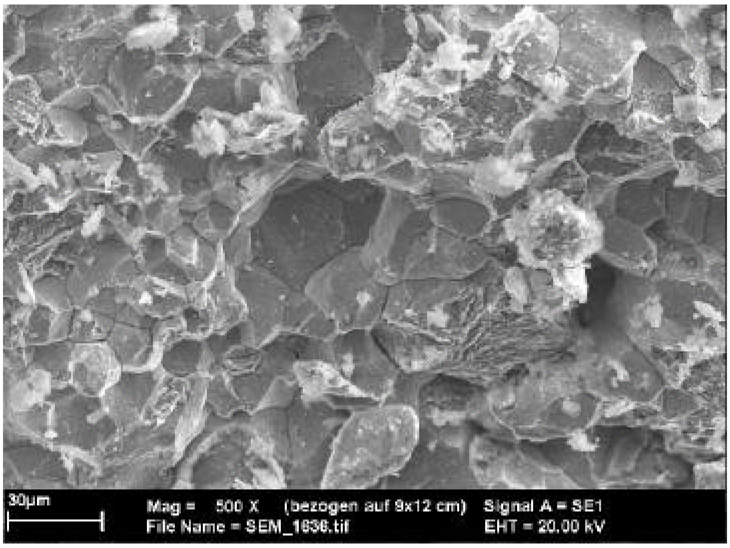

Figure 6 **Failed forward attachment bolt. The arrow indicates the corroded crack origin.**

Figure 7 **Microstructure in the corroded area of the bolt (crack origin). The crack is inter-crystalline and open gaps between the grain boundaries are visible.**

4.5 THE F/A-18 FLEET USAGE (S. Oesch)

Switzerland now has almost five years of experience with the operation of the F/A-18 fighter aircraft. The introduction of the fighter was completed successfully and today it is operated routinely. The aircraft's' structure to date has required a few small technical dispositions, but the young fleet, so far, has not been affected by fatigue damage. To prepare for future eventualities several engineering projects are in place: The Swiss Full Scale Fatigue Test is in an advanced engineering phase; the capability of performing fatigue analyses is being built up, and fatigue monitoring of the fleet is made regularly.

4.5.1 Capabilities of the fatigue tracking system

The built-in fatigue tracking system monitors every single flight of every F/A-18. Two main sources of data belong to the monitoring system. One source is the flight parameter data and the other source is the strain sensors. The collected data is downloaded after each flight for processing with the fatigue tracking software.
The strain sensors monitor the major component load paths of the aircraft structure. There are seven strain sensor positions in the F/A-18, but the one at the titanium carry-through bulkhead generates the most reliable results for the calculation of the fatigue life expended of the Swiss aircraft. This sensor allows individual aircraft tracking of the aircraft's main structural components. The fatigue calculation follows the rules of the strain-life approach.
Recent engineering efforts tried to define a calibration method for the Swiss specific strain sensors, which would allow the tracking of additional F/A-18 components.

The operational flight program was updated (OFP13), and the Swiss Air Force slightly changed the configurations that are actually flown. These two facts required an update of the Swiss fatigue tracking software. During this update the year 2000 capability and the introduction of improved data integrity checks were also addressed.
These software changes require a reprocessing of all Swiss F/A-18 fleet data, which is now in progress.

4.5.2 Results of the Swiss F/A-18 usage monitoring

The monitoring results were analyzed and compared to the Swiss design spectrum of the F/A-18. It was observed that the flight maneuver loads spectrum was less severe than expected while the dynamic flight environment, which generates the well-known buffet of some F/A-18 components, was more severe than forecasted.
Based on these findings, engineering activities were concentrated on buffet affected components of the aircraft. A first study was done for the aileron and trailing edge flap. The study was made using fleet usage data and showed results which will necessitate maintenance action for the fleet.
In a future study the situation for the other buffet-affected components will be evaluated, such as the vertical tail (see Figure 8 & 9) and the outer wing. The following two Figures illustrate typical F/A-18 vortex shedding, which is considered one of the primary origins of the buffet phenomena.

Figure 8: Swiss F/A-18 with visible vortices

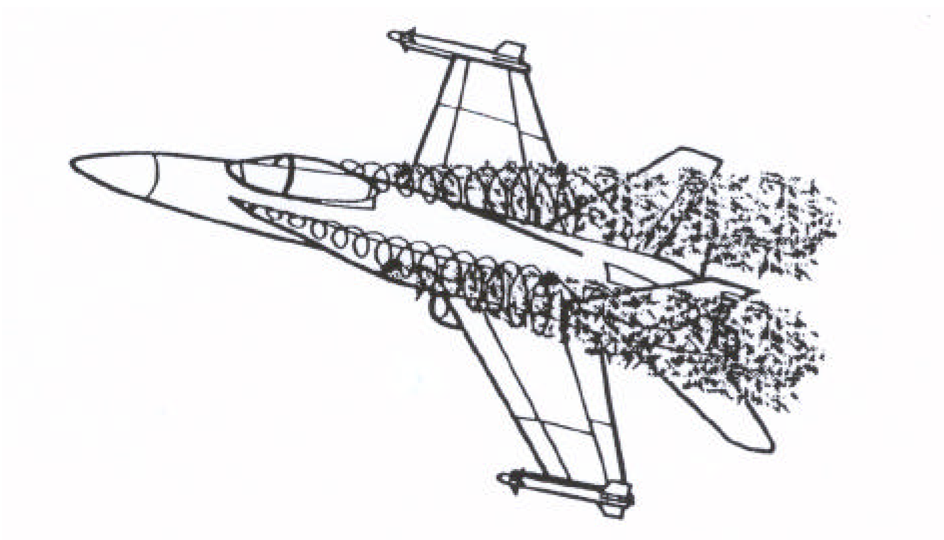

Figure 9: The F/A-18 LEX vortex is generated at high Angles-of-Attack

4.6 THE SWISS FULL SCALE FATIGUE TEST PROGRAM (M. Guillaume)

In the 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 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.

The redesigned Swiss F/A-18 structure based on the analytical approach used in the ASIP study shall be verified and validated:

– **First** the Swiss fatigue requirements of 10,000 service flight hours (SFH) of crack initiation, 10,000 SFH of crack growth and the residual strength load of 1.2 * P_{xx} have to be proven.

– **Second** the fatigue lives of those structural parts, which do not meet the Swiss redesign requirements, will be determined by test and validated.

– **Third**, the analytical approach used to redesign the Swiss F/A-18 structure will be verified, i.e. the pegging procedure described in ref. [4] will be checked by proving the following relationship:

Ana1ytical CI life / CI life determined by test = constant.
In the ASIP study it was assumed that this relationship would be true for the US NAVY and the Swiss design spectrum.
The Swiss FSFT program will be performed in the time frame between 1998 and 2004. These 6½ years are broken down in three major time periods with following milestones:

- **Pre-Testing-Activity Phase (Summer 1998-Fall 1999):**
  To assemble the FSFT article,
  To define a detailed test concept, the test master event spectrum (MES) and fatigue relevant load cases,
  To carry out a coupon test program to verify the test MES,
  To convert FE data from CGSA/ CASD into NASTRAN/ PATRAN models and
  To build up a full FEM for the F/A-18 aircraft to support the FSFT.

- **Test-Set-Up Phase (Fall 1999-Fall 2002):**
  To develop the test load spectrum and to set up the flight by flight test program,
  To determine the inspection concept and
  To build up the mechanical test setup including test equipment installation (hydraulics, jacks, control & monitoring and data acquisition system).
- **Test Performance Phase (Fall 2002-2004):**
  To simulate the 10,400 test hours and to perform inspection based on the requirements of the inspection concept.

The test should have been accomplished in a successful manner and at this time the data from the inspections during test should be processed.

4.6.1 ASSEMBLY OF TEST ARTICLE AND INSTRUMENTATION

The assembly of the FSFT article was started in November 1998 with the installation of the vertical tail at SF Emmen. The complete assembly was at SF Emmen within the production program. The final assembly of the test article was after the last F/A-18 a/c for the fleet. Due to the special configuration of the test article the assembly took some additional time. In September 1999 the test article was completed and applied with 1095 strain gauges for the test. For the acquisition of 1095 strain gauges and other data as e.g. the forces of the load cells, a data acquisition system, with 1800 channels will be required. At present SF is in the procurement phase for the combined control & monitoring and data acquisition system. The structural configuration of the FSFT article is recorded in detail, which is very important for a successful test and data management.

At present the test article is ready to be reinforced for local load introduction and for installation of the dummies, see Figure 10. All the dummies and fittings were designed by SF design office and will be released for manufacturing by the end of 2000. The dummies (radom, gun mount, nose landing gear, ctr line pylon, main landing gear, inboard wing pylon, engine, and arresting hook) are predominantly used as load introductions.

Figure 10: Test article with wings removed for pading

4.6.2 DEVELOPMENT OF TEST MASTER EVENT SPECTRUM

The FSFT will be performed with the Swiss design spectrum, which was used for the ASIP study to define the structural modifications. However, in order to reduce the number of load cases for the FSFT, it was necessary to perform a study to determine a test MES program. The loading test set up is defined by means of a "push-pull" actuator system, which has already been applied successfully in FSFT of other fighter aircraft.
Each event in a Master Event Spectrum (MES) is defined by a combination of an aircraft configuration and a point-in-the-sky (PITS). The loads on an aircraft are affected by these combinations. There are 25 combinations of PITS and configuration in the Swiss F/A-18 design criteria. During the Swiss F/A-18 ASIP study, the number of PITS/ Config combination was reduced during the fatigue loads development. The fatigue analysis of the ASIP study and the subsequent analysis to support manufacturing are based on 17 PITS/ Config combinations for symmetric maneuvers and 11 PITS/Combinations for asymmetric maneuvers. Therefore, the fatigue lives predicted by the 17/11 PITS/Config will serve as the basis of developing a simplified MES suitable to the FSFT (see Figure 11). The reason to perform this task is related to the amount of fatigue loads relative to each PITS/Config combination: the number of fatigue loads data increases with the number of PITS/Config combinations.

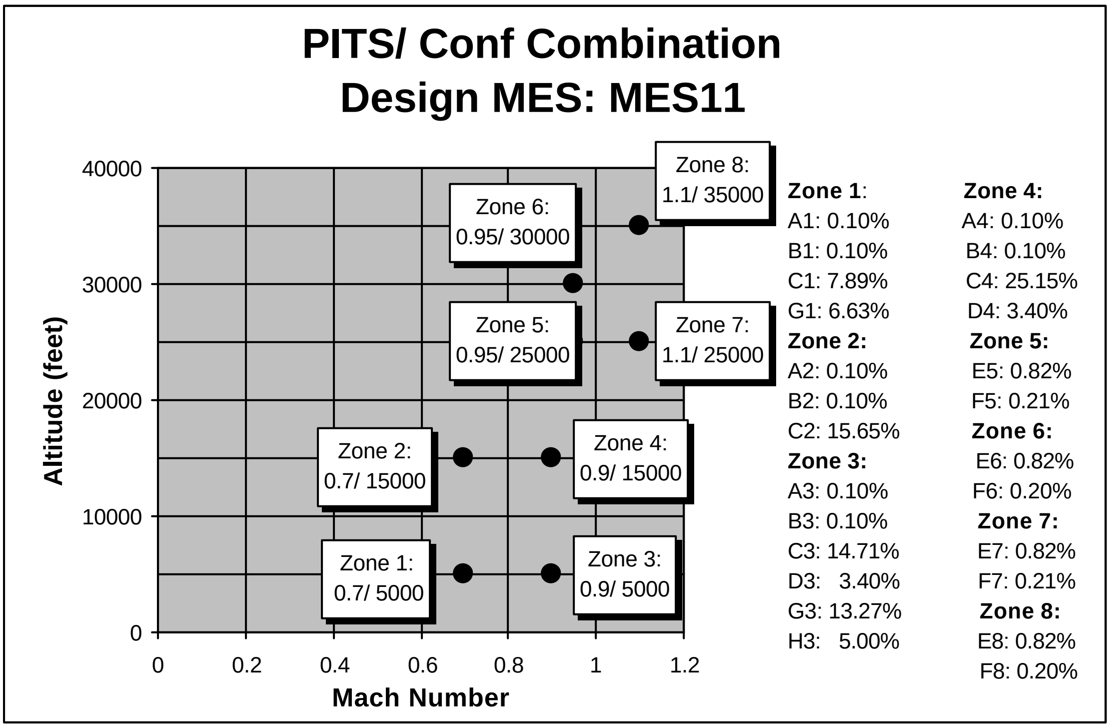

Figure 11: PITS/Config combinations for the Swiss design used in the ASIP study

For the ASIP analyses, fatigue loads were generated only for 17 selected locations in the F/A-18 airframe and for 17/11 PITS/Config combinations. However, for the FSFT fatigue loads of the entire aircraft need to be generated. The goal to reduce the number of combinations of PITS/ Config in the MES for the FSFT was to maintain the fatigue damage within ± 10% in terms of crack initiation (CI). CI is the primary fatigue design criterion for the F/A-18 aircraft. There were eight locations selected on the aircraft to develop the test MES (see Table 1). It mainly addressed the ctr section and the wing due to the changes and the redesign of the Swiss configuration.

1.	Wing root – Bending moment (Wrbm, FS475, Xw38)
2.	Wing fold – Bending moment (Wfbm; FS515, Xw162)
3.	Forward fuselage FS357 – Bending moment (Ff357bm)
4.	Center fuselage: fastener hole @ FS544 in upper main (dorsal) longeron – Local stress (Cf544dor)
5.	Center fuselage: free edge of skin flange @ FS488 in upper outboard longeron – Local stress (Cf 488obl)
6.	Center fuselage: free edge of ECS cutout @ FS428 in dorsal deck – Local stress (Cf428ecs)
7.	Center fuselage: FS488 bulkhead OML near lower lug – Local stress (Cf488oml)
8.	Aft fuselage FS557.5 – Bending moment (Af557bm)

Table 1: Selected 8 loacations for PITS/Config reduction

The development of the test MES was carried out in two steps:

1. A reduction of PITS/Config combinations was performed by comparing loads of the wing root location for those PITS/Config combinations causing major fatigue damage. Using this approach, three MES were generated with a computer program for 11, 8 and 6 PITS/Config as a starting point. Three Wing Root Bending Moment (WRBM) spectra were then created based on the MES of those PITS/Config combinations. Based on these stress spectra the crack initiation (CI) lives were afterwards determined for the most extensively used material in the F/A-18 aircraft, i.e. titanium, aluminum 7050 and aluminum 7075.
   The results are presented in Figure 12. The fatigue lives are normalized to the design spectrum consisting of 17/11 PITS/Config combinations. It revealed that further adjustments of combining PITS/Config was necessary leading into a second step with more detailed investigations in modifying the distribution of PITS/Config combinations.

2. The procedure mentioned above was used to determine the CI lives at eight different locations (see Table 1) of the F/A-18 aircraft for 6 PITS/Configurations combinations with different distributions. Several iterations were necessary in order to achieve the goal for the test MES spectrum with 6 PITS/ Config combinations.

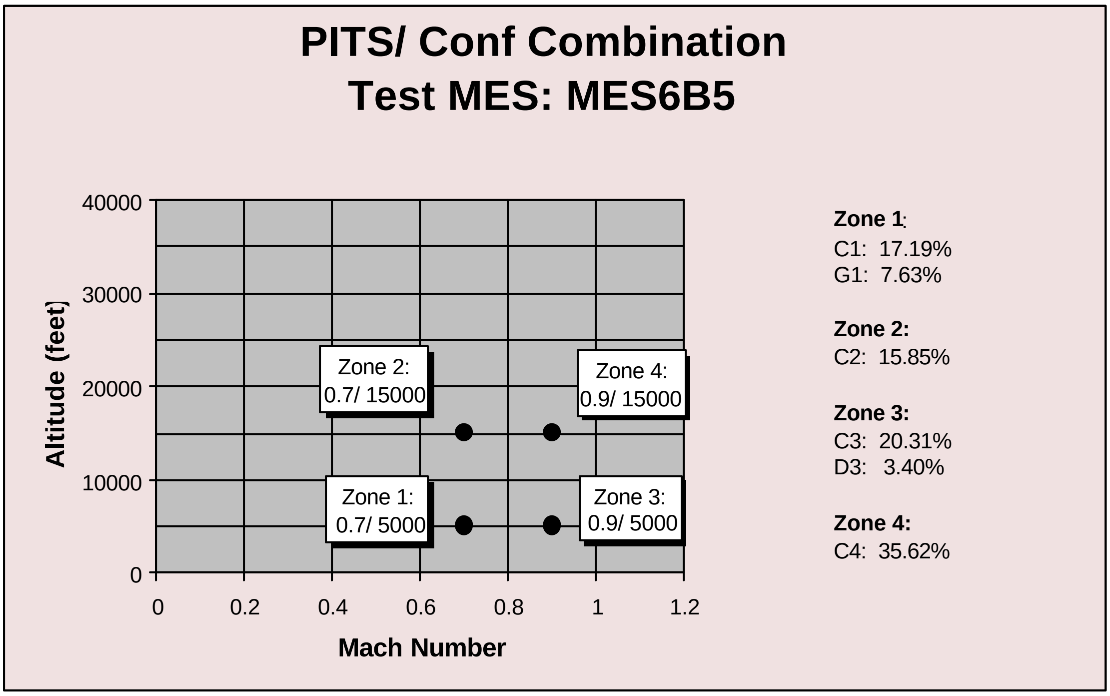

Figure 12: PITS/Config combinations for the test MES (called MES6B5)

Based on this extensive analytical study the following conclusions can be drawn:

- A test MES for the FSFT was created with 6 PITS/Config combinations (see Figure 3) meeting the ± 10% CI life criterion with small exceptions (worst case –17% reduction) at all eight fatigue relevant locations of the F/A-18 aircraft.
- A reduction of PITS/Config combinations based on one material can result in an erroneous MES. All major materials used in the airframe should therefore be included in the investigation.
- The influence of each PITS/Config on the load/ stress of the various major components in an airframe is significantly different. Thus, the PITS/Config reduction study should include a number of locations whose load/ stress can be representative of the major components.
- Some PITS/Config have a small percentage of usage, but have a major influence on the load/ stress of some locations. These PITS/Config must be identified and included in the MES to meet the goal of the intended CI life criterion.
- The analytically determined test MES has to be verified by a coupon test program using specimens made of all major materials of the aircraft and different spectra with different stress levels.

4.6.3 COUPON TEST

The Swiss Test MES was developed for the FSFT by reducing the number of PITS/ Config combinations without changing the number of events. Based on an analytical approach, described above the number of combinations were minimized while maintaining the crack initiation (CI) life within a tolerance range of ± 10% with small exceptions.
In order to qualify this Test MES program for the FSFT, verification by a coupon test program was deemed to be necessary. The goal of the coupon tests is to compare the Swiss Design Spectrum with the Swiss Test Spectrum. This comparison will be accomplished based on the CI life. The CI life is defined as the fatigue life for the initiation of a 0.01 inch crack.
Because it is very difficult to detect such a small crack, the following procedure will be used in this test program:
The test specimens will be cycled until a crack is automatically detected by an eddy current probe. A special eddy current probe was developed which was sensitive enough to detect a 0.01 inch crack. Then the test will be continued up to failure while measuring the crack propagation with an optical device (microscope). For a reference, a cracked section is shown, see Figure 13. The CI life is later determined by subtracting the crack growth (CG) life from the tested fatigue life. However, the CG life had first to be backtracked by analysis to an initial crack size of 0.01 inch.

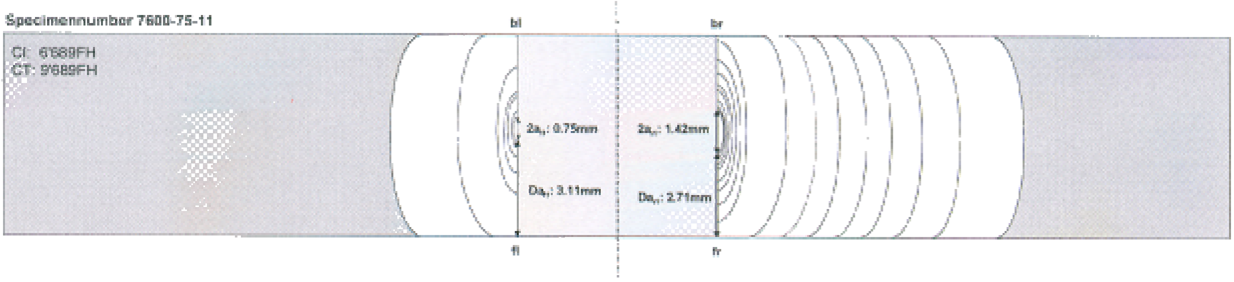

Figure 13: Crack growth result of coupon testing, material 7075 T7351

The test specimens are finite width specimens with a ¼ inch centered open hole (see Figure 14). In order to keep the number of specimens to a manageable size, a selection of parameters was required, resulting in the test matrix shown in Table 2.

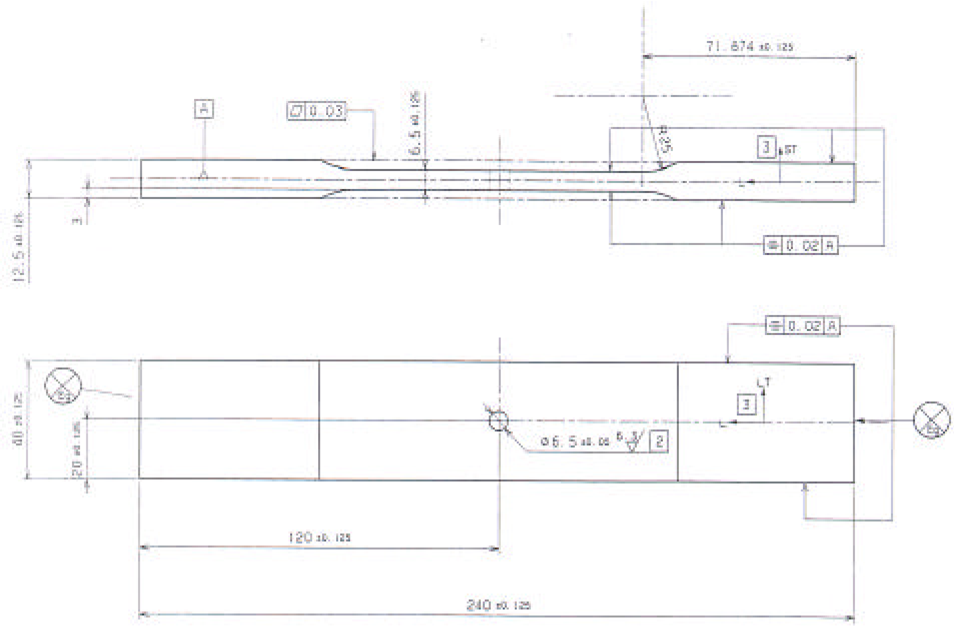

Figure 14: Coupon test specimen for MES testing

Type of Spectrum	Al 7075-T7351		Al 7050-T7451		Ti-6Al-4V (RA)	
	Stress for Lev 1 at 5‘000 FH	Stress for Lev 2 at 20‘000 FH	Stress for Lev 1 at 5‘000 FH	Stress for Lev 2 at 20‘000 FH	Stress for Lev 1 at 5‘000 FH	Stress for Lev 2 at 20‘000 FH
Wrbm Location 1 Design	3	3	3	3	3	3
Wrbm Location 1 Test	3	3	3	3	3	3
Cf544dor Location 4 Design	3	3	3	3	3	3
Cf544dor Location 4 Test	3	3	3	3	3	3

Table 2: Test matrix for MES validation; spectrum, stress level and number of specimens

It needs to mentioned that each location represents a typical major structural assembly of the F/A-18; i.e. location A with the wing root bending moment spectrum (WRBM) represents the inner wing and location B with the ctr fuselage bending moment spectrum (CFBM) represents the ctr fuselage. The coupon test program was finished in November 1999. The test data was analyzed in detail. The following conclusion could be made:

- The crack initiation analysis for the ASIP study seems to be on the conservative side (Kt = 3.24 was tested for an open hole).
- The following assumption in the ASIP study was made: $^{2}/_{3}$ o f the total life was crack initiation life up to 0.01 inch and the remaining $^{1}/_{3}$ of the total life was crack growth life. Our experimental results (tested crack initiation life and total life) confirmed the above assumption for all three materials for a crack geometry with an open hole. Using the crack growth code CG93 (Boeing crack growth software used for F/A-18 program) to back track from the total life, a life ratio of 1:1 compared to 2:1 was calculated between crack initiation and crack growth life. Figure 14 shows an example for titanium using the ctr section spectrum CF544. The theoretical back tracked result seems to be wrong because the stress intensity concept may not be valid for a small crack as 0.01 inch and furthermore the da/dN vs K data used in CG93 is very poor in the threshold regime.
- The experimental data of 7075-T7351 and 7050-T7451 confirmed the analytical results for the test MES quiet good. The Ti-6AL-4V (RA) showed more scattering in the life ratio of the test MES compared to the design MES. Some titanium specimen showed no crack initiation up to 50'000 flight hours where analysis would predict crack initiation life at 20'000 flight hours.

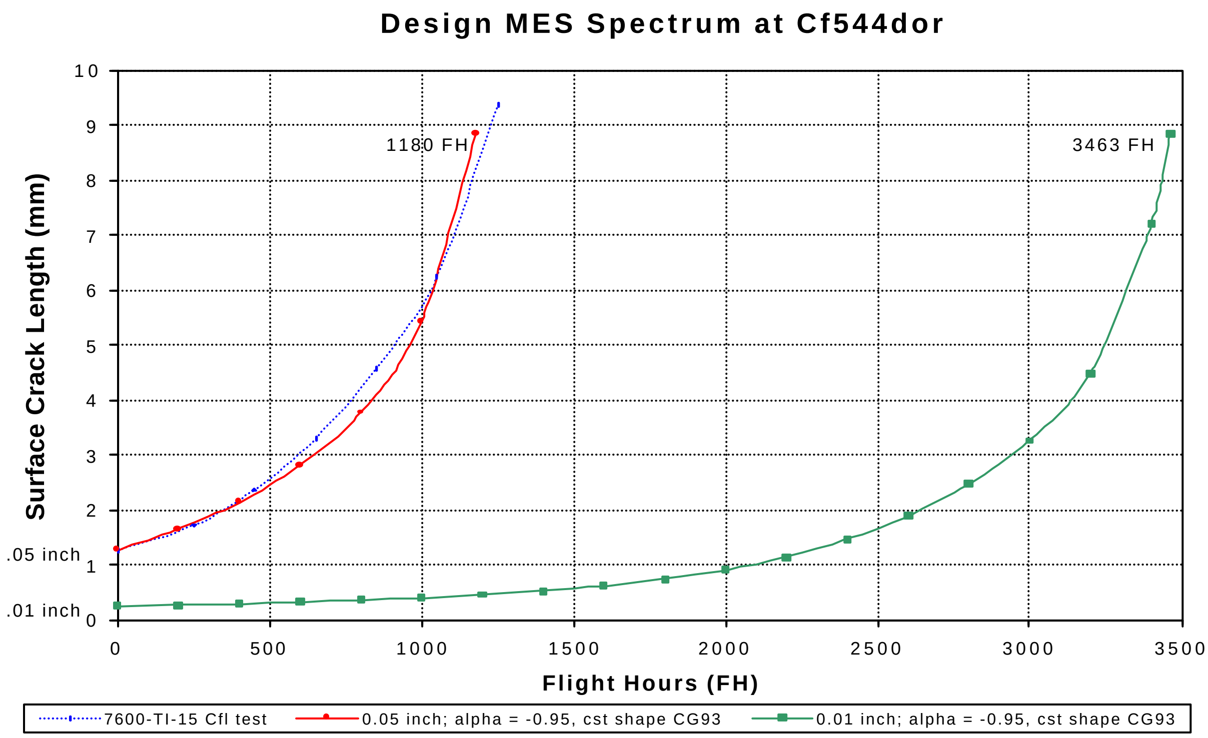

Figure 14: Crack growth curve for titanium at dorsal deck location cf544

4.6.4 FINITE ELEMENT ANAYSIS FOR THE SWISS F/A-18 FSFT

During the past three decades finite element analysis (FEA) has become an increasingly powerful tool for many technical or scientific applications. Computers, solvers and pre/post processing software have become more reliable, faster, and much more user friendly. FEA methods were employed in the development of the F/A-18 structure more than twenty years ago. Now SF is in the preliminary phase of an F/A-18 Full Scale Fatigue Test (FSFT) for the Swiss F/A-18 where FEA methods will again play a major role.
FEA is an important tool for the Swiss F/A-18 Full Scale Fatigue Test, see Figure 15. Finite element (FE) models generate internal loads that can be used for subsequent stress and fatigue analysis. One of the many advantages of FEA methods over hand calculated methods of years past is the determination of primary load paths within the structure as a function of the

relative stiffness of each of the structural components. All load carrying stringers, stiffeners, bulkheads, frames and skins will be reflected in the internal loads FE model. For a subsequent local stress analysis the stress analyst will have to decide, depending on the stress gradient, whether it would be necessary and beneficial to create a detail stress model with a finer mesh (more elements) or employ p-element formulation that would be more accurate for the location under consideration. SF's primary FEA tools for the Swiss F/A-18 FSFT are MSC/Patran/Nastran. This software package provides the full functionality of a state of the art FEA software package.

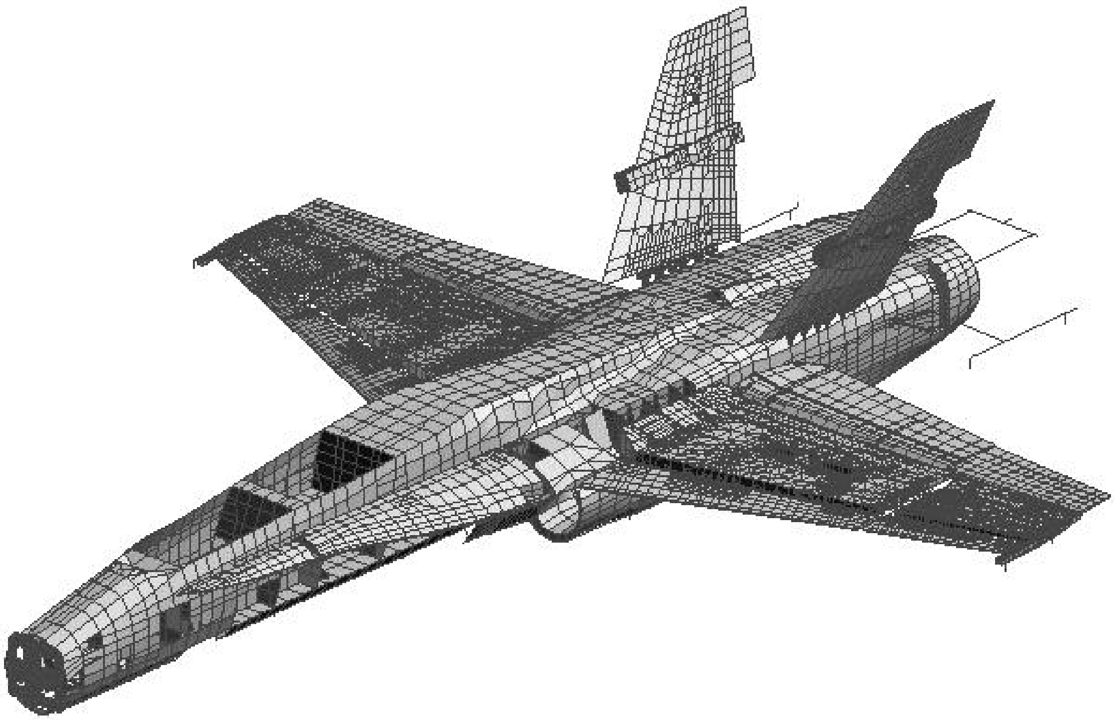

Figure 15: Full scale fatigue test FE model with dummies and contour board at the V-Tail

4.6.5 FINAL TEST CONCEPT

The selection of load introduction locations especially on the fuselage structure required major engineering investigations to understand the local impact of load distribution. SF will implement this analysis first on static load and stress assessment utilizing MSC/Patran/Nastran and then on a fatigue analysis which is based on the CI89 crack initiation software procured from Boeing St. Louis. For details of the development of the loading program and the load match requirements, see ref. [5].

In the test concept for the fuselage 20 jacks in z direction will be used to simulate the bending moment as well as possible. To introduce the high loads in the center section it was decided to use shear pads because local fittings would cause major problems at the local area. The shear pad technology will be approved in a special program prior to the FSFT fatigue cycling. To react, the side loads from the vertical tail, 4 jacks will be used, one in the aft splice fuselage section and one at the engine center of gravity and two on each inboard wing pylon.

The whole a/c will be restrained at the 3 landing gears (nose and main landing gear) in z (up) direction and at the engine dummy in x direction (flight direction), and at the arresting hook and the FS 286.5 fwd fuselage location for side direction.
For the free LEX two jacks in z direction will be used to simulate the bending moment and the shear force at the reference location FS395. The interface loads to the fuselage should be approximately matched. One jack will be used to simulate the net loads from the lower surface on the wing at the Fix LEX for each side.

The tension/compression pads will be used for the wing as load introduction. The pads will be grouped to 4, 6 or 8 and then connected with a stiff plate to a whiffle tree. For our test concept on each wing 14 jacks will be used, 5 for the inner wing box, 2 for the outer wing box, 2 at the wing tip, 1 for outboard leading edge flap, 1 for the trailing edge flap, and one for the aileron, 2 for the inboard leading edge flap. The load will be applied only to the lower surface by using push-pull actuators.

On the inboard wing pylon a x load introduction (flight direction) for drag simulation and a y load for side load introduction will be used. No real pylon loads will be simulated because 92.4% of the events only have stores at the wing tip (AIM-9) and 7.6% of the events have light bombs at the wing pylons.
For the vertical tail two jacks will be used with a horse collar to introduce the load. No real net-load will be simulated over the vertical tail surface. Only the bending moment and the torsion will be matched at the reference location F5 574.
For the horizontal tail a dummy structure will be used attached to the spindle. Two jacks will be used along the z direction to simulate the bending moment and the torsion at FS 652 location. One jack in x direction will be used to simulate the impact on the structure of the deflected position due to the maneuver.
The test hall at SF Emmen has no strong floor to react the test loads. Therefore structural steel floor will be used to react all the test loads. Figure 16 shows the test concept. Overall, 68 push-pull jacks will be used.
The final test concept was approved in Spring 2001 and is now under detail design of the loading rig and inspection platforms. The start of the construction will take place in August 2001 with the assembly of the structural floor. The actual planning is to start the test after commissioning by end of 2002.

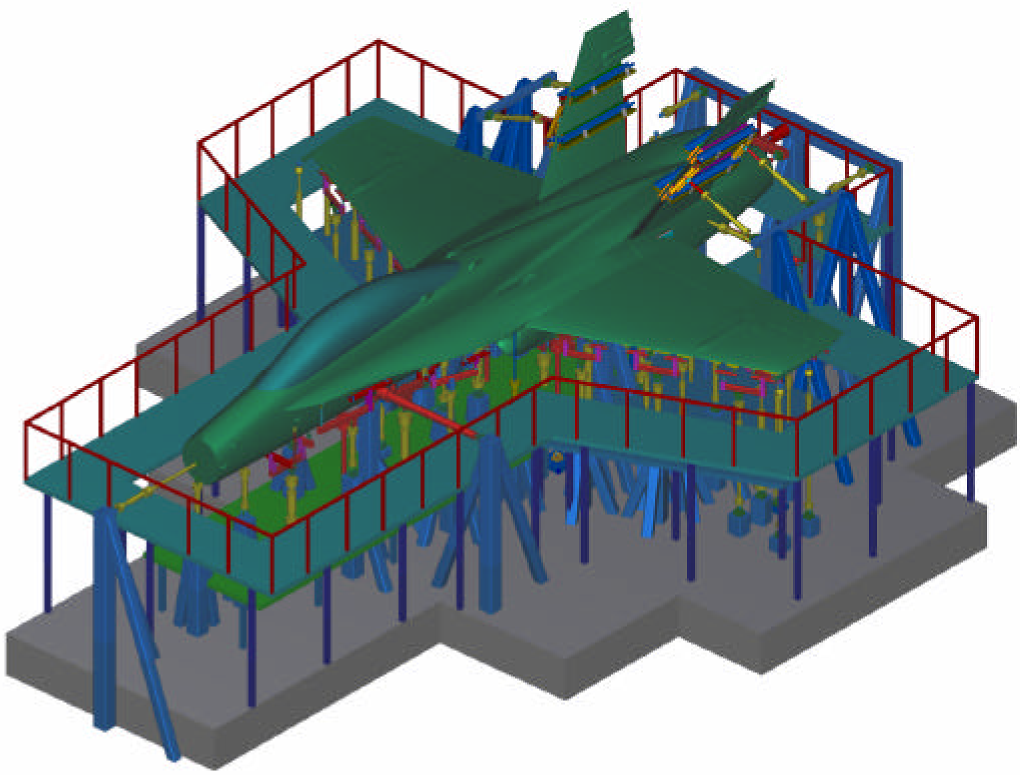

Figure 16: Test concept of the Swiss full scale fatigue test

4.7 A New Model to Predict Corrosion Fatigue Crack Growth in Aluminum Alloys (S. Michel, G. Soyka, EMPA Dübendorf and I. Kongshavn / RUAG Aerospace)

A new crack growth model has been proposed, which takes into account the effects of corrosion on crack growth in aluminum alloys in the near threshold regime [6]. The theoretical and experimental development of the model was presented in a poster at the ICAF Conference in 1997. The model has since been implemented in a software tool, '*Swiscrak2*', which uses a Weibull probability distribution to compute discrete crack advance through the oxide layer. The probability distribution functions have been determined from crack growth rate (da/dN) vs K_{eff} data in the near threshold regime, for coupons tested in air, nitrogen and vacuum. In a poster to be presented at ICAF 2001, predictions with the model are compared to experimentally determined crack growth lives of 7075-T651 center-holed coupons tested with a FALSTAFF (Fighter Aircraft Loading Standard for Fatigue Evaluation) sequence.

As described in [1], da/dN vs K_{eff} curves of aluminum alloys such as 2024-T351 and 7075-T651 tested in air and in nitrogen show a plateau-like region near the threshold regime (see Figure 17). As shown in Figure 18, this plateau-like region is not observed in coupons tested in a vacuum. It is proposed that this plateau is a result of oxide film fracture during the uploading part of a fatigue cycle, followed by the formation of a fresh metallic surface at the crack tip, which is immediately covered again by an oxide film. To verify this hypothesis, a new model that takes into account the formation and subsequent fracture of the oxide layer during crack growth has been proposed. The crack surfaces of 2024-T351 coupons tested in air, and 7075-T651 coupons in air, nitrogen and vacuum were first analyzed with the XPS method. Thickness, morphology and chemical composition of the oxide films were measured. Based on these measurements and a Weibull probability of failure, a crack growth model has been proposed for the quantitative description of these mechanisms. The predictive capabilities of the model have been compared to experimental crack growth lives of 7075-T651 coupons tested in air, nitrogen and vacuum under both constant and variable amplitude loading. The results indicate that for both loading types, the model accounts for the influence of the environment on crack growth. Under variable amplitude loading, however, an interaction occurs between crack retardation and oxide layer formation. This interaction is load level dependent, and is not fully accounted for by the model. As seen in Figure 19, a close prediction of the crack growth lives at a 100% and a 66.6% load level has been obtained for coupons tested in air. However, at the 50% load level, the predicted life is shorter than that of the test life, due to the influence of retardation on the formation and subsequent fracture of the oxide layer.

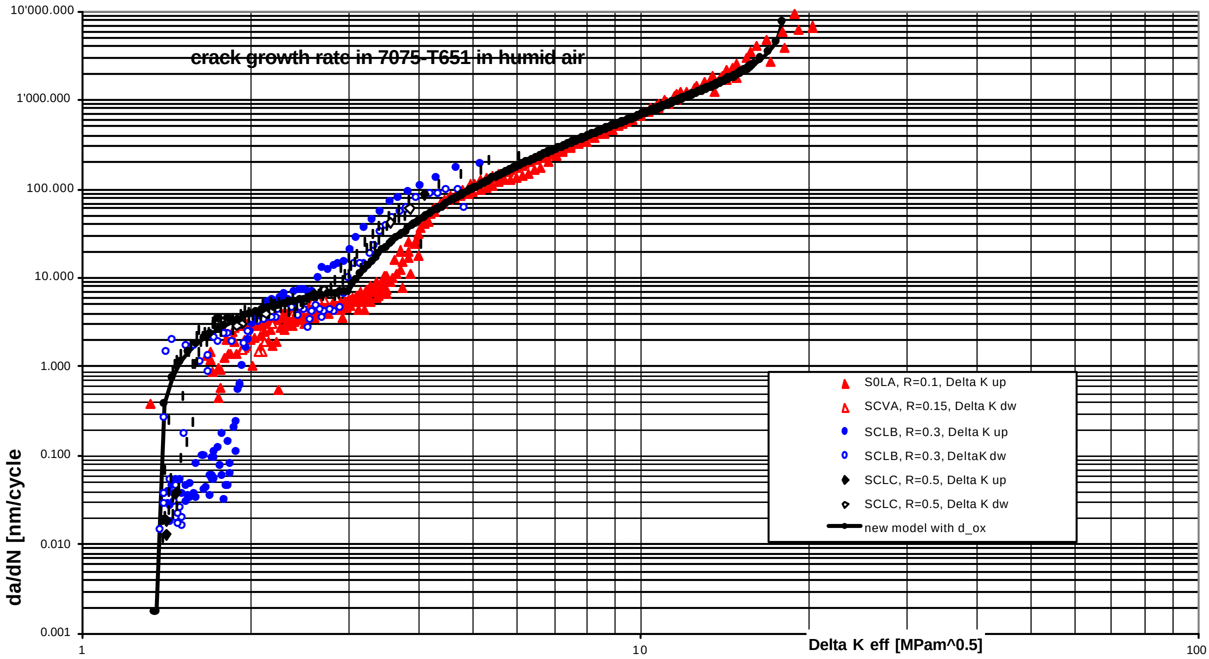

Figure 17: Crack growth rate (da/dN) vs K_{eff} plot of experimental data and the new model for 7075-T651 in humid air. The parameter *d_ox* accounts for the oxide film thickness in the crack propagation direction.

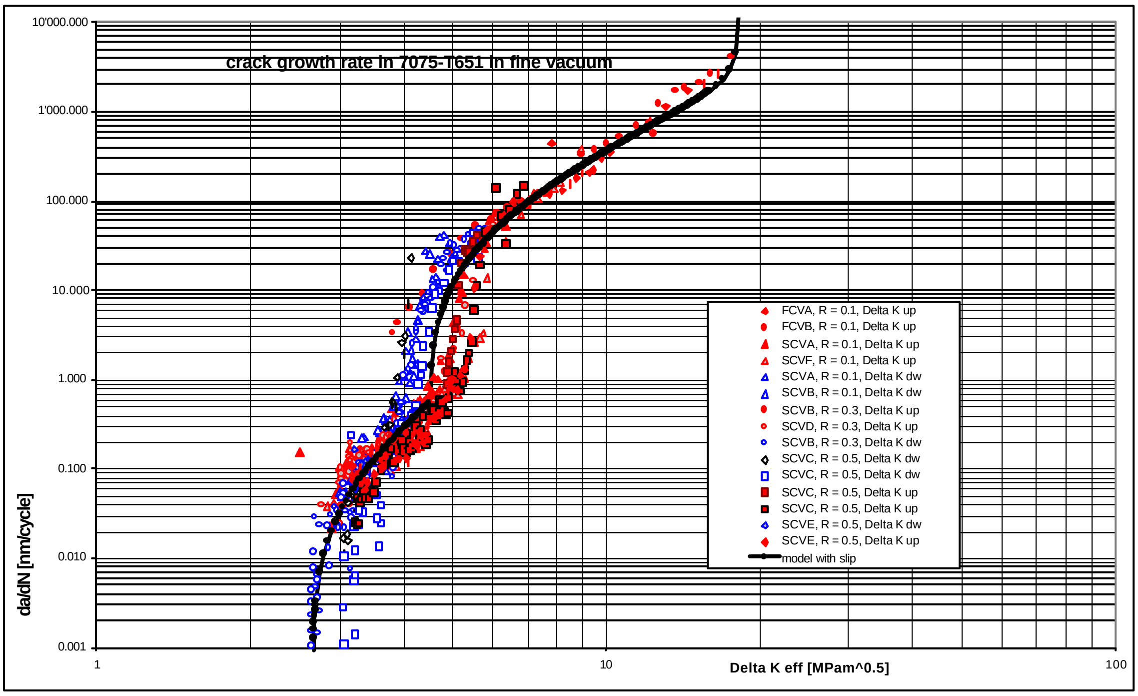

Figure 18: Crack growth rate (da/dN) vs ΔK_{eff} plot of experimental data and the new model for 7075-T651 in vacuum. In the near threshold regime, crack growth is a function of cyclic slip on a single plane.

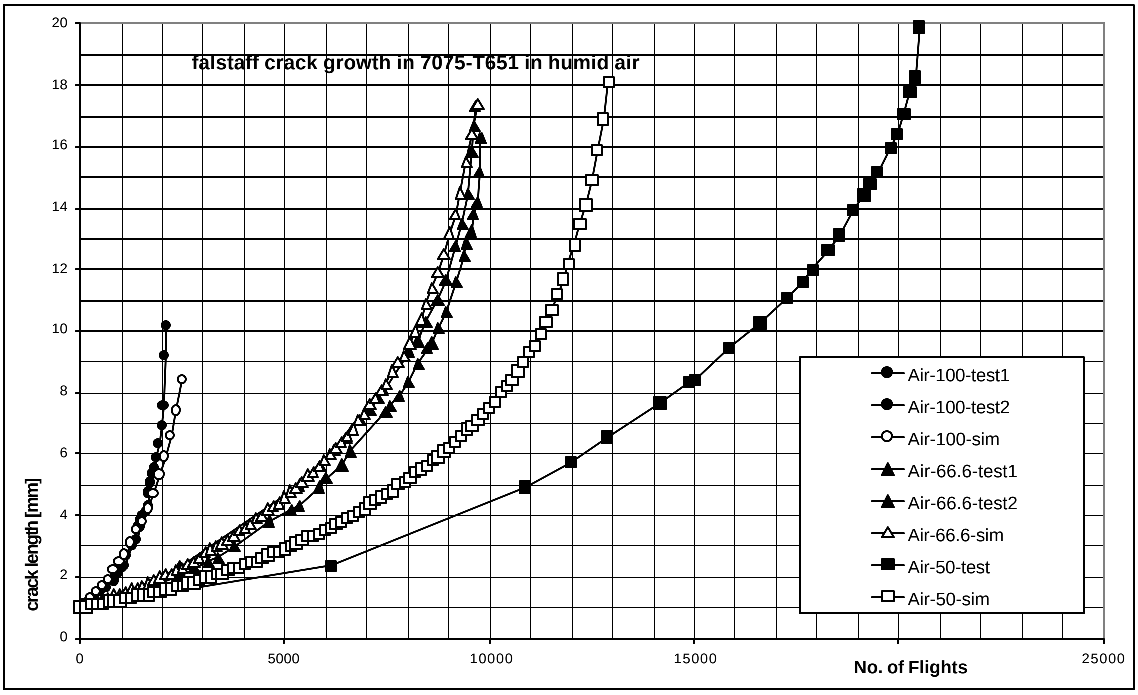

Figure 19: Test vs predicted crack growth lives for 7075-T651 coupons tested in air with the FALSTAFF spectrum. Three different load levels were applied, at 100%, 66.6% and 50% of the maximum load level in the spectrum.

4.8 REFERENCES

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