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Damage Tolerance Assessment of the Fighter Aircraft 37 Viggen Main Wing Attachment

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The Swedish fighter aircraft 37 Viggen, designed some 25 yr ago on a safe life basis, has been reassessed in terms of a damage tolerance evaluation. The aim was to ensure structural safety and to investigate the possibilities for extending the service life of the aircraft. This article deals with the reassessment of the newest wing attachment frame for the fighter version of the aircraft. The purpose of this article is to briefly show the extent and complexity of the assessment. Because of the original safe life design, resulting in rather high stresses, very extensive finite-element analyses were necessary in order to obtain accurate stress distributions and three-dimensional stress intensity factors in critical sections. Also, high demands have been placed on the accuracy of the crack growth predictions. Hence, extensive validation of the crack growth prediction technique was required. Structural testing, including artificial flaws, was carried out with the aim of obtaining crack growth data for correlation to the prediction technique. It is concluded that the state-of-the-art methodology used was successfully verified. Furthermore, damage tolerance of the considered parts were analytically proven and experimentally verified. Finally, extension of the original design life may be possible following further considerations.

Introduction

THE purpose of this article is to summarize some of the damage tolerance analyses and the testing performed on the main wing attachment frames of the Swedish fighter aircraft 37 Viggen (Fig. 1). There are four versions of the main wing attachment frame due to two different versions of the aircraft, attack (AJ) and fighter (JA), and due to major changes in the geometry made after serial production of several aircraft. In this article focus is on the latest version of the JA frame. Furthermore, the aim is to briefly show the complexity of a damage tolerance assessment in a case where the safe original life design has resulted in rather high stresses.

Geometry and Loading

The principal geometry of the JA main wing attachment frame is shown in Fig. 2. The frame can be described simply as an assemblage of two curved U- or I-shaped beams, located a distance of 160.0-mm apart and kept together by inner and outer cover sheets. The beams are made of an aluminum die forging, and the cover sheets are made of aluminum alloy sheet material. Between the two beams a structural detail, called the middle part, is located as indicated in Fig. 2. Depending on the aircraft version, the middle part is made of an aluminum die forging or a high-strength steel forging. Also, a lower rear engine attachment is mounted between the two beams (see Fig. 2).

The main wing attachment frame is loaded by the main wing spar. It is the only attachment frame that takes up the bending moment from the wings. Shear loads are distributed on several frames. The loads from the main wing spar are introduced into two attachment holes in each beam and two holes in the middle part. The purpose of the middle part is to distribute a portion of the load to two adjacent attachment holes in each beam, called middle part holes. Furthermore, the main wing attachment frame is loaded by the engine through the rear engine attachment, which transfers vertical loads. Finally, loads are introduced in the frame from the fuselage through the cover sheets. A principal sketch of the loads acting on a quarter of an isolated main wing attachment frame assemblage is shown in Fig. 3. The load, $P1z$, at the top of the frame is introduced in combination with extra loads on the engine attachment to compensate for the missing load transfer from the rest of the fuselage when an isolated assemblage is studied. Obviously, the magnitudes of the loads on the different versions of the main wing attachment frame are different.

Damage Tolerance Assessment

The approach taken has been to follow the military specification MIL-A-83444 issued by the USAF.[1] All of the components considered represent primary structures with few or no alternative load paths and have, therefore, been classified as slow crack growth structures. Initial flaw size assumptions follow the requirements of the specification.[1] A conservative residual strength requirement of 1.2 times limit load has been applied in terms of the linear elastic plane strain fracture toughness (K_{Ic}) irrespective of actual thicknesses, except for cover sheets where K_c values for actual thicknesses were used.

Load Spectrum and Stress Analyses

The load spectrum used for both analytical predictions and for the testing of some coupon test specimens and an isolated main wing attachment frame assemblage is shown in Fig. 4. As can be seen, the utilized design spectrum is more severe than the average usage spectrum of the aircraft.

The finite element (FE) analyses have been made in several steps where the initial step has been to make a global model of the complete component. These global models were able to describe stiffnesses correctly and to indicate where stress concentrations were located. In a few cases the global models were sufficiently detailed to give local stresses, but usually more detailed models were required. A typical example of a global model for a quarter of the main wing attachment frame assemblage (remaining after accounting for the symmetry) is shown in Fig. 5. The model, which is made using a substructuring technique, consists of approximately 75,000 degrees of freedom (DOF).

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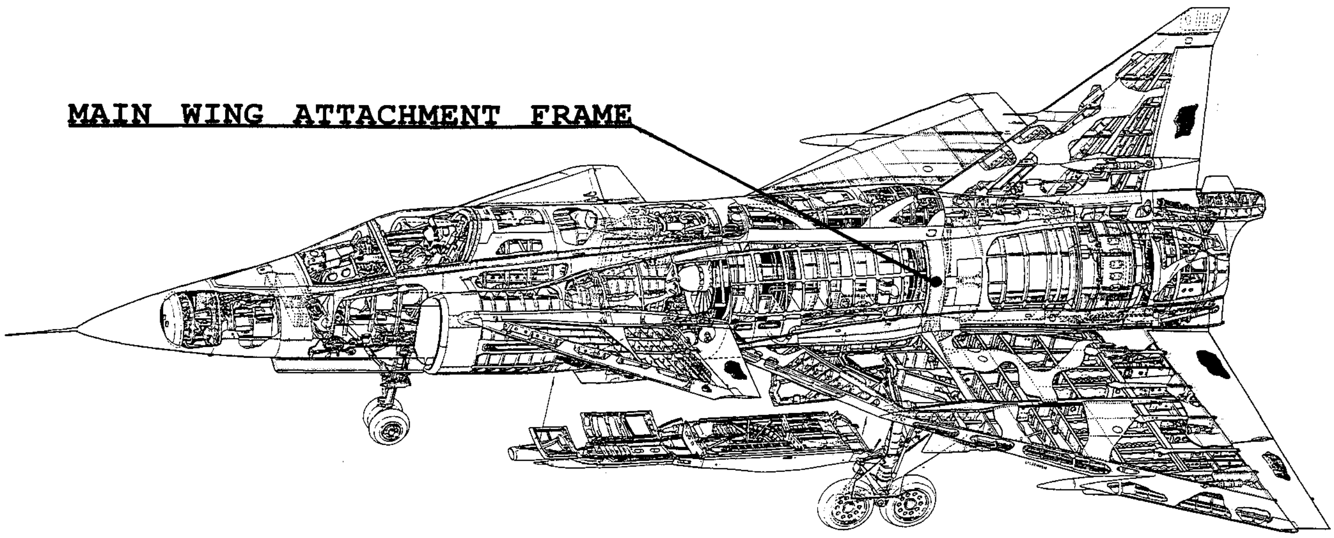


Fig. 1 Swedish fighter aircraft 37 Viggen.

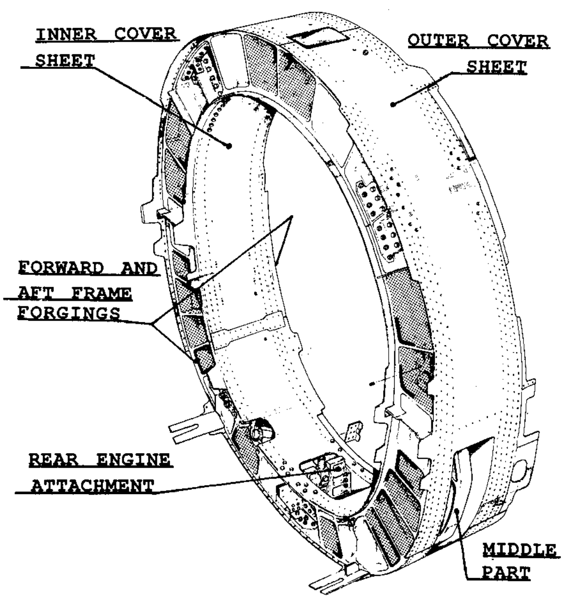


Fig. 2 Main wing attachment frame assemblage.

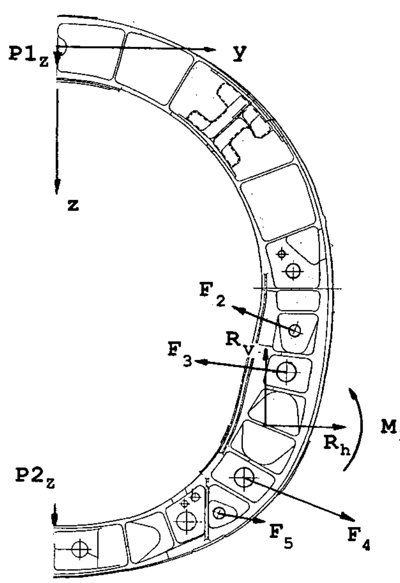


Fig. 3 Schematic showing the loads applied.

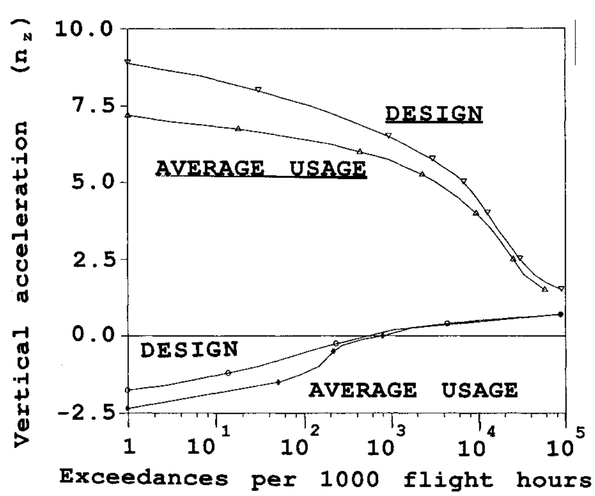


Fig. 4 Comparison between design spectrum and in-flight recorded spectrum for the average aircraft.

Based on the global FE results, critical areas were identified, which were then subjected to more detailed FE analyses, including such elements as accounting for bushings and solving contact problems. The boundary conditions for the detailed models were obtained from the global models in terms of displacements and rotations of nodes along the created cuts through the global models.

The region around the fuel pipe holes in the main wing attachment frame (see Fig. 6) was analyzed using the p-version of a newly developed self-adaptive FE code called STRIPE.[2] Also, some stress intensity factors were computed using this technique, which is believed to be the most accurate technique there is for such calculations.

Regions around major attachment holes were analyzed using conventional FE technique but, as already mentioned, contact stresses between wing bolts and bushings, as well as between bushings and frame forgings, were computed using an automatic iterative solution technique. In many cases the detailed analyses were repeated with considerations of more and more details, such as heads of bolts and stiffening effects of middle parts. The reasons for repeating the detailed analyses were seemingly improper or too large deformations and unrealistically high local stresses. For example, by including the bolt head in the model of the middle part hole region (see Fig. 7), the largest principal stress was reduced by 25% due to reduced bolt tilting.

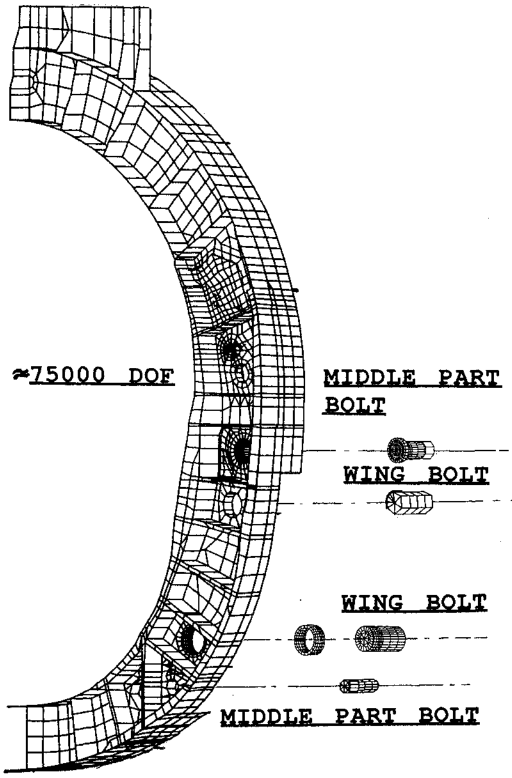


Fig. 5 FE model of the main wing attachment frame assemblage. Due to symmetry only one-fourth of the real structure is modeled.

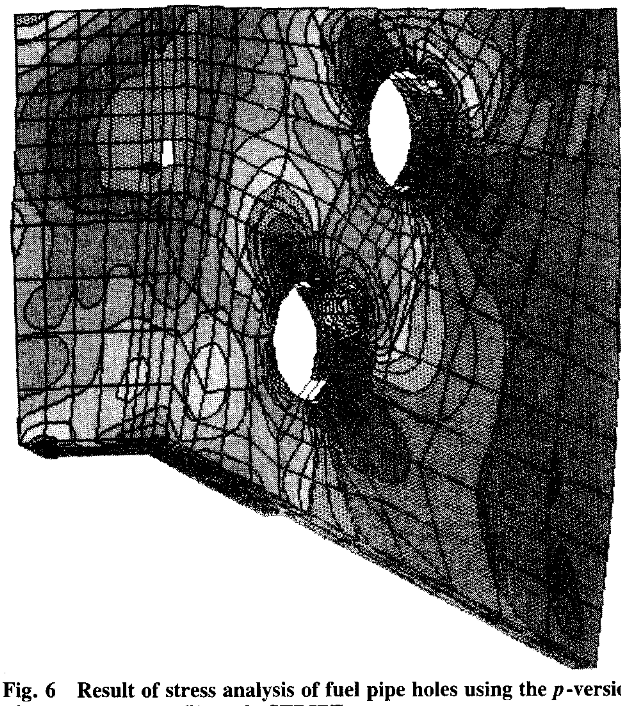


Fig. 6 Result of stress analysis of fuel pipe holes using the *p*-version of the self adaptive FE-code STRIPE.

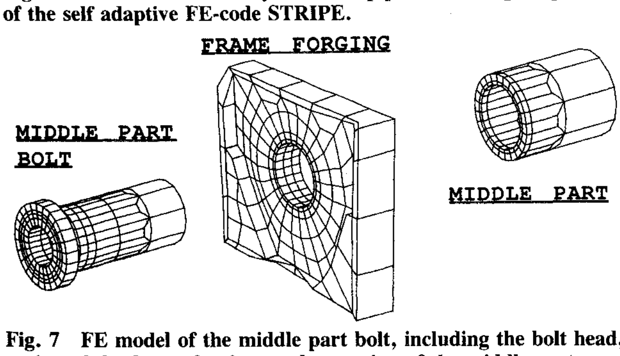


Fig. 7 FE model of the middle part bolt, including the bolt head, a region of the frame forging, and a portion of the middle part.

Fracture Mechanics and Fatigue Crack Propagation

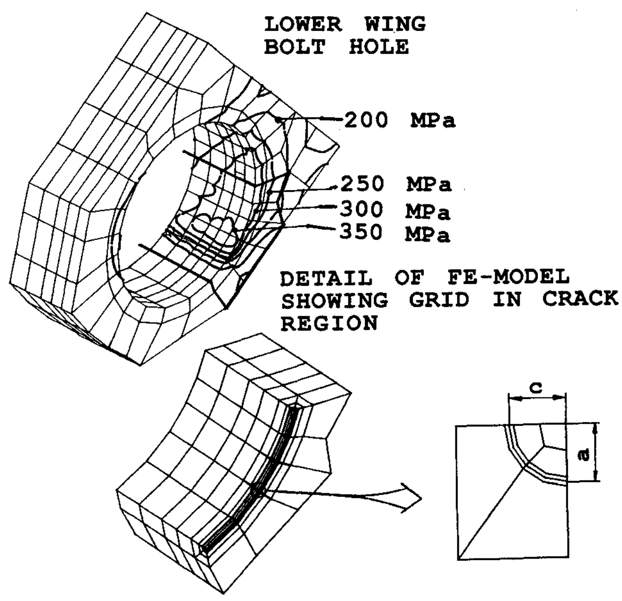
A difficulty in the damage tolerance analysis has been to obtain relevant stress intensity factors. In many cases the stresses at critical locations have been high (locally) with large gradients in both surface and thickness directions. Stress intensity factor solutions, based upon a remotely applied uniform loading, are generally not able to describe the local stress gradients. Besides, it is very difficult to define the remote uniform loading.

The most accurate stress intensity factors used were those computed using the adaptive FE technique (Fig. 8). However, there were too many critical locations for applying this technique everywhere. Second-best stress intensity factor solutions seemed to be those based upon the weight function technique, in which case local stress distributions could be described accurately. Such stress distributions were obtained directly from the FE analyses and they are valid as long as no major changes due to load redistributions occur. However, weight functions for three-dimensional geometries are scarce. For two-dimensional geometries it is easier to find accurate weight functions, but through-the-thickness cracks were, in general, a far too severe assumption for the critical locations studied.

The solution to the problem was to introduce correction functions based on the two-dimensinoal weight function solutions such that the three-dimensional solutions for remote loading were modified to approximately account for the local stress distribution in one of the two directions.

Fatigue crack growth predictions were performed using a cycle-by-cycle analysis technique without consideration of plasticity-induced load interaction effects. However, a method for extracting contributing load cycles from an irregular load sequence was used which can be described as a very simplified rain flow counting algorithm.

All available fracture mechanics and crack growth data for the aluminum alloys and the high strength steels involved, have been critically reviewed and collected in a data base for easy access. Complementary testing on coupon specimen level was performed to derive constant amplitude fatigue crack growth data in cases where data were lacking. Figure 9 illus-



STRIPE RESULTS, UNIFORM p=6			
a (mm)	c (mm)	K_{Ia} (MPa $\sqrt{mm}$)	K_{Ic} (MPa $\sqrt{mm}$)
1.3	1.3	472.8	572.5
1.3	1.6	514.6	602.9
2.0	2.6	621.9	749.2
2.0	3.0	647.4	778.8

Fig. 8 K_I determination using the *p*-version of STRIPE.

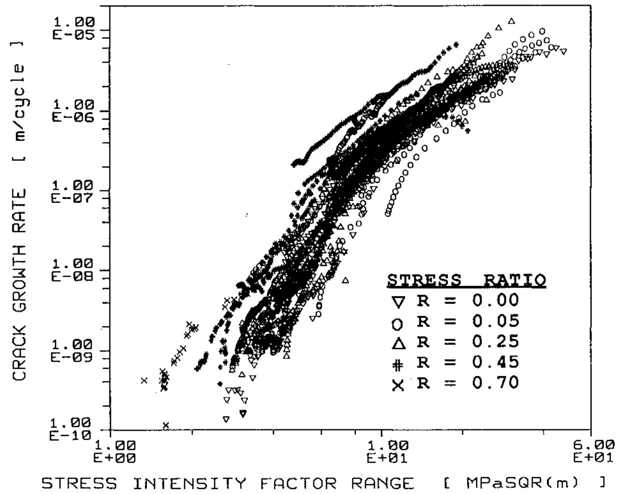


Fig. 9 Fatigue crack growth rate data for the 7075 type of aluminum alloy used in frame forgings.

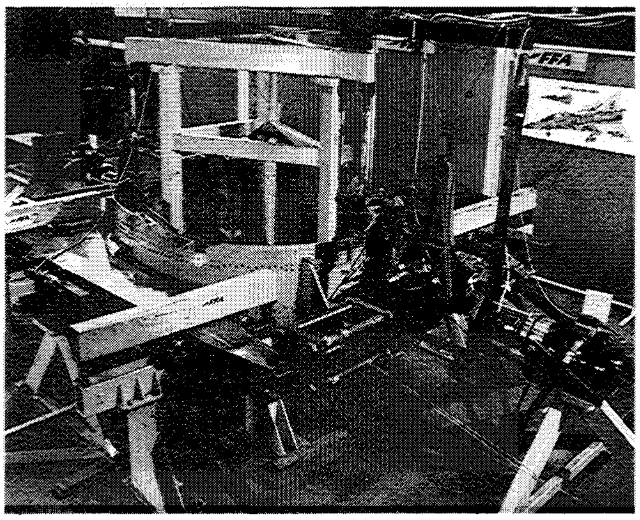


Fig. 10 Test setup for damage tolerance testing of an isolated main wing attachment frame assemblage.

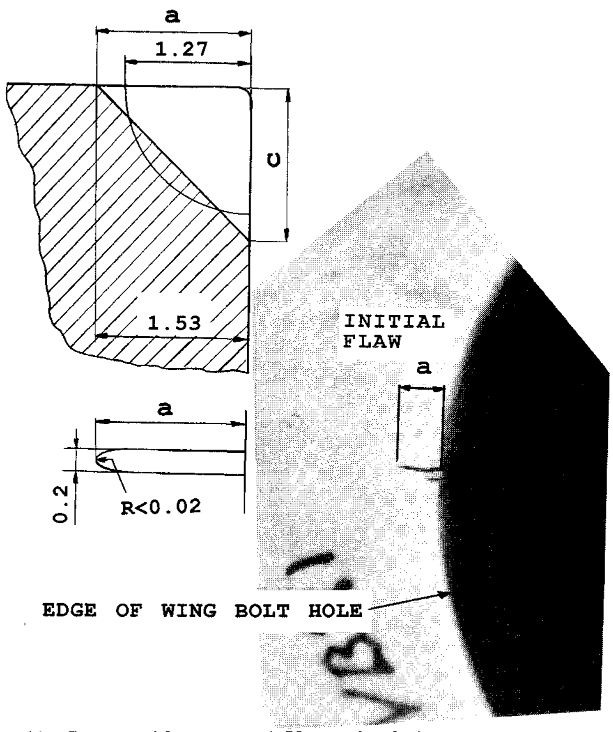


Fig. 11 Sawcut with $a = c = 1.53$ mm simulating a quarter-elliptical initial flaw with $a = c = 1.27$ mm.

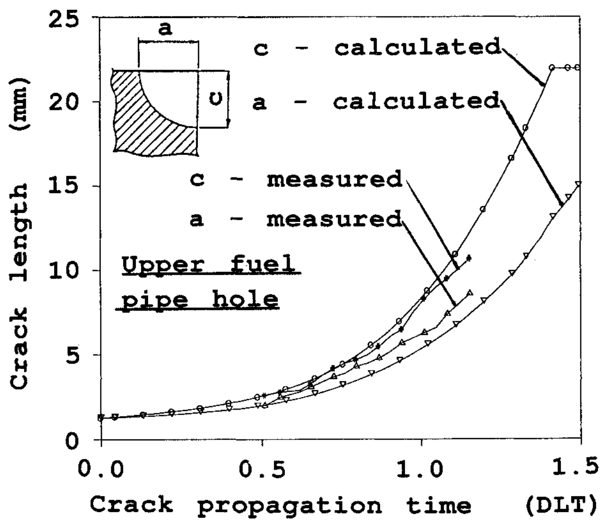


Fig. 12 Comparison between predicted and measured crack size as function of time (DLT = design life time).

trates the collected data for one of the relevant aluminum alloys.

Verification of Stresses and Crack Growth Prediction Technique

Initially, the FE stress analyses were verified by comparisons to experimental results obtained during the traditional static and fatigue testings of the components considered. The comparisons showed that stresses, in general, were computed very accurately even with the global models. Unfortunately, stresses really close to attachment holes could not be assessed due to the lack of experimental data.

Next, the technique of modifying three-dimensional stress intensity factors, using two-dimensional weight functions, was verified by comparing such stress intensity factor solutions to solutions obtained using the adaptive FE technique for a number of typical geometries.

Finally, the computer program LIFE,[3] used for the crack growth predictions, was verified as far as possible by comparisons to other computer programs (CRACKS IV, ESACRACK, EFFGRO, etc.) and to experimental results (on a coupon specimen level with well-known stress intensity factor solutions) from the literature as well as from test results (on a coupon specimen level) obtained in the current investigation. These comparisons involved both constant amplitude results and results from different types of spectrum loading.

Structural Testing

The main wing attachment frame assemblage mounted in the test rig is shown in Fig. 10. Due to relatively large displacements (wing root bending moments of up to 520 kNm were applied) the effective test frequency became around 0.2 Hz (testing was conducted at a constant displacement rate). The actual frame tested had already been subjected to four fatigue lives of spectrum loading during the traditional testing to support the original safe life design of 2800 flight hours. No fatigue cracks were reported after the traditional testing. In the present investigation another four fatigue lives were applied subsequent to introduction of defects in the frame. A total of 22 artificial flaws (crack tip radius <0.02 mm) (see example in Fig. 11), were introduced by sawing in critical locations of the frame.

The locations chosen for saw cuts were selected based on the FE results, a strain survey using a brittle coating technique, crack growth predictions, and inspection considerations. Very good agreement was found between the brittle coating results and the global finite-element analysis with respect to both critical locations and magnitudes of the strain field.

Crack growth was monitored both visually and by means of various NDI-techniques, primarily eddy current. First, cracks started to grow from defects introduced at the two fuel pipe holes. This growth was easily observed visually, and was therefore very useful in verifying the analytical crack growth predictions. A comparison of predicted and experimentally observed crack growth at the upper fuel pipe hole is shown in Fig. 12. Later on, after approximately 12,500 simulated flight hours, this crack—together with a crack at the lower fuel pipe hole—resulted in final failure of the frame, as shown in Fig. 13.

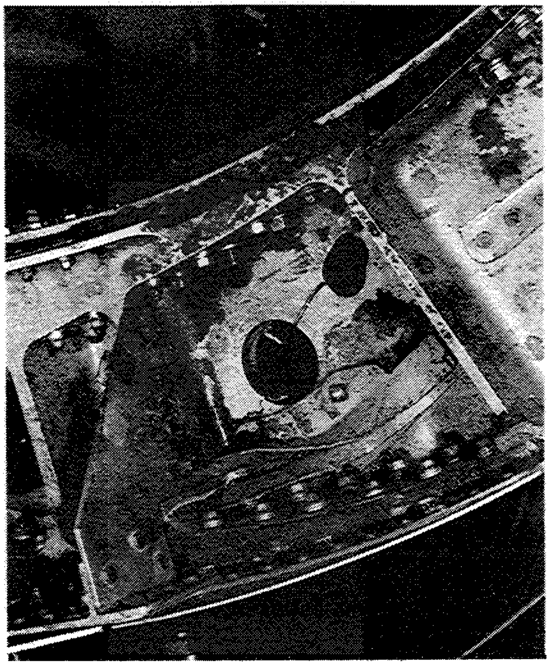


Fig. 13 Final failure at the fuel pipe holes.

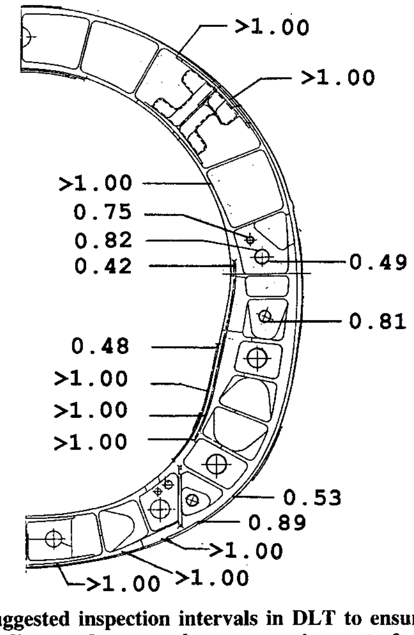


Fig. 14 Suggested inspection intervals in DLT to ensure structural safety according to damage tolerance requirements for slow crack growth structures.

After some 4000 simulated flight hours, crack growth was observed at one of the wing bolt holes. This is, according to numerical predictions, one of the most critical locations in the frame. However, the predictions for this location are assumed to be overly conservative as friction between the interference fitted bushing and the frame forging was disregarded. Furthermore, it is believed that the interference fitted bushing in the wing bolt hole restrains the crack mouth opening, and thereby, reduces the effective stress intensity factor range. This would explain the unexpectedly slow crack growth observed at this location.

Inspection Intervals

Based primarily upon crack growth predictions, but with due consideration to component testing results and also results concerning the prediction capability (accuracy for various structural details) of the computer program used, safe periods of crack growth have been established. Inspection intervals have been determined, assuming that the critical locations are depot or base level inspectable. The inspection intervals are then half of the safe crack growth periods. Figure 14 shows inspection intervals for the JA main wing attachment frame.

Conclusions

Based upon extensive damage tolerance analyses complemented by a damage tolerance testing of an isolated main wing attachment frame assemblage, and supported by fracture mechanics data testing and traditional fatigue testing, the damage tolerance assessment for the newest JA main wing attachment frame can be summarized as follows:

1) The state of the art methodology used has been successfully verified.

2) Damage tolerance has been analytically proven and experimentally verified.

3) Extension of the original design life may be possible following further considerations.

The final results do verify the structural integrity of the considered parts. It is of particular interest to notice that the analytical work carried out in the present investigation is much more refined than the work needed to verify the damage tolerance of the new Swedish fighter aircraft JAS 39 Griffin. The reason is that the latter aircraft is designed according to damage tolerance requirements, leading to significantly lower stress levels. Hence, first-order conservative estimates of stress levels and stress intensity factors are normally adequate to verify the design life.

Acknowledgments

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