A REVIEW OF AERONAUTICAL FATIGUE INVESTIGATIONS IN SWEDEN DURING THE PERIOD JUNE 2001 TO APRIL 2003

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CONTENTS

3.1 INTRODUCTION

In this paper a review is given of the work carried out in Sweden in the area of aeronautical fatigue during the period June 2001 to April 2003. The review includes aircraft loading actions basic studies of fatigue development in metals and composites, stress analysis and fracture mechanics, studies of crack propagation and residual strength, testing of joints and full-scale structures, and fatigue life predictions. A reference list of relevant papers issued during the period covered by the review is included.

Contributions to the present review are from the following sources:

– The Swedish Defence Research Agency (FOI), Aeronautics Division, FFA
  Sections 3.2.2, 3.2.3, 3.3.1, 3.3.2, 3.3.3, 3.3.4, 3.5.1, 3.5.2, 3.6.1, 3.6.2, 3.6.3, 3.6.4, 3.6.5, 3.7.1, and 3.7.2

– The SAAB Company
  Sections 3.2.1, 3.4.1, 3.4.2, 3.4.3, 3.4.4, 3.4.5, and 3.8.1

3.2 FATIGUE LIFE TESTING AND PREDICTION

3.2.1 Fatigue Life of High Speed Machined Surfaces

High speed machining (HSM) has been evaluated regarding its influence on fatigue strength of plate Al7010-T74. After having been demonstrated earlier for low Kt-values, it has now been showed also with specimens with Kt=2.5, that HSM causes less fatigue resistant surfaces than Conventional machining does. The cutting speeds in the "transition range" were tested as well as two high cutting speeds (>1000 m/min). Spectrum loading with MiniTwist was used and the peak stress was 240 MPa in all tests, see Figure 1. For Conventional milling a high speed steel tool was used and for the HSM milling a tool with carbide inserts. At the cutting speed 400 m/min Conventionally and HSM made specimens received similar spectrum life.

3.2.2 Definition of Shear Amplitude in Multiaxial Fatigue

In many failure criteria proposed in the literature for multiaxial fatigue loading of metals, the shear amplitude, defined in stress or strain, is the most important parameter. That means that the parameter has the largest impact on fatigue life and also that the parameter in some criteria is defines the critical plane. For a multiaxial load where all load components are varying in-phase, the shear stress varies along a line in all possible planes making the definition of shear amplitude trivial. But, if two or more load components are varying out-of-phase the shear stress will describe a two-dimensional path during one cycle. The definition of shear stress amplitude is no longer obvious and the choice of definition will affect the deduced fatigue life.

In Reference [1] five definitions of shear stress amplitude was described and compared. The definitions have earlier been proposed in the literature. The comparison was made using both numerical examples and experimental results. Two failure criteria were used for the experimental comparison. Only constant amplitude loading was considered not to involve the procedure of cycle counting.

It was found that the different definitions of shear amplitude could give differences in shear amplitude of up to 63 % for various shear stress paths. When the shear amplitude is used in a criterion, the choice of definitions of shear amplitude will therefore have a large impact on the deduced fatigue life. It was not possible to recommend a certain definition since that is probably material dependent. The choice of definition of shear amplitude has to be based on experiments that discriminates between the definitions, a choice that should be made together with the choice of criterion. For general waveforms, including random loading that is not discussed here, it is important that the definition has a consistent dependence on the amount of two-dimensional deviation from a linear shear stress path.

3.2.3 Automatic Crack Detection Using Optical Techniques

In fatigue testing the number of cycles up to crack initiation is an important parameter. After initiation it is desired to measure the crack propagation rate. The technique for detecting initiation should have a high accuracy regarding the number of fatigue cycles and during the propagation phase an accurate measure of the crack length is needed. It is also of advantage if the technique can be automatized since that for example would enable fatigue testing outside working hours. Various techniques to solve these measurement tasks have been proposed. Anyhow there is still a need to improve the measurement techniques.

In reference [2] two new optical methods that detects crack initiation and monitors crack propagation have been tested on notched aluminium specimens. The objective of the study has been to examine how well the methods work and to get experience needed to develop a fully automatic system in the future.

The idea of the method of laser speckle decorrelation is to illuminate the specimen surface by laser light and look for decorrelation of laser speckle patterns. Decorrelation will occur due to plasticity as for example around a crack tip. This method manage to determine the crack tip position with an accuracy of about 5 pixels that in the tests corresponded to a distance in the range of 0.02 - 0.05 mm. It also manages to detect an initiating crack at an early stage. But if the crack initiates inside the specimen surface it is difficult to know when the crack reaches the surface. Other disadvantages are that on curved surfaces it is difficult to achieve homogeneous illumination and that the method gets problems when many cracks propagate simultaneously. Figure 2 shows the growth of a fatigue crack that has initiated near the centre of a notch.

In the second method the specimen is illuminated by white light. The images of the surface roughness are analysed using digital speckle photography (DSP) algorithms to get the in-plane displacement field over the surface. A discontinuity in the displacement field indicates a crack. From the displacement field one also obtain a measure of the crack opening. The DSP method where crack tip positions are evaluated from displacement fields has an accuracy of about 3-5 pixels, which is slightly better than the other method. With this method it is easier to get a homogeneous illumination on curved surfaces such as a notch. The method also manages to analyse complicated crack patterns where many cracks appear simultaneously, this is illustrated in Figure 3.

3.3 JOINTS

3.3.1 Load Transfer and Secondary Bending in Two Mechanical Joints

Standard titanium Hi-lok fasteners have been instrumented with strain gauges for measurement of their axial load and their shear load, Ref. [3]. The fasteners having a diameter of 6 mm were of two types, countersunk and with protruding heads, see Figure 4. Also, two different designs of instrumentation were employed, internal and external strain gauges. The instrumented fasteners were calibrated and subsequently two of them were used to investigate the load transfer in two standard joints.

The purposes of the investigation were to evaluate the performance of the instrumented fasteners and to study the load transfer and secondary bending in two standard joints.

The calibration showed that the instrumented fasteners responded linearly with respect to axial loading and that a power function could represent the response to shear loading in a single shear lap joint.

The secondary bending and load transfer in a two column by two row lap joint was studied. The lap joint was made of 3 mm thick sheets of the aluminium alloy 2024-T3. The lap joint was heavily instrumented with strain gauges in order to measure the applied load, the secondary bending and the total load transfer. It was found that the total load transfer, at maximum applied load, was 54.5 % in the first fastener row and that the fastener load transfer was 35 % of the maximum load. Unfortunately the secondary bending could not be obtained due to failure of the strain recording system.

The second standard joint was the so called 1 ½ dogbone joint, see Figure 5. The joint had one fastener row and two fastener columns. The same sheet material was used for this joint as for the lap joint and the joint was provided with strain gauges for the same type of measurements as the lap joint. The local secondary bending ratio was found to be different for the left and the right hand sides of the fastener, se Figure 6. The total load transfer at maximum applied load was found to be 38.7 % of the applied load whereas the fastener load transfer only amounted to about 10 % of the applied load. The fastener load transfer ratio is shown in Figure 7.

For both joints the fasteners were installed using standard Hi-lok collars which for the instrumented fasteners sheared off at a installation torque between 6.9 Nm and 7.5 Nm corresponding to an axial clamping force of 7.5 kN and 11.3 kN, respectively. These rather high clamping forces results in a substantial friction between the mating surfaces which is evident from the load transfer measurement results, see Figure 8.

3.3.2 Mechanically Fastened Joints: Critical Testing of Single Overlap Joints

Three single shear overlap joints were subjected to spectrum loading using the FALSTAFF load sequence, Ref. [4]. The lap joints, having two by two fastener rows and columns respectively, were assembled using two instrumented Hi-lok fasteners (CSK1 and CSK2) and two ordinary Hi-lok fasteners. The material used for the plates was the aluminium alloy 7475-T761 in 3 mm thickness. The titanium fasteners had a diameter of 6 mm. Standard Hi-lok collars were used and tightened to their maximum torque. The total width of the test specimens was 50 mm, see Figure 9.

The spectrum fatigue testing was carried out in a servo-hydraulic testing machine having hydraulic grips. The grip displacement, corresponding to the peak to peak elongation of the specimen during one load cycle, was recorded after different number of passes through the load sequence. A drastic decrease in the grip displacement was found to take place during the first stage of spectrum fatigue loading. The grip displacement stabilized after about 20 passes through the spectrum, see Figure 10. A large part of the initial grip displacement comes from the relative movement between the plates. Thus, the drop in grip displacement range indicates a reduction in the relative movement between the plates caused by increased friction.

At the beginning of the spectrum loading a thin oxide layer covers the plate surfaces. This oxide layer suffers from degradation due to the mechanical wear process according to Szolwinski and Farris. This leads to the removal of the protecting oxide layer and the origin metal of the surfaces in contact starts to adhere. This process results in the initial accumulation of wear debris between the aluminium plates which increases the coefficient of friction.

The change of the absolute axial fastener load during the fatigue loading was not possible to obtain with the present test set up. However, the variation during a single load cycle and how this variation changed during the fatigue loading was studied. After 20 repeats of the FALSTAFF sequence the variation in axial load during a single load cycle had dropped to 20 % of the initial value obtained after one pass through the load sequence.

Also, the change of the absolute fastener shear load during the fatigue loading was not possible to obtain with the current test set up. The variation in fastener shear load during a single load cycle was measured at predetermined intervals., before the start of spectrum loading, after one pass, after two passes, after five passes, after 20 passes, after 60 passes and finally after 95 passes through the load sequence. The result, for the fastener CSK1, is shown in Figure 11 as the fastener load transfer ratio versus the number of passes through the load sequence. Because of the logarithmic scale the number of passes through the load sequence was increased by one. Thus, 1 pass actually corresponds to the situation before the spectrum loading was started. The fastener load ratio is defined as the ratio between the fastener shear load and half the total load applied to the joint. Only one half of the total load applied to the joint is used for the calculation of the fastener load ratio since there are two fastener columns.

Before the spectrum loading started the fastener load transfer ratios for the two instrumented fasteners CSK1 and CSK2 were about 47.5 % and 42.5 %, respectively. The fastener CSK1 was installed in fastener row number one while the fastener CSK2 was installed in fastener row number 2. After 20 passes through the load sequence the fastener load ratios had dropped to 10.8 % and 7.6 %, respectively.

A new measurement technique, with respect to lap joints, the digital speckle photography (DSP) was employed in this investigation. The idea was to measure the fastener translations and rotations during cyclic loading. It was found that the secondary bending had a relatively large influence on the measured fastener movement. However, despite the difficulties it was clear that the fastener movements were reduced as a result of the fatigue loading.

3.3.3 Fatigue Life Prediction of Composite Bolted Joints with Bolt Failure

Composite structures are often joined with bolted joints. One common failure mode in fatigue of composite bolted joints is bolt failure. Double lap bolted joints with six fasteners were constant amplitude fatigued at R=-1. The six bolt joints were FE-modeled and the stresses in the fasteners were calculated. Double lap bolted joints with two fasteners were constant amplitude fatigued at R=-0.2. Some of the joints were subjected to repeated overloads. The joints, which were subjected to repeated overloads, had a longer fatigue life than the joints, which were constant amplitude loaded. This behavior is typical for metal fatigue and shows that the fatigue life of the joints is determined by the fatigue life of the fasteners. The average peak tensile stress in the bolts was plotted against the fatigue life of the joints and results for joints with 4 mm and 6 mm bolts collapsed into one scatter band, see Fig. 12. From this it is possible to predict the fatigue life of bolted joints, Ref. [5].

3.3.4 Fatigue of Joints in Composite Structures

In Ref. [6] a review of fatigue behaviour in composite structures is given. Below, a short summary of conclusions for fatigue in adhesives and fatigue behaviour of bolted joints is presented.

During fatigue loading of adhesive joints one major crack usually propagates until failure occurs. The crack can propagate in the adhesive, at the interface between the adhesive and the adherend, or in the adherend as a delamination. The crack is usually modeled with fracture mechanics and a modified Paris law is used to describe the crack growth rate. The crack growth rate is often assumed to depend on the maximum energy release rate or the change in energy release rate. The fatigue life can be estimated by integrating the crack growth rate. Since the exponent in Paris law is large the fatigue life is sensitive to the applied load. Therefore, the threshold approach is often used in applications. The joint is designed such that no crack growth will occur.

There are four major failure modes in fatigue of bolted joints, hole elongation, bolt failure, net-section, and shear out. In general an increased clamping pressure will increase the fatigue life by increasing the amount of load transferred by friction between the plates. Thereby reducing the stresses in the bolts and in the composite close to the bolt hole. During hole elongation the bolt hole is elongated by wear between the bolt shank and the composite hole surface. At the beginning of fatigue loading the hole is elongated slowly and close to failure the hole elongation increases rapidly. Countersunk fasteners will reduce the fatigue threshold compared to protruding head fasteners. Bolt failure can be characterized as metal fatigue failure. The fatigue life of joints with bolt failure can be estimated by calculating the opening stress in the bolts and comparing it with a metal fatigue curve. An increased joint thickness of thin joints promotes bolt failure. Countersunk fasteners have a shorter fatigue life than protruding head fasteners. Net-section and shear out failure have been observed at high load levels for joints, which fail due to those failure modes quasi-statically.

3.4 STRUCTURAL EVALUATION

3.4.1 The Saab 340 Full Scale Fatigue Test

General

The Design Service Goal (DSG) for Saab 340 is presently 90000 flights or 45000 flight hours whichever occurs first. However, an authorized Service Life Extension Program allows the operators to extend the utilization to 60000 flight hours (after embodiment of subject Service Bulletins)

The Saab 340 Full Scale Fatigue test has been tested for 204000 simulated flights. Thereby, the cyclic testing of the airframe is completed. The first 180000 flights of the testing finalized the fatigue phase of the test. Consequently, the Design Service Goal (DSG) of 90000 flights has been verified using a scatter factor of two. Artificial damages were embodied after the fatigue testing in order to verify the damage tolerance characteristics of the airframe structure. This damage tolerance phase of testing with artificial notches has been completed and accordingly 24000 flights have been simulated. Thus, the longest inspection interval (12000 flights) outlined in the Maintenance Review Board Report has been verified with a scatter factor of 2.

Damage Tolerance Phase

The damage tolerance testing was conducted to verify the possible crack propagation and correlate the damage tolerance analysis of SAAB 340 in accordance with requirements presented in FAR/JAR 25.571

A total of 21 artificial damages were introduced before start of damage tolerance testing. The test specimen was subjected to fatigue testing for 24,000 flights. The damages can be divided into two different types:

- Damages where the intention is to study the crack growth rate.
- Damages in fail safe structure, where the primary load path is removed. In this case the ”residual fatigue life” of the remaining load paths, i.e. the time to initiation of secondary cracks, is of interest. The subsequent crack growth rate of possible secondary cracks will also be studied.

The following structural parts have been furnished with artificial damages:

- Wing: lower panels, lower spar caps, front and rear spar web, wing centre splice, wing to fuselage attachment.
- Fuselage: skin panels, skin splices, skin cut-outs
- Cockpit: pilot window posts
- Stabiliser: stabiliser spar/skin, stabiliser/fuselage attachment.
- Nacelle: upper longeron

The selected damages were considered to cover the relevant types of spectra, crack types and materials, in order to justify the complete damage tolerance verification of the Saab 340.

The experiences from the damage tolerance testing have been taken into consideration in the damage tolerance verification and accordingly the existing in service inspection program has been updated if required.

The damage tolerance testing have experienced that it is sometimes difficult to observe any natural crack propagation of artificial flaws or notches. The theoretical crack growth analysis indicates a portion of conservatism for the analysed crack propagation.

The damage tolerance testing will be completed by residual strength tests of the entire airframe. The fuselage vessel and wing and the empennage structure will be subjected to a number of static limit load cases in order to verify the airframe for critical crack length conditions.

Testing the Benefit of Cold Working

Cold working technique is accomplished on the interface between the horizontal stabiliser spars and the fuselage of Saab 340 Full Scale Fatigue test specimen. A limited number of fastener holes are cold worked and modification will be verified by additional fatigue testing of the horizontal stabiliser structure.

The purpose of this modification is to improve economic life of some of the components and thereby avoid future time and cost consuming repairs.

A service bulletin is released for all SAAB 340 aircraft, which calls out the implementation procedure of the cold working as retrofit modification. The requirement of continuing airworthiness is maintained by mandatory special detailed inspection tasks outlined in the Maintenance Review Board Report.

3.4.2 The Saab 2000 Full Scale Fatigue Test

General

The Design Service Goal (DSG) for Saab 2000 is presently 75000 flights or 60000 flight hours whichever occurs first. For the Saab 2000, no complete airframe will be tested due to the commonality with the Saab 340. Consequently, a number of full scale component tests are used to justify the long tem characteristics.

Stabiliser Fatigue Test

This test includes the horizontal stabilisers and the attachment structure to the rear fuselage.

The fatigue phase including the damage tolerance test is completed, comprising of 150000 flights in pure fatigue test with subsequent testing with artificial notches for further 24000 flights.

A total number of four artificial damages were inflicted on the test specimen. These artificial damages can be classified into two types as follows:

- Damages where the intention is to study the crack growth rate.
- Damages in fail safe structure, where the primary load path is removed. In this case the ”residual fatigue life” of the remaining load paths, i.e. the time to initiation of secondary cracks, is of interest. The subsequent crack growth rate of possible secondary cracks will also be studied.

The following structural elements of the horizontal stabiliser have been subjected to artificial cracks. This was made by mechanical means in terms of sawing, grinding or cutting.

- Stabiliser/fuselage attachment
  Study of crack growth rate for a crack in the stabiliser spar cap, from a fastener hole at the attachment to the fuselage frame.
- Rear spar web
  Study of crack growth rate for a crack growing from an inspection hole.
- Upper skin panel
  Study of crack growth rate for a crack growing from a fastener hole towards the honey comb core.
- Mid hinge
  Study of crack growth rate for a crack growing from a fastener hole (LHS), study of residual fatigue life and crack growth of possible secondary cracks after complete failure of one of two parts in the mid hinge (RHS).

The rationale for selecting the aforementioned damages is that the damages shall cover relevant types of spectra , crack types and materials in order to constitute a substantiation of the damage tolerance analysis of the Saab 2000 horizontal stabilisers.

Residual strength tests are ongoing for those damage which are considered to require substantiation of the residual strength as a result of complex geometry and/or loading. Fifty percent of this verification is completed with acceptable results, limit load capability has been demonstrated.

The experiences from the damage tolerance testing have been taken into consideration in the damage tolerance verification and if necessary the existing in service inspection program has been updated to reflect the results from the testing.

Wing/Fuselage Fatigue Test

The subject test includes the centre and the rear part of the fuselage, the complete wing torque box and the rear part of the engine nacelles.

The wing detail design is changed compared to the Saab 340 (machined spars with integral spar caps), and the wing/fuselage interface also. Furthermore the cabin pressurisation spectrum is more severe, more exactly twice as severe as the spectra for the Saab 340 aircraft. The flight and landing loads on the fuselage is also more severe due to the slender fuselage of Saab 2000.

The first part of Wing/Fuselage Fatigue test up to 150000 flights with fatigue loads is finalised. Subsequently, the damage tolerance testing will continue aiming to reach 2*12000 flights with artificial notches.

In conformity with all other tests, the wing/fuselage structure will be tested to demonstrate the damage tolerance characteristics in accordance with airworthiness requirements specified in FAR/JAR regulations.

This also includes residual strength tests up to limit load condition for selected and critical load cases.

Prior to commencing the test, a revised and a lesser truncated sequence need to be defined.

Due to some findings, a limited number of areas on the wing structure have been subjected to cold working in order to improve the economical life of the wing spars and skins. Hence, a retrofit Service Bulletin is called out to enhance the fatigue characteristics and consequently avoid future time and cost consuming repairs.

Engine Mount structure Fatigue Test

The Saab 2000 engine mounting structure is completely different in design compared to the Saab 340. Eighth steel struts attaching to the forward engine mounts to the nacelle structure basically build up the structure. Each of those eight struts is redundant in terms of continuing airworthiness if an arbitrary strut is failed.

The fatigue phase of the test is completed (verification of 150000 flights from fatigue point of view). A number of fail safe situations (simulating a broken strut or fitting) with respect to the fatigue behaviour are tested in order to check the fail safe characteristics of the entire trussgrid under the fatigue loading. This portion of the test verified the fail-safe concept for a time period of 3*12000 flights.

The damage tolerance phase with artificial cracks will be the final phase of this test programme.

3.4.3 JAS39 Gripen Fatigue Testing

The strength verification programme with large components was completed during 1994. The full scale fatigue test of the twin seater, 39B, was completed in 2000. The full scale fatigue test of the single seater, 39A, is still running and the full scale fatigue test of the twin seater export version, 39D, is just to begin.

3.4.4 Full Scale Fatigue Test of the Single Seater, 39A

The configuration of the major fatigue test is almost identical to that of the major static test, both with regard to the structure and the test arrangement, Figure 13. The test set-up has about 90 control channels and is monitored by acoustic emission, besides the inspection by conventional methods. The test article has more than 1,000 strain gauges installed and has today (April 03) been subjected to 25,000 flight hours (35,000 flights) testing. The test will continue to 32,000 flight hours. At 26,000 flight hours will the loading of pylons be changed in order to verify extended flight time with external stores.

Fatigue cracks have been found:

- in a lug for the attachment of the actuator for the main landing gear door.
- in parts belonging to the air brakes.
- at hole edges in webs of formed sheet frames.
- at tool holes in the web of a machined fin attachment frame.

The outcome of the test has been used to retrofit operational aircraft and to redesign parts for batch 3 for the Swedish airforce as well as for export aircraft. The lug for the main landing gear door actuator and the parts for the airbrake are redesigned. The formed sheet frames are replaced by high speed machined integral frames and the web of the machined fin attachment frame is made thicker. The cracking is not a flight safety issue but the rectifying actions have been made in order to avoid any cracking at all during the design life.

The objectives with the testing beyond the test verification goal of 16,000h is.

- to identify areas where fatigue, in the long run, may show up in the A and B versions.
- to verify extended flight time with external stores in pylons for the A and B versions.
- to verify wing and fin structure for an extended life for the C and D versions.

3.4.5 Full Scale Fatigue Test of the Twin Seater Export Version, 39D

The test object is a complete fuselage, Figures 14 and 15. Attachment loads from the wings, fin, foreplanes, landing gears etc. are applied via dummies. The whole test set-up has about 90 control channels (actuators and pressure valves) and the structure is equipped with more than 1,000 strain gauges.

The test is made in order to verify:

- Increased service life (8,000 flh)
- Changes due to part count reduction (e.g. introduction of high speed machined integral parts)
- Changes due to world wide climate adaptation (WWC)
- Increased cabin pressure
- Increased basic design mass
- Changes due to Air-to-Air Refuelling installation (AAR)
- Changes due to Radar Cross Section reduction (RCS)

A number of unit load cases and balanced load cases will be subjected to the test object before fatigue testing is started in order to measure strains. The measured data will be used to certify new structures for full flight envelope. Some of the load cases will be measured after every 1,000 flight hour block of fatigue testing as well.

The repetition test sequence for the fatigue test consists of about 400 flights representing 500 flight hours. The goal is to exceed 32,000 flight hours of test simulation. Start of fatigue testing is scheduled for August 2003.

3.5 STRESS ANALYSIS AND FRACTURE MECHANICS

3.5.1 Analysis, Optimisation and DTA of 3D Bolted Joints

A computational procedure for analysis, optimisation and damage tolerance analysis of 3D bolted joint problems was developed in a project financed by the European Union under the GROWTH program (project BOJCAS). The procedure is a further development of a mathematical splitting scheme [7], [8].

In the Splitting Method the problem to solve is splitted into subproblems and the final solution is obtained by superposition [9, 10, 11, 12, 13, 14]. Nonlinearities due to apriori unknown contact surfaces between bolts are efficiently handled by solving the full 3D problem repeatedly until a virtually exact solution is found. A key requirement was that the computational scheme developed must be so effective that it can be used as a computational module in optimisation, damage tolerance analysis and statistical analysis of bolted joints. Key activities in this development are reviewed below.

The bolted joint problem of interest was formulated as a set of partial differential equations with appropriate boundary conditions. The equations considered were the Navier 3D equations of linear elasticity with unknown contact surfaces and assumed friction free conditions. Damage formation was not considered in the present work although the splitting method developed is well suited for analysis of several kinds of damage in composite bolted joints [7].

The numerical scheme developed will provide virtually exact numerical solutions for pointwise stresses. Hence, it is important to know the general mathematical properties of the exact 3D solution. If *homogenized* material properties are used for the plates in the analysis, the exact mathematical solution is smooth everywhere in the bolted joint except in the neighbourhood of geometrical edges and vertices (if the detailness of the model is a ply-by-ply model, stresses are also singular at ply interfaces at the bolt hole surfaces). At washer-bolt head, washer-plate and bolt-plate edges (at hole surfaces), the displacements are of the type $u \sim r^{\lambda}, Re[\lambda] < 1, r$ being the distance to the edge (the singularity exponent λ depends on the position along the edge). Hence, stresses and strains are *infinite* at these edges for arbitrary small loads.

Figure 16 exemplifies a calculated radial stress distribution σ_r as function of the distance z from the contact surface between the two plates at the circumferential angle where radial stresses are highest. Results for both homogenised plate properties and homogenised ply-by-ply properties are shown (in the latter case stresses at half ply thicknesses are shown). The solutions are converged. The stresses in the 0-degree plies are the largest, while stresses in +45/-45 plies are much smaller and roughly the same (z-coordinates differs slightly). The stresses in the 90-degree plies are smallest. The large black circles shown is the *average stress* in a 45/0/-45/90 stack (plotted at average z for the four plies). This average is very close to the homogenised solution (open circles), except near the singular edges z=0 and z=t respectively. This imply that given a homogenised solution, the radial (*average*) stresses in the 45/0/-45/90 stack can be obtained (except close to the singular edges).

The major work in the project was the development a fast method for reliable solution of 3D nonlinear bolted joint problems of *real-life complexity* where pointwise stresses can be determined (for homogenised material data) with high accuracy, say, *a guaranteed* relative error in maximum pointwise stress less than 1 %. The performed work consisted of the following parts,

- Invention of *very fast* and accurate method for solution of nonlinear 3D contact problems
- Derivation of mathematical proofs for the existence of a solution to the splitting scheme, it's uniqueness and convergence properties,
- Implementation of the splitting scheme on a cluster of SMP-computers,
- *Verification* that the mathematical equations were solved correctly with control of error and with very high accuracy,
- *Validation* that the numerical solution of the full 3D equations provide stresses in good agreement with experimental data.

Solution of contact problem. A novel scheme based on linear elastic fracture mechanics was used for solution of the contact problem. The lines γ_i separating areas of contact and no-contact are used as primary unknowns in the computational scheme (Figure 17). By using a simple Newton scheme and the condition that the positions of the contact lines must satisfy $K_I(\gamma)=0, K_{II}(\gamma)=0$ and $K_{III}(\gamma)=0$, approximations to the exact solution are found with high accuracy. In order to be computationally efficient, the method requires a solver that can solve the full 3D problem, for apriori given contact surfaces, repeatedly to a very low computational cost/case. The splitting scheme is the basic tool used to achieve this objective.

Simple mesh design when employing the Splitting Scheme. In the splitting scheme, the bolted joint problem (with apriori assumed location of the contact surfaces) is splitted into subproblems and the solution is obtained by superposition. The discrete solution ${}^{n}\widetilde{u}$ to this problem is,

$$
{}^{n}\widetilde{u} = w_1 + \sum_{k=1}^{n} \beta_k \cdot w_2{}^{(k)} + \sum_{k=1}^{n} \beta_k \cdot w_3{}^{(k)} \tag{1}
$$

where β_k are scaling factors to be determined. The displacements w_1 and $\left\{ w_2^{(k)}, w_3^{(k)} \middle| k = 1,\ldots,n \right\}$ are solutions to *local* and *global* problems labelled $\boldsymbol{H}$, $\boldsymbol{A_1}$, and $\boldsymbol{A_2}$, respectively. Figure 18 shows a schematic picture of the splitting scheme in a 2D setting.

The mesh design is an expensive and time consuming part in normal FE-analysis. However, by employing the present scheme drastic simplifications are possible. Figure 19 (upper part) shows a generic mesh for the local problem (i.e. bolt and plate). By simply expanding, stretching/compressing and twisting the bolt/plate meshes (mapping is defined by parameters α , Figures 17, 19) and by adding a *hp*-type of mesh details at assumed contact lines γ_i the local mesh design can be completely automised.

Another great advantage in mesh design is that on the global level, the mesh might be *very* coarse and contact surfaces need not be modelled at all (perfect contact between bolts and plates are assumed in the global problems, an error which is corrected for exactly when using the splitting scheme). In fact, under most conditions the modelling of the bolts on the global model might be avoided too (see optimisation below).

Exponentially fast convergence when solving nonlinear contact problems. It was mathematically prooved that coefficients β_k in (1) are uniqely determined and converges exponentially fast to the exact mathematical solution when employing the *hp*-version of FEM. The nonlinear solution scheme developed, which is based on the condition that stress intensity factors are zero for the exact solution, do also converge extremely fast to the exact solution. In a case of a 20-bolt joint it was demonstrated that a relative stress error in maximum bearing stress of order 10^{-4} could be obtained in only four iterations (Figure 20).

The discretisation error can be estimated. By using the *hp*-version of FEM combined with the mathematical theory derived, the discretisation error in the nonlinear contact solution, with respect to the exact mathematical solution, can be controlled. This is simply done by deriving a sequence of solutions for increasing polynomial orders *p* and refined *local* meshes and monitoring the convergence in the quantity of interest, for example pointwise stress.

Verification of contact solution algorithm. In order to verify that the splitting scheme is correctly implemented, a benchmark example having 20 bolts, was also solved using a direct approach, i.e., an extremely fine mesh was created and solved using the *hp*-version of FEM using standard techniques. The two solutions were found to be in very close agreement.

Validation. The analysis procedure was validated by comparing with experimental data. Figure 6 shows that the calculated 3D strain solutions are in good agreement with the real-life strain distributions, in five different complex joints with 20 bolts each.

A system setup for large-scale analysis. The splitting scheme has been designed to solve truly large-scale problems on a cluster of SMP-computers. A special version has been designed for optimisation studies, statistical analysis and damage tolerance analysis. The system exhibits excellent *scalability*. The simulation system has been tested by solving problems with up to 700 bolts (Figure 22).

Optimisation, DTA and statistical analysis. Due to the efficiency of the computational scheme developed, more advanced *design approaches* can be used. To this end, the following analysis types were addressed in the BOJCAS project,

- *Optimisation* studies of a test specimen (Figure 21) in which the objective is to find the minimum number of bolts and indirectly the corresponding optimal bolt pattern. Design variables are bolt positions and number of bolts. Constraints are the fatigue life and the joint strength.
- Determine the *statistical* fatigue life and joint strength distribution for the *optimal design* for perturbations in bolt hole positions in adherends, bolt diameters etc.
- Determine *damage tolerance properties* of the optimal design with respect to lost bolts, i.e. determine the statistical and load bearing capacity due to loss of one or several bolts in randomly selected positions.

A meshless method for bolt modeling on global level. When numerous 3D solutions are needed, as in case of statistical analysis, optimisation etc, one *cannot* afford to frequently re-analyse the global problem for various bolt patterns or damage patterns. Hence, a method was needed where *a single global analysis is* sufficient for deriving the 10^4 to 10^6 virtually exact nonlinear solutions that might be required. To this end a meshless solution algorithm (on the global analysis level) to be used together with the splitting scheme was developed.

System for optimisation, statistical and DT-analysis of bolted joints. Figure 23 gives an overall view of the analysis tools created to be able to perform *advanced design of bolted joints*. The lower left corner in Figure 23 exemplifies a very simple global domain (the specimen shown in Figure 21). Note that bolts and bolt holes are not explicitly modelled on the global level. The local domains (Figure 23 upper left corner) contain all geometrical details. This made it possible to determine stresses accurately and with control of the error also in regions where stresses are singular. By using the meshless bolt strategy indicated the complex analysis types could be performed to feasible cost.

Bolt pattern optimisation. One main difficulty is the discrete and non-convex character of the optimisation problem. Hence, several minima might exist which require non-standard approaches. However, having access to a fast solver which can solve a bolted joint problem in very short time/case, general optimisation methods like genetic algorithms, extensive search algorithms etc can be afforded for simpler cases. The meshless (bolt) optimisation procedure developed uses a restricted search space where bolt centers $(x_i, y_i) \in (\mathbf{X}, \mathbf{Y})$ where (**X**, **Y**) are discrete sets containing allowable bolt coordinates (Figure 24). The reason for limiting the search space (which is indeed large with over 10^{90} bolted joints to select between) to a fixed set of coordinates is that the computational efficiency can be increased by several orders of magnitude. A gradient based type of optimisation algorithm is used for most of the studies. In the optimisation scheme used, bolts are moved in the fixed gitter of feasible bolt locations until an optimum (in most cases local) is found.

Figure 25 shows the iteration history for the case of a double lap joint with 28 bolts. The start solution and the bolt movements during subsequent iterations and the optimal solution are shown. The red filled circles marks the start bolt pattern. Open small circles shows bolt locations at each iteration and the large filled circles the optimal positions minimising (a measure) of the bearing stress (note maximum bearing stresses are locally infinite). The optimum solution, with error control, was obtained after 7 iterations, and solution of 190681 3D nonlinear bolted joint problems. Results from bolt pattern optimisations using different start solutions (resulting in different local optima) gave a quite clear picture on how the bolt pattern depends on the number of bolts. Optimisation studies for cases with 12-28 bolts shows that it is possible to find bolt patterns giving only slightly increasing maximum bearing stress with increasing number of bolts, assuming that the total load is proportional to number of bolts.

Damage tolerance properties in case of lost bolts. The damage tolerance properties of bolted joints for different number of bolts were investigated in case of loss of one, two and three bolts, respectively, *at arbitrary positions* in the joint. An extensive search strategy is economically feasible (when the splitting method is used) since loss of *k* of the *N* bolts in the optimal joint design can be removed in only $\binom{N}{k}$ different ways. Hence, for *k*=1, 2 and *k*=3 and N bolts only, $\binom{N}{1}+\binom{N}{2}+\binom{N}{3}$

configurations need to be analysed. For *N*=20 there are 20+190+1140 cases to analyse. For *N*=28 there are 28+378+3276 cases.

Figure 26 summarize the 1140 solutions corresponding to loss of three bolts in arbitrary positions in a joint with *N* =20 bolts. The worst (damaged) bolt patterns found correspond to a bearing stress magnification factors of 2.11. The worst damage scenario is marked with three large red circles. However, Figure 26 shows that there are very few joints (i.e. damage patterns) that exhibit such high magnification factors. If damage tolerance constraints are included in the optimisation scheme, it seems likely that damage tolerance properties can be very much improved.

Future work. The work briefly reviewed are documented in a number of internal reports and will be published internationally. New research has been initiated to solve the combined 3D contact problem and fracture mechanics problem (i.e. merge developments in SMAAC and BOJCAS projects). The objective in the first step is to be able to do more reliable statistical multiple-site fatigue crack growth analysis [13] of complex aircraft structures (Figure 27). Computations will be performed on a Linux-cluster with TFLOP-capacity.

3.5.2 Stress Intensity Factor Equation for Quarter-Elliptical Corner Cracks at an Open Hole

Numerical stress intensity factor solutions are continuously improved as the computer capability increases. Also, improved finite element mesh generators make it faster than before to investigate extended ranges of geometry parameters. In many cases it is sufficient to have a numerical data base from which stress intensity factors can be calculated by simple linear interpolations. However, in some cases it is an advantage to have an equation to calculate the stress intensity factor.

An equation for the stress intensity factor of two symmetrical corner cracks at an open hole in a plate subjected to uniaxial loading has been developed, Ref. [15]. The geometry is shown in Figure 28. The equation is made to fit the numerical data developed by Lin and Smith using the finite element method. Furthermore, the general form of the equation is based upon the equation developed, in the beginning of the eighties, by Newman and Raju.

The equation developed for the stress intensity factor is written,

$$K_I = \sigma\sqrt{\frac{\pi a}{Q}}\left[M_1 + M_2\left(\frac{a}{T}\right)^2 + M_3\left(\frac{a}{T}\right)^4\right] f_\phi f_{OH}\, grf_{a/c} \quad (2)$$

where σ is the remote uniform tensile stress, a is the crack depth in the thickness direction, Q is the square of the complete elliptical integral of second kind and T is the plate thickness. M_i are functions of the aspect ratio a/c where c is the surface crack length measured from the hole edge. Based upon the equation by Newman and Raju it was assumed that the M_i:s for a/c≤1 could be written in the following form,

$$\begin{aligned} M_1 &= m_{11} + m_{12}(a/c) + m_{13}(a/c)^2 \\ M_2 &= \left(m_{21} + m_{22}(a/c) + m_{23}(a/c)^2\right)/\left(m_{24} + a/c\right) \\ M_3 &= \left(m_{31} + m_{32}(a/c) + m_{33}(a/c)^2\right)/\left(m_{34} + (a/c)^{\omega_1}\right) \end{aligned} \quad (3)$$

The equation for M_3 is a simplification of the equation used by Newman and Raju. For a/c>1 it was assumed that,

$$\begin{aligned} M_1 &= \left(m_{11} + m_{12}(c/a) + m_{13}(c/a)^2\right)\sqrt{c/a} \\ M_2 &= 1/\left(m_{21} + m_{22}(a/c)^{\omega_2} + m_{23}(a/c)^{\omega_3}\right) \\ M_3 &= m_{31}/\left(m_{32} + (a/c)^{\omega_4}\right) + m_{33}/(a/c)^{\omega_5} \end{aligned} \quad (4)$$

where the equations for M_2 and M_3 are modified as compared to the corresponding equations by Newman and Raju. Using the method of least squares combined with a trial and error procedure the coefficients m_{ij} and exponents ω_i were found. Their values are presented in Table 1.

a/c	i	m_{i1}	m_{i2}	m_{i3}	m_{i4}	ω_1	ω_2	ω_3	ω_4	ω_5
≤1	1	0.9981	-0.0485	-0.034						
	2	4.7766	-9.1728	5.2176	1.35					
	3	-0.6269	0.9276	-0.3767	0.70	1.55				
>1	1	0.9099	-0.0143	0.0206						
	2	0.9204	8.6463	-6.7058			3.50	3.65		
	3	-1.3164	1.3000	0.5276					6.10	3.40

Table 1. Coefficients and exponents for the M_i-functions.

The function f_ϕ is given by

$$\begin{aligned} f_\phi &= \left[(a/c)^2\cos^2\phi + \sin^2\phi\right]^{1/4} \quad a/c \le 1 \\ f_\phi &= \left[\cos^2\phi + (c/a)^2\sin^2\phi\right]^{1/4} \quad a/c > 1 \end{aligned} \quad (5)$$

f_{OH} is the stress intensity factor for two symmetric through the thickness cracks at an open hole as suggested by Schijve,

$$f_{OH} = \left[\lambda^2/\left(0.609\lambda^2 - 2.07\lambda + 2\right) + (\lambda + 2)/2\right] \quad (6)$$

where

$$\lambda = 1/(1 + c/R) \quad (7)$$

and R is the hole radius. In the present investigation the same modification of λ is applied as used by Newman and Raju, that is,

$$\lambda = 1/\left(1 + (c/R)\cos(\mu\phi)\right) \tag{8}$$

where μ is a fitting parameter which was set to 0.85.

By studying the g-functions used by Newman and Raju it was assumed that the following function would remove most of the angle dependence for a/c≤1.

$$g = \begin{pmatrix} G_1 + G_2(a/T)^{\kappa_1} + \left(G_3 + G_4(a/T)^{\kappa_1}\right)(1-\cos\phi)^{\gamma_1} + \\ \left(G_5 + G_6(a/T)^{\kappa_1} + G_7(a/T)^{\kappa_2} + G_8(a/T)^{\kappa_3}\right)(1-\sin\phi)^{\gamma_2} + \\ \left(G_9 + G_{10}(a/T)^{\kappa_1} + G_{11}(a/T)^{\kappa_2} + G_{12}(a/T)^{\kappa_3}\right)(1-\sin\phi)^{\gamma_2}(1-\cos\phi)^{\gamma_1} \end{pmatrix} \tag{9}$$

and it was furthermore assumed that

$$\begin{aligned} G_7 &= G_7(a/c \le 1)/(a/c) \\ G_8 &= G_8(a/c \le 1)/(a/c) \\ G_{11} &= G_{11}(a/c \le 1)/(a/c) \\ G_{12} &= G_{12}(a/c \le 1)/(a/c) \end{aligned} \tag{10}$$

for a/c>1.

The numerical values of the coefficients and exponents are

γ_1=4.8 ; γ_2=1.8
κ_1=3.2 ; κ_2=0.2 ; κ_3=2.1
G_1=0.9426 ; G_2=-0.0619 ; G_3=0.1177 ; G_4=0.1542 ; G_5=-0.0233 ; G_6=-0.0570
G_7=0.1734 ; G_8=0.2846 ; G_9=78.5017 ; G_{10}=127.4589 ; G_{11}=-56.6870 ; G_{12}=-178.0167

There is no explicit dependence on the R/T-ratio in the equation by Newman and Raju. Lin and Smith investigated four different R/T ratios. In the present investigation it was found that the function r could describe the R/T-dependence rather well.

$$r = 1 + \beta_1\left(1 - e^{\nu_3 a/c}\right)\left(1 - \frac{1}{R/T + 0.5}\right)^{\nu_4}\cos(\beta_2\phi) \tag{11}$$

the values of the parameters β_1, ν_3, ν_4 and β_2 were found to be,

β_1 = 0.194 ; ν_3 = -1.525 ; ν_4=1.60 ; β_2 = 0.54

Finally, a function of the aspect ratio and the elliptical angle was introduced to cut some of the rather large differences observed between the values computed using the proposed equation and the numerical data by Lin and Smith.

$$f_{a/c} = 1 - \left((1-\cos\phi)^{\eta_1} / \left(B_1 + B_2(a/c)^{\eta_2}\right)\right) \tag{12}$$

The accuracy of the stress intensity factors computed using the developed equation with respect to the numerical data by Lin and Smith is within ±6 % for 95 % of the data points, see Figure 29.Lin and Smith used approximately the same number of finite elements in their model as Newman and Raju did. Thus, the accuracy of the numerical solution should be approximately the same.

3.6 FATIGUE CRACK PROPAGATION AND RESIDUAL STRENGTH

3.6.1 Experimental investigation of the effect of plastic deformation on fatigue crack initiation and propagation

One of the most important properties of metallic materials is their capability to tolerate large plastic deformation. Obviously, forging, stamping, rolling, and other deformation techniques may benefit from the plasticity of material. A less obvious aspect while being very important for the fatigue of metallic materials is the role of plasticity in reduction of stress concentration in complex structures. The local plastic yield reduces stress concentration so that high stress concentration may be tolerated in a structure. This aspect is seldom addressed when fatigue is considered. The yield may, however, lead to a different fatigue resistance in the affected zone. As a part of the HSM project to provide reference material data, an investigation is performed to study how the plastic deformation may change fatigue crack initiation and propagation property. An aluminium alloy, 2024-T3, is used throughout the research activity reported, Ref. [16].

An investigation has been performed to evaluate the effect of plastic deformation on the fatigue property of metallic materials. Fatigue tests have been performed under the constant amplitude loading with single notched specimens made of 2024-T3 aluminium sheets, both under original and plastic deformed condition.

The fatigue tests indicate that plastic deformation has a considerable effect on the fatigue life of material, depending on the stress level, see Fig.30. A reduction of more than 20% in fatigue strength has been observed near fatigue limit while a less or no reduction may be expected at higher stress levels.

The crack initiation and propagation is monitored with either laser or white light illuminated CCD camera, see Fig.31, and the fatigue crack growth is evaluated based on the image analyses. Microscopic and scanning electronic microscope inspections have been made both on the specimen surface and on the fatigue and fracture surfaces to identify fatigue crack initiation and growth mechanisms for original as well as plastic deformed material.

The research shows that plastic deformation affects fatigue life through the creation of a large number of micro cracks on the surface of material, see Fig.32. As a result, the crack initiation is accelerated and the fatigue life is reduced. The micro cracks created by plastic deformation also reduce considerably the scatter in fatigue life. Although there are indications that plastic deformation may affect the crack growth rate, the fatigue tests showed that the crack growth rate is only marginally affected, see Fig.33. In an engineering treatment, the effect of plastic deformation on fatigue crack growth rate may be ignored.

The investigation indicates that different initial crack size may be needed in analysing the fatigue crack growth depending on the stress level and how many cracks may participate crack initiation process.

3.6.2 Cyclic and Static Fatigue Crack Growth Thresholds

Mathematical foundation for the new double threshold concept is investigated for the analysis of fatigue crack growth, Ref. [17]. The model consists of two major elements; an intrinsic crack growth threshold which corresponds the material resistance to the fatigue crack growth due to the reverse yielding at the crack tip, and a maximum stress intensity factor threshold which contributes to the possible change of crack growth mode and the crack closure mechanism when the tensile plastic deformation ahead of crack tip is very small. These two thresholds are proposed as material parameters to determine fatigue threshold condition. The mathematical model is developed based on the double threshold concept so that only three parameters are required to determine the threshold for various different materials. The model is successfully used to characterise the crack growth threshold for varieties of materials with significantly different features. The model is also randomised to account for the scatter in fatigue crack growth thresholds. The statistical model has successfully accounted for and explained the widely observed phenomenon that the scatter in the experimentally measured thresholds may increase considerably for low stress ratios.

A new double threshold model is proposed in this investigation for the analysis of fatigue crack growth, see Fig.34. This model is based on two elements; an intrinsic cyclic threshold which corresponds the material resistance to the fatigue crack growth due to the reverse yielding at the crack tip, and a maximum stress intensity factor threshold which contributes to the possible change of crack growth mode and the crack closure mechanism when the tensile plastic deformation ahead of crack tip is small. These two thresholds are proposed as material parameters to determine the threshold condition though they may not be directly measurable for some materials.

The benefit of using such parameters as material constants is that they have their physical interpretations. The intrinsic crack growth threshold corresponds a condition at which the crack closure effect can be eliminated so that the threshold represents a closely micro structurally related property; the resistance of material to cyclic crack growth driving force. Here, the driving force for a fatigue crack growth can be rationalised by using the reverse yielding at the crack tip under the cyclic loading in a stabilised condition (no primary plastic deformation at the crack tip). The maximum stress intensity factor threshold on the other hand represents a condition at which the crack closure may be changed due to the possible change of crack growth mode and the increasing effect of micro irregularities and environment, resulting in a reduce of crack growth driving force. Together with the intrinsic crack growth threshold, the maximum stress intensity factor threshold determines the crack arresting condition for low stress levels for which the crack tip plastic deformation is small. This parameter is also closely related to the material property, see Fig.35.

A system of mathematical model is developed based on the double threshold concept so that only three parameters are required to determine the threshold for various different materials (not limited to metallurgical materials). The model is successfully used to characterise the crack growth threshold for varieties of materials with significantly different features. The model is also randomised to account for the effect of scatters in both the cyclic and static thresholds on the crack growth arresting condition, see Fig.36. The statistical model has satisfactorily accounted for and explained the observed phenomenon that the scatter in the experimentally measured thresholds may increase considerably for low stress ratios.

3.6.3 Impact of Local Yield On Evaluation of Fatigue Crack Growth Under Spectrum Loading Condition

A system of analytical and numerical solutions is developed to deal with fatigue crack initiation and propagation when local yielding occurs, Ref. [18]. The method is based on the detailed finite element analyses under spectrum fatigue loading. The finite element analyses are used to identify a shakedown condition where a stable residual stress field may be created after a period of fatigue loading. Under the shakedown condition, the analyses of fatigue crack initiation and propagation are performed based on a cycle-by-cycle crack growth evaluation according to a crack closure solution that accounts for both the small and long crack growth behaviour. In this way, the post yield fatigue crack initiation and propagation behaviour is evaluated.

The consideration of post yield fatigue is necessary for complicated structural details that may involve plastic yielding under fatigue loading. Without this consideration, it is very difficult to evaluate fatigue problems based on fracture mechanics methods. Under the elastic shakedown condition, a strip yield crack closure model is extended to solve the fatigue problems due to its capability of dealing with load interaction and small crack growth problems. The model is based on the non-linear finite element analyses when cracks are not involved. Extension of the model to post yield problems is demonstrated and verified with the non-linear finite element solutions. Several practical problems are presented to illustrate the solution and to verify its effectiveness. It is shown that with the correct solution, significant improvements, in at least one order of magnitude, may be achieved in evaluating fatigue life based on the fracture mechanics method even when gross local yielding has occurred. The method is significant in helping to understand fatigue behaviour in complex structural details when local yield occurs, and in promoting the fracture mechanics method into a much wider area of crack growth problems.

3.6.4 Fracture Mechanics Evaluation of Post Yield Fatigue Crack Initiation and Propagation

Spectrum loading differs from the constant amplitude loading in that the peak load spikes, though they seldom occur, affect significantly the fatigue behaviour of the rest of small load cycles with large numbers, Ref. [19]. It is usually a great challenge to translate fatigue data for constant amplitude loading into fatigue behaviour under the spectrum loading condition. This presentation will highlight several aspects that should be considered if the fatigue behaviour of spectrum loading is to be addressed. In addition to the conventional load interaction mechanisms, the authors would like to point out that the local gross yield due to peak cycles as well as the stress state transition should be considered. Some possible solutions are discussed and several examples are presented

To account for the spectrum effect on fatigue crack growth, Ref. [19] shows that it is not enough to consider the conventional load interaction effect alone. The local gross yield and the stress state transition are the additional important aspects. These effects can considerably change the fatigue crack growth behaviour under a spectrum loading condition since the seldom events of high loads can produce a different, more complicated condition, leading to the profound post yield effect.

An integrated solution has been presented based on detailed analyses of elastic-plastic stress around a stress concentration location and around the fatigue crack tip. In this solution, Elber's crack closure model is used for the fatigue crack growth and a modified Dugdale model is used to determine the yielding condition around a fatigue crack when the residual stress field are considered due to the post yield effect. A system of approximate weight function solution is developed to realise the crack closure analyses. Some examples are presented. The investigation indicates that a reliable evaluation of fatigue growth may be realised even though the stress can be complicated and local residual stress field and gross yielding may occur. Based on such analyses, various aspects of fatigue crack growth are considered under the spectrum loading condition based on the local gross yield or the residual stress field analyses since these aspects may interact with the crack growth process.

3.6.5 Initial Flaw Concept and Short Crack Growth

In the Swedish ICAF review for the period 1999 to 2001 the Swedish part of the work carried out within the GARTEUR Action Group 26 was reported. Since then the work has continued with a study of a unified model for short and long crack growth. The short crack growth is assumed to be governed by the stress ahead of the plastic zone being able to operate dislocation sources in the next grain. The following parameter is introduced,

$$k = a/\left(a + r_p\right) \tag{13}$$

where a is the crack length and r_p is the size of the plastic zone. As the crack grow, while the plastic zone is blocked by a grain boundary or a grain mis-orientation, the parameter k increases towards a critical value k_c. The critical value is given by,

$$k_c = \cos\left(\frac{\pi}{2}\frac{\left(\sigma - \sigma_{fl}/\sqrt{i}\right)}{S_u}\right); \qquad i = 1,3,5,.... \tag{14}$$

where σ is the local stress σ_{fl} is the fatigue limit stress and S_u is the ultimate stress. The counter "i" counts the number of half-grains the crack spans. At the point when $k=k_c$ the stress concentration reaches a level sufficiently high to activate dislocation sources in the next grain such that the plastic zone extends right across the next grain. Assuming that the short crack growth rate is equal to the shear decohesion of the shear bands due to the shear strain, then the short crack growth rate can be expressed as,

$$da/dN = \Delta\gamma_p r_p \tag{15}$$

where γ_p is the plastic shear strain. From Eqs.(13) and (15) it is obvious that the crack growth rate drops while the plastic zone is blocked by the grain boundary. When the plastic zone extends across the next grain the crack growth rate jumps to a new higher value than the value at the previous grain boundary. Thus, the crack growth rate as function of the crack length will have a saw-tooth appearance with increasing saw-tooth hight. This behaviour is dominant as long as the crack length is small compared to the plastic zone size.

At some point the crack becomes large enough such that LEFM conditions controls the crack growth. Then the crack length is large compared to the plastic zone size. Furthermore, the crack growth rate is proportional to a power of the plastic zone size, as shown below,

$$r_p \propto \left(\frac{K_{max}}{\sigma_y}\right)^2 \Rightarrow \quad K_{max} \propto \sigma_y\sqrt{r_p} \tag{16}$$

$$\Delta K = K_{max}\left(1 - R\right) \tag{17}$$

$$da/dN = C\left(\Delta K\right)^n \tag{18}$$

Combining (16), (17) and (18) yields,

$$da/dN \propto C\left(\sigma_y\left(1 - R\right)\right)^n \left(r_p\right)^{n/2} \tag{19}$$

The unified crack growth rate model provides a smooth transition from Eq.(15) to Eq.(19).

The model was applied to the round robin data used in the GARTEUR project. However, the short crack growth rate, as obtained by Eq.(15), was far too high in this case resulting equivalent initial flaw sizes smaller than the numbers that could be represented by the computer.

3.7 COMPOSITE MATERIALS

3.7.1 Fatigue Testing of Impact Damaged Composite Specimens

An extensive fatigue testing programme on impact damaged carbon fibre/ epoxy composite laminates is ongoing at FFA in co-operation with SAAB. The objective is not only to evaluate the fatigue resistance of such structures but also to study the mechanisms that control the fatigue life. To aid this two measurement techniques are used: Ultrasonic C-scan and Digital Speckle Photography (DSP). With the Ultrasonic measurements, one can see the distribution of delaminations after impact and which delaminations that grow during the fatigue life. The optical measurement technique DSP enables one to obtain the 3D-displacement field as the specimen is loaded. DSP is used mostly to study the shape of the buckles during compressive load. By combining these tools it has been found that the fatigue life is controlled by delamination growth in the direction transverse to the load. The delamination growth is driven by buckling that occurs during the compressive part of the load cycles.

In Reference [20] it was studied how a change in impact energy affects the fatigue life and fatigue mechanisms. Specimens 156 mm wide and 6.2 mm thick were impacted with 19, 28, and 41 J and fatigue loaded with R=-1 to failure. It was found that the fatigue life was reduced when the impact energy was increased, see Figure 37. At low impact energies close to the threshold for causing a harmful impact damage there was a large scatter in fatigue life. The results suggest that it might be possible to predict the fatigue life of specimens impacted at different energies by normalising with the quasi-static strength. For high energy impact damages the buckling occurs through the whole specimen thickness centred at the impact point and directed to the backside. This buckling is facilitated by the indentation caused by the impact. At lower impact energies there is a smaller probability for buckling through the whole specimen thickness, this can partly be caused by the smaller indentation. Instead the harmful buckle might appear only on the frontside and grow at a fast rate during the last part of the fatigue life. It is also seen that buckling and delamination growth in the outer one or two plies are not harmful as long as the buckling occur outside the impact point.

3.7.2 Spectrum Fatigue of Impact-Damaged Composites

In applications composite structures are subjected to spectrum fatigue loading. The objective of the investigation is to develop a method to predict the fatigue life of an impact-damaged composite loaded with an arbitrary load spectrum. Elimination of load cycles from a load spectrum should be studied both experimentally and theoretically. The large scatter in fatigue lives should also be studied. Impact-damaged composites are block loaded and spectrum loaded with two different load spectra. Load cycle elimination studies are performed and fatigue life is predicted with Miner's rule. By using a constant amplitude fatigue curve as input for the Miner's rule predictions the fatigue curve for an arbitrary spectrum can be predicted. Good agreement between fatigue life predictions and experimental data was obtained, see Fig. 38. From testing and analysis it can be suggested that at least a 50% peak to peak load cycle elimination can be used without greatly affecting the fatigue life, see Fig. 39. Measurements suggest that the buckling z-displacement for the first load cycle can be used to predict if the fatigue life will be unusually short or long, Ref. [21].

3.8 AIRCRAFT LOADS

3.8.1 Load Spectrum Survey of Learjet LR-35

Saab Nyge Aero operates Mitsubishi MU-2 (twin turboprop) and Learjet LR-35 (twin jet) aircraft for special flight operations such as EW training and target towing. The operation of those aircraft differs from the operational profile they once was designed for and therefore requires special attention.

A load spectrum survey for the MU-2 aircraft has been running for a couple for years. In Figure 40 is the design spectrum for LR-35 shown together with a measured load factor spectrum from the MU-2 survey. The similarity in terms of operation between the two aircraft types calls for load spectrum survey also for LR-35.

An 8-channel MAS MICRO-II recorder system from SWIFT GmbH has been installed in one aircraft. The system supports, besides one accelerometer for the cg load factor, 4 strain-gauge bridges in the wing on two spares and 3 strain-gauges in the horizontal stabilizer.

The calibrations flights were done in February and the load survey is ongoing. The measurements will at least cover one year of operation. The objective is to verify current maintenance program.

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