A REVIEW OF AERONAUTICAL FATIGUE INVESTIGATIONS IN ISRAEL

APRIL 2003 – MARCH 2005

Compiled by:
Abraham Brot
Engineering Division
Israel Aircraft Industries
Ben-Gurion Airport, Israel

abrot@iai.co.il

SUMMARY

This review summarizes fatigue and fracture-mechanics investigations that were performed in Israel during the period April 2003 to March 2005. The review includes contributions from Israel Aircraft Industries Ltd. (IAI), Israel Air Force (IAF), Technion – Israel Institute of Technology, Tel-Aviv University and Ben-Gurion University.

Presented at the 29th ICAF Conference
Hamburg, Germany
7 June 2005

TABLE OF CONTENTS

A REVIEW OF AERONAUTICAL FATIGUE INVESTIGATIONS IN ISRAEL APRIL 2003 – MARCH 2005

11.1 INTRODUCTION

The Israel National Review summarizes work performed in the field of aeronautical fatigue in Israel during the period April 2003 to March 2005. The previous National Review [1] covered aeronautical fatigue activities up to March 2003. The following organizations contributed to this review:

Israel Aircraft Industries Ltd. (IAI)
Israel Air Force (IAF)
Technion – Israel Institute of Technology
Tel-Aviv University
Ben-Gurion University

The National Review was compiled by Abraham Brot of IAI (abrot@iai.co.il).

11.2 FATIGUE ANALYSIS, TESTING AND LIFE EXTENSION

11.2.1 An Evaluation of Several Retardation Models for Crack Growth Prediction under Spectrum Loading (A. Brot and C. Matias, IAI)

The previous Israel National Review [1] summarized a study performed at IAI in which the results of coupon tests, made from four different alloys and loaded by five distinct spectrum types, were compared to several crack retardation models. The results indicated that the NASA Strip-Yield model gave the best correlation to the test data, but was not sufficiently accurate for some spectrum types.

The continuation of this activity was performed by Dr. J. C. Newman Jr., of Mississippi State University, the developer of the NASA Strip-Yield model. The model was improved by Dr. Newman, and the results were compared to IAI test results for 7075-T7351 coupons under three aircraft load spectra and two crack configurations. The results for the improved model correlated much better to the test data than the original model, with the calculated crack-growth lives all within ± 20% of the test results. The results of the study were presented at the 2003 Aging Aircraft Conference [2] and were later published, in more detail, in a journal [3].

11.2.2 Determining Crack Growth and Arrest Capability of Integrally Machined Structures by Analysis and Testing (A. Brot, IAI)

IAI has begun a study into the crack growth and arrest capability of integrally machined structures. The aim of the study is to determine parametrically the optimum geometric configuration of the structure (rib height, width and spacing) to provide reliable crack-arrest behavior.

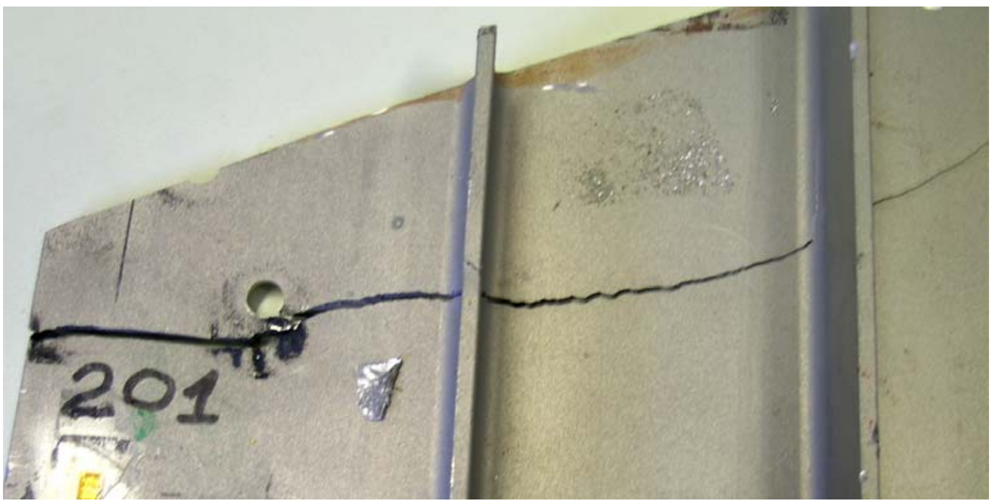


Figure 1: Crack Growth in an Integrally Stiffened Plate

Figure 1 shows the somewhat unusual behavior of crack growth in an integrally stiffened plate, which was observed during a fatigue test. The crack originated from a fastener hole, then it crossed and broke the first integral stiffener. It continued to grow as a slanted "mixed-mode" crack. When it reached the second integral stiffener, it bifurcated into two cracks. The first crack grew, as a shear crack, parallel to the stiffener and along its root. The second crack tunneled under the stiffener, and continued to grow as a slanted crack along the plate, without ever breaking the integral stiffener. As is shown in Figure 1, the second stiffener remained intact. This unusual behavior was thoroughly investigated by fractography.

IAI has joined an international consortium in a project called **DaToN** *(Innovative Fatigue and* ***Da****mage* ***To****lerance Methods for the Application of* ***N****ew Structural Concepts),* which is partially funded by the European Commission. This project is aimed at developing new damage tolerance assessment tools that will deal with specific problems introduced by the use of integral structures manufactured by high-speed cutting, laser-beam welding or friction stir welding. The project began in April 2005.

11.2.3 An Evaluation of Various Methods for Producing Initial Flaws for Fatigue Testing (A. Brot, IAI)

A common problem exists for component and full-scale *damage-tolerance* tests. Initial flaws are inflicted at critical locations on the structure by mechanical or electrical means. However, the transition of these flaws to fatigue cracks can take tens-of-thousands of cycles. As a result, the damage-tolerance test is delayed or terminated prematurely, and insufficient data on crack growth is generated. IAI has performed a comparative study on methods for inflicting initial flaws in structures. 7050-T74 material coupons were tested at a maximum stress of 15 ksi and R = 0.05. Two methods were used to produce the damage: Electro-Discharge Machining (EDM) and mechanical cutting. The results showed that flaws introduced by EDM transitioned to fatigue cracks within 10,000 cycles, while flaws introduced mechanically, required approximately 50,000 cycles to transition to fatigue cracks.

EDM equipment, while relatively effective, is not portable and cannot be used to inflict damage on an assembled full-scale fatigue test-article. As a result, IAI has initiated another set of coupon tests to investigate the effectiveness of introducing flaws using a very fine (8/0 blade) jeweler's saw compared to those inflicted by EDM. This study will compare the cycles to crack-initiation for various sized initial flaws that have been produced by the two methods.

Figure 2 illustrates a typical initial corner flaw that can be inflicted in aluminum using an 8/0 jeweler's saw (flaw width ≅ 0.25mm).

Figure 3 shows the stress field around a 2.54mm long initial flaw at a 6.35mm diameter hole. The stress field was obtained by building a 2D model using *StressCheck,* p-version, finite-element software. The results will be used to estimate the stress-concentrations that exist at the tip of an initial flaw.

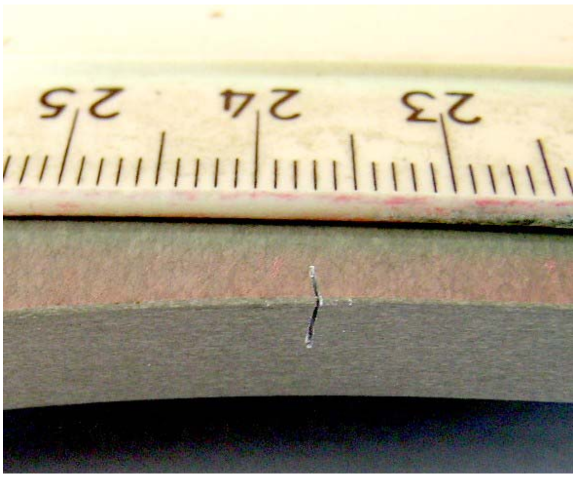


Figure 2: Artificial Corner-Crack Inflicted by an 8/0 Jeweler's Saw Blade (scale in cm)

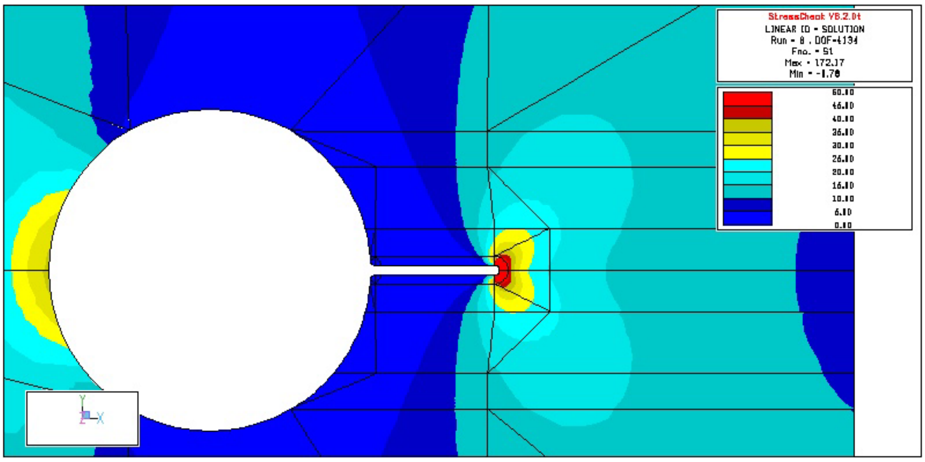


Figure 3: Stress Field Around a 2.54mm Long Initial Flaw at a 6.35mm Diameter Hole

11.2.4 Effect of Shot-Peening on Crack Growth (C. Matias and A. Brot, IAI)

Structures designed and certified to the damage-tolerance requirements, must be inspected periodically for cracks. These inspections usually result in significant aircraft downtime and may be expensive to implement. In order to reduce the cost of inspections, without compromising aircraft safety, the benefits of shot-peening at critical locations is being investigated.

It is well known that shot-peening of highly-loaded structures extends their life to crack-initiation. However, the literature has only limited data establishing a connection between the shot-peening process and crack growth rate [4]. It is expected that the shot-peening process will reduce the crack growth rate of relatively small cracks. By establishing the effect of shot-peening on crack growth rate, the initial (threshold) inspection of highly-loaded structures could be safely delayed.

In order to examine the influence of shot-peening on crack-initiation and crack growth life, a test program was initiated. The test program includes AL2024-T351 and AL7075-T7351 specimens, having symmetrical semi-circular edge notches, which are subjected to constant-amplitude load cycles, as is shown in Figure 4. The effects of steel ball shot-peening, ceramic ball shot-peening and flap-peening are being investigated for their effect on crack-initiation and growth. Testing is in progress and the final results will be used to establish empirical parameters that will allow the threshold inspection to be safely delayed.

Early results of this study will be presented in a poster, at the 23rd ICAF Symposium [5].

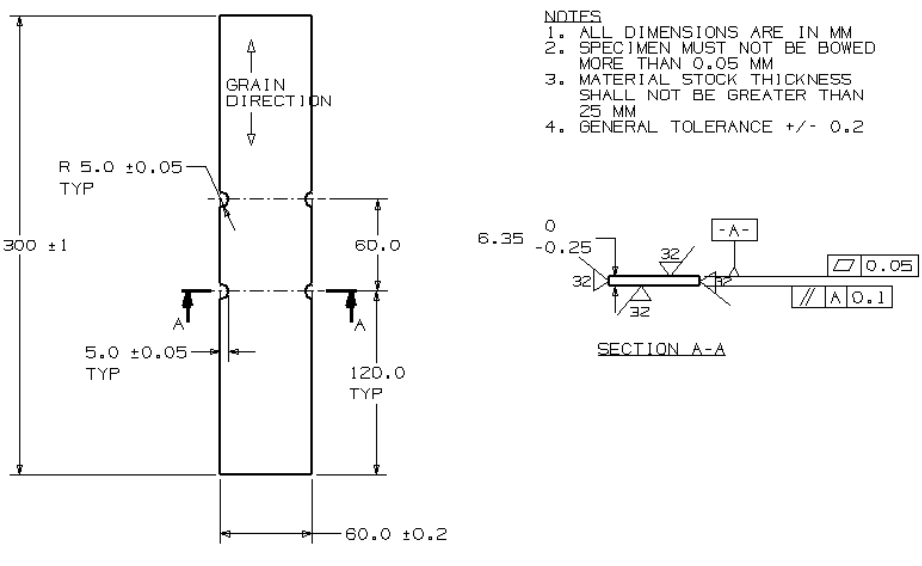


Figure 4: Specimen Used For Testing the Effect of Shot-Peening on Crack Growth

11.2.5 Size Effect of Microdamage Growth and its Relation to Macro Fatigue Life (E. Altus, Technion)

In its initial evolution stage, fatigue damage consists of many microdamage sites, having random sizes and locations. The way in which these sites grow and coalesce has a crucial effect on the macro fatigue life. A statistical micromechanic fatigue model has been developed, in which the material is composed of microelements of random strength with a certain probabilistic dispersion parameter (β). In addition, the model takes into account local interactions between damaged microelements and their first neighbors by considering a failure sensitivity factor (c), which is the probability that the neighbor will survive the local (micro) stress concentration.

It was shown analytically in previous studies that β is proportional to the S-N power intensity, and ln(1-c) is proportional to the macro endurance limit. In this study, the analysis is generalized to the case where the growth of each micro-damage is size dependent, i.e., each damage site grows at a rate that depends on its current size. The strength of this rate-size relation controls the order of the governing differential equation. It was found that certain "microdamage growth laws" still preserve the macro power law, so that the power on the S-N diagram can be directly related to the local microdamage evolution. While the analytical micro-macro relation is still under current study, a numerical simulation of fatigue damage evolution has been obtained and revealed that the macro S-N power law prevails in spite of the noticeable complexity [6].

Figure 5 illustrates graphically the damage morphology progression. In Figure 5, colors reflect the damage "age": blue for damage initiation sites (old) and brown for damage just before total failure. Upper pictures show fractography just before failure, for non-probabilistic local behavior (top right) versus probabilistic one (top left). Lower pictures show morphology progression of a single microdamage on hexagonal net: non-random (right) versus random (left).

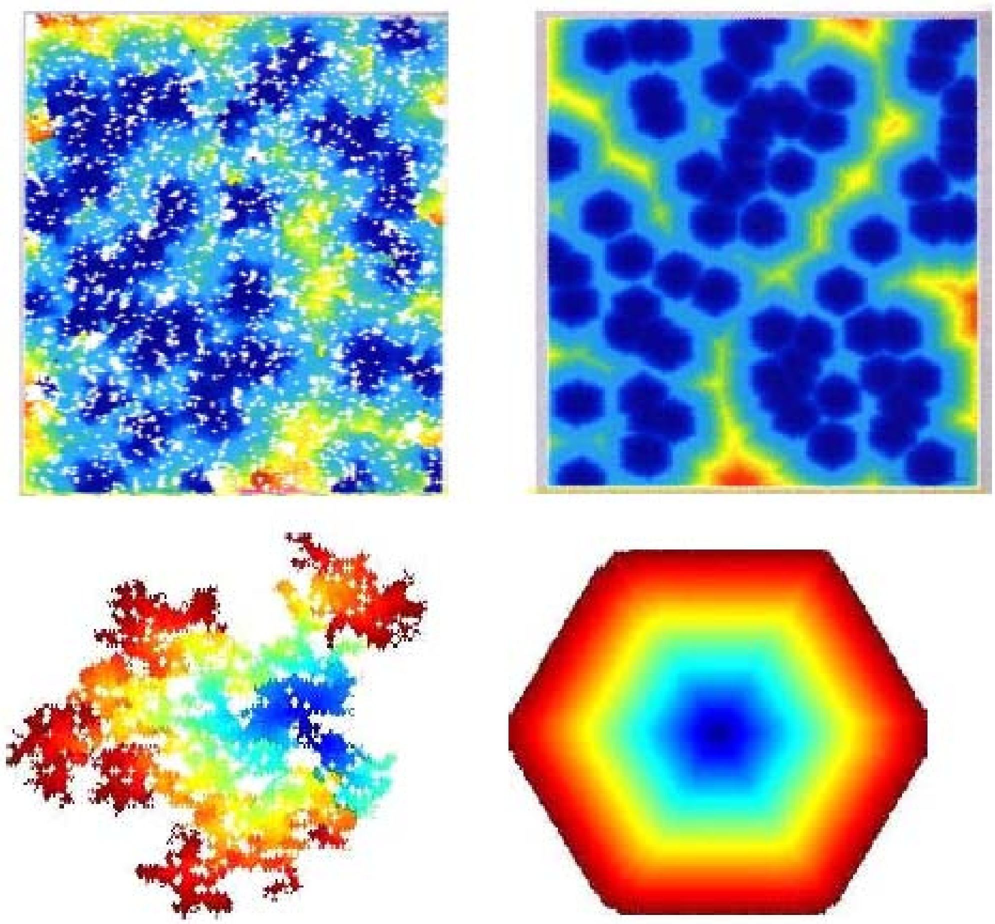


Figure 5: Simulation of Damage Morphology for Uniform Microdamage Rate

11.3 AIRCRAFT PROJECTS

11.3.1 G200 Executive Jet Fatigue Substantiation Program (S. Afnaim and A. Brot, IAI)

The G200 (formerly known as the Galaxy) is a wide-body executive jet, developed by Israel Aircraft Industries and subsequently incorporated into the Gulfstream Aerospace Corporation family. The G200 has a transatlantic range and a maximum cruise speed of Mach 0.85. It can transport up to 18 passengers in a corporate configuration and up to nine passengers in an executive configuration. The G200 is powered by two PW306A jet engines. The G200 primary structure is metallic except for the ailerons, elevators and rudder that are manufactured from composite materials.

11.3.1.1 G200 Full-Scale Fatigue Test

The progress of the G200 full-scale fatigue test was reported in previous Israel National Reviews [1]. After the completion of two lifetimes of fatigue testing (40,000 flights), a residual strength test was performed on the pressurized cabin, by successfully applying a differential pressure of 10.5 psi. There were no indications of any structural failure. This was followed by a comprehensive teardown inspection of the test-article.

Both wings have been disassembled from the fuselage at the wing-root and their lower skins have been disconnected to permit access to the all wing elements (See Figure 6.). The fuselage has been cut into five sections, to facilitate removal of critical structures such as bulkhead elements, sill beams, etc. and to facilitate detailed inspections under laboratory conditions.

During the teardown inspections, some additional cracks were detected, mainly in the same areas as previous cracks (pressure bulkheads, wing Rib 2 area, etc.)

After analytical evaluation of the cracked areas, the results of the teardown inspection will be implemented in the revision of the inspection and maintenance procedures of the aircraft.

Figure 6: G200 LHS Wing During FSFT Teardown Inspection

11.3.1.2 G200 Empennage Fatigue Test

The progress of this fatigue test was reported in previous Israel National Reviews [1]. The G200 empennage test has completed two lifetimes (40,000 flights) of fatigue and one lifetime (20,000 flights) of damage-tolerance testing. Crack growth, resulting from the artificial flaws that propagated, has been measured and evaluated. The composite material elevators and rudder had initial damage inflicted at critical locations, in order to simulate impact damage. No flaw growth was detected at these locations.

At the end of the damage-tolerance testing, a total of ten residual strength tests, covering all air and ground (mainly roll and yaw moments caused by engine reverse thrust) load types, have been performed. The tests have been conducted up to limit load without any indication of structural failure. The teardown inspection of the various empennage elements is currently in progress.

11.3.1.3 G200 Wing Rib 2 Component Fatigue Test

The progress of this component test was reported in previous Israel National Reviews [1]. A second Rib 2 component, having a different fillet radius at the critical location, has been tested for damage-tolerance behavior. Crack growth data has been acquired. At the end of the second test, fractographic analysis of the cracked area has been conducted and also compared to two cracked areas obtained from the RHS and LHS wings in the FSFT.

All the data will be processed and will support the update of the damage tolerance analysis of this detail.

11.3.2 G150 Executive Jet Fatigue Substantiation Program (A. Hermelin, S. Afnaim and A. Brot, IAI)

Israel Aircraft Industries and Gulfstream Aerospace Corporation are currently jointly developing the G150, midsize executive jet. The G150 will have transcontinental (USA) range and a maximum cruise speed of Mach 0.85. It can transport up to eight passengers in an executive configuration. The G150 will be powered by two TFE 731 jet engines. The G150 primary structure will be metallic except for the ailerons, elevators and landing gear doors that are manufactured from composite materials.

In the past, sizing of fatigue sensitive structural members during the detailed design phase was performed on the basis of *allowable stresses* that were provided by the Fatigue Department. These fatigue allowable stresses were defined in terms of *design ultimate stresses*. The stress analyst would calculate stresses under the ultimate load cases and would verify that the fatigue allowable stresses were not exceeded. Since many of the FAR-25 ultimate load cases represent malfunctions or artificial conditions, this procedure was not always successful.

For the detail design of the G150 Executive jet, an innovative procedure for specifying allowable stresses for fatigue was developed. *Typical* fatigue load cases were defined, which include: a typical vertical gust load case, nominal cabin differential pressure, dynamic landing at a typical sinking speed and a typical lateral gust load case (for aft fuselage and empennage). The fatigue analyst selects the fatigue load case that is appropriate for the location to be designed, and calculates the allowable stress for that load case that would result in sufficient fatigue and damage-tolerance life. These fatigue allowable stresses are then transmitted to the designers and stress analysts. The finite-element model of the structure is then run for all of the *typical fatigue loading cases,* and the stress analyst verifies that the fatigue allowable stresses have not been exceeded. In this way the structure can be optimized to meet fatigue and damage-tolerance requirement, without using artificial allowable stresses.

The G150 will be certified to the FAR-25 Regulations, and FAA certification is planned to be completed by December 2005. Figure 7 is a photo of the first flight of the G150 executive jet, which took place on 3 May 2005.

Figure 7: First Flight of the G150 Executive Jet (3 May 2005)

11.3.2.1 G150 Full-Scale Fatigue Test

As part of the damage-tolerance certification, a G150 test-article will be tested for two lifetimes (40,000 flights) of fatigue and approximately one lifetime (20,000 flights) of damage-tolerance testing.

The test-article will include the fuselage and both wings. The tested areas will consist of the forward fuselage and cabin to the aft pressure bulkhead and fuel tank. Both entire wings will be fatigue tested. Since the aft fuselage and empennage have not changed significantly from the G100, they will not be fatigue tested and will act only as load application elements. The test aircraft will be mounted to the test fixture at the nose landing gear attachment and at the engine mount fittings.

The fatigue spectrum loading consists of randomly selected flight-by-flight sequences, reflecting the anticipated usage of the aircraft. A flight consists of the various flight and ground events that the aircraft will experience. 20 – 25 events per flight will be included in the 2000 flight spectrum block.

The test-article will be divided into 20 loading zones, each of which is independently loaded during each event of the spectrum, using 30 servo-hydraulic actuators. In addition, the passenger cabin is pressurized, using compressed-air during the airborne events of the spectrum. The zone loading for each event will be determined using a "constrained least-square error method" which minimized deviations in loading of the important structural parameters. Approximately 500 strain-gages will be bonded to the test-article, mainly to monitor the onset of cracking.

At the end of two lifetimes of fatigue testing, artificial flaws will be introduced by saw-cut at approximately 50 critical locations and damage-tolerance testing will be proceed.

At the end of the damage-tolerance test (total of approximately 60,000 flights), residual strength tests, with the application of limit load and cabin pressurization of 10.5 psi, will be performed to the aircraft structure, in the presence of large cracks at several critical locations. This will be followed by a selected teardown inspection.

11.3.3 Composite Patch Repair for the F-16 Upper Fuselage Skin Fatigue Cracks (I. Kressel, G. Ghilai and N. Hackman, IAI; M. Ben-Noon, IAF and S. Gali, Consultant)

This repair was first reported upon in the previous Israel National Review [1] and was presented, as a paper, to the 22nd ICAF Symposium in 2003 [7].

Figure 8: Boron Patch Repair for IAF F-16C

The F-16 upper fuselage skin suffers from fatigue cracks, initiating from fastener holes and access panel corners. The Israeli Air Force (IAF) developed and applied a bonded, co-cured boron composite patch repair for the F-16A aircraft at the beginning of 2003. The repair was applied by IAI technicians, using field repair equipment.

This aircraft is still flying and the repair is regularly inspected. It currently has flown more than 250 flight-hours with no indications of debonds or delaminations. The main advantages of such a bonded repair, as compared to a metallic bolted one, are its smoother load transfer, the elimination of stress concentration due to additional fasteners, the excellent fatigue and damage-tolerance behavior of the composite materials, and its easy application on curved areas. The detailed repair design and substantiation process was presented in [7]. The main activity since 2003 was the application of the same repair concept on an F-16C aircraft, as is shown in Figure 8.

This specific aircraft already had cracks that extended beyond the allowable size, and therefore, could not be flown. The conventional repair approach, which is based on replacing the damaged skin and performing the original manufacturer's bolted repair, was non-feasible due to a shortage of new skins at that time. Although the boron patch was originally designed for the F-16A, it was found suitable to be used as a solution for actual cracked F-16C. The geometrical difference between the two F-16 models was not a problem since, in the co-curing approach, the raw composite plies are laid, cured and bonded to the metallic structure in a single process, providing excellent geometrical fit to the metallic substrate. The cracks were removed and the repair was completed in two days, demonstrating that such repairs are much easier to apply, compared to the alternative metal repair.

The F-16C repair was removed during a planed major maintenance inspection in March 2005. No indications of corrosion or delaminations were found, following the 170 flight-hour service life.

One of the challenges that had to be addressed during this project was the need to apply the patch on an aluminum skin (Figure 9) that was bolted by steel fasteners to the fuselage frames and stringers. The bonding surface treatment, selected for the aluminum skin, did not ensure bonding between the composite patch and the fastener heads.

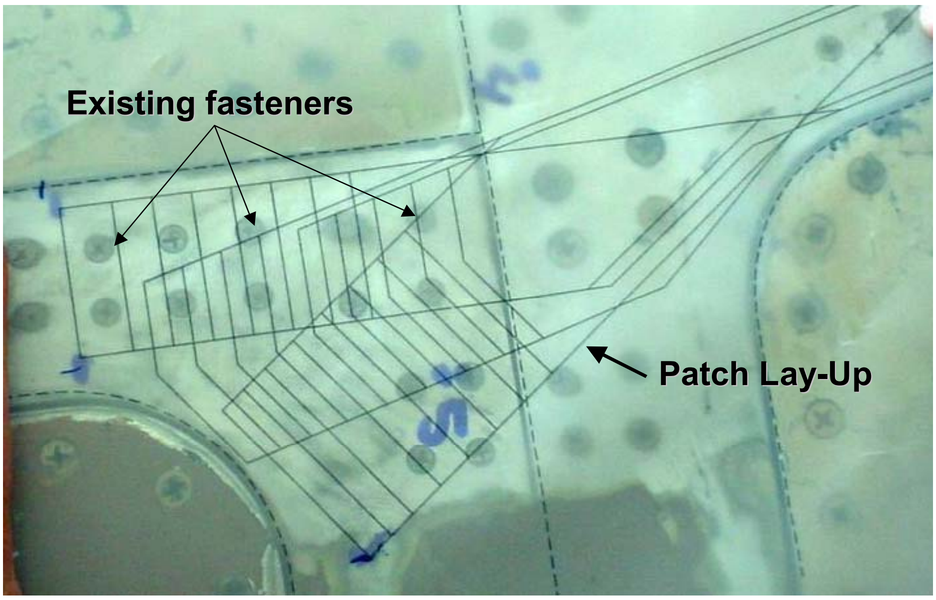


Figure 9: Bonded Repair Lay-Up and Existing Fastener Locations

Analytical evaluation of bonded-bolted structure was performed, using an MSC-MARC finite-element analysis. The FE model included the skin, original splice and the boron patch. The model included both the non-linear behavior of the adhesive and the contact between the fasteners and the metal parts, as is shown in Figure 10. Typical results are shown in Figure 11.

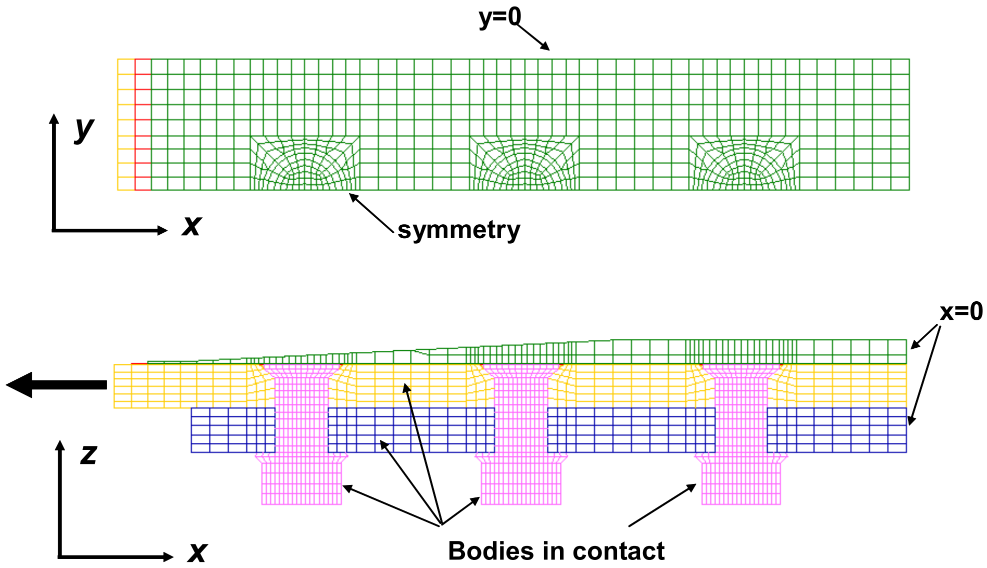
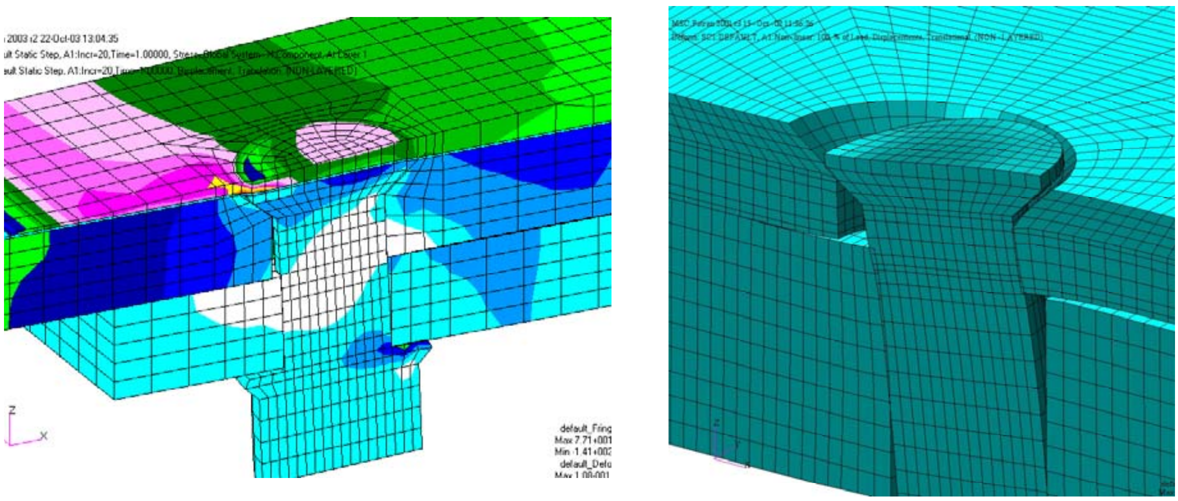


Figure 10: Bonded-Bolted Finite-Element Model and Boundary Conditions

IAI has performed a fatigue test on a specimen representing the F-16C bonded repair with a Grit-Blast surface treatment. The specimen successfully demonstrated *ultimate load* residual strength, after a fatigue test simulating 2000 flight-hours.



(a) **(b)**

Figure 11: Finite-Element Analysis of Bonded-Bolted Joint
(a) Bonded-bolted joint (b) Deformed shape of bolted repair

11.4 COMPOSITE MATERIALS

11.4.1 Fatigue of Composite Materials (Z. Granot , IAI and S. Gali, Consultant)

Work continued on this topic, which was reported in the previous Israel National Review [1]. As part of an ongoing process of striving for lighter and more reliable high performance composite aircraft structures, it is necessary to predict the structure's fatigue life and residual strength. An investigative program is underway at IAI to establish the preferred fatigue life and residual strength analysis techniques to be implemented. Such analytical techniques are required to demonstrate that the structure is tolerant to all flaws that are below the readily detectable size, for the entire projected service life. To effectively design, size, and substantiate the structure, it is important to understand and quantify the structure's fatigue life and residual strength behavior.

It is necessary to show compliance with the specified strength requirements, the target service life and applicable loading spectrum, and the relevant critical environmental factors. It is aimed that these analytical techniques will be used in the design stage, in selecting the most suitable material system, the definition of preferred design details, the local lay-up, and other key design details, of each critical structural element, including the allowable strain or stress levels, and to specify fatigue tests to demonstrate compliance of the design.

The durability approach being implemented is called the “no growth” method. In this method, as is illustrated in Figure 12, one is required to ensure that, in the presence of initial sub-critical flaws, the structure’s original static strength does not degrade below ultimate strength throughout the service life. The analytical tools used here are empirical, based on fatigue test results for test coupons, and can, in principle, be used to deal with all types of flaws, such as delaminations, impact damage, notches, or equivalent geometrical notches.

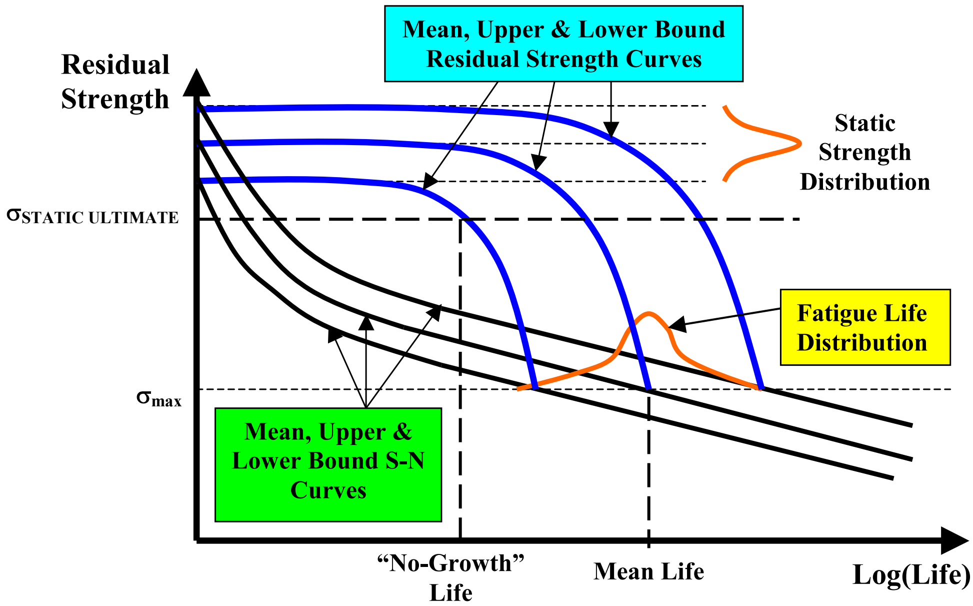


Figure 12: Determination of “No-Growth” Life for Composite Material Structures

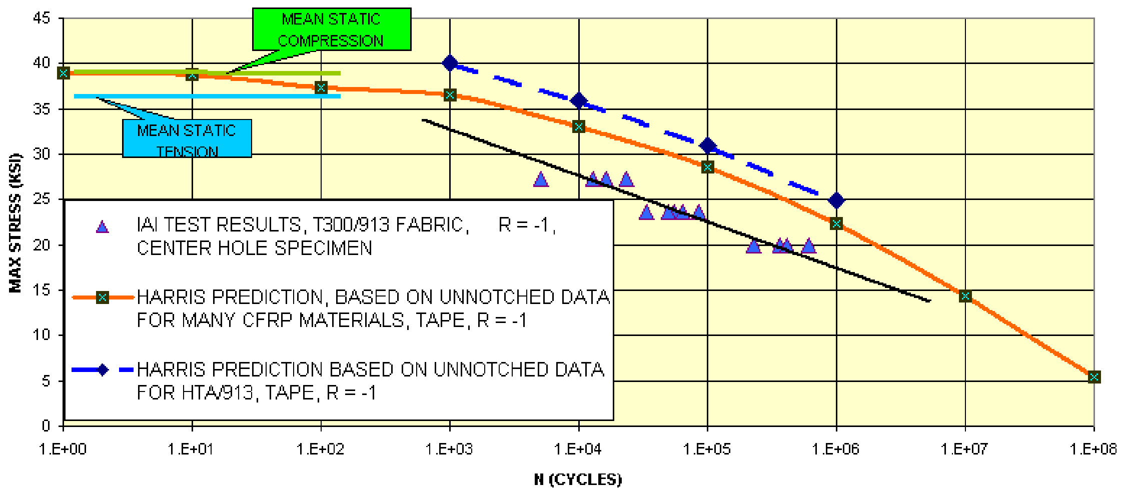


Figure 13: IAI Test Results Versus Harris Equation Predictions

Figure 13 presents some preliminary fatigue test results for small center hole laminated coupons made of T300/913 fabric prepregs. This test program is mainly for the purpose of evaluating different influencing variables and practical analytical approaches for implementation. For comparison purposes, test result data are also given, as derived from data supplied by B. Harris, and from [8], for un-notched HTA/913 laminated specimens made from tape.

A summary of the results was presented at the 45th Israel Conference on Aerospace Sciences [9], and will be presented, in detail, in a paper that will be given at the 23rd ICAF Symposium [10].

11.4.2 Damage-Tolerance Analysis of Bonded Composite Repairs – Analysis and Testing (I. Kressel, IAI)

The concept of using bonded composite repairs for the maintenance of aging metallic aircraft has been proven both as a preventive measure and as a method for retarding future growth of existing damage. The main advantages of a bonded repair, as compared to a metallic bolted one, are its smoother load-transfer, the elimination of stress concentration due to additional fasteners, the excellent fatigue and damage-tolerance behavior of the composite materials, and its easy application on curved areas.

One of the important issues associated with bonded repairs is the presents of defects. Defects such as delamination or debonding may be generated during the bonded repair application or may develop during service. In order to assure the safe operation of the repaired structure, it is important to assess the repair integrity over time. This work presents experimental and analytical work aimed towards establishing acceptance criteria for bonded repair allowable defects. Special attention was given to the crack propagation rate of a cracked metallic structure, repaired by a composite patch with a debonded area over the crack. The effect of residual stress induced by the thermal expansion mismatch between the composite patch and the metal substrate was also considered.

The specimen examined in this work represents a typical 100x100mm co-cured bonded patch repair. It comprises a carbon epoxy, double-sided patch, bonded to an aluminum plate (2024-T3, 1.6mm thick). Each patch was made of three plies of Fiberite 753 style 3K-70-P carbon/epoxy plies and one ply of Fiberite 753 style, 120 fiberglass/epoxy prepreg. An adhesive ply was placed between each patch and the aluminum substrate. An 8mm hole with an initial crack of 3mm was introduced into the aluminum plate.

The patches were co-cured and bonded to the aluminum plate, over the crack, inside a vacuum bag at 120°C for two hours. The average heating and cooling rates were 1˚C/min.

Two types of specimens were manufactured: one with an artificial debonded area placed over the crack and the second type, a reference specimen, with no defects. Tensile and fatigue tests were performed in order to measure the crack propagation rate at environmental conditions of RTD and 70˚C/wet.

The analytical model used for crack growth prediction is based on the “Rose Model”. Finite-element analysis is being performed, in order to verify the results. Figure 14 describes the 3D finite-element model that was constructed in order to study the crack growth behavior of the repaired structure. Figure 15 shows the deformations of the cracked plate without a repair, with a bonded repair and with a bonded repair having a debond defect.

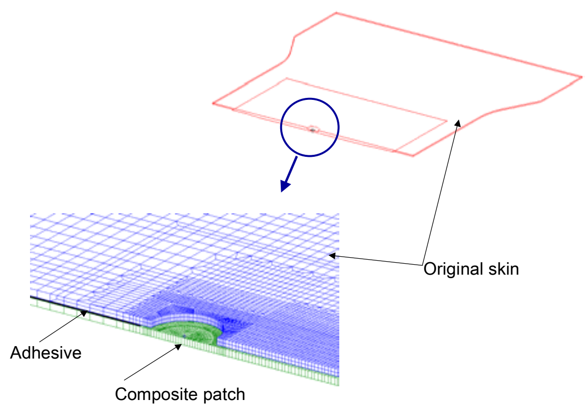
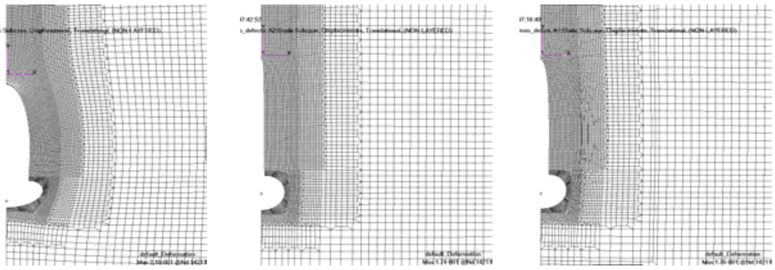


Figure 14: 3D Finite-Element Model of a Bonded Repair



a *b* *c*

Figure 15: Analysis of Bonded Repair for a 29.8 mm Crack
(a) unrepaired, (b) bonded repair, (c) bonded repair with a debond defect

A nearly constant crack growth rate was observed for all the repaired specimens, with or without defects, as is expected from the analytical Rose model. The crack growth of specimen having a debonded area, however, was higher compared to a similar specimen without a defect. All the tested specimens demonstrated slow crack growth rate, compared to a reference unrepaired specimen, demonstrating adequate damage tolerance behavior of the bonded repair concept. The results will be presented in a poster at the 23rd ICAF Symposium [11].

11.5 PROBALISTIC STUDIES

11.5.1 Evaluating Probability of Second Layer Detection by Using Low-Frequency Eddy-Current NDI (C. Matias, A. Brot and W. Senderowits, IAI)

Aircraft structures, designed and certified to the damage-tolerance requirements, must be inspected periodically for cracks. These inspections usually result in significant aircraft downtime and may be expensive to implement. In order to reduce the cost of inspections, without compromising aircraft safety, the use of low-frequency eddy-current (LFEC) non-destructive inspection (NDI) techniques is being investigated.

Often, in a highly loaded joint, the hidden second layer must be inspected for cracks. One possibility is to use an X-Ray NDI method, which is imprecise and expensive. Another possibility is to detect the crack only after it emerges from beneath the hidden layer. In both cases, safety considerations will dictate very short inspection intervals. A third possibility is to use the LFEC technique. This crack detection inspection method, allows the examination for cracks of the hidden second layer of the structure. As a result, it allows us to track the crack while it is still hidden, as opposed to detecting a crack only when it becomes visible. The use of this technique has the potential of significantly increasing inspection intervals. Unfortunately there is insufficient published statistical data to determine parameters for the detection capability of the LFEC method. The decisions for setting the crack inspection periodic intervals, resulting from damage tolerance analysis, relates strongly to the crack detection inspection method and its statistical probability of detection.

Crack detection by means of eddy-current non-destructive inspections for an uncovered layer (first layer) is widely used, thoroughly investigated and well established in the aircraft industry. This method uses a technique of high-frequency eddy-current (HFEC). That technique has an extensive statistical database, and a reasonably accurate, well-correlated Weibull three-parameter statistical model of crack detection probability. On the basis of that model, and the results of tests conducted in the framework of this study, we were able to set statistical parameters, in order to establish probabilities of detection correlations to the LFEC NDI. This will allow us to implement the LFEC NDI benefits to the damage-tolerance considerations for inspection intervals.

A test program was conducted in order to examine the LFEC capability of crack detection. A 7075-T7351 base-plate, with crack sizes varying from 2.5 to 12.7mm, was installed beneath a 7075-T7351 cover plate having a thickness ranging from 1.0 to 6.35mm. The base-plate has 100 holes, from which 30 of them have a crack that emerges out of the hole. The 30 cracked holes are randomly arranged. The cracks appear at five different lengths, and at different orientations, which are of typical for in-service inspections. The cover plates are of six different thicknesses, containing 50% plain holes and 50% countersunk holes. The inspections are being applied around the holes, with the present of HI-LOK fasteners. These features are of typical structural aircraft details and of typical in-service inspections. Inspectors performed the LFEC inspections, using their own instruments and methods (probes, oscilloscopes, calibration etc.). In order to maintain an objective test, the holes were numbered and the cover plates were rotated relatively to the base-plate for each test differently, so that the inspector will not be aware of the specific crack locations.

It was shown that a correlated Weibull distribution can be used to express the crack detection probabilities, for the LFEC NDI, as a function of crack length and cover plate thickness, based on the statistical data gathered at the tests done in this study. (See Figures 16 and 17.)

In this way, a statistical model that will predict the probability of crack detection as a function of crack length and cover plate thickness was established. This will allow the use of the LFEC NDI technique with a relatively high degree of reliability for rationally determined inspection intervals. These inspection intervals will be significantly longer than those determined by the alternate inspection methods. The results of this study were presented at the 45th Israel Conference on Aerospace Sciences [12] and will be presented, as a poster, at the 23rd ICAF Symposium [5].

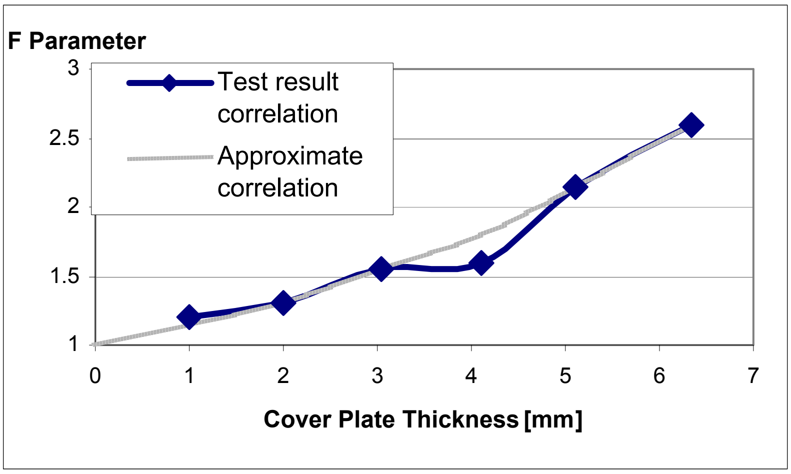


Figure 16: Detectable Crack Size Factor as a Function of Cover Plate Thickness

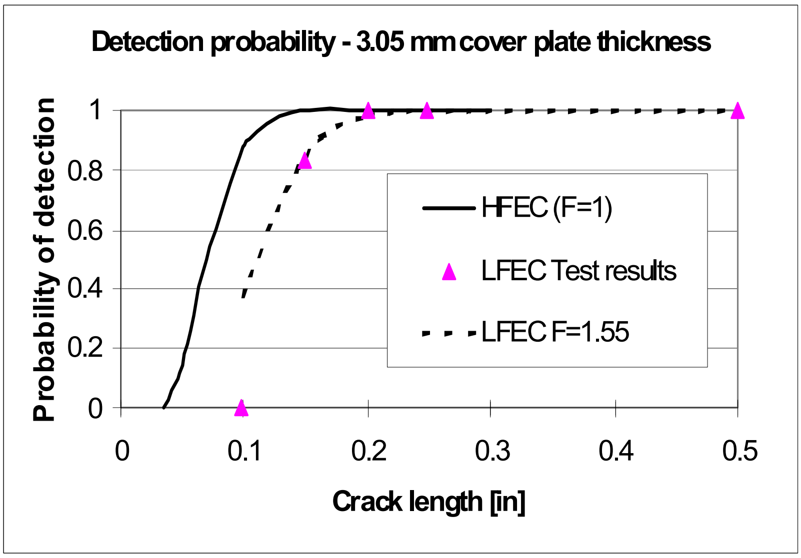


Figure 17: Typical Probability of Detection Result for LFEC Inspection

11.5.2 Evaluating Strategies for Minimizing Fatigue Failures (A. Brot, IAI)

The fatigue or damage-tolerance analyst needs to develop a strategy in order to minimize the probability of a fatigue failure in service. This can be achieved this by selecting suitable alloys, design stress levels, design features, target life and NDI methods. In many cases, a probabilistic analysis, which considers all aspects of the fatigue process, is needed in order to optimize the selection of options. This optimization process would not result from a conventional fatigue or damage-tolerance analysis, but only from a probabilistic simulation that considered all aspects of the fatigue process.

The ***INSIM*** computer program has been developed in order to simulate the *entire* fatigue environment that a structure must withstand. ***INSIM*** simulates, in a probabilistic manner, service life variation, service load severity, time to crack-initiation, crack growth history and NDI detection capability. There are three, mutually exclusive, outcomes of the fatigue process: The aircraft may reach the end of its operational life and be retired from service; a crack may be detected during scheduled maintenance operations; the structure fails in service. Crack initiation, crack growth and crack detection characteristics of a *specific structural location* are input into ***INSIM,*** which performs the simulation of the fatigue process for every aircraft in an large *virtual* fleet. Cracks initiate at various times and grow at variable rates in each aircraft. Inspections are performed according to a predetermined schedule. Cracks are detected during these inspections according to the statistical expectation of detection. As the simulation proceeds from aircraft to aircraft, cracks are detected, aircraft are retired from service or failures occur. The computer acts as a scorekeeper, amasses the statistics and summarizes the results. Based on these simulations, ***INSIM*** calculates the probability that failure has occurred.

The first of five examples considered in the study, is whether it is advantageous to design aircraft structures with single or multiple load-paths. ***INSIM*** simulations were used to study the benefits of multiple load-path design. The results indicate that the designer must weigh the benefit of a longer inspection interval against the increase in weight, cost and complexity for the dual load-path structure.

Present damage-tolerance methodology is ***not*** suited for selecting an optimum strategy for defining inspection thresholds (initial inspections); a probabilistic analysis is required in order to do so. A series of ***INSIM*** simulations were performed, the results clearly show that the inspection threshold can be delayed, beyond the usually accepted criteria, without compromising safety. Figure 18 shows the results of probabilistic simulations used to optimize the inspection threshold.

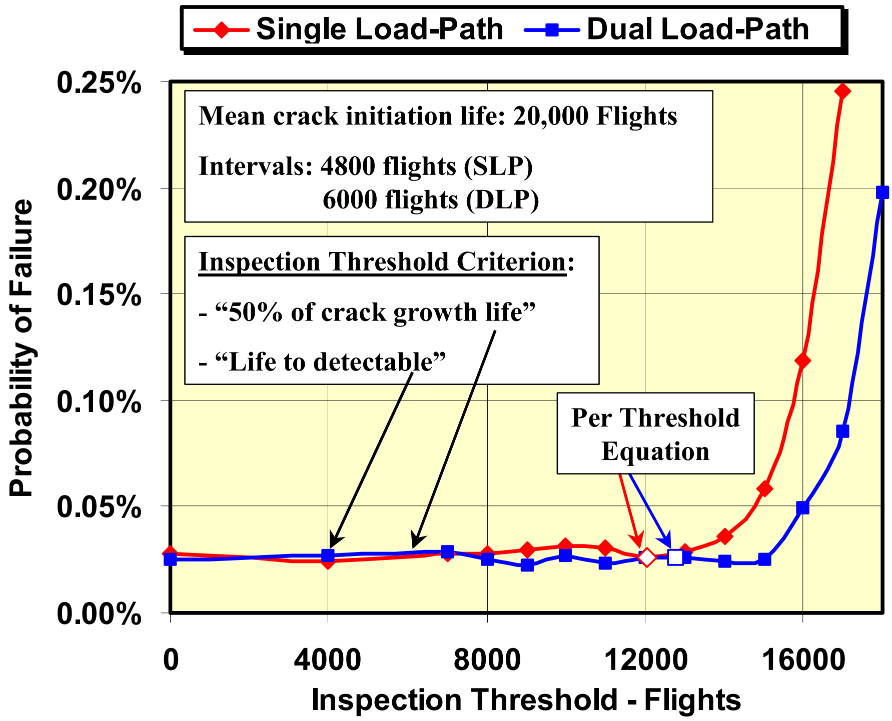


Figure 18: Effect of Inspection Threshold on the Probability of Failure

One of the important features of damage-tolerance methodology is that it provides a rational manner for determining inspection intervals. ***INSIM*** simulations were used to determine whether the inspection intervals determined by damage-tolerance methodology offer a constant level of safety. It was concluded that the damage-tolerance methodology does not always arrive at the optimum inspection interval.

A popular notion exists within the fatigue and damage-tolerance community, that aircraft designed to the damage-tolerance regulations are unaffected by increased service life (aging). Present damage-tolerance methodology is not equipped to evaluate the loss of safety as the aircraft ages. ***INSIM*** simulations clearly demonstrate that, as the fleet ages, the probability of failure increases, even though the inspection interval was selected on the basis of damage-tolerance criteria. Only a probabilistic analysis, such as that performed by ***INSIM***, will disclose the aging effect, and help define the required "aging program".

When an aircraft structure is shown to have an insufficient fatigue life, two approaches can be followed: frequent periodic inspections or a redesigned part (terminating action). Manufacturers and operators often prefer the first approach since the cost of designing, manufacturing and purchasing the improved part is eliminated. ***INSIM*** simulations were used to evaluate both approaches. This example illustrates how probabilistic analyses can be used by the manufacturer, operator or certifying authority to evaluate the relative benefits of periodic inspections vs. a terminating action.

The results of this study were presented at the ASIP 2004 Conference [13] and at the 45th Israel Conference on Aerospace Sciences [14].

11.5.3 Risk Analysis for IAF F-16 Wing Attachment Fitting (R. Halevi, IAF)

Traditional force management planning uses deterministic durability and damage-tolerance methods to plan aircraft maintenance. While deterministic methods generally provide an adequate margin-of-safety and are easy to implement, they do not quantify risk level or predict when an actual repair is needed. By using stochastic methods to define the relationship between risk level and maintenance schedule, methods can be established to more accurately plan and allocate maintenance resources.

Lockheed Martin has developed a software program identified as PC-based Structural Tracking Risk Assessment Procedure (PCSTRAP). PCSTRAP is used to perform tailored structural risk assessments, stochastic structural maintenance planning and structural maintenance scenario comparisons, based on Individual Aircraft Tracking (IAT) data and analysis results from a variety of aircraft tracking programs. IAF funding was used to upgrade the software and to tailor it to IAF requirements.

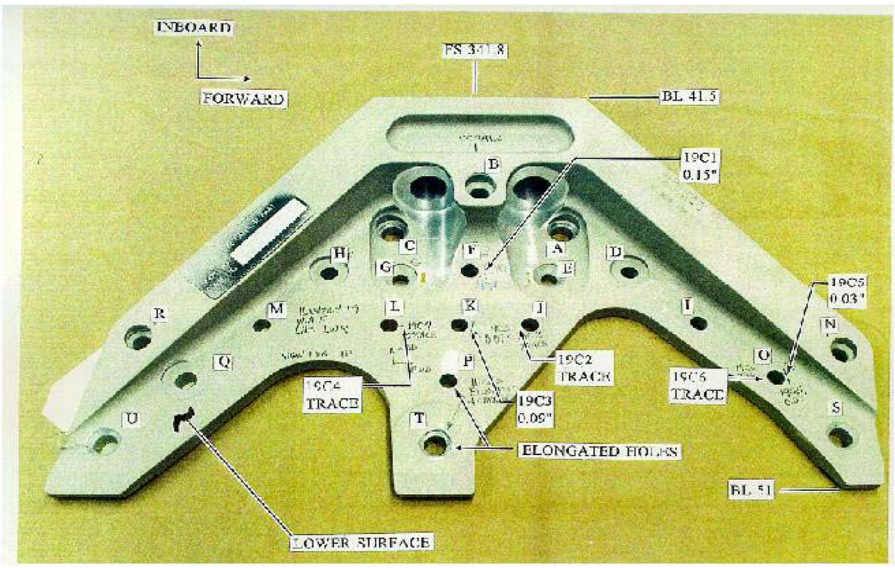


Figure 19: F-16 Wing Attachment Fitting Cracks

The F-16 has sixteen Wing Attachment Fittings (WAF) on every aircraft (8 per wing – 4 upper and 4 lower). The fitting transfers the loads between the wings and the fuselage main bulkheads. The IAF (and other users) have found cracks in almost every one of its Block 30 aircraft fittings, as is shown in Figure 19. Using deterministic methods, the IAF implemented a damage-tolerance analysis (DTA) based maintenance policy and initiated a WAF replacement line. In order to evaluate the maintenance policy and plan the replacement line, a risk assessment was performed using PCSTRAP. The DTA crack growth curves, which were calibrated to fleet findings and the IAT data, gathered throughout 20 years of usage, were used to generate the fleet usage variation. Analysis results supported a maintenance policy change, which helped keep high aircraft availability and safety. A typical result of the risk analysis is shown in Figure 20. The results of this study were presented at the ASIP 2004 Conference [15] and at the 45th Israel Conference on Aerospace Sciences [16].

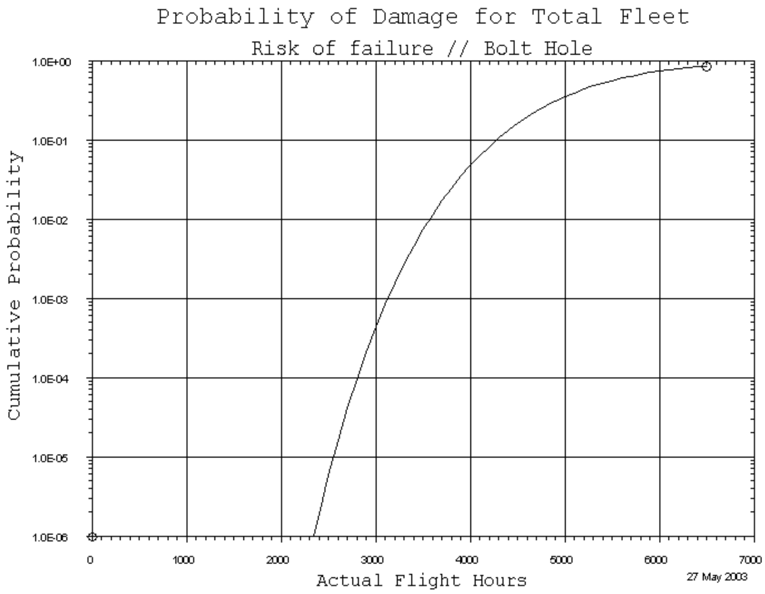


Figure 20: Typical Results Resulting From the Risk-Analysis Study

11.6 MISCELLANEOUS

11.6.1 Computation of Stress-Intensity Functions in the Neighborhood of Edges (Z. Yosibash, Ben-Gurion University)

The computation of stress-intensity functions in the neighborhood of edges in a three-dimensional linear elastic body is of major importance in engineering practice, and some methods for extracting these from finite element solutions have been earlier proposed. However, a detailed mathematical framework of the 3-D edge singularities seems not to be available, and the methods are not the most efficient and accurate due to the need of extracting the Edge Flux Intensity Functions (EFIFs) very close to the edges.

Towards developing efficient and accurate methods, we first show in [19] that the commonly used J-integral is path dependent in 3-D. In [17] we propose a general framework of a very efficient extraction of Edge Flux/Stress Intensity Functions along edges by using the "quasi-dual function method". It is proven mathematically based on generalization of the dual-singular extraction method in 2-D domains (or the so called Contour Integral Method), and in [18] its application to the general scalar elliptic problem in conjunction with the p-version of the finite element method is provided. The application of this very efficient and accurate method for elasticity problems is very promising, and numerical evidence will be provided shortly.

11.6.2 Solutions of Linear Elastostatic Problems in the Vicinity of Reentrant Corners or Multi-Material Interfaces (Z. Yosibash, Ben-Gurion University)

Another research project reported in [20] and [21] is directed to solutions of linear elastostatic problems in the vicinity of reentrant corners, or multi-material interfaces, and the formulation of a failure criterion for these problems.

The validity of three ‘best" failure criteria proposed in the last decade for prediction of failure initiation at V-notch sharp tips has been examined and compared with experimental observations, and a new and simplified one is proposed in [21]. The experiments include: loading of specimens made of two kinds of elastic materials (PMMA - a polymer, and Al_2O_3-7%ZrO_2, a composite ceramic) in three and four-points, in order to produce mode I stress field in the vicinity of the notch tip. To quantify the influence of V-notch tip radius on the failure initiation, specimens having different tip radii have been selected.

All failure criteria assume a mathematical sharp tip, however a small blunt tip (with a tip radius denoted by ρ) shows a higher generalized stress-intensity factor as compared to the predicted values.

Nevertheless, both the Novoshilov-Seweryn and Leguillon criteria seem to predict well the observed failures, however, as the opening angle increases, their validity deteriorates. Leguillon criterion outperforms the Novoshilov-Seweryn criterion, and it has been refined to include ρ dependency in order to match better the experimental observations - see [20]. Generalization of the Leguillon and the new failure law to mixed mode fracture is under investigation.

11.6.3 Structural Airworthiness Assurance of Historical Aircraft (C. Rodberg and M. Myara, IAF)

Historical aircraft, kept in flying condition and operated by the IAF on behalf of the IAF Museum, include the Supermarine Spitfire Mk. IX (Figure 21) and the Stearman PT-17 (Figure 22). These aircraft are more than 60 years old, and the IAF found it difficult to ascertain the continued airworthiness of the aircraft from a structural point of view. Historical aircraft operating in the U.S are certified by the FAA under the "experimental" category, and therefore their continued airworthiness is the responsibility of the aircraft operator.

Technical manuals were located, which were used when the aircraft was in regular service in the IAF, but these manuals do not take into account the aging of the aircraft structure.

Moreover, since the subject aircraft are from an era in which structural fatigue was not a design consideration, and in which NDI techniques were practically non-existent, no engineering information specific to these airplanes exist in these fields. Also, some NDI techniques require specific material identification, and some also require calibration specimens, that were not available.

After being tasked with ascertaining the structural condition of these historical aircraft, the IAF Depot Engineering Division tackled the issue in a methodical and thorough way. Structural data on both aircraft were gathered, critical structural components were identified, the structural materials were identified and confirmed by conductivity testing, and non-destructive testing procedures for these components were developed.

For the Spitfire Mk IX, a mostly metallic aircraft, X-ray radiography proved to be a particularly useful tool in looking for possible corrosion, cracks or material flaws. In ferromagnetic parts, like the main landing gear support eyebolts and the main landing gear axles, magnetic particle inspection was used. When components could be dismantled from the aircraft (like control surfaces), real-time radioscopy was employed at the IAF Depot NDT lab. Ultrasonic inspection was also used to check for possible cracks in the engine support truss attachment bolts.

Figure 21: Spitfire Mk IX (S/N 057) in Flight

Figure 22: Stearman PT-17 (S/N 031) Undergoing NDI

The Stearman PT-17 has wooden wings covered with fabric. This posed a challenge to aerospace engineers groomed in an era of all-metal and composite aircraft, and forced the IAF Depot Engineering Division engineers to learn new inspection techniques for wooden aircraft structures. The inspection of the wooden structure on the Stearman's wings was performed using a video-borosope, which allowed the inspectors to inspect the inner wing structure without removing the fabric skin. Bolts attaching main structural components like MLG-to-fuselage, upper to lower wing attachment trusses, and the horizontal stabilizer were checked for cracks using ultrasonic inspection.

Based on these techniques a complete non-destructive testing handbook was written for the Spitfire Mk. IX aircraft, and specific NDI inspections were developed for the Stearman PT-17. After these procedures were implemented and results analyzed, the condition of the aircraft structure was determined with a reasonable degree of confidence. Based on these results, IAF Headquarters authorized the continued flying of these historical aircraft.

Further activities in this field will involve the use of NDI to ascertain the structural condition of a third aircraft, a Harvard trainer from WWII vintage, that, although preserved in a flying condition, has not actually flown for quite a long time.

The results of this study were presented at the 45th Israel Conference on Aerospace Sciences [22].

11.6.4 Fracture Mechanics of Bonded Joints (L. Banks-Sills, Tel-Aviv University)

The previous Israel National Review [1] reported about work being performed at Tel-Aviv University on the subject of bonded joints. During the last two years, investigation of the behavior of cracks along an interface between two isotropic and two fiber reinforced (transversely isotropic) materials have continued. In addition, we have considered bimaterial notches, cracks in homogeneous, anisotropic material and modeling failure of metallic glasses as a result of hydrogen embrittlement.

In one study [23], the conservative *M*-integral was extended to treat thermal-elastic, mixed mode problems. With it, stress intensity factors may be obtained for cracks in homogeneous, isotropic materials, as well as isotropic and anisotropic, bimaterials. Excellent agreement was found between results determined in this study and those found in the literature. In addition, new results were obtained for interface cracks for a wide range of material properties and for a delamination in a composite material.

Another conservative integral, based on the Betti reciprocal principle, was developed to obtain stress intensity factors for a bimaterial notch in which the body is subjected to a thermal load [24]. The bonded materials are linear elastic, isotropic and homogeneous. According to the linear theory of elasticity, stresses in the neighborhood of the notch tip are generally singular as a result of the mismatch of the elastic constants. Eigenvalues and eigenfunctions depend upon the mechanical properties and wedge angles. They may be real, complex or power-logarithmic. Real and complex eigenvalues were considered in this study. The stress intensity factor represents the amplitude of the stress singularity and depends upon material properties, geometry and load or temperature. Because of the highly singular behavior of one of the integrals that is part of the conservative integral, the former was carried out by a hybrid analytical/numerical scheme. The finite-element method was employed to obtain displacements caused by the temperature distribution in the body. The conservative integral was applied to several problems appearing in the literature. Both good agreement between those results and the ones obtained in this study, as well as path stability for all problems were attained.

Another study [25] described here is that of a model which was developed to describe the expansion of high-pressure hydrogen bubbles and propagation of cracks between them in the absence of external loads. The focus was on cracks that form during electrochemical hydrogen charging of amorphous $Fe_{80}B_{11}Si_9$ ribbons. A coupled diffusion/fracture mechanics approach was developed, allowing determination of the time to failure. Finite element analyses were carried out to determine the values of the stress intensity factor for cracks of different lengths, assuming linear elasticity. In addition, the volume of a bubble with edge cracks was related to the internal pressure. The relation between critical pressure and crack length was obtained from an appropriate value of the fracture toughness. A criterion was proposed to obtain the pressure and volume as a function of the number of hydrogen moles within the bubble with edge cracks. Finite element analyses were also used to calculate the hydrogen concentration and hydrogen diffusion flux as a function of crack length and time. The time to failure as predicted from this model is of the same order of magnitude as that observed experimentally.

In another study [26], the elastic two-dimensional problem of contact with friction was formulated. Two-dimensional equilibrium equations and boundary conditions in an orthogonal curvilinear co-ordinate system were written explicitly. The above formulation was solved with the aid of the finite-difference technique. An iterative algorithm, which does not require load increments, was employed for solving interface fracture problems with contact and friction subjected to a monotonically increasing load. The *J*-integral was extended for problems in which there is friction along the crack faces. Stress intensity factors were calculated by means of the integral, as well as an asymptotic expansion of the tangential shift. Two problems were analyzed: (1) a crack in homogeneous material in the presence of friction involving stationary contact and (2) an interface crack in the presence of friction involving receding contact. Results were compared to those found by analytical and semianalytical methods which are presented in the literature, as well as to those obtained by means of the finite element method. The accuracy of the results established the reliability of the finite difference analysis, as well as the post-processors. In addition, a problem involving stick conditions was considered. It was observed that with increasing friction, the normal gaps and tangential shifts decreased. The size of the contact zone increased and values of the stress intensity factor decreased.

Another crack problem, that in general anisotropic material under LEFM conditions was considered. In Part I [27], three methods were presented for calculating stress intensity factors for various anisotropic materials in which $z = 0$ is a plane of symmetry. All of the methods employ the displacement field obtained by means of the finite element method. The first one is known as displacement extrapolation and requires the values of the crack face displacements. The other two are conservative integrals based upon the J-integral. One employs symmetric and asymmetric fields to separate the mode I and II stress intensity factors. The second is the M-integral, which also allows for calculation of K_I and K_{II} separately. All of these methods were originally presented for isotopic materials. Displacement extrapolation and the M-integral were extended for orthotropic and monoclinic materials, whereas the J_I and J_{II} -integrals were only extended for orthotropic material in which the crack and material directions coincide. Results were obtained by these methods for several problems appearing in the literature. Good to excellent agreement was found in comparison to published values. New results were obtained for several problems. In particular, results were obtained for isotropic, cubic and orthotropic material. It may be observed that the stress intensity factors for the cubic and isotropic material with similar mechanical properties are quite similar. For the orthotropic material where $E_{22}/E_{11} = 10$, the normalized K values differ from those obtained for the other materials. In Part II, the M-integral will be extended for more general anisotropies. In these cases, three-dimensional problems must be solved, requiring a three-dimensional M-integral.

Another project, which we have been working on for several years, involves predicting delaminations in laminate composites. Two interfaces are being considered: 0°/90° and ±45°. Experiments have been carried out to determine the delamination toughness for a crack along the interface between two transversely isotropic materials in 0°/90°directions [28]. The material chosen for study consists of carbon fibers embedded within an epoxy matrix. Brazilian disk specimens shown in Figures 23a and 23b were employed in the testing. Finite element analyses were carried out to determine stress intensity factors arising from the applied load. Residual stresses resulting from the curing process create transverse cracks in the 90° layer. Stress intensity factors pertaining to this stress field were obtained, as well. These stress intensity factors were superposed with those from the applied load to obtain the total stress intensity factor. From the load at fracture, the critical interface energy release rate G_{ic} as a function of phase angle ψ was determined, and results were compared to a fracture criterion as shown in Figure 24. Two lay-ups of the carbon fiber/epoxy material were examined.

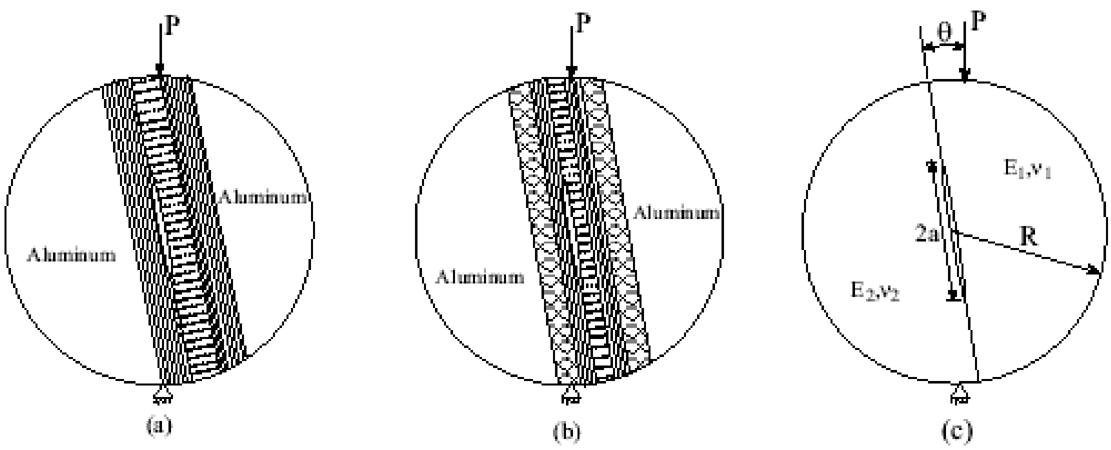


Figure 23: Brazilian Disk Specimens
Composite Specimens containing (a) 0°/90°/0° strip and
(b) with ±45° outer stiffening layers. (c) Original bimaterial specimen

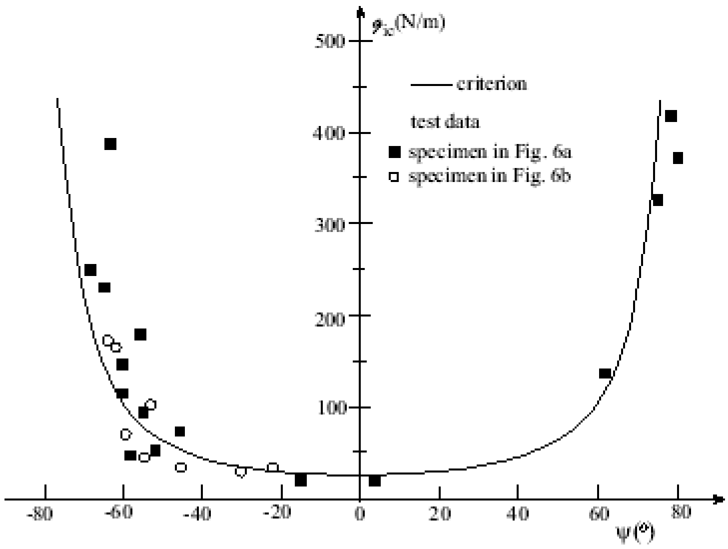


Figure 24: Delamination toughness G_{ic} along a 0°/90° interface as a function of mode mixity ψ (L = 100μm) for graphite/epoxy (AS4/3502).

Continuing with the ±45° interface, a study [29] was conducted to analytically determine the first term of the asymptotic displacement and stress fields for a straight through crack along this interface. The upper and lower materials are transversely isotropic. Since with this configuration, there is full coupling between the modes, this problem requires a three-dimensional treatment. To calculate stress intensity factors, a three-dimensional *M*-integral was derived using the asymptotic fields as auxiliary solutions. The displacement extrapolation method was derived, as well, and used to check the results obtained by the *M*-integral. Two numerical test cases were considered to examine the accuracy of both methods. Results obtained for other mechanical problems were presented, as well. Work is continuing with experiments, together with development of a thermal *M*-integral to treat the residual curing stresses.

11.7 REFERENCES

1. Brot, A., "Review of Aeronautical Fatigue Investigations in Israel, April 2001 – March 2003", *Minutes of the 28th ICAF Conference,* Lucerne, Switzerland, 2003.

2. Newman, J. C., Jr., Brot, A., and Matias, C., "Improved Crack-Growth Predictions in 7075-T7351 Aluminum Alloy Under Various Aircraft Load Spectra", *Proceedings of the Seventh Joint DoD/FAA/NASA Conference on Aging Aircraft*, New Orleans, USA, 2003.

3. Newman, J. C., Jr., Brot, A., and Matias, C., "Crack-Growth Calculations in 7075-T7351 Aluminum Alloy Under Various Load Spectra Using an Improved Crack-Closure Model", *Engineering Fracture Mechanics*, Vol. 71, 2004.

4. Everett, R. A., Jr., Matthews, W. T., et al, "The Effects of Shot and Laser Peening on Crack Growth and Fatigue Life in 2024 Aluminum and 4340 Steel", *Proceedings of the USAF Aircraft Structural Integrity Program Conference*, 2000.

5. Matias, C. "Investigation of Two Methods for Increasing Inspection Intervals for Highly Loaded, Fatigue-Critical, Aluminum Structures", *Proceedings of the 23rd ICAF Symposium*, 2005.

6. Altus, E., Size Effect of Microdamage Growth and its Relation to Macro Fatigue Life", *Journal of ASTM International*, Vol. 2, No. 6, 2005.

7. Kressel, I., Cohen, I. et al, "Design Substantiation and Application of a Boron-Epoxy Patch Repair on an F-16 Upper Fuselage Skin", Proceedings of the 22nd ICAF Symposium, 2003.

8. Harris B., "A Parametric Constant-Life Model for Prediction of the Fatigue Lives of Fibre-Reinforced Plastics", *Fatigue in Composites*, Edited by B. Harris, CRC 2003.

9. Granot, Z. et al, "Predicting Fatigue Life and Residual Strength of Composite Material Structures", *Proceedings of the 45th Israel Annual Conference on Aerospace Sciences*, 2005.

10. Granot, Z., "Simplified Methods for Predicting Fatigue Life of Composite Material Structures", *Proceedings of the 23rd ICAF Symposium*, 2005.

11. Kressel, I., Damage Tolerance of Bonded Composite Repairs – Analysis and Tests", *Proceedings of the 23rd ICAF Symposium*, 2005.

12. Matias, C., Brot, A. and Senderovits W., "Increasing Fatigue Crack Detection Inspection Intervals by Means of Low-Frequency Eddy-Current Non-Destructive Inspections", *Proceedings of the 45th Israel Annual Conference on Aerospace Sciences*, 2005.

13. Brot, A., "Evaluating Strategies for Minimizing Fatigue Failures in Metallic Structures", *Proceedings of the USAF Aircraft Structural Integrity Program,* 2004.

14. Brot, A., "Using Probabilistic Simulations in Order to Minimize Fatigue Failures in Metallic Structures", *Proceedings of the 45th Israel Annual Conference on Aerospace Sciences*, 2005.

15. Halevi, R. et al, "Risk Analysis for IAF F-16 Wing Attach Fitting", *Proceedings of the USAF Aircraft Structural Integrity Program,* 2004.

16. Halevi, R. et al, "Risk Analysis for IAF F-16 Wing Attach Fitting (WAF)", *Proceedings of the 45th Israel Annual Conference on Aerospace Sciences*, 2005.

17. Costabel M., Dauge, M. and Yosibash Z., "A Quasidual Function Method for the Computation of Stress Intensity Factors Along Edges", *SIAM Journal on Mathematical Analysis*, **35**, No 5, 2004.

18. Omer N., Yosibash Z, Costabel M. and Dauge, M., "Edge Flux Intensity Functions in Polyhedral Domains and their Extraction by a Quasidual Function Method", *International Journal of Fracture*, **129**, 2004.

19. Omer N., Yosibash Z, "The J-Integral in 3-D IS path Dependent ", Submitted for publication, 2005.

20. Leguillon D. and Yosibash Z., 'Crack Onset at a V-Notch. Influence of the Notch Tip Radius", International Journal of Fracture, 122, 2003.

21. Yosibash Z., Bussiba A. and Gilad, I., ``Failure Criteria for Brittle Elastic Materials", International Journal of Fracture, 125, No 3-4, 2004.

22. Rodberg, C. and Myara, M., "Structural Airworthiness Assurance of Historical Aircraft", *Proceedings of the 45th Israel Annual Conference on Aerospace Sciences*, 2005.

23. Banks-Sills, L., and Dolev, O., "The Conservative M-Integral for Thermal-Elastic Problems", *International Journal of Fracture*, 125, 2004.

24. Banks-Sills, L. and Ishbir, C., "A Conservative Integral for Bimaterial Notches Subjected to Thermal Stresses", *International Journal for Numerical Methods in Engineering,* 60, 2004.

25. Eliaz, N., Banks-Sills, L. et al, "Modeling Failure of Metallic Glasses due to Hydrogen Embrittlement in the Absence of External Loads", *Acta Materialia*, 52, 2004.

26. Dorogoy, A., and Banks-Sills, L., "Shear Loaded Interface Crack Under the Influence of Friction – a Finite-Difference Solution*", International Journal for Numerical Methods in Engineering,* 59, 2004.

27. Banks-Sills, L., Ingraffea, A. R., et al, “Methods for Calculating Stress Intensity Factors in Anisotropic Materials: part I”, to appear in: *Engineering Fracture Mechanics*.

28. Banks-Sills, L., Boniface, V. and Eliasi, R., “Development of a Methodology for Determination of Interface Fracture Toughness of Laminate Composites – the 0°/90° Pair”, *International Journal of Solids and Structures*, 42, 2005.

29. Freed, Y. and Banks-Sills, L., “A Through Interface Crack Between a ±45° Transversely Isotropic Pair of Materials”, to appear in*: International Journal of Fracture*.