REVIEW OF CANADIAN AERONAUTICAL FATIGUE WORK 1999-2001

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SUMMARY

This paper provides a review of Canadian work associated with fatigue of aeronautical structures during the period 1999 - 2001. All aspects of structural technology are covered including full-scale tests, loads monitoring, fracture mechanics, composite materials and engine fatigue.

Organisation Abbreviations Used in Text:
AVRS - Air Vehicles Research Section
AMRL - Aeronautical and Maritime Research Laboratory (Australia)
BARA - Bombardier Aerospace Regional Aircraft
BADS - Bombardier Aerospace Defence Services
BGF - BF Goodrich
CF - Canadian Forces
DGAETM (FT) - Director General Air Equipment Technical Management (Fighters and Trainers)
DND - Department of National Defence
DRDC – Defence Research and Development Canada
DTA - Directorate of Technical Airworthiness
IAR - Institute for Aerospace Research
IMI – Industrial Material Institute
NRC - National Research Council of Canada
QETE - Quality Engineering Test Establishment
RAAF - Royal Australian Air Force
RMC - Royal Military College of Canada
SMPL – Structures, Materials and Propulsion Laboratory
UTIAS - University of Toronto Institute for Aerospace Studies

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INTRODUCTION

Canadian industry, universities and government agencies were solicited for information describing their fatigue technology related activities over the period 1999 to 2001. This review covers work performed or being performed by the following organizations:

Acsion Industries

Bombardier Aerospace
- Defence Services
- Toronto Site

Department of National Defence (DND)
- Canadian Forces (CF)
- Director for Technical Airworthiness (DTA)
- Quality Engineering Test Establishment (QETE)
- Air Vehicles Research Section (AVRS)
- Royal Military College (RMC)
- Director General Air Equipment Technical Management (Fighters and Trainers) (DGAEPM(FT))

IMP Group Limited
- IMP Aerospace

Indal Technologies Inc.

National Research Council of Canada
- Institute for Aerospace Research (IAR/NRC)

BF Goodrich (BGF)
- Landing Gear Division

Messier-Dowty Inc.

Spar Aerospace Limited

Carleton University, Department of Aerospace and Mechanical Engineering

University of Toronto Institute for Aerospace Studies (UTIAS)

Names of contributors and their organizations are included in the text of this review.

Full addresses of the contributors are available through the Canadian National ICAF Delegate at:

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FULL-SCALE TESTING

F/A-18 International Follow-On Structural Program (IFOSTP) Center Fuselage Test

J. Dubuc, Bombardier

Bombardier has been contracted by the Canadian Forces (CF) to perform a full-scale fatigue test on the F/A-18 (Canadian Forces designation CF188) Center Fuselage. The original objective of the test was to demonstrate the economical life of the F/A-18 aircraft Center Fuselage under a CF and Royal Australian Air Force (RAAF) representative spectrum. This objective has lately been reduced to the demonstration of the original design life of 6000 flying hours. The fatigue cycling of the test article called FT55 started in May 1995. In May 2001 the test article has reached its test goal of 15,800 hours (15,100 spectrum flight hours plus 700 SFH of flying before retirement from fleet). A residual strength test in accordance with the DEF-STAN -970 standard, followed by a teardown inspection is being planned.

In accordance with the CF Lifing Policy for the F/A-18, the milestone of 15,000 hours represents the certification of most of the components on the Center Fuselage. The 12,000-hour milestone had certified all the secondary structure as well as the durability critical areas exhibiting good inspectability. Following a very extensive inspection program at 15,000 hours, FT55 has now demonstrated the original design life of the remaining of the Center Fuselage structure. This inspection program comprises approximately 300 cards including a visual inspection of 100% of the test article. All known critical areas are inspected using NDI techniques such as eddy current and ultrasound.

Several failures were discovered since the 13,050 hours landmark, in March 1999. The most significant ones were the:

Bulkhead web taper line at Fuselage Station 453;
Bulkhead Pockets below the control hole at Fuselage Stations 470 and 488;
Bulkhead Pockets below the upper lug at Fuselage Stations 470 and 488;
Bulkhead at Fuselage Station Y488 and B.L. 20 and;
Upper Outboard Longeron near Fuselage Station 490

Note that these last three failures were never seen in the previous full-scale fatigue tests done by the aircraft OEM, Boeing (formerly McDonnell Douglas). It has been established, through numerous analyses and coupon test programs, that FT55 has reached the equivalent severity of the OEM center fuselage test (ST16) between the 12,000 and 13,000 hours landmark and thus failures not observed in OEM test could now be expected. On FT55, a bonded graphite epoxy patch on both faces of the bulkhead repaired the first failure whereas shot peening is planned for the fleet. Extensive 3D FE modeling was required to develop this repair.

The residual strength test is planned for March 2002 after extensive inspection of the aircraft and the re-enforcement of the critical areas representing a risk for the test. The teardown inspection is scheduled for January 2003.

As part of IFOSTP, Bombardier also participates in the Wing (FT245) and Aft Fuselage and Empennage (FT46) full-scale fatigue tests which are performed respectively by the IAR/NRC and AMRL in Australia. This participation now consists mainly in developing repairs and modifications for these two test articles.

F/A-18 Wing Fatigue Test – IAR/CF/Bombardier/RAAF

R.L. Hewitt, SMPL, IAR, NRC

The set up and development of the FT245 wing fatigue test was described in earlier ICAF reviews. The derived spectrum consisted of about 175,000 load lines in a block, 135,000 of which were independent. Because of control system limitations and the computation time required to calculate actuator loads for so many load cases, it was necessary to reduce the number of unique load lines to less than 50,000. Similar loads were grouped into bins and a single representative load case from each bin used to represent all the load cases in a bin. It was demonstrated that this had no significant effect on fatigue life for the twelve major loads on the wing. The final test spectrum then contained about 135,000 lines.

The first loads were applied to the test article in November 2000 without incident. The loads and test loading system were validated by calculating and applying some load cases from a series of validation flights for which some sixty strain gauge outputs had been measured. Agreement between test measured strains and flight measured strains was excellent.

Fatigue cycling commenced on Australia Day (January 26) 2001 and again was uneventful. Strain data from about 600 gauges was collected at every end point during the first block to provide strain spectra for analysis. Thereafter, it is really only necessary to record data when an output changes for a given applied load case because this signifies either a problem with the gauge, the loading or the structure. Conventionally, this is done by analyzing end point data off-line, but because this is very difficult with such large amounts of data, it is often only done after a problem is encountered. A new trend monitoring system was therefore developed [1] for this test that monitors strain at particular end points in the spectrum and checks for changes. The system allows limit checking on a very large number of different end points with individual limits for every data channel assigned to each end point. Thus data can be checked on any load condition that is critical for any section of the structure. The system also automatically plots recent trend data for any limit exceedence.

Another type of data that is often requested is load data at every end point so that the analyst or certifying authority can check that every load condition has been applied correctly. Again, the problem is the extensive storage facilities required and the difficulty of checking the data. In addition, checking the load levels at some nominal end point does not ensure that the loads were not exceeded just before or just after the recording point. It is therefore more appropriate to develop on line end level verification and only record selected end points in conjunction with the data checking. A new software system has therefore been developed [1] to check the loads throughout the loading cycle. This system notifies the operator whenever an end point is missed and keeps track of all missed end points.

As of May 31, 2001, the test has accumulated 1467 flight hours. The test is shown in Figure 1.

Figure 1. Overall View of FT-245 Wing Fatigue Test.

Recent Fatigue and Damage Tolerance Work at the Toronto Site of Bombardier Aerospace

B. Leigh, Bombardier Aerospace

Over the past two years, fatigue and damage tolerance work at Toronto site has largely consisted of certification analysis and fatigue tests for the recently introduced DHC-8 Series 400 aircraft. Although it is a derivative of the earlier Series 100 and 300 aircraft, certain changes in design and local increases in loads have made this a challenging task. On the other hand, it has been possible to use essentially the same analysis methods as those used on the earlier Dash 8 models. Component fatigue tests have either been completed or are being conducted on the nacelle, (Figure 2), horizontal stabilizer (Figure 3), and the inboard and outboard flaps (Figure 4). A complete aircraft fatigue test is also under way (Figure 5). Detail tests for various smaller components, for which analytical damage tolerance methods are considered to be inadequate, are under way or have been completed. A component fatigue test of the horizontal stabilizer of the Series 700 Canadair Regional Jet is being conducted at the Toronto site in support the Montreal site's certification program for that aircraft (Figure 6).

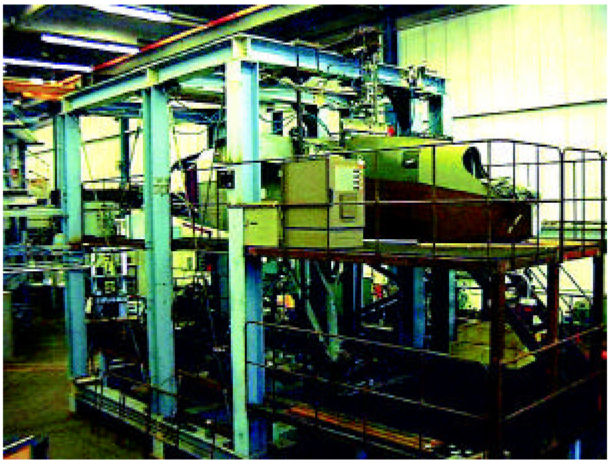


Figure 2. Dash 8-400 nacelle fatigue test.

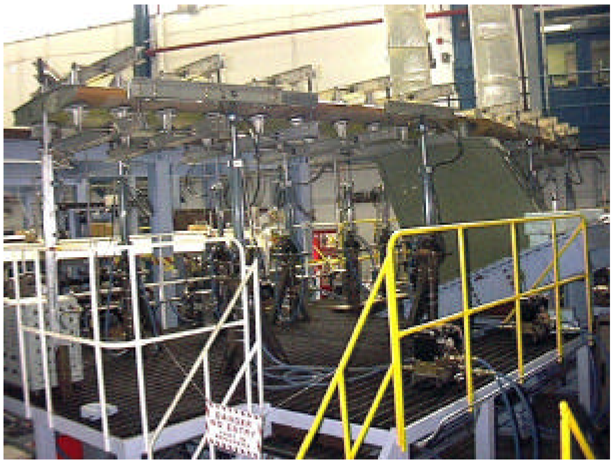


Figure 3. Dash 8-400 horizontal stabilizer fatigue test.

(a) (b)

Figure 4. Dash 8-400 flap inboard (a) and outboard (b) fatigue tests.

The Toronto Site of Bombardier Aerospace is currently preparing for full scale fatigue and damage tolerance barrel tests of Fibre Metal Laminate fuselage panels.

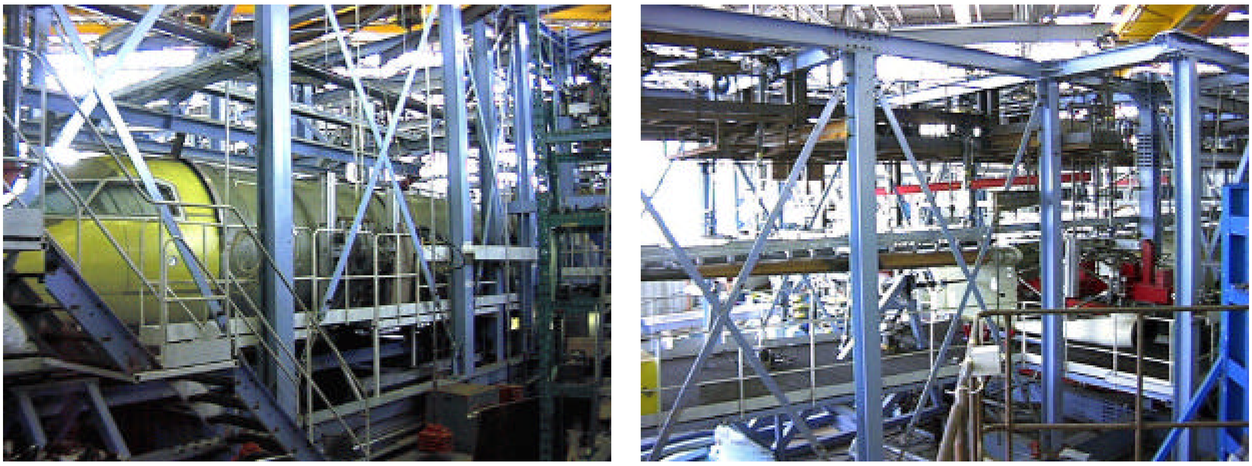


Figure 5. Dash 8-400 full-scale fatigue test.

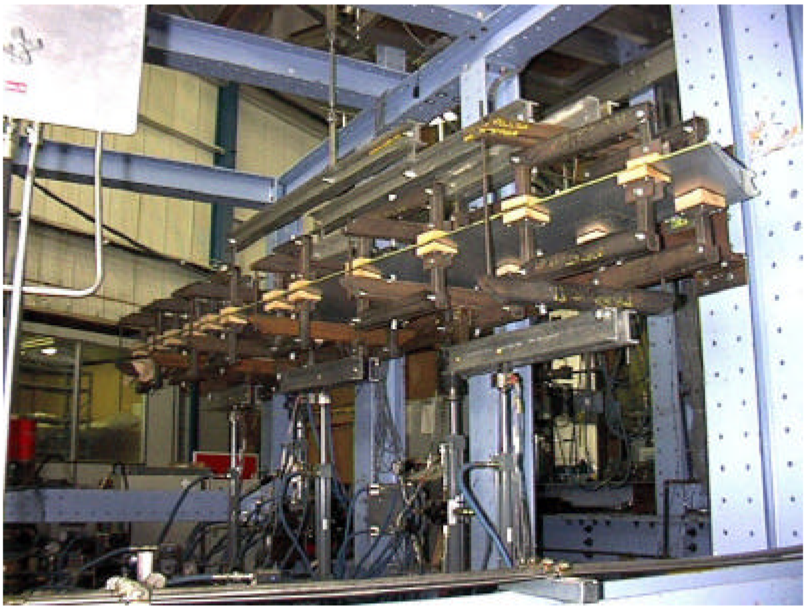


Figure 6. CRJ-700 horizontal stabilizer component fatgue test.

Structural Fatigue Testing Technology - IAR/MTS Systems Corporation

R.L. Hewitt, SMPL, IAR, NRC

Modern full-scale aircraft structural fatigue tests are extremely complex from a control systems viewpoint. There are usually a large number of actuators with significant interactions between them and control is made more difficult because the load cells usually move with the actuators. Work has been underway at IAR for a number of years under a contract to MTS Systems Corporation, Minneapolis, to develop a model of a full-scale structural test system. The aim of this work is to better understand the complete structural test system and thereby reduce both the set-up and operation time for a full-scale test. Certain elements of this program have recently been released for publication and are described below.

State space models of a hydraulic actuator, servo-valve, load cell and structure have been developed and combined into a single state space model. The model has been experimentally verified for a single channel system by a comparison of frequency responses for a number of actuator/servo-valve/structure combinations. This work is reported in [2] and an example of predicted versus measured response is shown in Figure 7. The model should be useful for test engineers as a tool for improving their understanding of the systems they must control and for estimating the relative stability of systems with different components. A multi-channel model has been developed using modal analysis concepts. This model uses the outputs from a dynamic FE analysis to characterize the structure. Experimental verification of this model is planned.

The structural model has been used to investigate dynamic strain errors in structural fatigue testing. It has been shown [3] that significant dynamic strains can be developed in a structural test specimen when the test cycling frequency is above about 10% of the first natural frequency of the specimen. Since the loading in a full scale fatigue test is normally assumed as quasi-static, with associated static strain levels, these dynamic strains can lead to over-testing. The mass of the fixturing used to apply the loads increases the dynamic strains and the situation is made worse by static counterbalancing. It was recommended that fixture mass be kept to a minimum and that all tare loads and counterbalancing be done via active systems.

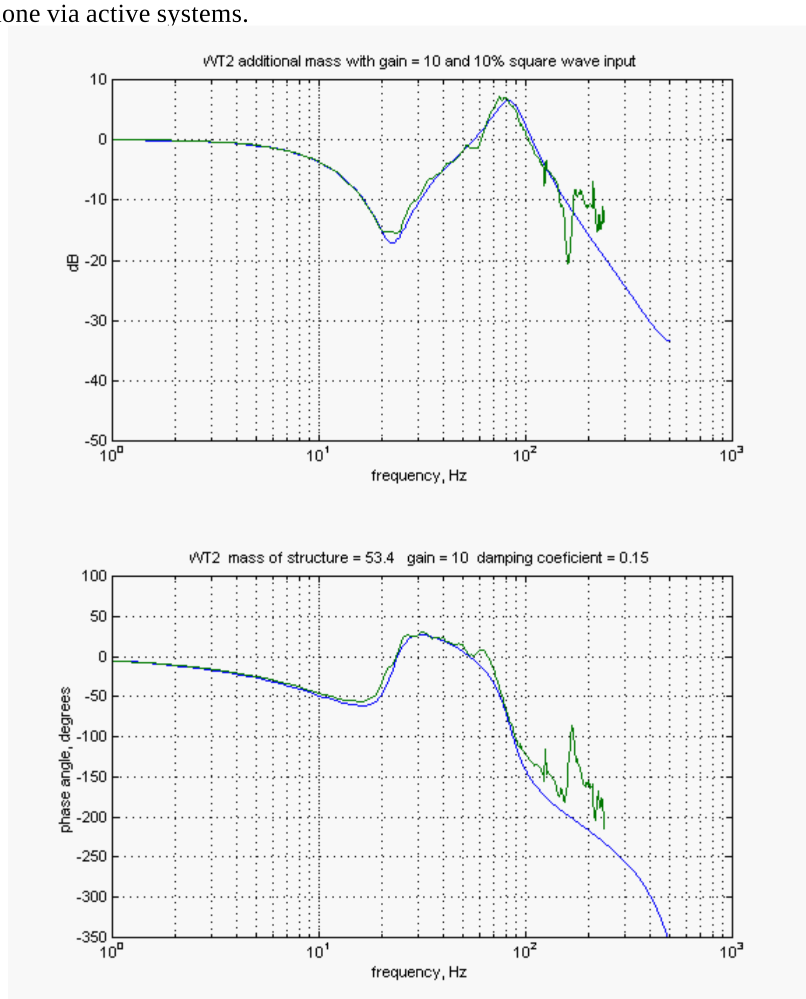


Figure 7. Comparison of Measured and Predicted Frequency Response of a Single Channel System.

Further work [4] investigated a method by which these models could be interfaced with a structural test controller using real time processors to allow a test engineer or operator to run a virtual test prior to assembling test hardware. It was demonstrated that it is possible to simulate the response of a cantilever beam with a servo hydraulic actuator at the tip and mid span in real time using a very simple processor. Representing the beam with four modes resulted in a total of 16 states for the system. Execution times for this system were less than 100 μs while stable solutions could be achieved with integration step sizes of greater than 1 ms. It therefore appears feasible to simulate more complex systems with this simple processor. Very large systems will require faster processors working in parallel.

Large integration step sizes require the use of the transition matrix solution method, which makes use of the linearity of the system. This solution technique gives accurate solutions for any step size if the input is constant. The input to the system using a digital servo controller is constant over the update period. The simulation should therefore accurately reproduce a real test using a digital servo controller as long as the integration step size is less than or equal to the update rate of the controller.

At its present level of development, the system shows excellent potential for use as an educational tool. When interfaced with a servo controller, it is possible to demonstrate the effects of the various control parameters as well as the interaction effects of actuators. With additional development, it may be possible to provide the test engineer with a “virtual test” that can be tuned prior to setting up the physical test.

LOADS AND USAGE MONITORING

Damage Tolerance Characteristics of Ageing CP140 Aircraft Using SDRS Data

A. Dabayeh, IMP Aerospace

In 1983 the CP140 Aurora Structural Life Evaluation Program (SLEP) review was initiated under the CP140 Aircraft Structural Integrity Program (ASIP). In 1993 three CP140A Arcturus aircraft joined the CP140 fleet as training role aircraft. As the CP140 fleet approaches the end of its design service life, the Canadian Forces (CF) called for a fleet life extension to at least the year 2010 in anticipation that the USN led Service Life Assessment Program (SLAP) will identify any issues that must be addressed to take the aircraft to 2025. To support this goal, the fatigue life and damage tolerance characteristics of the existing airframe are to be evaluated, and the structural modifications required to attain the extended service life goal will be identified.

The Structural Data Recording Set (SDRS) is an advanced airborne structural recording set, which is mounted on every aircraft of the CP140/140A fleet. The SDRS records various flight parameters including the wing strains through three strain sensors at three different locations on the right hand wing, and the horizontal tail strain through a strain sensor on the right hand horizontal stabilizer.

IMP performed crack growth analyses to predict the crack growth life of critical points 2 and 12 of the CP140/140A aircraft using AFGROW software and the SDRS data collected [5][6]. Critical point 12 is the lower front web and spar cap attachments at WS 209, while critical point 2 is the lower front spar cap and plank attachments at WS 71.2. The various elements needed for the crack growth analysis of critical points 2 and 12 include:

1. The stress intensity factor solution for crack growth paths obtained using "MSD" software at IMP.
2. The residual strength analysis.
3. The stress spectra generated using the collected CP140/140A SDRS data.
4. The material properties obtained from crack growth tests performed at ambient temperature and at 95% relative humidity.
5. Rain-flow cycle counting technique to identify damaging events.
6. A Wheeler retardation model to account for crack growth retardation.

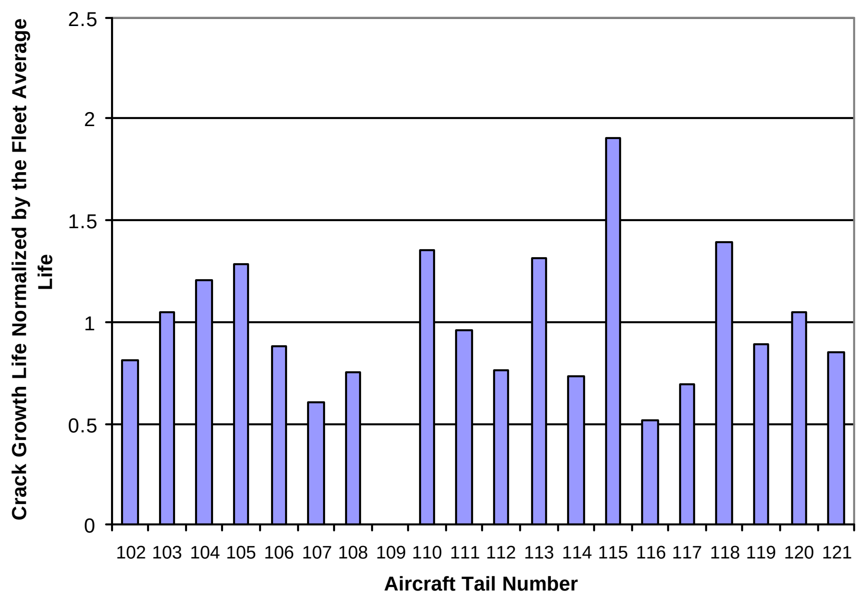


Figure 8. Individual aircraft crack growth life of critical point 12 for the 1996-1998 period.

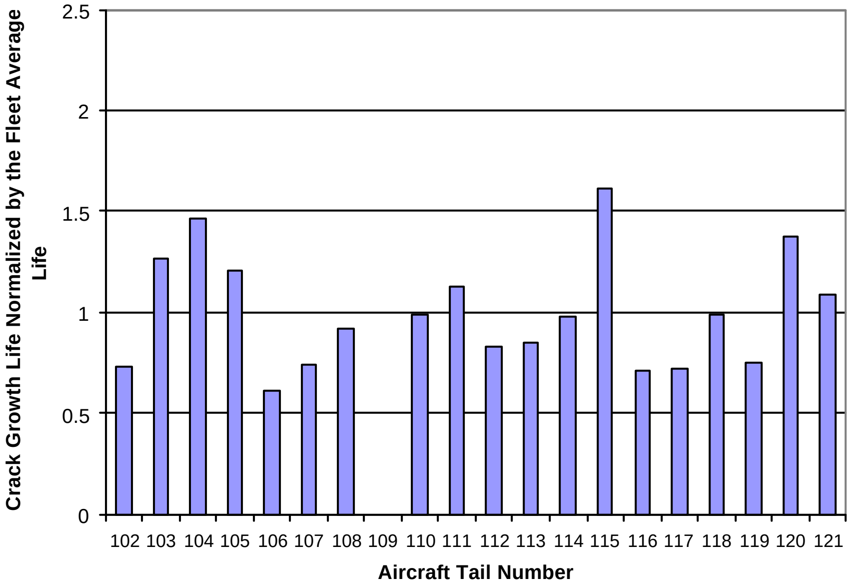


Figure 9. Individual aircraft crack growth life of critical point 12 for the 1996-1999 period.

Crack growth analyses of critical points 2 and 12 were performed using the SDRS data collected over two periods of time. The first period selected was from January 1996 to April 1998 and contained 3,464 SDRS data hours for the CP140 fleet and 876 hours for the CP140A fleet. The second period, from January 1996 to June 1999, started from the same start time of the first period, but contains 8,984 SDRS data hours for the CP140A and 1,986 hours for the CP140A fleet. Crack growth results of critical point 12 for each individual aircraft in the CP140/CP140A fleet are shown in Figure 8 and Figure 9 for the first and second periods, respectively. The results shown in these figures are based on inspection without bolt removal (i.e. initial crack length a_i = 0.25 inch). The individual aircraft crack growth data for the CP140 fleet were analyzed statistically to obtain a measure for the variation in crack growth lives between aircraft. The statistical analysis showed a strong variation in crack growth lives amongst the individual aircraft for the first period (1996-1998) and less variation for the second period (1996-1999). Or in other words as more SDRS data is used in the crack growth analysis the variation in crack growth lives amongst the individual aircraft decreases. Figure 10 shows the statistical analysis for the CP140 fleet in terms of the mean and ± one standard deviation for the two periods studied. It is expected that as more SDRS data are used and the fleet continues flying in a consistent manner, the standard deviation will stabilize. This is because every aircraft in each group type, i.e. CP140 and CP140A, is expected to be flying in a similar utilization, and is anticipated to have, over an extended period of time, similar average crack growth life.

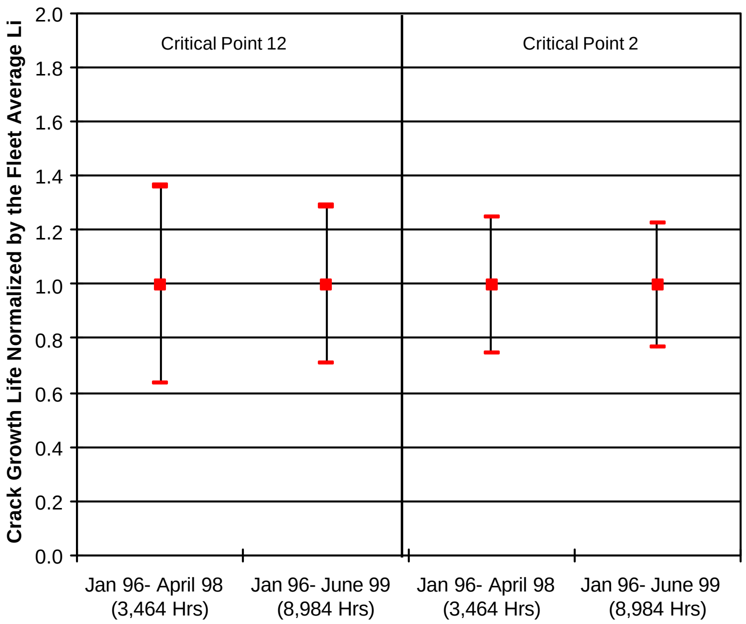


Figure 10. Mean and standard deviation for crack growth lives of the CP140 aircraft.

CF188 Fatigue Life Tracking

J. Dubuc, Bombardier

Efforts to improve the accuracy of the CF188 Fatigue Tracking for the aircraft Center fuselage were completed in 1998, followed by a complete re-processing of the fleet. The fatigue usage of each aircraft is now referenced to the Canadian Forces (CF) Baseline Operational Spectrum (BOS). Efforts are now targeted at establishing a proper tracking strategy for the empennage. This is difficult as the horizontal and vertical tails sustain dynamic loading and exhibit fairly variable responses from one aircraft to another. In the meantime, a sampling inspection program of the vertical tail attachments is currently being implemented to establish the relative severity and scatter between the IFOSTP test (FT46) and the fleet.

The CF are studying the possibility of replacing the Mission Computer with one now used by the US Navy. The solid state MSDRS tape will also be replaced with a new, more robust data storage unit. Details of these are not known at this time but these changes will undoubtedly improve the data capture rate.

CT114 Tutor Operational Loads Monitoring Program

J. Dubuc, Bombardier

The Tutor aircraft has been the primary jet trainer of the Canadian Forces until the arrival of the NATO Flying Training in Canada program. DND intend to keep approximately 20 Tutor aircraft in the Snowbird Demonstration Team role for another 5 years.

A combined Safe Life and Safety-by-Inspection philosophy is used to manage the Tutor fleet. Bombardier`s General Integrated Fatigue Tracking System (GIFTS) is used to track both wing and tail components usage severity. These results are presented on a monthly basis to the fleet manager for aircraft rotation, component swapping and aircraft retirement decisions.

Coupon testing of wing critical locations was performed at QETE in order to calibrate the damage tolerance analyses model and determine the inspection intervals. The USAF developed AFGROW software is used to perform the damage tolerance analyses.

CT133 Silverstar Operational Loads Monitoring Program

J. Dubuc, Bombardier

The Canadian Forces have reduced the number of CT133 Silverstars in operation. They intend to operate the CT133 during the next decade in roles ranging from CF188 training support to multi-purpose support flying.

The aircraft usage severity is tracked using Bombardier`s General Integrated Fatigue Tracking System (GIFTS). Results are gathered in the form of periodic report and presented to the fleet manager.

A coupon test program on wing critical locations is in progress at QETE. Material qualification will also be conducted using coupons from used and unused wings. Results from these programs will be used to determine the inspection intervals for critical locations.

Development of a CC130 Operational Loads Monitoring / Individual Aircraft Tracking (OLM/IAT) Data Analysis System (QETE, DTA, RMC, Martec, Spar)

M. Sova, QETE

The CC130 Hercules (OLM/IAT) DAS program is a multi-organizational activity to develop and validate a Data Analysis System for application to critical primary structures on CC130 aircraft. The system will be using actual flight load data collected by the Marconi Operational Loads Monitoring / Individual Aircraft Tracking Systems installed on the CC130 fleet. The objective is to rationalize operations, fleet management, maintenance and inspection schedules/priorities and consequences of decisions using individual aircraft flight data that will be approximately 1 month old. Currently the best information available is collated, analyzed and reported on a yearly basis. Since the reporting turnaround of this data is 6 months, the most current information is approximately 1.5 years old.

Responsibilities and activities for the system are as follows: Spar has overall responsibility; Martec is undertaking the finite element analyses required to develop the load transfer functions and load module for the system; QETE has selected and is in the process of validating the damage tolerance crack growth module to calculate damage progression and determine appropriate inspection/maintenance intervals, as well as undertaking material testing to generate a representative materials and operational service usage spectrum database; RMC is supplementing the testing capability.

The program is currently in Phase V where coupons simulating 5 critical location design details have been machined and are being tested to mild, medium and severe mission mix spectra developed for each specific location. By comparing the crack growth behaviour exhibited by the mission mixes, Spar will be able to provide mission severity factors into the crack growth module to further refine the prediction of appropriate maintenance / inspection intervals based upon actual service usage on a per aircraft basis. This final addition of information will allow the fleet operators to better manage their fleet and enable the maintenance contractor to better schedule / predict priorities for depot level maintenance and inspections [7] to [12].

DHC-8 Service Loads Monitoring

B. Leigh, Bombardier Aerospace

DHC-8 service load monitoring is continuing for the two DHC-8-200 aircraft currently operated in Australia in a coastal patrol role. This data (rainflow counted vertical C. of G. acceleration data) will be used to determine what changes are required to the fatigue crack inspection intervals specified for aircraft operated in the normal passenger carrying role.

FATIGUE LIFE PREDICTION

Applicability of Neuber’s Rule

R.L. Hewitt, SMPL, IAR, NRC and Carleton University

The study on the applicability of Neuber’s rule in the local strain method of fatigue life prediction that was reported in the last review has been published in the International Journal of Fatigue [13].

Kaman Probe Fatigue

D. Spencer, Indal Technologies Inc.

Background

The Royal Australian Navy will operate Kaman SH-2G(A) aircraft on its new ships in elevated sea states using dedicated aircraft securing and handling equipment. The on-deck equipment for the ANZAC vessel is the Indal Technologies Inc. (ITI) RAST (Recovery Assist Secure and Traverse) system. The system interfaces with the aircraft by capturing and securing a probe projecting from the underside of the aircraft. The RAST RSD (rapid securing device) and a typical probe are illustrated in Figure 11. The cantilevered nature of the probe, application of securing forces near its tip, and the repetitive nature of loading on the probe motivate the need for investigation of the fatigue life of the probe.

The securing forces acting on the probe have time varying components in the longitudinal, lateral, and vertical directions. The characteristics of the securing forces vary with the phase of aircraft operation, the aircraft configuration, and the environment in which the aircraft is secured. The environment includes such factors as the ship heading, ship speed, wind speed, and wind direction. The main objective of the fatigue methodology is to consider the significant factors affecting the probe to quantify the fatigue loading the probe is likely to experience during its life and use this description to estimate the probe fatigue life.

Analysis and Testing

The analysis was conducted in three distinct parts. The first involved computer-simulation-based dynamic interface analysis of typical probe securing forces for the SH-2G(A) embarked on the ANZAC Frigate. This was performed

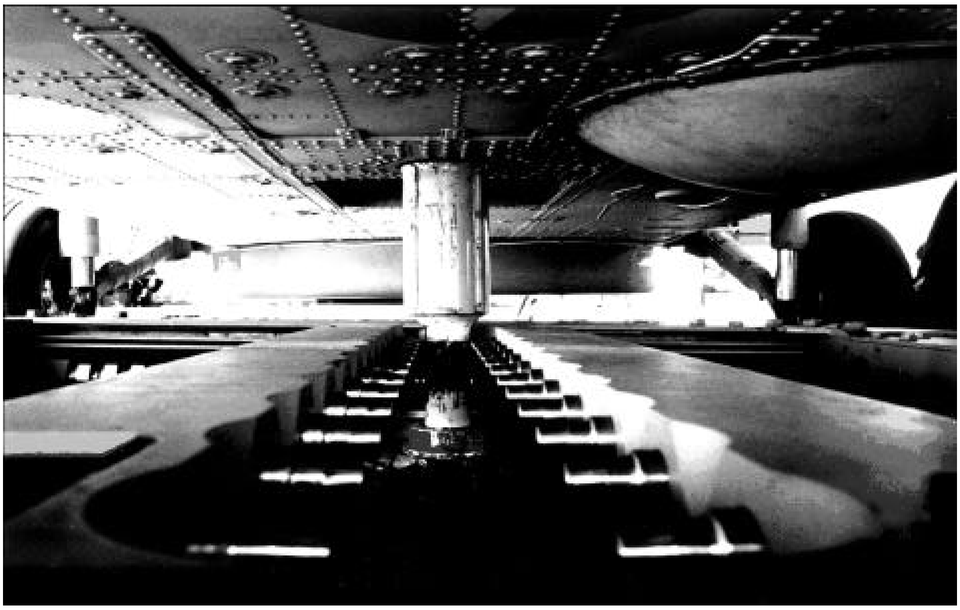


Figure 11: ITI RAST RSD and a typical probe.

using ITI's proprietary *Dynaface* simulation software. The results quantified the load spectrum expected to be applied to the in-service probe. The second portion of the analysis used a block loading approach to conduct a full-scale fatigue test to failure of the complete probe system. The test results indicated that the mode of the failure was fatigue of the aluminium upper flange. The loads to failure data were used to calibrate a generalised component S-N curve for the probe system. In the third and final portion of the analysis, the in-service load spectrum, which resulted from the dynamic interface analysis, was combined with the safe-life curve, which resulted from the fatigue test to perform Palmgren-Minor cumulative damage analysis on the probe. The results revealed that fatigue damage on the probe during its life is minimal and that fatigue failure of the probe system, based on the data available at the time of analysis, will not occur during the 30 years design life of the probe. The report contains all necessary data such that the probe estimated fatigue life could be updated should the aircraft's in-service operating profile change. [14].

FRACTURE MECHANICS AND CRACK PROPAGATION STUDIES

CT114 Safety by Inspection Coupon Test Program

M. Yanishevsky, M. Sova, QETE

The Canadian Forces (CF) CT114 Tutor aircraft fleet, procured in 1962, is in the process of being retired from its current role as lead-in fighter trainer. However, the Canadian Forces have decided to continue operating several of these aircraft in the Snowbird aerobatic role. The CT114 Tutor aircraft was originally designed, monitored, and maintained according to the Safe Life philosophy. Recent re-analysis using current aircraft/fleet usage data from on-board Operational Loads Monitoring and Individual Aircraft Tracking systems, as well as recent field detection of cracks in fuselage and wing structures, indicated that the Safe Life of the fleet in general has been consumed. The Original Equipment Manufacturer, Bombardier, and the CF DTA and QETE organizations have collectively developed and conducted a Safety by Inspection testing program in order to keep selected aircraft from the Tutor fleet operational for the Snowbird aerobatic team using Damage Tolerance and Crack Growth principles. Since the fuselage and tail structures have recently seen a full scale durability and damage tolerance test, it was urgent to generate component level spectrum test data in order to validate that the change from the Safe Life to the Damage Tolerance philosophy for the wing attachment fittings was indeed possible. Analysis, service experience, and results from previous full scale tests identified five locations that were most critical (Figure 12). Three aircraft were decommissioned in order to recover components for testing purposes. Electron Discharge Machining was used to introduce damage in critical design details of these components to simulate worst case unsuccessful inspections. These components were then monitored to establish crack growth behaviour while being fatigue tested in uniaxial servohydraulic material test load frames. The components were subjected to the current Snowbird aerobatic usage spectra in order to develop/validate a Safety by Inspection program for formation and solo aircraft. To date, most locations have demonstrated that they will be able to satisfy the life extension requirement. Only Location 8 has required to have a repair doubler designed and implemented to allow the wing front spar lower cap to achieve its expected life. Additional tests will be performed on specimens where initial damage introduced was “too severe” [15], [16].

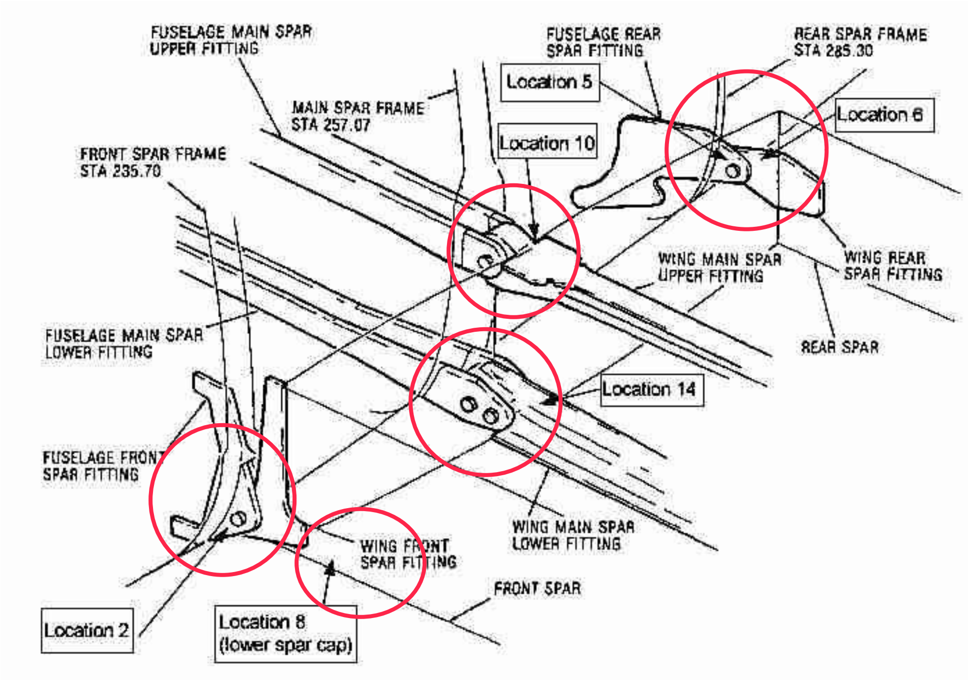


Figure 12. Main Fuselage to Wing attachment fittings and spar components tested to establish appropriate inspection intervals to enable implementation of a Safety by Inspection program on CT114 Snowbird aircraft.

CT133 Safety by Inspection Program

M. Yanishevsky, M. Sova, QETE

The Canadian Forces CT133 (Lockheed T33 - Silver Star) first entered Canadian military service in the early 1950's in the role of fighter lead-in trainer. During its career, the role of the CT133 has changed several times, evolving to the current service demands for target towing, electronic warfare, executive transport, and most recently for an ejection seat test bed. Projection for the CT133 aircraft fleet is that it will be remaining in active service at least until the year 2010, requiring extension of its structural life far beyond the original design parameters. The Canadian DND has developed an action plan to ensure continuing structural airworthiness based upon Safety by Inspection and the Damage Tolerance philosophy to enable prediction of the operational life of the airframe. Operational Loads Monitoring systems have been installed on fleet aircraft, data from which were used to assess and monitor the flight loads experienced in each of the CT133's roles. This system is also being used to continually update the load spectra for analytical purposes. The wing is the key life-limiting structural element of the aircraft. Therefore, an accurate assessment of the available fatigue life in critical wing locations is required to prove / validate the continued viability of the fleet. To achieve this, DND

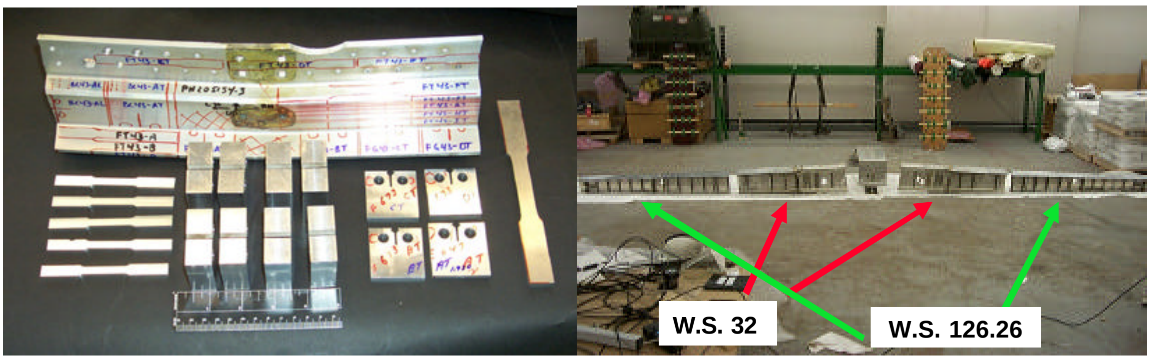


Figure 13. Specimens removed from CT133 Rear Spars for evaluating static and damage tolerant material properties (left) and CT133 wing sections containing rear spars that will undergo component level spectrum fatigue tests.

has undertaken a coupon test programme using spar caps from retired wings to generate experimental material and sub-component level test data, that will be used to substantiate / maintain the Canadian Military Airworthiness Type Certificate (CMATC) of this aircraft.

To develop the material property database, tensile, fracture toughness, and fatigue crack growth test coupons are being extracted from existing spar caps (Al 2014-T6 extrusion) from a time expired wing and a wing that has not seen service. This material can no longer be purchased and fracture mechanics data on Al 2014-T6 does not exist. To develop appropriate inspections and inspection intervals, two critical locations will be cut from 3 aircraft (6 spar caps) for generating component level design detail crack growth data, loaded to representative service usage spectra (Figure 13) [17].

Three Dimensional Crack Growth Analysis in Isotropic and Non-Isotropic Materials

M. Oore, IMP Aerospace

The cracking in thick material (for example lugs, landing gear components) is often characterised with a crack front profile where the stress intensity factor may vary along the crack front. Such configurations are referred to as 3D cracks. The object of this analytical study at IMP Aerospace is to investigate the basic 3D crack shape assumption (elliptical) under uniform stress distribution and assess its applicability to fatigue crack growth and stress corrosion cracking, and also, to further the understanding of crack shape development.

Using a closed form solution approach and assuming that the Paris' crack growth equation is applicable, it is shown in [18] that the evolving crack shape is dependent on the power m of Paris' crack growth equation for the specific material. This analysis considers the local growth rate along the crack contour in relation to the elliptical shape growth pattern. In the case of isotropic materials the elliptical growth assumption is shown to be exact or conservative for m=2 and m>2 respectively. Figure 14 is an example of semi-elliptical stress corrosion crack found in service. For non-isotropic materials the solution is found to be more complex. However, for some non-isotropic material properties distribution, the elliptical growth pattern is shown to be valid. These analytical results can assist in determining the conservatism involved in making crack shape assumptions under certain conditions. Also, the present results can be used to validate or calibrate software algorithms designed for predicting the propagation of 3D cracks.

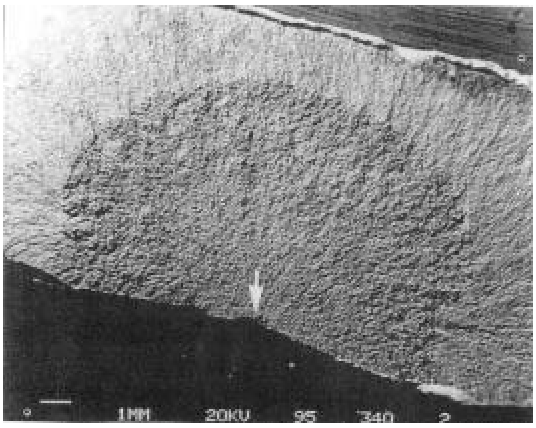


Figure 14. Semi-elliptical stress corrosion crack in Boeing 707 landing gear truck beam, QETE Project A027288.

Shot Peening Certification Program for CF188 Y470.5 Bulkhead Location X19

M. Yanishevsky, M. Sova, QETE

DGAEPM(FT), DTA, QETE, RMC and Ecole Polytechnique are participating in a program to evaluate the benefit of shot peening in a critical location of the Y470.5 bulkhead of the CF188 aircraft (Figure 15). During the original McDonnell Douglas ST16 full scale test, this location developed cracks. The current test program is an attempt to quantify the benefit of shot peening at different stages of fatigue damage accumulation, i.e. shot peening pristine material, shot peening after accumulation of 40% of the baseline crack formation life, shot peening after accumulation of 65% of the baseline crack formation life, shot peening after accumulation of 80% of the baseline crack formation life, as well as the benefit of repetitive shot peening processes (first shot peening done after the accumulation of 65% of the baseline crack formation life, second shot peening following an additional 88% of the baseline crack formation life) including the benefit of polishing the surfaces prior to shot peening. A special coupon (Figure 15) has been developed simulating the X19 location and the IARPO3a spectrum, representative of the joint CF and RAAF F/A-18 International Follow-On Structural Test Program, IFOSTP, full scale test program is being used. It has been determined that there is at least a 2.4 times increase in the baseline crack formation life after shot peening of "crack-free" surfaces irrespective of the prior history and accumulated damage. However, it has been demonstrated that cracks with depths less than the shot peen layer thickness can severely compromise the life improvement effectiveness of the shot peening (Figure 16 and Figure 17). Statistical analysis of the limited current data, including those coupons which exhibited cracking below the threshold of inspectability of conventional Eddy Current NDI pencil probes, has been undertaken. With 8 to 10 data points per condition and using a 1/1000 crack formation risk level, statistical analysis indicates that the Shot Peening Life Improvement factor ranges from 4.1 for coupons shot peened at time 0% of Baseline Crack Formation Life (BCFL); 3.8 for coupons shot peened at 40% of BCFL; 3.3 for coupons shot peened at 65% of BCFL; and 1.2 for coupons shot peened at 80% of BCFL. While these tests results show some optimism about shot peening repair, DND is currently assessing how this information will be used for fleet-wide management purposes. Further testing is underway to supplement the material database [19] to [22].

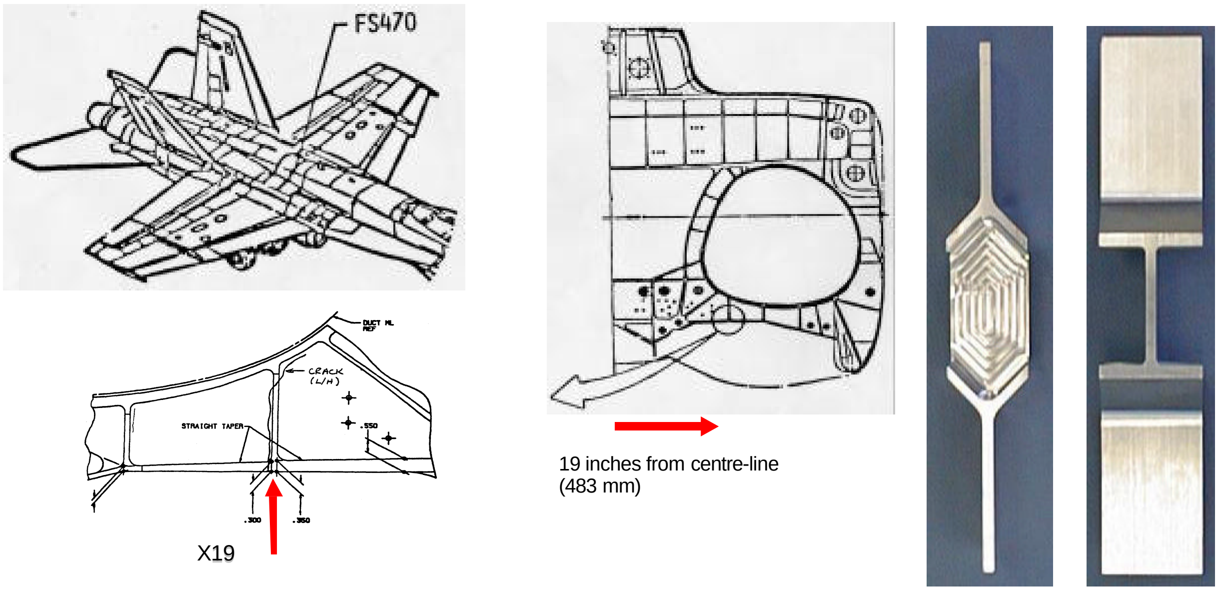


Figure 15. Location X19 on the CF188 Y470.5 Bulkhead (left) and the coupon developed to simulate the web / flange radius (right).

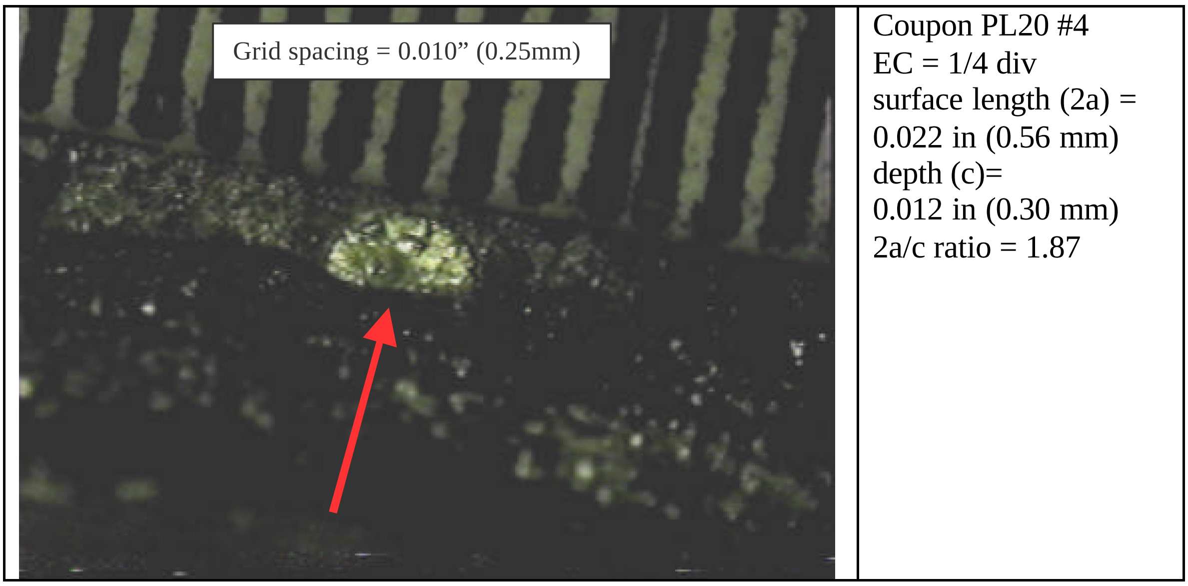


Figure 16. Opened surface crack in X19 coupon radius on the threshold of inspectability by eddy current.

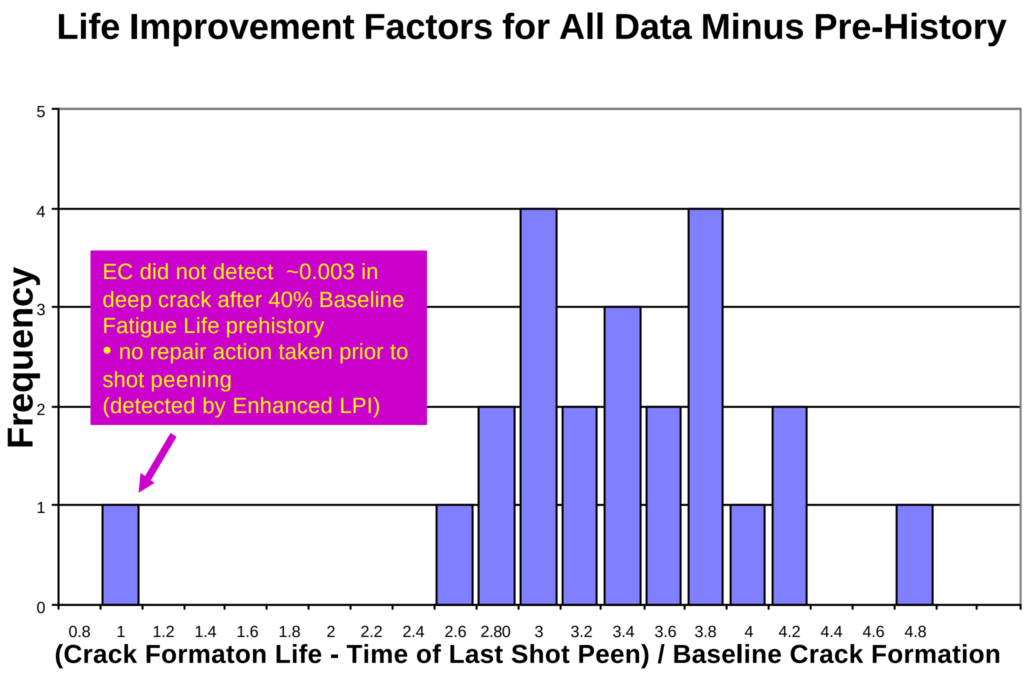


Figure 17. Life Improvement ignoring previous history. Note that cracks below the conventional eddy current threshold of inspectability severely compromise shot peening effectiveness.

Investigation of the Effects of Heat Cycles on the Relaxation of Shot Peening Residual Stresses
M. Sova, QETE

Shot peening is a commonly used technique to improve the fatigue life of components. This is the result of the build up of a layer of compressive stresses when the component's surface is plastically deformed by the peening particles. Many CF188 aluminum alloy components have been shot peened. Unfortunately, exposure to heat may induce a relaxation of the residual stresses. In particular, it was found that the heat cycles used for bonded repairs resulted in relaxed residual stresses. QETE recently evaluated the effects of two heat cycles, simulating the cure cycles used to adhesively bonded doublers, on the shot peening residual stresses and on the fatigue life of 7050-T7451 Al alloy coupons. The coupons were shot peened to an Almen intensity of 0.008A ± 0.001A with 200% coverage. They were then fatigue tested to detectable crack life using the IARPO3a spectrum.

The two heat cycles were as follows:

Heat cycle #1	175°F for 30 min
	1 hour cool (at room temperature)
	250°F for 1 hour
	1 hour cool (at room temperature)
	185°F for 8 hours
	final cool (at room temperature)
Heat cycle #2	175°F for 30 min
	1 hour cool (at room temperature)
	250°F for 1 hour
	1 hour cool (at room temperature)
	250°F for 2 hours
	final cool (at room temperature).

The first heat exposure at 175°F is associated with the silane treatment, the second heat exposure at 250°F with the cure of the primer, and the third heat exposure with the cure of the epoxy adhesive.

The residual stresses were measured using a dedicated X-ray stress analyzer model AST X2001. The surface residual stresses and the distribution of the stresses with depth were measured in one as-received coupon and one coupon each after heat cycles #1 and # 2. To accomplish this, material was electropolished away in increments of 0.001 inch. The average surface residual stresses in the as-shot peened specimens were around -38 ksi. The compressive residual stresses reached a maximum of around -55 ksi below the surface, at a depth of about 0.004 inches. The layer with significant compressive stresses was at least 0.007 inches. The maximum relaxation of the residual stresses occurred at the surface. The surface stresses were reduced between 25% and 27% after heat cycle #1 and between 28% and 31% after heat cycle #2 compared to the as shot peened condition.

Relaxation of residual stresses increases with temperature and time at temperature. It is believed that for both heat cycles, most of the relaxation would occur during the exposure at 250°F. To understand the kinetics of that process, the residual stresses were measured as a function of time at 250°F, for times up to 8 hours. The measurements were also performed at different depths. The rate of residual stress relaxation was the highest up to 2 hours and then decreased at longer times. This explains why the surface stresses were slightly more relaxed after heat cycle #2 because of the longer exposure time at the temperature of 250°F [23].

FAILURE INVESTIGATIONS

Fractographic Investigation of the CC130307 (Hercules) Right Hand Drag Angle

Martin Roth, QETE

The special inspection performed on CC130 drag angles revealed fleet wide cracking along the radius at F.S. 497 and to a lesser extent at F.S. 517, 537, 557 and 577. The CC130307 R/H drag angle, which had the longest crack at F.S. 497, was removed for replacement and forwarded to QETE for detailed inspection and fractographic examination in order to determine the mechanisms of crack initiation and crack growth, and, if possible, the rate of crack growth.

The drag angle called on its drawing the Attach Angle - Wing to Fuselage, Center Wing Station 61.625, was installed on CC130307 in October 1972 as part the center wing installation. Its TSN was 27,769 hours. The part was manufactured from a 7075-T6 aluminum alloy extrusion. The part was given a coloured chemical film treatment and then coated with two coats of zinc chromate primer.

The 7.5 in. long crack at F.S. 497 was clearly visible. Eddy current inspection of the radius along the length of the drag angle confirmed the presence of cracking at F.S. 537 over a length of 1.0 in. Eddy current inspection also indicated the presence of a 0.7 in. long crack and two smaller ones on a slightly different plane at F.S. 577.

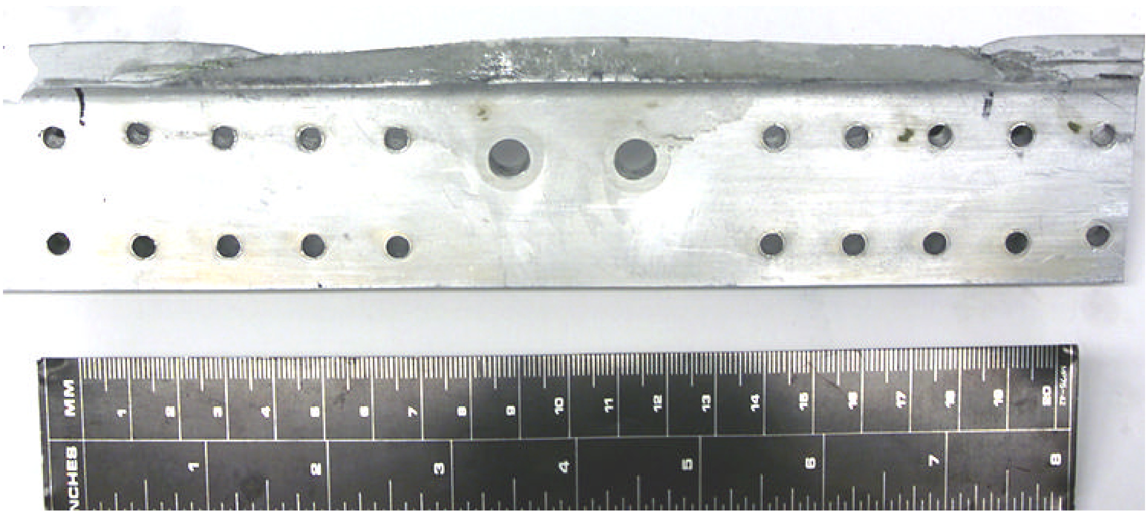


Figure 18. Overall view of the surface of the 7.5 in. long crack at F.S. 497 on the horizontal member of the CC130307 drag angle. With the part as shown, the aft end of the drag angle is towards the left.

The 7.5 in. long crack was opened up (Figure 18). Its surfaces were covered with paint and sealant residue and had to be extensively cleaned prior to visual and scanning electron microscope (SEM) examination. Macroscopic examination revealed the presence of adjacent small fatigue cracks which joined up to form a shallow nearly semi-elliptical fatigue crack, 1.625 in. long and 0.084 in. deep (Figure 19). However, detailed SEM examination of the fatigue cracks revealed that the initial crack propagation was intergranular, up to a depth of about 0.017 in., indicating that the early stages of cracking were by stress corrosion cracking (SCC). The fatigue crack was nearly in a horizontal plane and was centered at the mid-plane between the larger holes shown in Figure 18. Crack growth by overload followed, through the thickness and aft, about 3.5 in., and forward, about 3 in. (Figure 18). The orientation of the overload fracture surface changed from close to zero at the fatigue crack to about 60^0 at the aft end and 30^0 at the forward end. At the curved aft overload crack front, a number of SCC cracks had initiated at varying depths and propagated in planes parallel to the surface of the part (Figure 20). The longest of these SCC cracks was about 1 in. long. On the last 1 in. of the forward side of the crack, the overload was interspersed by a few fatigue bands (Figure 21). At that crack front, there were also SCC cracks, similar in location and size to those at the aft crack front (Figure 21).

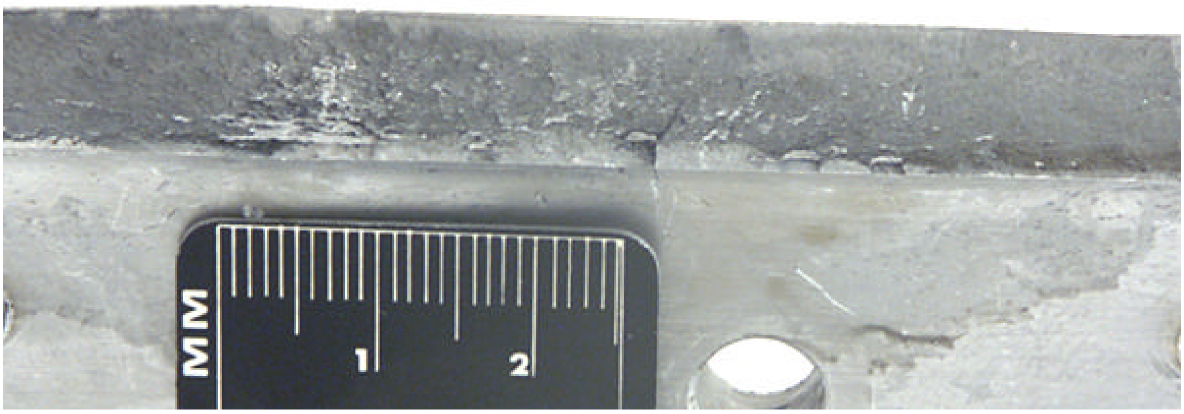


Figure 19. Central portion of the above crack showing the shallow nearly semi-elliptical SCC plus fatigue crack, 1.625 in. long and 0.084 in. deep. That crack was in a horizontal plane.

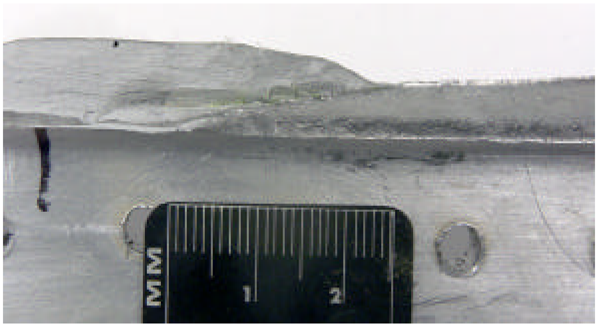


Figure 20. Aft end of the Figure 18 crack showing SCC cracks in planes parallel to the surface of the part.

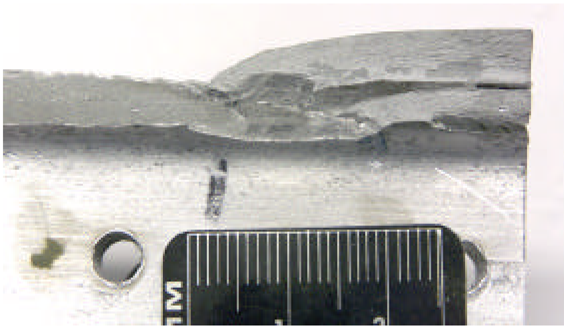


Figure 21. Forward end of the Figure 18 crack showing fatigue bands over the last 1.0 in. of the overload zone and SCC cracks in planes parallel to the surface of the part.

The detailed SEM examination of the fatigue cracks had revealed that the initial crack propagation was intergranular, up to a maximum depth of about 0.017 in. Fatigue features were observed between the end of the SCC cracks and the fatigue/overload transition. The fatigue features, where the crack was the deepest, were investigated in detail. Fairly evenly spaced fatigue striations were observed over the first half of the fatigue zone. In the second half, there was a mixture of fatigue zones with striations and rougher tear areas. The average striation spacing generally increased from about 20 μin. (0.5 μm) at the beginning of the fatigue crack (i.e. at a depth of 0.017 in.) to about 40 μin. (1.0 μm) at the end of the fatigue crack (i.e. at a depth of 0.084 in.). From these striation spacing measurements, it was estimated that about 2400 cycles (likely ± a few hundred cycles) were associated with growth of that fatigue crack.

The sequence of events leading to the large crack at F.S. 497 was the following: possibly pitting, SCC, fatigue, overload, unstable fatigue crack growth, and SCC. Pitting was clearly observed at the origin of the cracks at F.S. 537, but this could not be confirmed in the case of the F.S. 497 cracks because of surface damage. SCC first occurred in the layer with the equiaxed grain structure, which is expected to have a low threshold stress for SCC. Because of the evenness of the intergranular features, it is not possible to determine the rate of SCC crack growth. It should be noted that the SCC to fatigue transition occurred where the microstructure changed to flattened grain, where, for a crack with that orientation, the threshold stress for SCC would be higher. The relatively evenly spaced fatigue striations indicated cyclic loads of similar amplitude, such as those due to the pressurization cycles. It was estimated there were only about

2400 fatigue cycles before the critical crack size was reached. The critical crack, 1.625 in. long and 0.084 in. deep, appeared to be significantly smaller than that reported for a Danish failure and slightly larger than one investigated by the USAF. It was estimated that at least a few thousand cycles were associated with fatigue crack growth at the forward end of the crack at F.S. 497. Considering this figure, it was surmised that the cycles responsible for crack growth were also pressurization cycles.

The authors of the USAF failure report blamed the fatigue cracking on high residual stresses, mainly from manufacturing, more specifically the stretch forming operation, and to a lesser extent from assembly. In the CC130307 case, if there had been high residual stresses from manufacturing, the presence of many SCC cracks along the length of the part would have been expected. As such, it is still believed that assembly stresses played an important role in this failure. The different critical crack sizes between the Canadian, USAF, and Danish cracks could possibly be the result of varying levels of assembly stresses. It might be possible to confirm this hypothesis by correlating the goodness of the fit of drag angles being removed with their critical crack sizes. Further details of the investigation can be found in reference [24].

PROBABILISTIC AND RISK ANALYSIS METHODS

Probabilistic Buckling Analysis of Fibre Metal Laminates under Shear Loading Condition

G. Shi, SMPL, IAR, NRC

Experimental work at the IAR/NRC has shown a relatively large scatter in FML shear properties when compared to those of traditional aluminum alloys. For shear buckling tests, it is not easy to apply a pure shear load on an FML panel. As a consequence, some randomness has been observed during recent test work on the shear buckling of two FML panels carried out at IAR/NRC. This phenomenon makes it difficult to accurately simulate the loading condition on a FML panel.

In order to take the uncertainties associated with the material properties and the loads into consideration, a probabilistic analysis methodology for FML panels under a shear loading condition was proposed. Under this methodology, the shear buckling design allowable will be presented in the form of distributions rather than fixed values. In this work, the elastic Modulii of ARALL3-3/2 and GLARE3-2/1 in the longitudinal and transverse directions are considered as random variables. The scatter in the test data is characterised using standard distribution functions. The variations in the applied load to the panels are characterised by a parameter, which is also assumed to be a random variable. The shear buckling load of the panel, as a response function of the three random variables, is developed based on limited finite element results using the Response Surface Methodology (RSM). Three RSM design methods: Full Factorial Design, CCD Design and Saturated D-Optimum Design, are applied to construct the response function. The distributions of the shear buckling load are predicted and compared with independently generated finite element data for the purpose of verification. The obtained results indicated that it is necessary to use a probabilistic method for the FML buckling analysis and the methodology developed is useful and efficient for the probabilistic buckling prediction [25].

Risk Assessment Codes

M. Liao, SMPL, IAR, NRC

In recent years, risk assessments that directly address the stochastic nature of structural integrity have been increasingly used as an important fleet management tool to ensure the flight safety, maintain readiness, and reduce maintenance costs. Analytical capabilities for the risk assessment of aircraft structures continue to be developed in academia, industry, and government laboratories. The research efforts have resulted in several computer codes for performing risk assessments of ageing aircraft structures. Two computer codes for risk assessment were developed by Canadian companies under the support of the DND. The code PRISM (Probabilistic Crack Growth Analysis for RISk Management) was developed by Bombardier Aerospace and the code PROMISS (PrObabilistic Maintenance and Inspection Scheduling System) was developed by Martec Limited. A project is currently underway at the IAR/NRC with the objective to evaluate the capabilities of the Canadian codes PRISM and PROMISS. In the evaluation work, the USAF code PROF (PRObability of Fracture) was used as a benchmark tool. Results from the evaluation of PRISM are presented here.

The methodologies used in PRISM were reviewed and the input requirements, the data format, and the input/output (I/O) interfaces of PRISM were discussed from a user's point of view. Two application examples, which used two components including a T-38 aircraft lower wing skin and a CF188 upper outboard longeron, were worked out to examine various features of PRISM as compared to those of PROF. Particularly, the computational efficiency and the quality of results of PRISM and PROF for the two examples were compared. Based on the evaluation results, the following conclusions can be drawn:

PRISM has most of the features and capabilities of PROF (Figure 22) and the two codes provided close predictions for the two examples (Figure 23 and Figure 24). PRISM provides a capable tool for probabilistic durability and damage tolerance analysis and risk assessment for aircraft structural integrity.

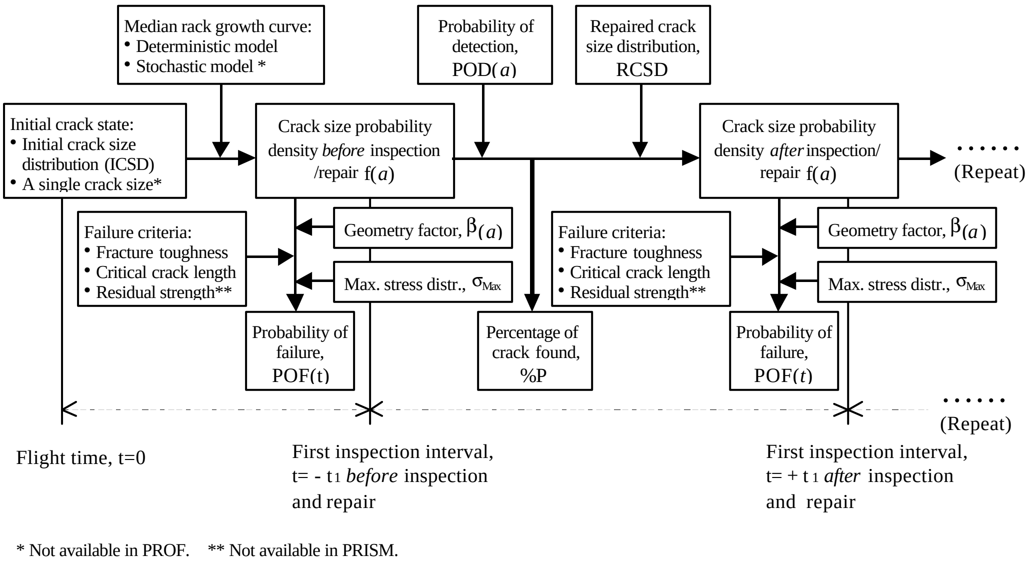


Figure 22. Methodologies used in PRISM and PROF.

The input data preparation has a great influence on the risk assessment results. The most influential input data were the initial crack size distribution (ICSD) and the median crack growth (CG) curve. Different forms of input distributions and different integration methods used in PRISM and PROF affect the accuracy of the calculated results.

In contrast to PROF, PRISM provided the capability to use either a deterministic or stochastic crack growth model for the median crack growth curve. More studies are required to select the deterministic or stochastic crack growth model in a risk assessment since these two models can produce very different results (Figure 23 and Figure 24).

More practical data sets are necessary to further evaluate the quality of results between PRISM and PROF. Further details can be found in references [26] to [28].

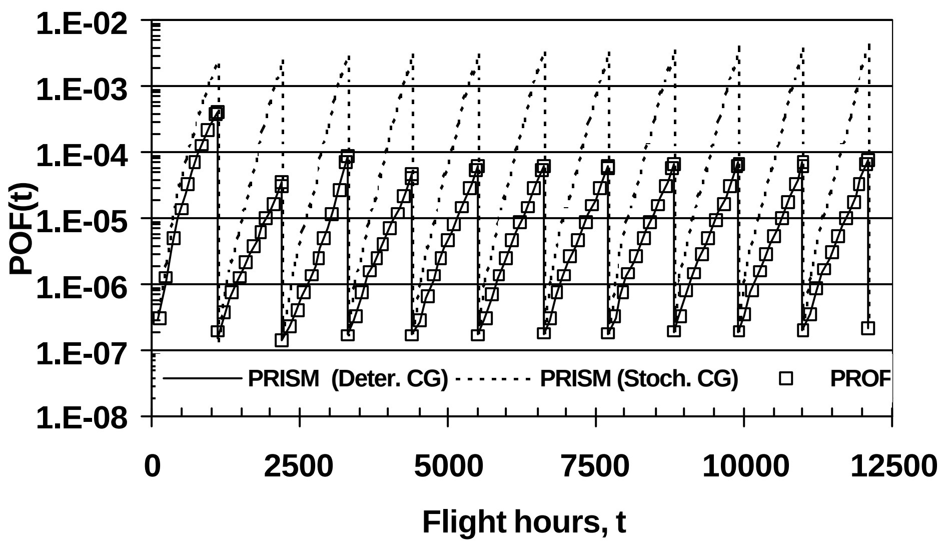


Figure 23. POF(t) results by PRISM and PROF (Example 1 Fastener hole in T38 aircraft lower wing skin).

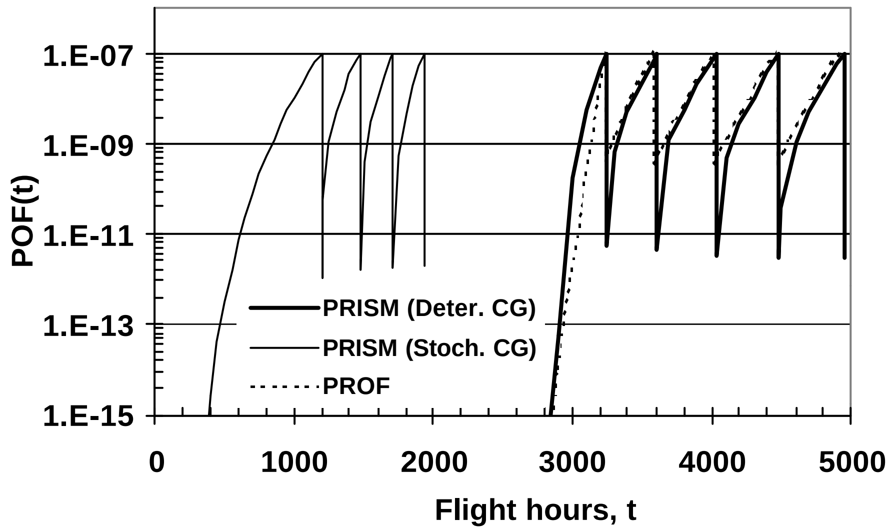


Figure 24. POF(t) results by PRISM and PROF (Example 2 Fastener hole in CF188 upper outboard longeron).

AGEING AIRCRAFT ISSUES

Summary of Spar Aerospace Limited Ageing Aircraft Initiatives

A. Baig, Spar Aerospace Limited

Ageing aircraft are an enormous challenge to Fleet Managers from the standpoints of flight safety, operational availability and the economics of fleet maintenance. The CF operate the highest time military C130 fleet in the world and Spar Aerospace Limited plays a pivotal role in maintaining and strategizing how to cost effectively manage these ageing Hercules aircraft. Spar is currently conducting three comprehensive life improvement technical evaluations on behalf of the CF focusing on aircraft rewire and wing and fuselage structural improvements.

Rewire Program

The E Model Fleet wiring is over 40 years old - the majority of the aircraft electrical system wiring, associated hardware, and RF cables are original equipment that have deteriorated over the years, due to normal wear, exposure to environment extremes, and contamination. A requirement therefore exists to rewire.

A C130 Rewire Program is being defined by Spar to address this requirement. To meet the needs of the Canadian Forces, the program has been designed for completion in two distinct phases. Phase I of the program is aimed at replacing all the wiring, components, and RF cables in the wings and the wiring from the wings to the cockpit. Phase II of the program will replace all remaining wiring.

Fuselage Improvement Program (FIP2)

The fuselage of the C130 Hercules has remained virtually unchanged over its 40 year history and has proven to be an effective and damage tolerant structure. Working closely with the Canadian Forces, Spar has undertaken a number of technical initiatives over the years to ensure that the fuselage will remain airworthy - notably, the Fuselage Improvement Program (FIP1) in the late 1970's. Some twenty years later the fuselage has again reached a stage where further structural improvements are necessary to ensure Canadian Forces fleet airworthiness until the year 2010.

The improvement technical evaluation is divided into three phases. Phase I, which Spar has now completed, consisted of identifying potential problem areas for the fuselage and empennage structures. Additionally, two in-depth fuselage sampling inspections were developed by Spar. Damage review uncovered 976 potential problem locations. Phase II of the program (which has now also been completed by Spar) entailed the identification of potential solutions (repair, replace, redesign), cost estimation activities, the performance of additional fuselage sampling inspections, and the identification of the methodology for life assessment of the fuselage and empennage structures in the absence of fuselage full scale test (FST) data. Phase III of the program is about to commence and involves the detail design of improvements and completion of a prototype modification program.

Wing Improvement Program (WIP)

The original C-130E center wings were designed for a low service life - this drove the requirement for a Canadian Forces replacement program in 1972 when Spar integrated new H model configuration center wings with the C-130E airframe. After some 28 years, this replacement center wing has accumulated 24,000 to 30,000 wing hours and significant damage is being observed. In response, a Center Wing Service Life Extension Program (CWSLEP) study has recently been undertaken by Spar with the intent of determining the feasibility of safely and economically reaching the Canadian Forces desired ELE of 2010. This technical evaluation examined three options for completing a CWSLEP - these options included a Center Wing Improvement Program (CWIP), Center Wing Replacement Program (CWRP), and a Continuing Maintenance Program (CMP). Following Spar's recommendations, the Canadian Forces have recently selected the Wing Improvement Option (which now also includes some outer wing elements) as the optimum solution from a cost, operational availability and safety perspective.

Retrogression and Re-ageing of Al 7075-T6 Parts

D. Raizenne, X. Wu and C. Poon, SMPL, IAR, NRC

Corrosion is one of the most pressing issues associated with ageing aircraft. All C-130, and USAF C-141 and C-5 fleets have experienced, and continue to experience, stress corrosion cracking in primary structural components manufactured from Al 7075-T6 forgings, extrusions and plate products. To improve the corrosion resistance of these components, retrogression and re-ageing (RRA) heat treatments have been developed. The RRA process is an intermediate heat treatment (retrogression at 195^{0}C for 40 minutes followed by re-ageing at 120^{0}C for 24 hours) which when applied to Al 7075-T6 material results in improved corrosion resistance approaching the T73 temper, while maintaining the high strength of the T6 temper. Heat treatments are normally carried out using forced air furnaces and water quenching.

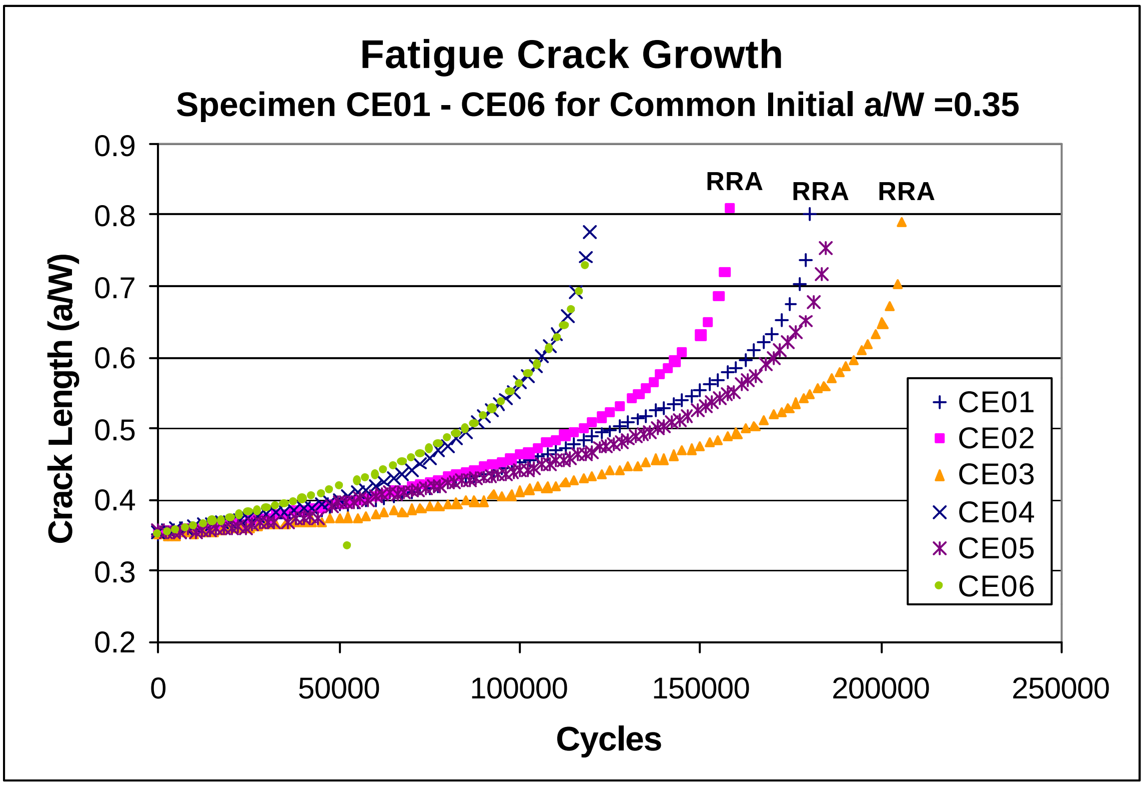


Figure 25. Comparison of fatigue crack growth in 7075 T6 and RRA material.

The RRA heat treatment process for aluminum alloys was first developed in the early 1970's. Since that time, one of the leading authorities in the RRA process is the IAR/NRC. In the last two years, extensive work has been carried out at IAR/NRC in the use of the RRA process for structural components on the C-130 aircraft - in particular, the 26 foot long sloping longeron. An 'in-situ' RRA treatment process was developed to heat treat selected areas of the longeron.

Two sections of a sloping longeron were heat treated using a Zimac Laboratories ALC-HBS-24 hot bonder and an IAR/NRC developed spray quenching technique for rapid cooling. A significant database consisting of hardness, electrical conductivity, tensile, exfoliation, stress corrosion, fracture toughness and fatigue crack growth results has been established. The test results to date indicate that an in-situ RRA treatment is practical for sections of Al 7075-T6511 extrusions up to 0.75 inches thick [29]. Optimized RRA treatments were developed using 'real time' kinetics modelling. Electrical conductivity and hardness were used successfully as screening criteria for corrosion resistance and strength. Extensive tensile, corrosion, fatigue and fracture tests were carried out to develop the required confidence levels to satisfy MIL-HDBK-5H requirements (Figure 25). A process specification was written to address protective coatings and quality control issues associated with depot level requirements.

Environmentally Assisted Cracks in 2024-T3 Fuselage Lap Joints

N.C. Bellinger, SMPL, IAR, NRC

During a study carried out at the IAR/NRC to determine the effect that corrosion pillowing has on the structural integrity of fuselage lap joints, pillowing cracks were found using x-ray techniques in both retired and operational aircraft. The majority of the cracks that were found had not penetrated through the thickness to reach to outer surface. A fractographic investigation of the fracture surfaces revealed extensive intergranular cracking with numerous secondary cracks and some fatigue striations were found near the crack tip. To better understand the process involved in the formation of these cracks, a study was carried out to determine the faying surface damage caused by corrosion. Naturally corroded lap joints were disassembled and the corrosion products removed using a chemical/ultrasonic cleaning process. Sections were then taken from different areas containing different levels of thickness loss, vacuum cold mounted and progressively polished. For the lightly corroded sections where the damage present had not yet penetrated through the clad layer, corrosion pits of various sizes were present, Figure 26. The results have suggested that the clad layer may be partially removed due to flaking and some cracking has been found between pits, Figure 27.

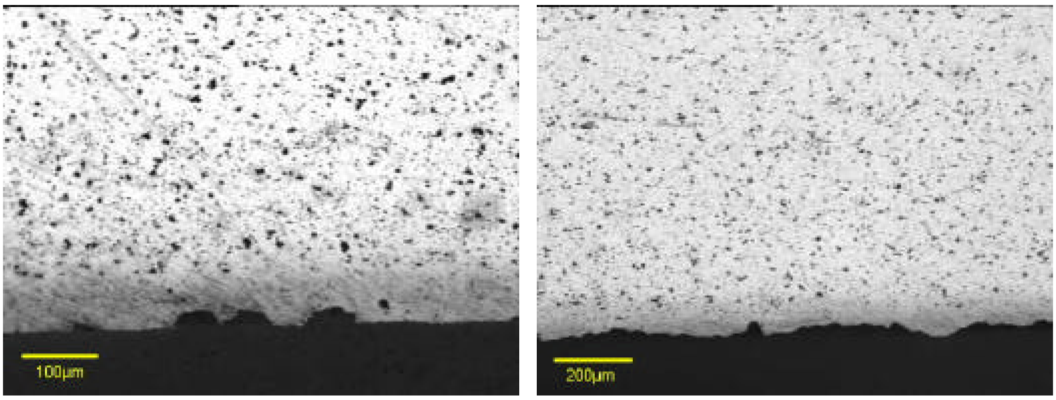
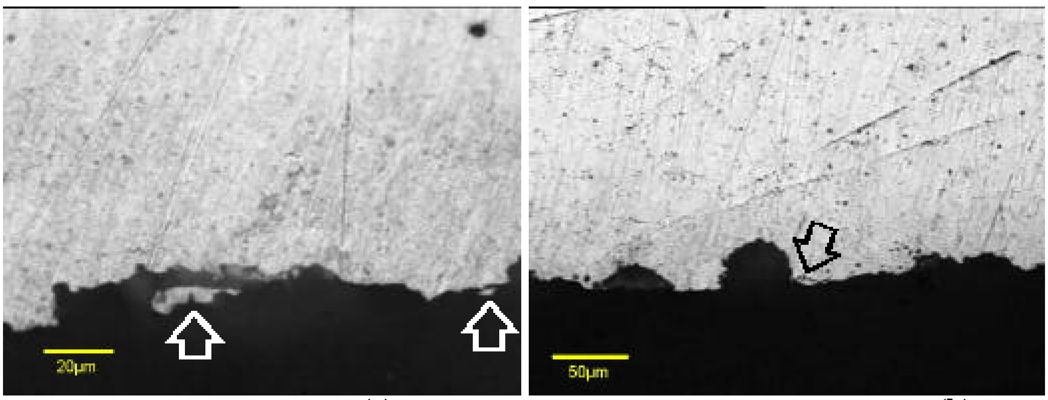


Figure 26. Optical micrographs of damage along faying surface of lightly corroded sections. Pits have not penetrated clad layer. The black spots are due to the polishing.



(a) **(b)**

Figure 27. Optical micrographs of damage along faying surface of lightly corroded sections. (a) Flaking of the clad layer, (b) Small crack between pits.

For the sections taken from the more severely corroded lap joints where the damage had penetrated the clad layer, the results showed the presence of pitting, intergranular attack and environmentally assisted cracking under sustained stress (EAC_{SS}), Figure 28. Since EAC_{SS} was only present near rivet holes, the sustained stress that caused this type of damage, Figure 28d, was associated with the corrosion pillowing, which is caused by the presence of the corrosion products in the lap joint.

The results from the polishing have suggested that a correlation may exist between the average thickness loss and the surface topography which is important for corrosion metrics.

Role of Surface Topography in Corroded Fuselage Lap Joints

To aid in determining whether a correlation exists between the average thickness loss and the surface topography, coupons that were tested in Phase I of the Corrosion Fatigue Structural Demonstration (CFSD) Program (Northrop Grumman Corporation) were examined using IAR/NRC Canada's NDIAnalysis software to determine the thickness loss distribution. These coupons were machined from a retired Boeing 727 lap joint with 56,784 hours / 49,256 cycles. Thickness maps were developed from radiographs taken of the different coupons as shown in Figure 29.

To determine the thickness loss distribution in the gauge length of each coupon, the x-ray film was digitized with a resolution of 800 dpi (1 pixel = 0.00125 inch). Using IAR/NRC Canada's NDIAnalysis software a histogram plot of the thickness loss was obtained for each coupon. Figure 30 to Figure 32 show the results for three of these coupons.

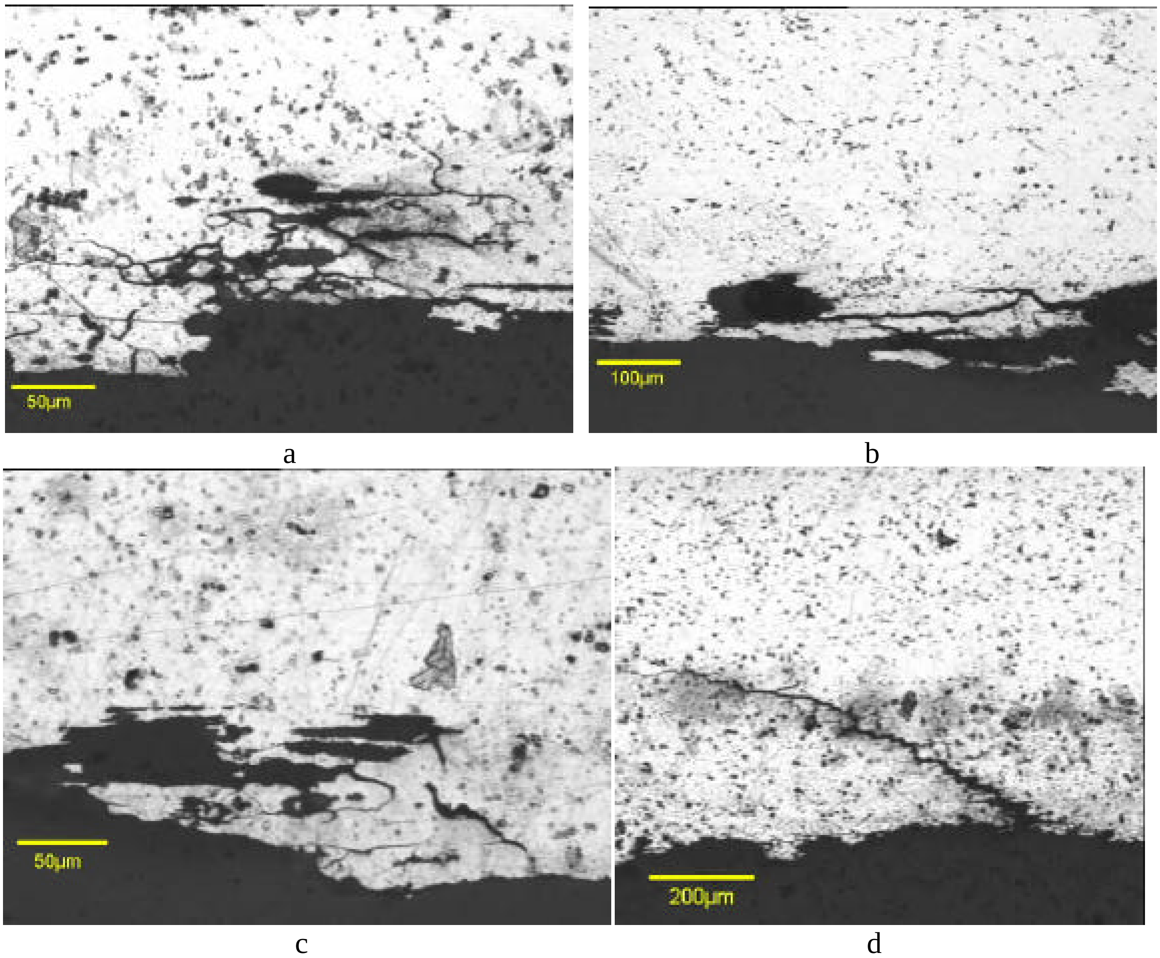


Figure 28. Optical micrographs showing damage along faying surfaces. (a) corrosion pit with intergranular attack at base, (b) intergranular attack at faying surface and corrosion pit intersecting a void in the base 2024-T3 material, which may have been caused by the corrosion consuming a discontinuity, (c) corrosion pits and intergranular attack, (d) environmentally assisted cracking under sustained stress.

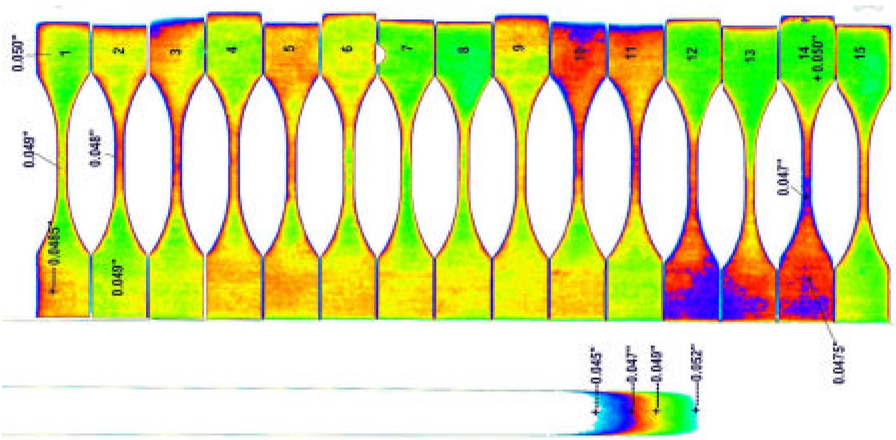


Figure 29. Thickness maps developed from radiographs of coupons.

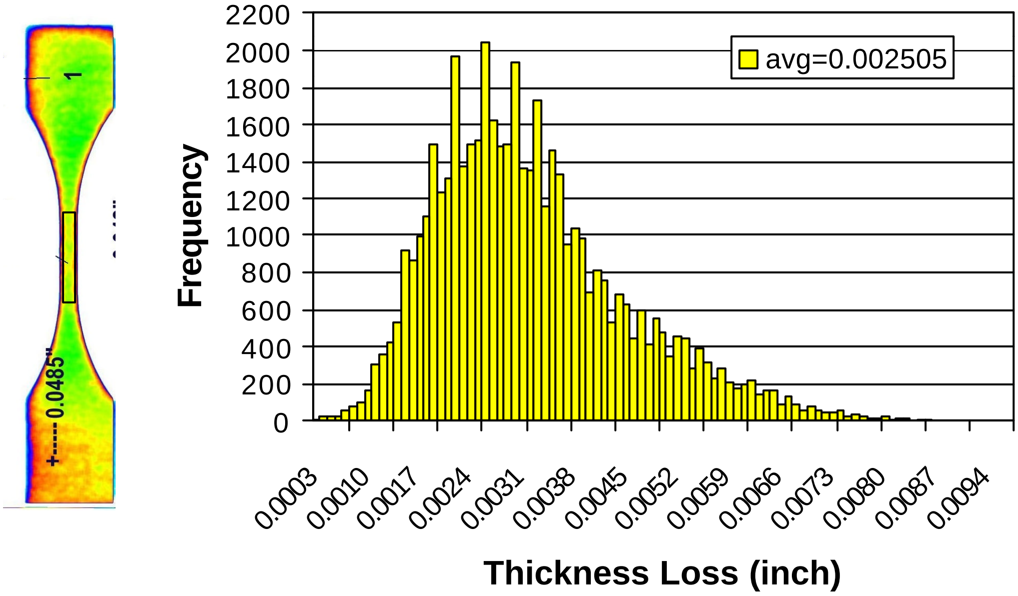


Figure 30. Histogram plot of thickness loss of coupon with average thickness loss on gauge length of 4.85% in a 0.0516 inch thick skin.

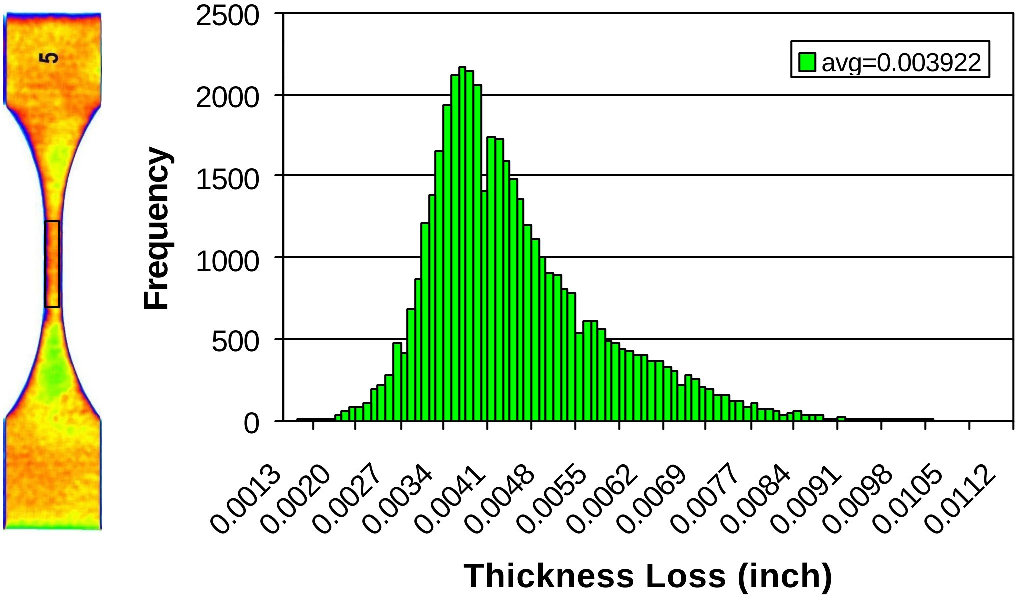


Figure 31. Histogram plot of thickness loss of coupon with average thickness loss on gauge length of 7.60% in a 0.0516 inch thick skin.

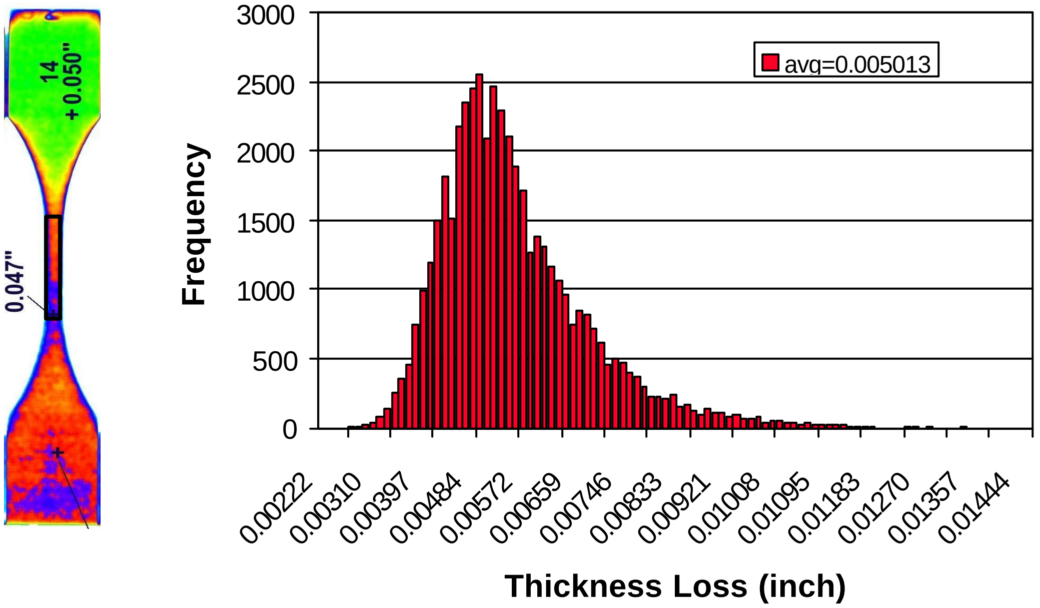


Figure 32. Histogram plot of thickness loss of coupon with average thickness loss on gauge length of 9.72% in a 0.0516 inch thick skin.

A log normal distribution was fitted to the histogram data that was generated for each coupon and the results for three of these coupons are shown in Figure 33. Using the parameters obtained from the log normal distributions, the coefficient of variation for each coupon was calculated by dividing the standard deviation by the mean. The results are plotted in Figure 34. As can be seen from this figure, as the thickness loss in the gauge length increased the coefficient of variation, which is a measure of the scatter in the surface topography, decreased.

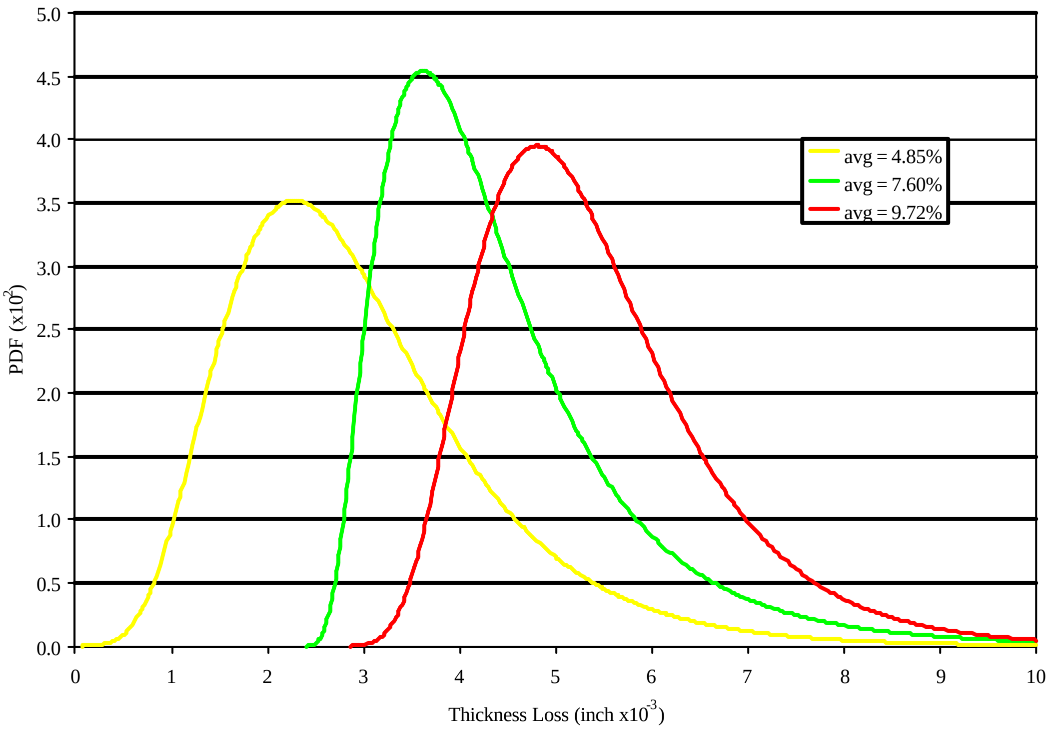


Figure 33. Probability density function plot of thickness loss for three coupons.

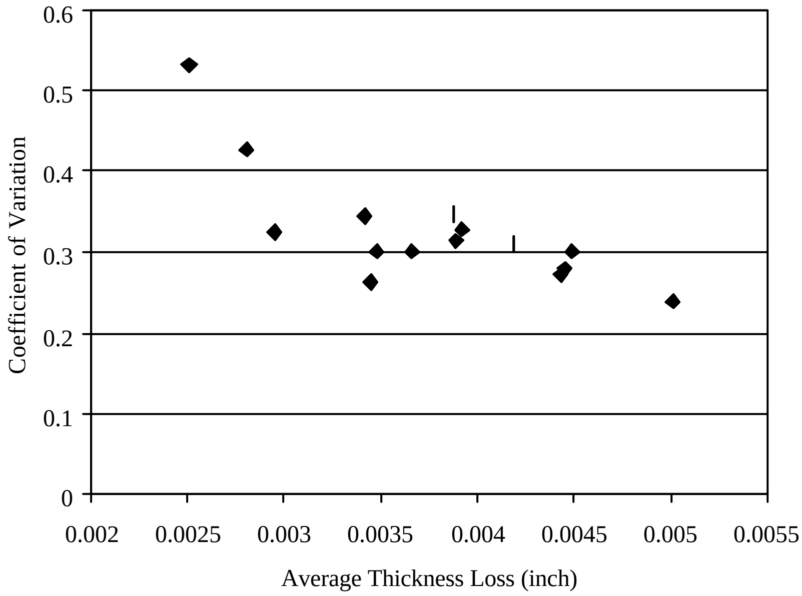


Figure 34. Coefficient of variation plot.

During the CFSD program, these coupons were subjected to cyclic load until failure and the fracture surfaces were examined using a scanning electron microscope (SEM) to determine the crack nucleation sites. The nucleation sites were located on the thickness map and the average and maximum thickness loss in the vicinity was determined. Table 1 to Table 3 show the results from this study for the three coupons shown in Figure 30 to Figure 32. It is interesting to point out that although the average thickness loss in the vicinity of the nucleation site was higher than the average loss in the gauge length, the maximum thickness loss did not occur in the vicinity of the crack origin.

The SEM results were used to measure the pit size that was present at the nucleation site for each coupon. These measurements, length along faying surface and depth were then used as input into the AFGROW crack growth rate to calculate the number of cycles to failure. The results are shown in Table 4. As can be seen from this table, the predicted values were consistently over-estimating the number of cycles to failure. However, for all the coupons examined, an area was present around each nucleating pit that was unique to the fracture surface as shown in Figure 35 for the 4.85% coupon. When the depth of this discontinuity was added to the depth of the pit and the number of cycles to failure was re-calculated, the predicted results were very close to the experimental results as shown in Table 4. When an energy dispersive x-ray analysis was carried out in this area for the 4.85% coupon, it was found that the levels of copper, magnesium and manganese were all below the levels that are normal for the 2024-T3 base material. However, away from the pit and the discontinuity, these levels returned to normal. It is not known at this time what caused these particular elements to decrease.

Location	Average (inch)	Maximum (inch)
Nucleation Site (area .038 x .038)	0.004250	0.005602
Gauge Length (area .15x.45)	0.002505	0.014108

Table 1. Thickness loss results for 4.85% coupon.

Location	Average (inch)	Maximum (inch)
Nucleation Site (area .038 x .038)	0.005313	0.006602
Gauge Length (area .15x.45)	0.003922	0.010862

Table 2. Thickness loss results for 7.60% coupon.

Location	Average (inch)	Maximum (inch)
Nucleation Site (area .038 x .038)	0.008005	0.008602
Gauge Length (area .15x.45)	0.005013	0.014108

Table 3. Thickness loss results for 9.72% coupon.

It should be pointed out that the current version of ECLIPSE assumes the existence of such a discontinuity at the bottom of crack nucleating pit. ECLIPSE is a software package based on a holistic life assessment including time and cyclic effects. It is a framework used to analyse aircraft structures to determine the impact of corrosion and fatigue on structural integrity [30].

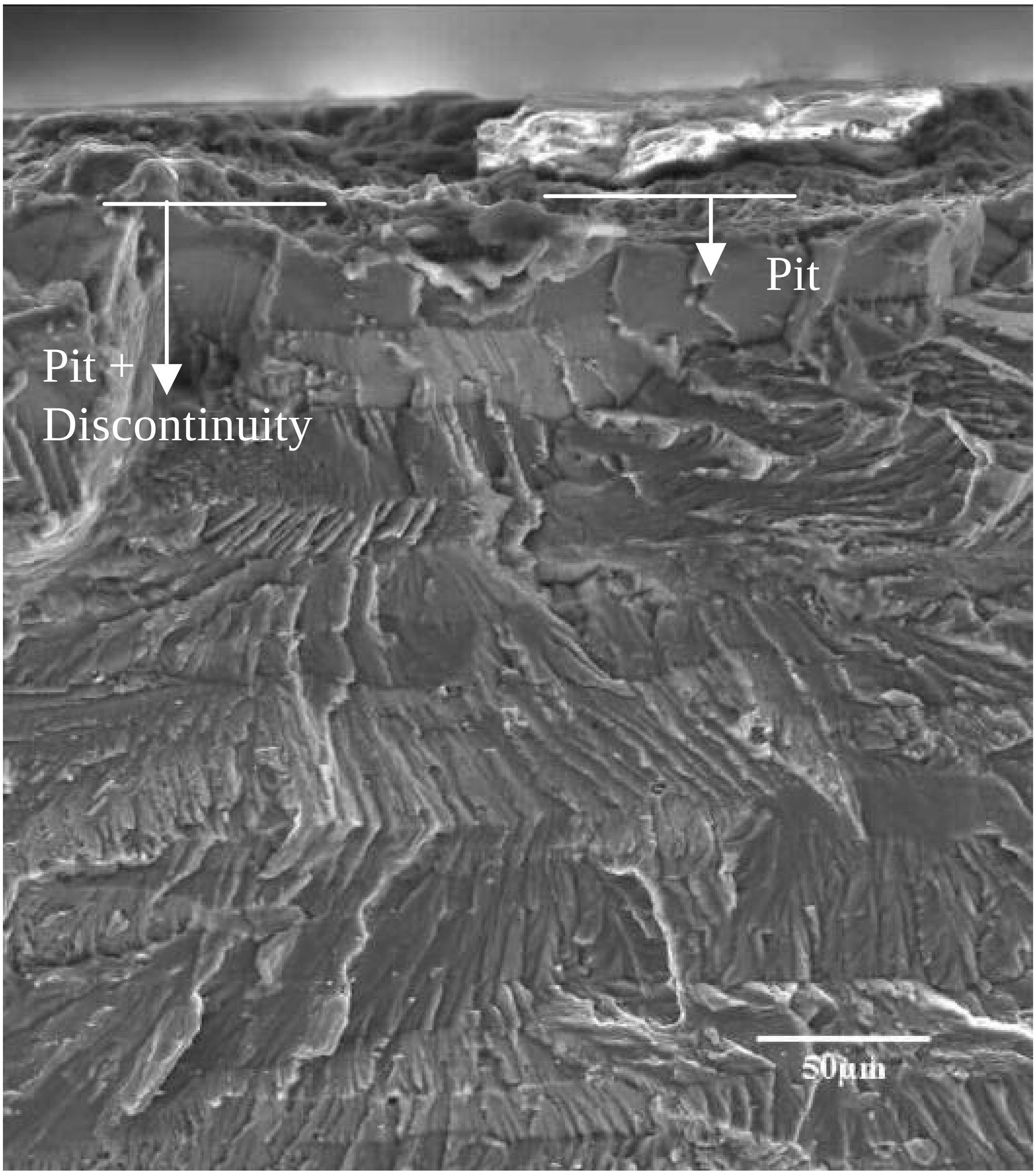


Figure 35. Scanning electron micrograph of a nucleation site.

Coupon	Predicted Cycles to Failure		Experimental Cycles to Failure	Percentage Difference	
	Pit Depth	Pit + Discontinuity		Pit	Pit + Discontinuity
4.85%	295,100	252,00	251,711	17.2	0.11
7.60%	341,602	215,387	249,146	37.1	-13.5
9.72%	277,100	221,800	210,447	31.7	1.11

Table 4. Predicted number of cycles to failure for three coupons.

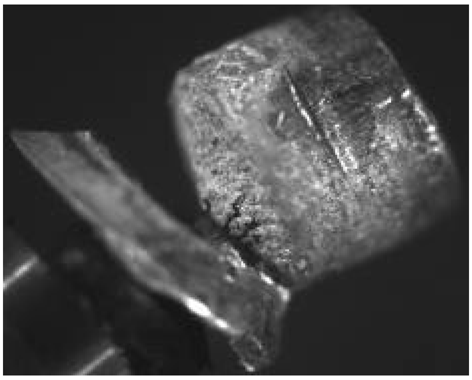
Failed and Failing Rivets - The Relationship to Corrosion Pillowing

R.W. Gould, SMPL, IAR, NRC

A number of naturally corroded aircraft fuselage joints originating from retired airframes have been inspected with traditional and novel non-destructive techniques. These joints have then been disassembled to precisely determine the corrosion damage to the skins and to assess the accuracy and utility of the applied inspection techniques. During on-aircraft maintenance damaged rivets are typically drilled out and replaced individually. In some of the joints selected for disassembly at IAR/NRC, corrosion pillowing had resulted in visually detectable failure or partial failure of one or a number of rivets. Examples of previously replaced rivets were also included indicating failure of rivets at lower corrosion levels. Careful disassembly for laboratory analysis has permitted the assessment of all of the removed rivets. Many rivets were found to contain cracks. These cracks occur primarily at the skin sheet interface rather than at the base of the countersunk type rivet head. These cracks are partial or completely around the rivet shank. The defective rivet population ranged from 50% to 75% of the rivets in the joints selected for disassembly (Figure 36). Work is being carried out to determine the residual strength of these damaged rivets. The disassembled and cleaned joint skins were x-rayed and digitally mapped to precisely determine local thickness loss. An attempt is made to associate corrosion damage and corrosion pillowing to rivet damage. General maintenance practices might have to be altered to require replacement of more than the visibly failed rivets to ensure joint integrity.

Of a total of 85 rivets, 3 are popped and 32 are cracked. The damaged percentage by row is 50% top row, 45% middle row and 27% bottom row.

Of all the cracked rivets, about 5 are 360 degree cracks, and all others are 45-180 degree cracks. All cracks lie at the base of the countersink.



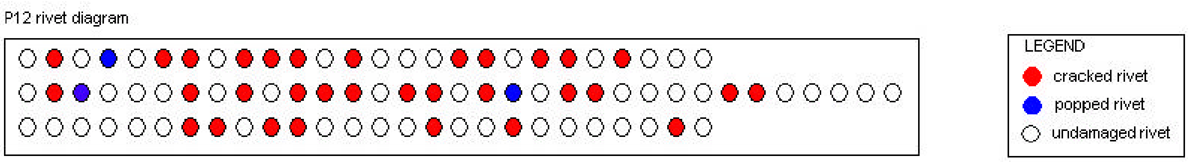


Figure 36. Section of a corroded lap joint from a retired aircraft with large population of failed (popped) or cracked rivets. Photograph shows a partially cracked rivet.

Multiple Site Damage Corrosion/Fatigue

G.F. Eastaugh, SMPL, IAR, NRC

The Multiple Site Damage Corrosion Fatigue Project of the IAR/NRC has been a collaborative effort involving Carleton University with some funding support from the Canadian Department of National Defence (DND). As reported in previous Canadian National Reviews, the work started several years ago with the development of special MSD specimen to simulate aircraft conditions and the MSD failure mode relatively inexpensively in a uniaxial load frame. This specimen was first used to investigate the MSD failure mode. To assist in this work, a computer model based on compounding of stress intensity solutions was developed to simulate visible MSD crack growth. Later, a procedure was developed for performing corrosion/fatigue testing using the same specimen. This incorporated an accelerated corrosion procedure previously developed by the IAR/NRC for research into NDI methods and corrosion pillowing. This procedure is applied to the finished specimen and so simulates the damage, pillowing stress and other corrosion effects experienced in service. To check that the corrosion procedure was sufficiently representative for corrosion/fatigue testing, corrosion in fuselage splices cut from retired aircraft was compared macroscopically and microscopically with accelerated corrosion in MSD specimens [31].

Exploratory testing using pre-corrosion/fatigue and alternating corrosion/fatigue sequences was performed between 1994 and 1997. In 1998 a statistically designed parametric study of the effect of pre-corrosion on the durability and MSD crack growth characteristics was started. All specimens were manufactured under carefully controlled conditions in a single batch to minimise nuisance variables. A total of twenty-three specimens out of this manufacturing batch were used for a randomised paired comparison test program. Testing of these corroded and non-corroded specimens has been proceeding over the past three years. Initially, funding was provided by the IAR/NRC and the DND. Future testing will be funded under the USAF/Lockheed Martin Corrosion Fatigue Structural Demonstration (CFSD) program. The basic fatigue test results from the first phase of the program were summarised in Reference [31]. They point to a 50% reduction in durability (cycles to the detection of the first crack) in specimens corroded to a level of 5% to 6% average thickness loss per sheet. At this level of corrosion the cladding has been consumed in large regions and localised pitting has occurred in the core metal.

Non-corroded specimens invariably failed as a result of MSD crack growth from nucleation sites near the rivet holes in a region of fretting. In corroded specimens, MSD was less apparent, since the failure mode tended to be dominated by crack growth from one nucleation site. This site was usually located at pitting in the core 2024-T3 alloy. In two specimens the dominant crack originated at pitting away from rivet holes in the inner sheet below the lower rivet row (in aircraft orientation). On an aircraft this location would be difficult to inspect. Other important insights into corrosion/fatigue interaction in fuselage splices are discussed in Reference [31]. Experimental data of this kind can be used to construct statistical equations that model the initiation and growth of detectable MSD cracks in corroded and non-corroded splices [32]. Such equations can be used directly in the durability and damage tolerance analysis of fuselage structure.

In view of the association between pitting and crack nucleation in corroded specimens, an investigation has been started of the relationship between the topography of corrosion damage and fatigue crack nucleation and growth. Cost-effective techniques have already been developed for measuring the topography at a high enough resolution for whole life crack growth analysis using the corrosion/fatigue modelling approach proposed by C. Brooks of AP/ES Inc. [33]. These involve x-radiography of the corroded sheets after the splice has been dismantled and chemically cleaned of corrosion product. Improved radiographic and data reduction procedures have been developed, based on a thickness mapping technique developed earlier by the IAR/NRC. Chemical cleaning procedures have been developed that avoid any further damage to the metal. In addition, prototype software has been developed for manipulating the large data files and characterising the topography in statistical forms useable in probabilistic prediction of whole life crack growth.

These methods are also being applied to the calibration and improvement of NDI methods used to measure corrosion damage in intact MSD specimens prior to fatigue testing. This NDI calibration work is intended to maximize the advantages of predictive corrosion/fatigue modelling in improving reliability, reducing maintenance costs and increasing operational flexibility.

The technology for topographical measurement and analysis developed by the IAR/NRC and Carleton University will be used for modelling and NDI calibration purposes on aircraft specimens as well MSD specimens in the CFSD program. This work will be done under contract with Lockheed Martin Aeronautics (R. Bell) and the in collaboration with other subcontractors, primarily APES Inc. (C. Brooks), University of Utah (Professor D.W. Hoeppner), SAIC (R. Grills), and Northrop Grumman Corporation (D. Carmody).

Participation in USAF/Lockheed Martin Corrosion Fatigue Structural Demonstration Program (CFSD)

G.F. Eastaugh, SMPL, IAR, NRC

The IAR/NRC was a major participant in Phase 1 (1999) of the Corrosion Fatigue Structural Demonstration Program initiated by the US Air Force Research Laboratory. This participation has continued in Phase 2 (2000-2002). Lockheed Martin Aeronautics Company, Marietta is prime contractor. The IAR/NRC is performing work in the following areas:

- the nucleation of fatigue cracks in aluminum alloys and the correlation of nucleation sites with intrinsic discontinuities;
- the development of cracks in pitted and exfoliated structure, particularly fuselage skin splices and wing skins;
- the characterization of the various stages of corrosion damage and the development of metrics for corrosion/fatigue modeling;
- and the development of relationships between these metrics and current NDI capabilities.

In addition, the IAR/NRC will be contributing models of crack growth at exfoliation that are currently being developed under another project.

Risk Analysis of Fuselage Splice Joints with Multiple Site Damage (MSD)

M. Liao, SMPL, IAR, NRC

A risk analysis has been carried out for fuselage splice joints containing MSD and corrosion using the software PRISM (Probabilistic Crack Growth Analysis for RISk Management) developed by Bombardier Aerospace Inc. The crack growth data, which were obtained in a previous IAR/NRC test program, were used in preparing the input data for PRISM and for validating the analysis. The MSD crack in the specimens was characterized by a single aggregate crack and the median crack growth curve was derived for this analysis. The initial quality state of the splices was represented by an equivalent initial flaw size distribution (EIFSD) (Figure 37), which was backtracked using the distribution of time to crack initiation (TTCI) (Figure 38). The critical crack length criterion was employed to predict the probability of failure (POF) at first linkup and final failure. For comparison purposes, the USAF software PROF (PRobability Of Fracture), was also run to predict the POF using the same set of input data [34][35].

The analysis results indicated;

- The predicted POF results for uncorroded and corroded specimens were in good agreement with the test data (Figure 39).
- PRISM provided almost identical POF results to those from PROF, and PRISM was found to be an easy-to-use and capable tool for probabilistic durability and damage tolerance analysis and risk assessment for ageing aircraft.
- The accuracy of the POF predictions of the MSD specimens was also found to be largely influenced by the Equivalent Intial Flaw Size Distribution (EIFSD) used. A significant improvement in the POF prediction was achieved using the EIFSD input derived from accurate Time To Crack Initiation (TTCI) distributions (Figure 37 to Figure 39). The input data preparation for the risk assessment codes played a key role in obtaining accurate predictions of POF.
- Corrosion, which caused the material loss and pillowing effects in fuselage splice joints, can have a significant influence on the POF.

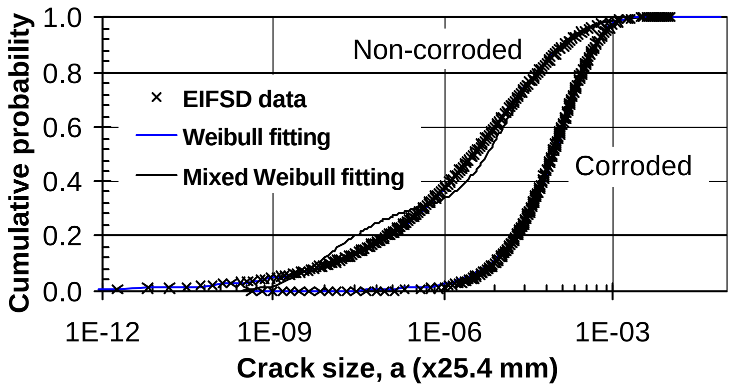


Figure 37. EIFSD for uncorroded and corroded MSD specimens.

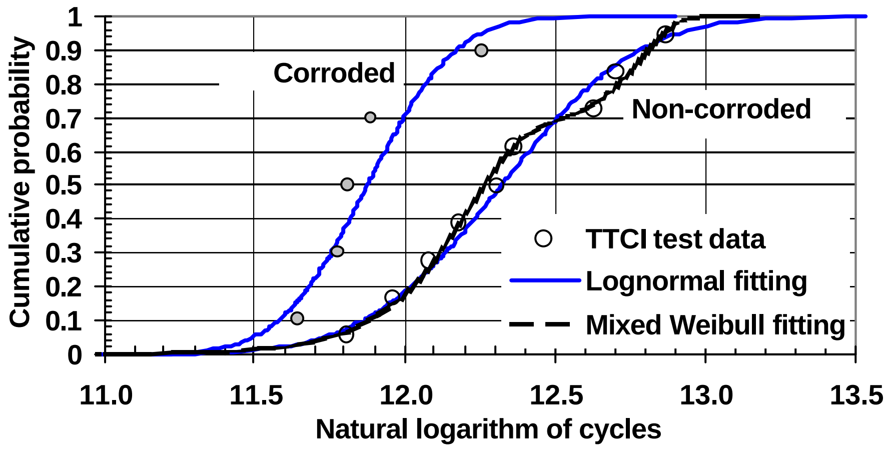


Figure 38. TTCI Distributions for uncorroded and corroded MSD specimens.

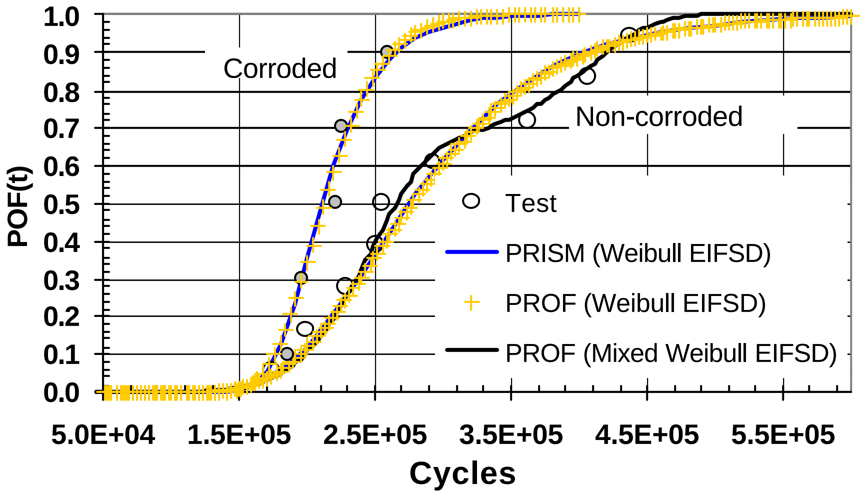


Figure 39. POF results calculated by PRISM and PROF.

Prediction of Fatigue Life Distribution of Fuselage Splices

M. Liao, SMPL, IAR, NRC

Due to the inherent uncertainties in manufacturing, loading, geometry and material properties, the onset of multiple site damage (MSD) of fuselage splices in ageing aircraft has a large variability. This variability can be best described using a statistical distribution, and this distribution is one of the most important parameters for risk analysis of fuselage splices in ageing aircraft. IAR/NRC has, for several years, investigated the fatigue characteristics of fuselage splices containing MSD and corrosion. Extensive test data have been obtained for corroded and uncorroded MSD specimens. Research work is underway at IAR/NRC to model MSD and corrosion with the objectives to: 1) develop engineering approaches to predict the fatigue life of uncorroded fuselage splices, and 2) extend these methodologies to evaluate the impact of corrosion on the structural integrity of fuselage splices.

At the beginning of this work, methodologies to predict the fatigue life distribution for the uncorroded MSD specimens, measured as the number of cycles to visible cracks, were developed as follows:

- Modeling procedures using 3D nonlinear finite element analysis (FEA) were developed to obtain the stress distribution at the rivet hole. The squeezing force (SF) resulting from the riveting process and the coefficient of friction (CoF) used for the contact surfaces were taken as random variables.
- Analytical expressions for local stress as a function of SF and CoF were developed using a response surface technique along with limited FEA.
- Based on the calculated local stresses, a strain-life approach, Smith-Watson-Topper (SWT) model, was employed to predict fretting fatigue crack nucleation at the rivet hole.

- Subsequent crack growth to a visible crack was predicted using the crack growth life prediction program, AFGROW.

A Monte Carlo simulation was developed, which integrated the random variables (SF and CoF) into the models, to determine the fatigue life distribution to visible cracks (Figure 40).

	Squeeze Force (SF)	Coefficient of Friction (CoF)	Fatigue life distribution		
			Prediction	Test	Error
Distribution	Type-I	Normal	Lognormal	Lognormal	
Mean	3520 **	0.5	271807	271724	0.031%
CV *	5%	5%	33.9%	34.6%	-2.136%

* Coefficient of variation.
** Obtained from *Trail and Error* method, this value is close to the direct test measurement made by R.P.G. Müller for the similar riveted lap joint.

Table 5. Numerical comparisons between prediction and test results.

Results from the simulation showed that the predicted fatigue life distribution correlated very well with the existing test data (Table 5 and Figure 41). The developed methodologies can be used to predict the fatigue life distribution of uncorroded fuselage splices. Further sensitivity studies indicated; 1) the rivet squeeze force has a stronger influence on the life distribution than the coefficient of friction, and 2) the scatter in the squeezing force and the coefficient of friction have a stronger influence on the life distribution than that of material properties [36][37].

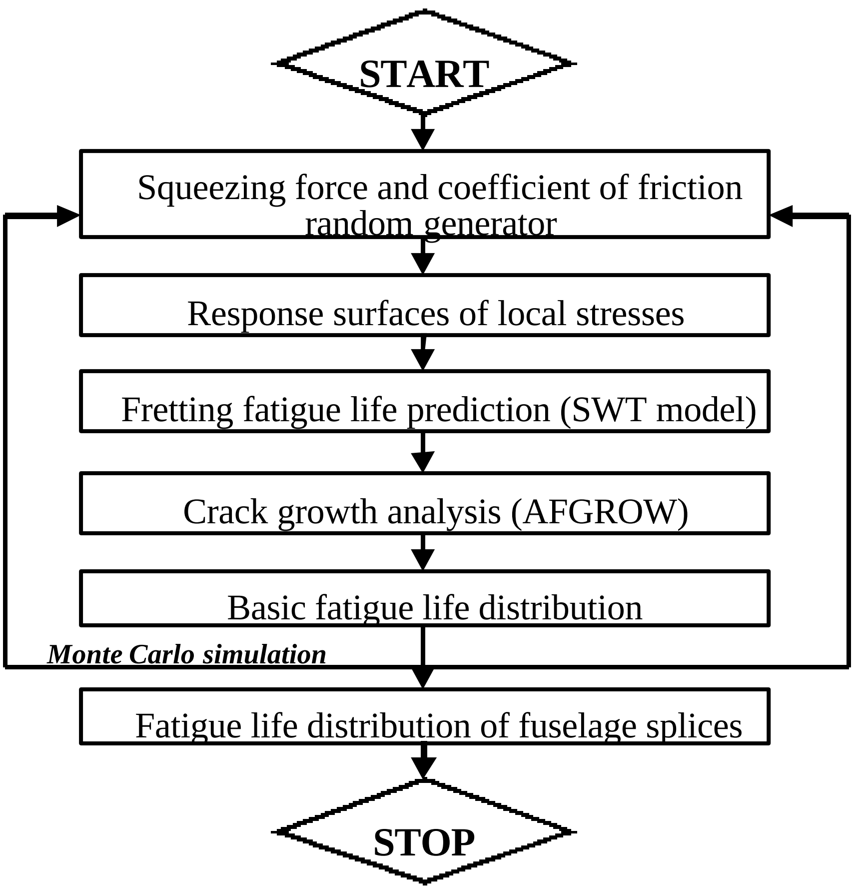


Figure 40. Schematic flowchart of Monte Carlo simulation.

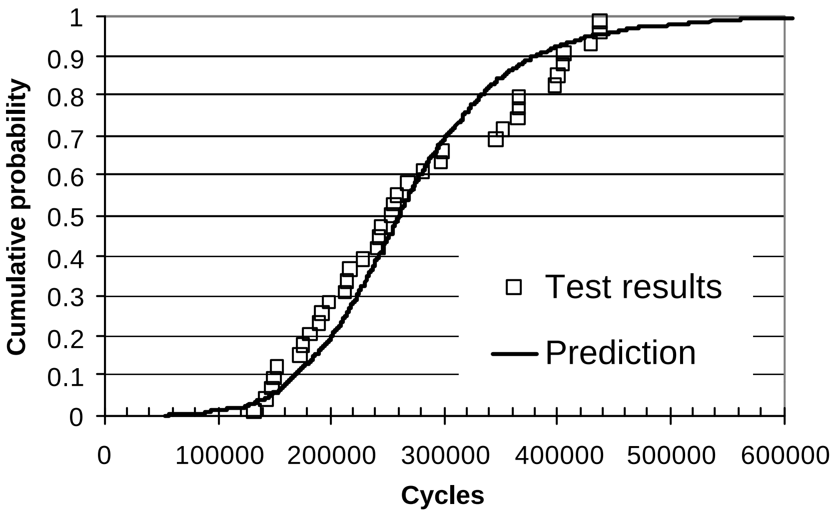


Figure 41. Distribution of fatigue life to visible cracks.

CF188 Aircraft Life Extension (ALEX) Program

J. Dubuc, Bombardier

Numerous fatigue problems have been discovered in-service or on the IFOSTP full-scale fatigue tests since 1995 that were not previously addressed in the OEM (Boeing) Engineering Change Proposal (ECP) program. In 1997, the ALEX program was instated by the Canadian Forces (CF) to address these problems on fleet aircraft with either repairs or preventive modifications. As some modifications are considered urgent while others will only be identified in upcoming teardown inspections on IFOSTP, a two phase approach is being used. The first phase includes a total of 23 modifications and has been in full production since January 2000 in Bombardier`s third line facility located in Mirabel, Quebec. The second phase includes 24 modifications thus far, with a projected total of at least 60 by the end of the IFOSTP tests. The ALEX Phase 2 is currently in the development/prototype phase and is scheduled to enter full production in early 2004.

Several of the fatigue problems addressed by the ALEX program are quite challenging as the CF188 airframe includes several features which enable it to operate at fairly high stress levels, e.g. monolithic machined 7050 plate, cold expansion of holes, interference fit fasteners, use of IVD as opposed to anodizing. For several design details, like fillet radii of machined pockets, the only viable alternative for preventative modification is to use shot peening. Both the OEM and Bombardier have conducted comprehensive coupon test programs (e.g. Y470 Bulkhead X19 area) in the past for the CF in order to demonstrate the benefits of shot peening. A more recent application, the Y453 Web Taper Area, has been found to be more difficult to access during third –line maintenance than the X19 area and impossible to access for follow-on inspections. Hence, the CF and Bombardier have mutually agreed to enter a joint R&D program to develop an in-situ robotic shot peening system, which shall provide a higher level of quality and consistency in peening.

Bombardier uses CI89 (under license from Boeing), a strain based crack initiation software, as a basis for the analysis of these modifications. Coupon testing is often used where analysis and generic test data alone appear to lack the desired level of accuracy for certification. This is usually the case for shot peening as each area has a unique combination of material and stress distribution. One coupon test program was already initiated with QETE for the above Web Taper area and at least two more programs will be launched in 2001. It is planned that the effect of the pre-IVD etching will be assessed where applicable as recent reports from AMRL in Australia, an IFOSTP partner, have indicated that this process affects the crack initiation mechanisms and response to shot peening of aluminium.

Two cases have recently been identified where cold expansion of holes occurs in non-standard conditions and thereby require dedicated coupon testing. The first is a case where cracks were discovered both at the edge and at the periphery (radial but away from the hole) of a cold-worked hole in a wing-carry-thru bulkhead (see next section). The problem is caused by the unique combination of a very high stress field interacting with the pre-IVD etching process. The second is a case of a cold-expanded hole in seal-groove in a wing spar 38. In both cases, Bombardier has tasked Fatigue Technology Inc (FTI) to study these problems and perform the testing in order to develop well-adapted cold-expansion techniques.

CF188 ALEX Modification to Y453 and Y488 Crease

J. Dubuc, Bombardier

The subject modifications were first designed as repairs to the FT55 test article which displayed very large crack in these areas. These areas of the wing-carry-thru bulkheads sustain moderate compressive loads but are highly curved and thereby develop cracks parallel to the main loading because of the induced out-of-plane deformations. The repair concept in this case is to reduce the amplitude of the deformation using contoured steel doublers on the mold-line of the aircraft.

Safe life is the desired certification basis for non-inspectable fracture critical areas. Hence, a comprehensive certification program was initiated in parallel for these modifications, including the AETE PD98/26 flight test program as well as a qualification program for inspecting and blending cracks in fleet aircraft. The qualification program was aimed at training the personnel involved and determining the sensitivity of the eddy current inspection technique as well as the accuracy of the blending which can be achieved with the specially developed tooling.

Bombardier has instrumented the flight test article (aircraft 188701) and has modified it on one side in order to provide comparative data from a crack free specimen, which will form the new basis for the final certification. Flight testing has started at AETE in Cold Lake, Alberta in January 2001. The data will be analyzed and combined with Finite Element Modeling to determine safe-life limits for the various blending depths used on fleet aircraft.

CF188 Y453 Bulkhead Web Taper Coupon Test Program

M. Sova, QETE

Full scale structural testing of the CF188 has highlighted fatigue cracking problems in the region of the Web Taper at the Y453 Bulkhead. BADS has developed a structural repair for this region that consists of a shot peening application to enable the component to achieve a full fatigue life. QETE has initiated a coupon testing program to verify the effectiveness of this repair and to certify its fatigue life. As part of the process, the design detail surfaces were prepared using a chemical surface etching process which simulated that implemented by the OEM for the application of Ion Vapor Deposition coatings. The process included a chemical degreasing, an acid etch (aluminum pickling), deoxidizing bath and final cold water rinse, similar to the process documented by DSTO ARL for their component level fatigue tests of F-18 bulkheads. Preliminary results indicate that during fatigue testing to the IFOSTP spectrum pertinent to this area, fatigue cracks are developing at the predicted number of spectrum blocks. Follow-on tests to determine the effectiveness of proposed shot peened repairs to this fatigue sensitive area are in progress [39].

EXPERIMENTAL AIRCRAFT

Ornithopter

J. DeLaurier, University of Toronto, Institute for Aerospace Studies.

The Ornithopter project was described in the previous Canadian ICAF National Review. It was expected that the first wing-flapping piloted aircraft would be flown in the fall of 1999. However, just as it was undergoing final taxi runs the right vertical link buckled and broke and the right wing disintegrated. The unbalanced lift then caused the aircraft to flip and tumble, skidding to a halt upside-down on the runway. The pilot was not injured. The cause was as follows:

In the course of designing the aircraft, a structural analysis was performed in which the vertical links were sized to withstand flight loads, along with a generous safety margin. However, the subsequent taxi trials showed that successful ground-handling strategy requires a nose-down attitude to suppress the lift buildup and consequent bouncing. In other words, a downward lift is needed to keep the aircraft planted on the runway while the speed builds up to the value for flight. At that point, the nose is lifted and the aircraft takes off. In the flight condition, the loads on the vertical links are mainly tensile (pulling). However, the nose-down ground run gave excessive compressive (pushing) loads that the links were incapable of withstanding. The inertial-reaction forces probably added to these loads from whatever ground bouncing occurred. Thus these loads were more critical than the in-flight loads (plus safety margin).

Since the accident the aircraft has been repaired and in some areas design changes were incorporated. The flapping wing aircraft is currently being readied for flight (Figure 42).

Figure 42. Repaired Ornithopter spring 2001.

JOINING TECHNIQUES

Friction Stir Welding

A. Merati, SMPL, IAR, NRC

Friction Stir Welding (FSW) is a new process for joining metals without melting. It has been used successfully with aluminium alloys, including alloys that have been considered to be non-weldable by conventional means, and therefore could be used in aircraft construction (as well as in other manufacturing industries such as rail vehicle construction and shipbuilding).

IAR/NRC is participating in a project coordinated by The Technical Co-operation Program TTCP MAT-TP-1 Operating Assignment O-28, "Stress Corrosion Control in Friction Stir Welds in Aluminium Alloys" focused on documenting properties of joints produced by friction stir welding in aluminium alloy structures. The objective of the project is to assess the viability of the FSW process for use in assembly or repair of military transport systems.

In support of the TTCP initiative, coupon fatigue tests and CT crack growth rate tests are being performed on FSW parts provided by other participants as part of the study to investigate the effects of the welding technology on the fatigue behavior. Figure 43 shows an example of fatigue performance in a transverse direction comparing samples removed from parent material AA 7050-T7451, with bead on plate FSW specimen.

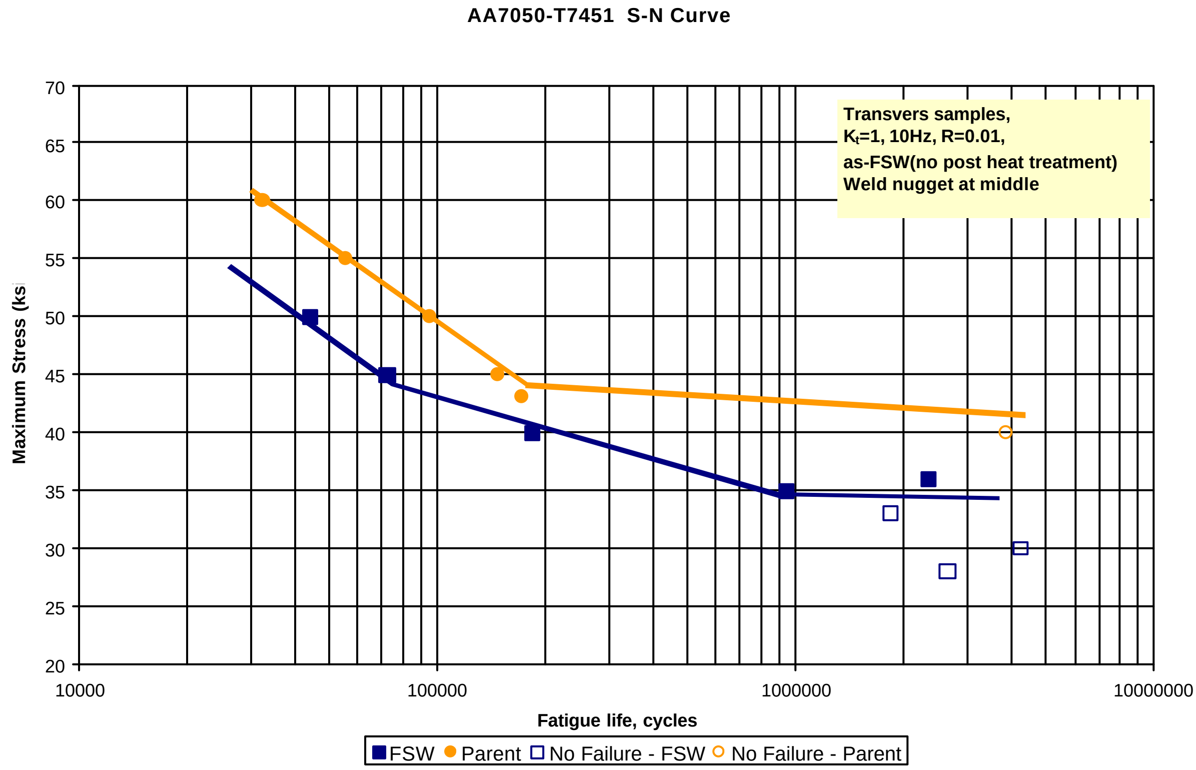


Figure 43. The fatigue lives of 7050-T7451 aluminum plates, for parent material and as friction stir welded conditions in transverse direction is compared.

Characterization of Laser Welded Joints

C. Poon, SMPL, IAR, NRC

Laser welding is a relatively recent development which promises several advantages over conventional riveted construction technology. In compression dominated areas, such as the lower fuselage, the welded structure will be on the order of 10% lighter, due the more efficient use of stringer material and the elimination of sealant and fasteners. The welded structure will be more resistant to corrosion due to the use of appropriate alloys and the elimination of faying surfaces. Lasers can produce joints at the rate of 10m/min or more, on the order of 100 times faster than an automatic riveting machine. This will result in significant reductions in the cost of manufacture.

In order to develop laser welding technology sufficiently to permit its use in certificated commercial aircraft, a comprehensive research program is required. This involves a variety of engineering disciplines, including materials science, metallurgy, structures and manufacturing.

Bombardier Aerospace has initiated a research and development program in this area in collaboration with Automated Welding Systems Incorporated (AWS) and the IAR/NRC. The main task of IAR/NRC is to characterize the microstructure and mechanical properties under static and cyclic loading of laser welded aluminum joints. The objective of the characterization work is to provide technical data for optimizing the laser welding process and for developing fracture mechanics models that can be used for life prediction of laser welded joints [40].

Preliminary work has begun to determine the tensile properties of laser welded Al6013-T6 sheets with variations in the filler wire. Three different filler wires were used in the laser welding process. The filler wires used on the Al6013-T6 sheets were as follows: Al4043, Al4047, and Al5556. In addition, a test panel manufactured from Al2219 sheets with an Al2319 filler wire was also tested. The laser welding was performed using a 4 kw Nd:YAG robotic laser system procured by IAR/NRC from AWS.

Constant amplitude fatigue tests are underway to determine the crack nucleation and propagation characteristics of laser welds. Crack growth measurement is being done using a travelling microscope. Different nondestructive inspection techniques including X-ray, eddy current, and edge of light will be used to improve crack detection and measurement.

Finite Element Design and Analysis of Composite Bolted Patch Repair

C. Poon, SMPL, IAR, NRC

The research on bolted patch repair has been underway in the past four years with an objective of developing a capability for design optimization of a patch repair for damaged composite plates using mechanical fasteners. This type of repair, so-called "temporary and quick fix", is mainly used for air battle damage because it can be easily implemented and removed. Although a patch can help recover the strength of the damaged part, the use of fasteners causes new localized stress concentrations at fastener holes. The design challenge is to achieve a balance between the strength recovery and localized stress risers and the key parameters to be considered include patch thickness, number of fasteners and spacing, diameter of fasteners and their locations. In order to obtain an optimum design, the localized failure at a fastener hole must be predicted using appropriate failure criteria integrated into the analysis tool [41].

The current focus of the investigation is a biaxially loaded composite cruciform specimen containing a central damaged hole of 25.4mm (1 in) diameter with the central area repaired by a GLARE patch using fasteners, as demonstrated in Figure 44. The 203.2mm x 203.2mm (8 in x 8 in) test section at the central area consists of 48 plies of AS4/3501-6 graphite/epoxy and the loading arms are of a combination of FM300K film adhesive and AS4/3501-6. The patch is made of GLARE 3 with different layers. Two-dimensional finite element analyses using MSC/NASTRAN were carried out for the repaired specimen. In addition, biaxial tests are being carried out to verify finite element results.

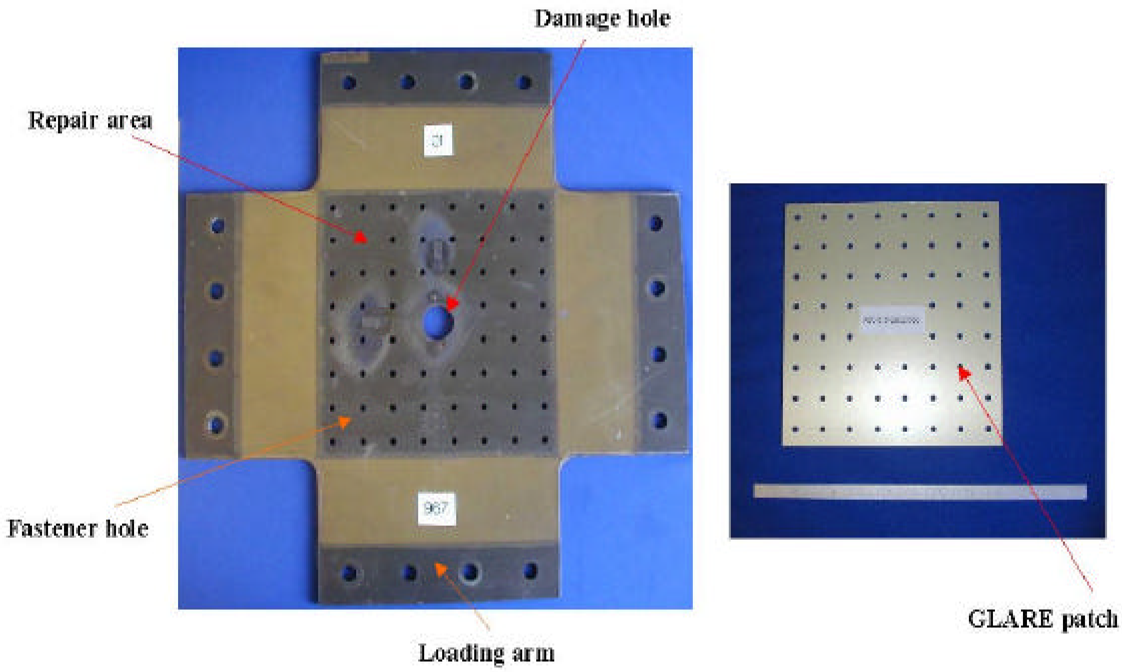


Figure 44. Composite Cruciform Specimen and GLARE Patch.

LANDING GEAR

Landing Gear Testing

M. Perrella, BFGoodrich (BFG) - Landing Gear Division

BFGoodrich (formerly Menasco Aerospace) has continued its extensive fatigue testing of its landing gear products. For details the reader is encouraged to review the 1997-1999 Canadian ICAF National Review. The typical test involves four life times, with 70,000 flights per life. These challenging tests run 24 hours per day 7 day a week. Test equipment specifications include high-response servo-valves, dual-bridge fatigue rated universal load cells, low friction hydraulic actuators for servo-control service, and sophisticated control and data systems. Fatigue tests run continuously and unattended other than periodic lubrication, maintenance and inspection. The CRJ-700 Main Landing Gear (MLG) Fatigue test is shown in Figure 45.

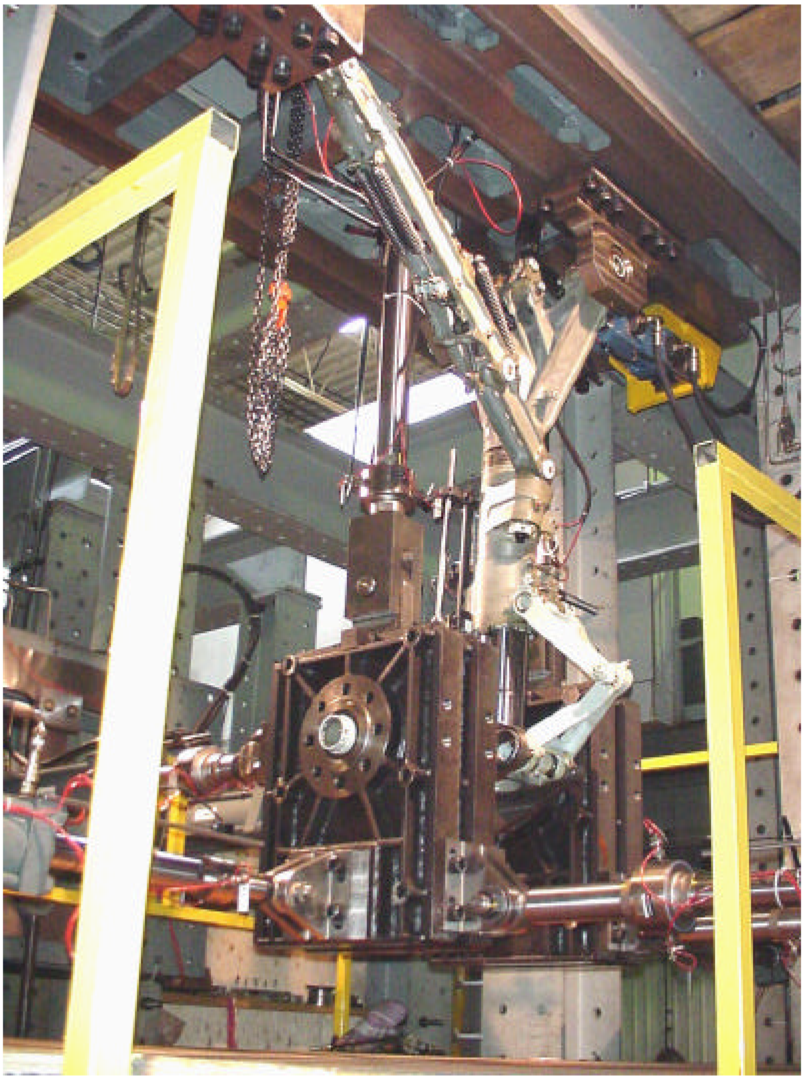


Figure 45. The CRJ-700 Main Landing Gear Fatigue test.

Compression Fatigue

R. Lee, Messier-Dowty

Background

During a routine inspection of a commuter aircraft, a crack was visually detected on the right hand main landing gear after approximately 7500 flights. This crack was located at the aft surface of the main fitting at the shock strut attachment lug to main fitting transition region.

Results from a metallurgical examination verified that the cracked main fitting was manufactured to the proper material specifications, that is, 300M steel, heat-treated to an ultimate tensile strength between 280 ksi and 300 ksi (1,930 – 2070 MPa). The fractography of the fracture face confirmed the presence of multiple semi-circular cracks. It was also concluded that the crack was a Mode I type fatigue failure.

A review of the design stress spectrum at the crack location showed that for typical operational loads, the stresses were predominantly compressive in nature. The finite element models showed stresses typically ranging from -235 ksi to 30 ksi (-1620 to 207 MPa), below the yield strength of the material. A flight test program was initiated to validate these analytical results. The main landing gear on a test aircraft was instrumented with strain gauges. These strain gauges were placed at the crack location as confirmed from both photoelastic survey and finite element analysis. Landing and ground manoeuvring conditions were performed at the extremes of the design envelope. The results from the flight-testing agreed with those predicted by the finite element analysis; stresses were primarily compressive.

Messier-Dowty's crack initiation analysis is based on a flight-by-flight sequence method that accounts for the structure's life history. To increase the accuracy of the life prediction, cycle counting using the Rainflow method is applied which accounts for hysteresis. The fatigue analysis using a plastic strain and elastic stress approach predicted insignificant fatigue damage at the failure location. Since the full-scale fatigue test article had successfully completed 480,000 flights, attention was focused on conditions beyond the design specifications.

With the occurrence of a second main fitting failure, the urgency of understanding the cause of the failure(s) had increased. Based on the geometry of main landing gear, any vertical load would create a compressive stress at the critical location. The occurrence of an aft acting load (i.e. spin-up or braking) would compound to the compressive stress. The likelihood of any significant tension stresses being created even from abnormal conditions was extremely remote. For this reason, research became focused on the phenomenon of compression fatigue.

Compression Fatigue

A structure subjected to a compressive overload, that is, above the yield strength of the material, at a stress concentration would induce a residual tension stress every time it is unloaded. These residual tensile stresses may be responsible for some fatigue failures in structures subjected to nothing but compressive loads. The observed failure was analysed for overload and load interaction effects using Heywood [42], Mordfin and Halsey [43] and Potter [44].

The sequence of the load application is critical for fatigue life of notched coupons. The Figure 46 below shows the results of an experimental investigation. In general, the specimens with the positive half cycle preceding the negative half cycle showed much shorter lives than their companion specimens subjected to the reversed sequence of preload. Different types of residual stresses (tensile or compressive) from the last overload are responsible for this effect.

Residual stress existing in a structure can be divided into two components at any given cycle in the structure's lifetime [44]. One component, the equilibrium part, is that portion of the residual stress, existing only because of the present nominal loading. The other component, non-equilibrium or relaxing part, represents the remaining, subjected to relaxation, amount of the residual stress that results from the preceding load history. This component is responsible for load interaction and sequence effects, influencing the local stresses for the following cycles.

The relationship between these two components is that if the equilibrium residual does not exceed the relaxing component, then this relaxing component continues to affect the structure and decay with a proper rate. Alternatively, if the equilibrium residual exceeds the relaxing part, the relaxing component is redefined with the value of the equilibrium residual from the overloading cycle.

In summary in a compression dominant stress spectrum, a compressive overload will result in a tensile residual stress upon unloading. For the subsequent load cycle, the local stress level and therefore the local mean stress in the spectrum tends towards a zero mean stress level. The extent of the effects from the residual stress on the entire load history depends on the cyclic relaxation of the material. If the tensile residual stress level remains constant during continued cycling, the fatigue life decreases. The fatigue crack would propagate under residual tensile stresses combined with tension stresses from the spectrum, if any.

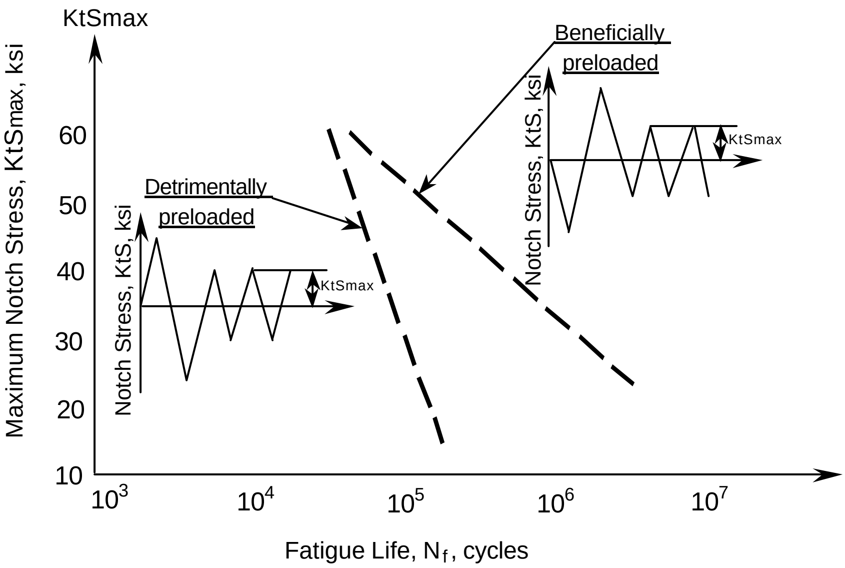


Figure 46. Effect of reversing the order of overload on the fatigue life of notched coupons, tested at R = -1 (1).

Conclusions of the investigation

Load conditions resulting in repeated yielding in compression are required for fatigue failures in compression-dominant stress spectra.

The existence of compressive overloads and cyclic relaxation provides explanation for the early in-service fatigue failures.

The existence of tensile residual stresses provides an explanation for the fact that the fracture face in the described failure has not been compromised under the compressive dominant load spectrum.

The tensile stresses from the overloads are able to propagate the crack under predominantly compressive loading conditions.

HVOF Coatings Fatigue Research

R. Lee, Messier-Dowty

Over the next five years it is anticipated that both military and commercial end users shall decline to take deliveries of chromium plated articles or use chromium electro-deposits for repair and overhaul on their aircraft. As a consequence, a comprehensive U.S./Canada project called HCAT (Hard Chrome Alternative Technologies) was established. The purpose of this project is to develop and share the technology of alternative chromium coatings. HVOF (High Velocity Oxygen Fuel) coatings have been selected as one of the most promising alternatives and as a participant in the HCAT project Messier-Dowty's contribution will be the fatigue testing of a military nose landing gear (NLG), the F/A-18 E/F NLG. Tasks being conducted by other participants in the program include civil landing gear testing, produceability criteria determination and design data and repair scheme development.

The F/A-18 E/F nose landing gear (NLG) that will be tested by Messier-Dowty will have HVOF coating on components that are normally plated with hard chrome. The major chrome plated components include the drag brace, shock strut piston and various pins. The fatigue testing activity is intended to provide the cognizant airworthiness authority with data which can be used to evaluate the replacement of chromium electro-deposits by HVOF coatings on landing gear parts for both existing and new aircraft. Direct application is anticipated for the US Department of Defence landing gears with a spin-off effect for the civilian landing gears after dialogue with prime aircraft manufacturers and airworthiness authorities.

The fatigue testing of the HVOF coated F/A-18 E/F NLG will take place in a newly constructed universal test rig. The rig is capable of performing two major or four medium size tests (strength or fatigue) simultaneously. The new rig is of modular design, which allows for rapid set-up, reduced rig design time, and reduced rig manufacturing and assembly time and cost. The rig has been designed for ease of operation and maintenance and, as well, the safety of the test article and personnel played a key role in the design of the new rig.

COMPOSITE MATERIALS AND STRUCTURES

Residual Stress Reduction in Bonded Patch Repair

A. Johnston, SMPL, IAR, NRC

IAR/NRC, in collaboration with DRDC, IMI, and the University of New Brunswick, is proceeding with the measurement and modelling of residual stress development during bonded composite patch repair. The overall objective for this work is to develop software tools to assist the user of bonded composite patch repair to develop "optimized" patch/adhesive curing cycles which will minimize stress and cycle time and maximize patch performance. To date, the team has characterized and developed models for the primary composite materials and adhesives of interest, developed closed-form solutions for stress development during cure, and measured patch warpage during cure for a number of different processing scenarios [45] to [47].

Low Cost Composite Aircraft Structures

A. Johnston, SMPL, IAR, NRC

IAR/NRC, in collaboration with Profile Composites of Sydney, BC., is investigating composite seaplane float fabrication using VARTM. The objective of this project is to design, fabricate and certify a low-cost composite seaplane float for the deHavilland Beaver. Composites provide an attractive alternative for this application due to the potential improvements provided in both aircraft capacity, due to lowered weight, and durability due to improved fatigue performance, and resistance to impact and corrosion. Vacuum Assisted Resin Transfer Moulding (VARTM) was chosen as the process to fabricate the floats. This out-of-autoclave process, while having been used for some time in marine applications is new to commercial aircraft structures. Initial processing trials showed that, with process optimization, VARTM could be used to produce high quality, high volume fraction laminates. However, subsequent testing demonstrated that the room temperature cure epoxy selected for this work could not provide the required hot/wet performance for this application. Work has thus been temporarily halted while alternatives are pursued.

EB Processing for the Repair of Aircraft

L. Biggin, Acsion; A. Johnston and D. Raizenne, SMPL, IAR, NRC

Repairs to composite and metal structures using current thermal cured bonding technologies have limitations. On aircraft bonded repair requires high temperature cure cycles in order to produce bonded repairs with optimum mechanical performance, particularly at elevated temperature. Thermal stresses induced during the repair process in the bonded patch and parent material can cause premature failure and/or reduced fatigue life as well as lowered patch effectiveness. In addition, adjacent aircraft systems and structure inhibit application of thermal heater blankets and vacuum bagging used for the repair. Some composite honeycomb structures that have absorbed moisture while in-service cannot be repaired using conventional high temperature processes without extensive drying operations because the repair process actually results in more damage to the part.

A potentially revolutionary technology which may be employed to address these issues is electron beam (EB) curing. EB curing uses high energy electrons to initiate crosslinking and polymerization of polymer adhesives and composites, eliminating the requirement for high-temperature curing processes. As described in reference 48, EB curing has a number of other potential advantages over conventional thermal processes which make it an attractive process for both manufacturing of new components and for aircraft repairs. Acsion has worked with OEMs in the aerospace and aircraft industries on EB curing of advanced composite products for the past 15 years and have developed EB processes and materials that have the potential to meet aircraft structure bonded repair requirements.

Three years ago Acsion Industries and Air Canada undertook a program to investigate the use of EB processing for the repair of composite structures used on commercial aircraft (Figure 47). During this time the technology has advanced to the stage where EB-process repaired non-flight critical parts are now flying on commercial aircraft. One of the repairs, to a wing-to-body fairing, has been flying for greater than 4500 flying hours and was declared a permanent repair in December 2000. The final step in developing the technology will be to certify the repair process and materials with Transport Canada and the aircraft manufacturer. This certification process is on schedule to be completed this year for non flight-critical parts.

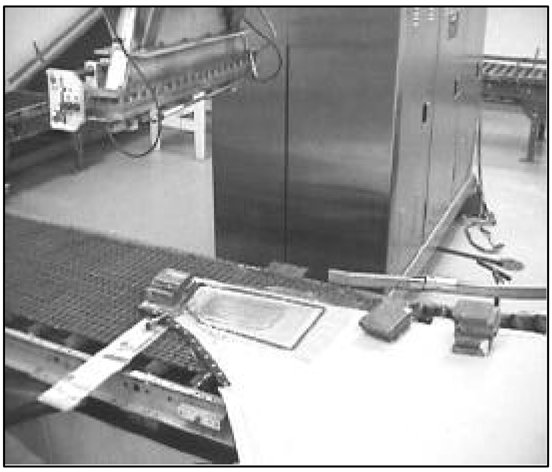
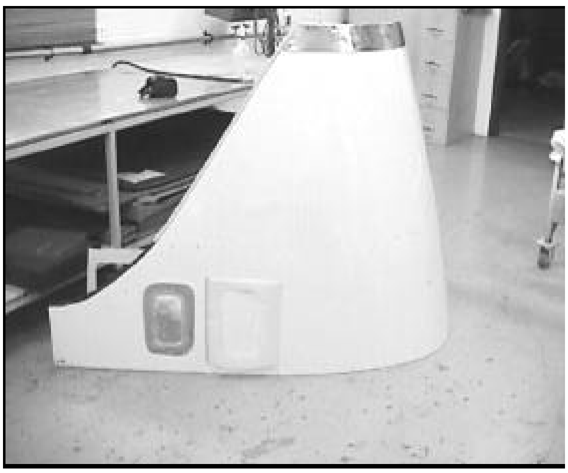



Figure 47. EB cured patch repair on an A320 wing-to-body fairing.

IAR/NRC is involved with 2 aspects of EB technology work: assessment of the potential of the technology for application to repair and reinforcement of tertiary, secondary, and primary military aircraft structures, and modelling of the kinetics of EB curing. Initial work on patch repair has focussed on issues related to process control and QA and adhesive bonding with additional work on kinetics modelling now in its early stages.

Early in 2001 Acsion Industries and the Department of National Defense completed a project to demonstrate the feasibility of EB technology for repairs to F-18 structures. The next phase of the project will be to review the requirements for testing and evaluation of materials properties. If this phase proceeds, the work will be completed with participation from IAR/NRC.

Acsion studies show that using EB technology could save between 21 and 65% of repair time as compared to traditional repair processes. The minimum of 21% reduction is based on actual data from repairs done as type-trials on commercial aircraft parts done off-aircraft. The higher 65% figure is based on large components requiring multiple repairs, being cured on the aircraft. The other benefit is the reduction of aircraft downtime due to these repairs. For typical small repairs there is a minimum reduction of 4 hours. For the complex on-aircraft repairs, reduction of one to two weeks is being forecast. Because of the anticipated improved performance versus conventional low-temperature cure repairs, a corresponding reduction in re-visits for re-doing past repairs is expected. Additionally, curing at reduced temperatures minimizes the risk of components, particularly honeycomb sandwich structures, sustaining additional damage during repair due to imperfect drying. Furthermore, because the resin systems are stable at room temperature, large quantity buying of prepreg, resin film and wet-layup resins can reduce the cost of these materials. The loss of these materials because of time-out is greatly reduced giving further cost benefits. The use of refrigeration to maintain the shelf-life of traditional resin and prepreg systems is eliminated further giving fiscal benefit to the process.

Future Developments

Acsion will be focusing on completing work in the following areas over the next year:

- Commercialization of an EB repair process for aircraft structures,
- Manufacturing aircraft components,
- Continue epoxy adhesive and resin development programs,
- Initiate research programs for repairing primary composite aircraft structures.

Durability of Fiber Metal Laminates

C. Poon, J. Laliberté SMPL, IAR, NRC and P. Straznicky, Carleton University.

Fiber metal laminates (FML) are hybrid laminates of metal sheets with fiber-reinforced polymer layers. This provides a combination of low density, fatigue damage growth resistance, impact damage tolerance and high static strength that is extremely well suited for aerospace structures [49] to [51]. In addition to being used as secondary airframe materials in applications such as control surfaces, cargo hold floors and doors, they are also being considered for primary structural airframe applications in next-generation aircraft such as the Airbus 380. A collaborative project between Bombardier Aerospace, the IAR/NRC and Carleton University was established to characterize the durability of three variants of GLARE (GLAss REinforced) laminates.

Post Impact Fatigue

One of the main focuses in the area of fatigue is on the post-impact fatigue propagation behavior under normal and off-axis loading conditions. Following the impact event there was a plastically deformed dent that acted as a stress concentration and as an initiation site for damage growth under fatigue loading. Due to the residual stresses in the panels the growth mechanism observed was significantly more complicated than that reported by other researchers in flat GLARE panels with holes or notches. The GLARE panels showed longer cycle counts to failure than the aluminum. The aluminum panels failed suddenly without any evidence of crack nucleation. In contrast, the GLARE panels showed stable crack growth in the aluminum sheets after crack nucleation in the impact dent. The growth of the cracks to the edges of the panel did not lead to catastrophic failure because the glass layers were sufficiently strong to support the fatigue loading. Most of the GLARE panels finally failed in the curved region of the dog-bone specimen away from the gage section due to the growth of nuisance cracks nucleated in the curved region after the cracks nucleated from the impact dent reached the edges of the panel. These nuisance cracks were common to all GLARE panels tested. One specimen was removed from the load frame early to prevent failure due to the nuisance cracks. A summary of the above findings for normal loading condition is given in Table 6.

One particular laminate of interest is a GLARE-4-3/2 variant which consists of three layers of 2024-T3 aluminum and six unidirectional plies of S2-glass reinforced epoxy in the following lay-up [Al/(0^o/90^o/0^o)/Al/(90^o/0^o/90^o)/Al]. In GLARE-4-3/2, the dent was accompanied by the cracking of the aluminum face sheets parallel to the “strong” axis of the panel with the inner S2 glass layers remaining mostly intact. Within the range of impact energy used in this investigation, the length of the crack resulting from impact showed a linear relationship with the impact energy. A series of post-impact fatigue tests was carried out using coupons fabricated from GLARE-4-3/2 material. Impacts were performed on a 292mm x 292mm flat coupon to produce a crack with a length of approximately 25mm. A 100mm x 280mm coupon for fatigue testing was machined from the impacted coupon with the crack oriented 45^o from the loading axis. Crack propagation in the aluminum face sheets under fatigue loading was measured using two travelling microscopes one on each side of the fatigue coupon. A series of baseline fatigue tests was conducted on 100mm x 280mm coupons fabricated from 2024-T3 aluminum. The aluminum coupons contained a central notch with a similar

Material	E_{imp} J	Cycles to Initiation	Cycles to Full Linkup	Cycles to Failure or end of Test	Comments
GLARE-3	52.06	No growth	No growth	127000	Punctured Specimen, failed at grips
GLARE-3	16.14	33000	52000	97200	Failed near radius after full growth of cracks that initiated alongside dent
GLARE-4	26.74	28000	49000	57000	Failed near radius after full growth of cracks that initiated alongside dent
GLARE-4	53.67	18000 (vertical crack, non-critical), 37500 (critical crack)	55000	61200	Stopped before full failure
GLARE-5	36.23	30000	65000	84000	Failed near radius after full growth of cracks that initiated alongside dent
GLARE-5	46.49	48000	55000	71750	Failed near radius after full growth of cracks that initiated alongside dent
2024-T3	57.67	No visible cracks, sudden failure	No visible cracks, sudden failure	25500	Failed suddenly, no warning
2024-T3	62.34	No visible cracks, sudden failure	No visible cracks, sudden failure	24000	Failed suddenly, no warning

Table 6. Summary of post-impact fatigue results.

length oriented 45^o from the loading axis. The test data were used to quantitatively establish the beneficial effect of fiber bridging in reducing crack growth rates. Based on experimental observation, damage growth mechanisms under fatigue loading in FMLs were identified. It was observed that the crack growth rate in the GLARE specimens decreased with increasing crack length [52]. A better understanding of the damage growth mechanism is believed to be important in the development of design and repair criteria for FMLs.

Multi-Site Damage

Another focus in the investigation on the fatigue behavior of GLARE was the multi-site damage (MSD) testing performed using an IAR proprietary MSD panel with and without exposure to corrosive environment. The MSD tests were conducted using a 440 kN MTS hydraulic load frame. A frequency of 10 Hz was used with a load ratio of 0.02 and a maximum load of 32.0 kN (7219 lbs). The configuration of the MSD panel, shown in Figure 48, was developed at IAR/NRC for testing monolithic aluminum sheet and was directly applied to the testing of GLARE as well. Strain data for the MSD fatigue tests was acquired using six resistance strain gauges mounted on either face of the panel. Periodically the tests were stopped and an in-situ x-ray machine was used to acquire x-ray images of cracks in the panels. A summary of the fatigue (baseline) and corrosion/fatigue test results is presented in Table 7. The total number of cycles indicates test termination. The inner glass layers of the GLARE sheets remained intact following complete link-up of the cracks in the aluminum layers. These glass layers allowed the MSD panels to retain a significant portion of their initial strength.

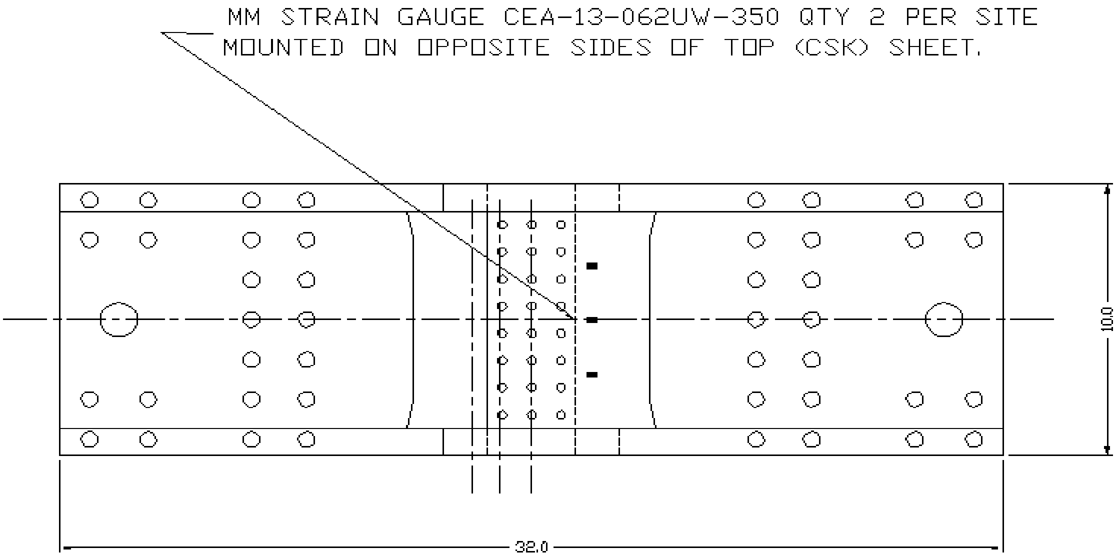
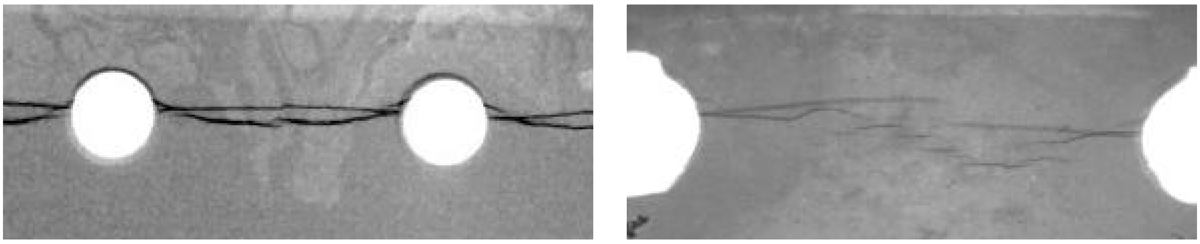


Figure 48. Multi-site damage simulated lap-splice specimen.

Specimen Number	Type of Test	Total Corrosion Time	Front Face link-up	Faying Face Link-up	Total Number of Cycles
MSD-581-001	Fatigue only	0	150000	140000	190000
MSD-581-003	Fatigue only	0	160000	150000	309000
MSD-581-005	Corrosion/Fatigue	35 days	180000	180000	220000
MSD-581-006	Corrosion/Fatigue	35 days	160000	160000	200000
MSD-581-007	Corrosion/Fatigue	85 days	230000	220000	280000
MSD-581-004	Corrosion/Fatigue	85 days	190000	180000	210000

Table 7. Summary of fatigue/corrosion tests on GLARE-3 MSD panels.

The fatigue test results show that the number of cycles to crack link-up in both the front and faying face aluminium sheets is higher in the corroded panels when compared to the non-corroded panels. This indicates that corrosion has no effect on the fatigue performance of the GLARE materials since the glass inner layers play the dominant role in fatigue resistance, not the aluminium face sheets. As a result, the corrosion damage sites in the aluminium sheets are not an important factor in crack link-ups. The link-ups of fatigue cracks initiated at the rivet holes at the front and faying faces were detected by X-ray. As shown in Figure 49, the propagation of the front and faying face cracks in the aluminium sheets for the fatigue-only panels is confined to a single path after crack initiation. For the corroded panels, cracks were found to initiate at the rivet holes as well as at corrosion damage sites leading to a staircase link-up pattern. The link-up of cracks in a staircase pattern is delayed by complicated load redistribution in a corroded panel, see Figure 49b. For monolithic aluminium sheets, corrosion significantly reduces fatigue performance since there is no glass reinforcement for the load redistribution mechanism to take place. The link-up of cracks in monolithic aluminium was found to be the main factor leading to catastrophic failure. The bridging effect of the glass inner layers has been shown to retard crack propagation in not only non-corroded materials but also in materials containing severe corrosion multi-site damage.



(a) Non-Corroded Panel (b) Corroded Panel

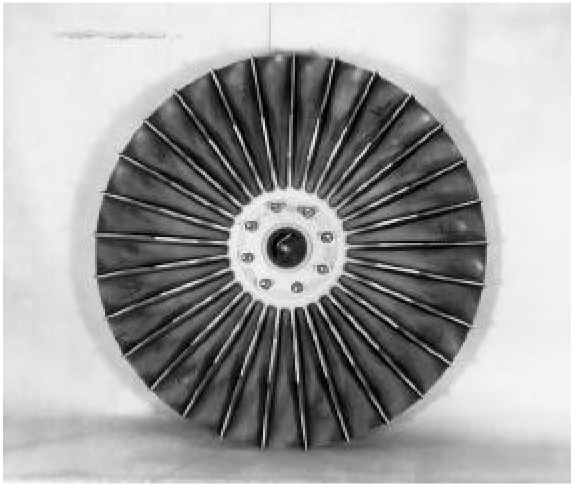
Figure 49. X-ray images of cracking in (a) non-corroded and (b) corroded MSD panels.

GAS TURBINE ENGINES – FATIGUE AND DAMAGE TOLERANCE

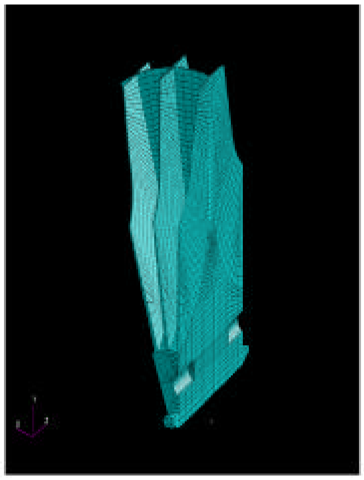
Damage Tolerance Assessment of Nene X Components

W. Beres, SMPL, IAR, NRC

Work on a damage tolerance assessment of the turbine disc, the impeller (Figure 50) and the turbine shaft of the CF Nene X engine was initiated in 1998. The objectives of the project were to predict a safe inspection interval (SII) for two major components and to apply damage tolerance based life cycle management concepts to maintain these critical components in the field. The work on the turbine disc and impeller was completed. The IAR/NRC deterministic and probabilistic fracture mechanics based damage tolerance algorithms were used to predict safe inspection intervals for the disc and impeller and to assess the risk associated with various inspection strategies. It was concluded that the turbine discs could be maintained using damage tolerance based inspection concepts at depot level. The impellers can also be maintained using the damage tolerance concept but the application of a different inspection strategy at the depot level is required. The use of an eddy current technique was recommended for impeller inspections [53] to [55].



Nene X impeller



Finite element model for the Nene X impeller

Figure 50. Nene impeller and FEM model.

Development of Damage Tolerant Microstructure in Turbine Disc

P. Au, SMPL, IAR, NRC

A modified heat treatment process has been developed for the nickel-base superalloy, PWA 1113, to create a damage tolerant microstructure (DTM) using mechanistic microstructural design concepts. This project is a continuation of the successful microstructural design philosophy developed at IAR/NRC implemented for such disc alloys as IN-718 and Merl-76 [56]. The DTM was designed with the aim of imparting improved fatigue crack growth and creep resistance without forfeiture of other vital properties such as tensile strength, stress rupture life, and low cycle fatigue lifetimes. This was achieved by optimizing the material's grain size, grain boundary morphology, and the intragranular precipitate size and distribution. Mechanical testing has demonstrated that when compared to the conventional microstructure (CM), the short crack growth rate for the DTM was slower by a factor of 3 at room temperature, and 2 ½ times slower at the service temperature [57]. Creep test results showed that at 690 MPa (100 ksi) and 705 ºC, the creep-rupture life was extended by a factor of almost 4 for the new DTM. Tensile test results indicated minimal strength losses for the DTM with respective YS and UTS values of 80% and 90% of the CM baseline values at both test temperatures.

Thermomechanical Fatigue of Superalloys

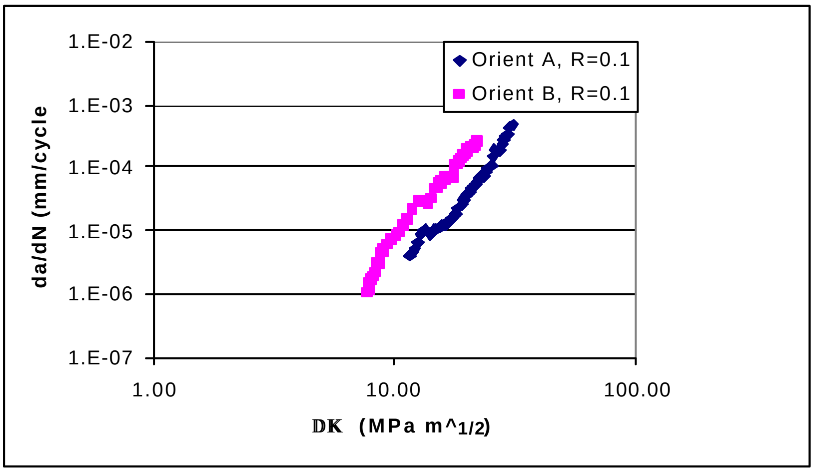
P. Au, SMPL, IAR, NRC

Components in the hot sections of gas turbines undergo complex temperature-strain-time histories while in service. This leads to fatigue crack growth under conditions of varying temperature and cyclic strain, referred to as thermal-mechanical fatigue (TMF). A computerized TMF testing facility and testing procedures are described. The TMF testing facility is capable of conducting fully reversed strain-controlled, in-phase (IP) and out-of-phase (OP) TMF, and isothermal low-cycle fatigue (IT-LCF) in the 600ºC to 1000ºC temperature range. TMF and IT-LCF tests were performed on bare IN738LC nickel-base superalloy at strain ranges from 0.4 to 1 percent to commission the TMF testing facility. The tests were conducted in air, through a temperature range of 750ºC to 950ºC at a constant strain rate of 2x10-5 sec-1. The results of these tests demonstrate the successful application of the testing procedures and prove the testing system is capable of cycling specimens under various programmed thermal-mechanical loading histories. The fatigue lives were found to differ with strain-temperature phasing and strain range. The results of IP-TMF tests correspond well to the cyclic life observed during IT-LCF loading. Predominantly intergranular crack initiation and propagation were observed under all loading conditions [58].

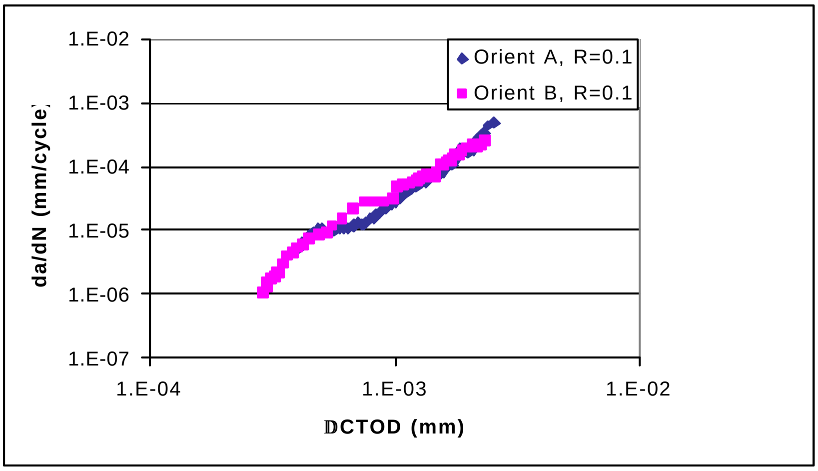
Elastic-Plastic Fracture Mechanics of Single Crystal Materials

X. Wu, SMPL, IAR, NRC

Single crystal blades are used extensively in advanced gas turbine engines for improved performance. There is a need to understand the cracking behaviour of single crystal blades at high stresses and high temperatures. Experiments have shown that crack propagation in a single crystal superalloy depends on a number of factors including (i) crystal orientation, (ii) anisotropy of crack-tip plasticity, and (iii) environmental effects. These factors are not explicitly described in conventional fracture mechanics treatments of isotropic materials with long cracks. For example, the stress intensity factor range, K, is a macroscopic and isotropic fracture mechanics parameter. Unlike large cracks in fine-grained polycrystalline materials, the correlation of single crystal crack growth rate with K cannot uniquely represent the material's crack growth resistance, because of anisotropy of the material. Therefore, appropriate fracture mechanics formulations are needed for design of damage tolerant single crystal components.



(a)



(b)

Figure 51. a) FCGR vs. DK and b) FCGR vs. DCTOD for single crystal Udimet 720 at room temperature. Orientation A represents [001]/[110] and B represents [001]/[010], as [loading]/[crack].

So far there has been little work on elastic-plastic theoretical fracture mechanics of single crystals. Work is underway at IAR/NRC to develop such a formulation for f.c.c. single crystals (e.g. Ni-base single crystal superalloys). The elastic-plastic analysis is based upon the use of the continuously distributed dislocation theory (CDDT) and the Stroh formalism. A closed form solution for the problem has been obtained for cracks under combined Mode I, II loading [59]. In application to fatigue crack growth phenomena of single crystal materials, it is found that the crack tip opening displacement (CTOD) is more suitable for correlation with the fatigue crack growth rate, as shown in Figure 51. This will also shed lights into understanding the short crack growth behaviour of polycrystalline materials, because small cracks would first form and grow in a single grain.

NON-DESTRUCTIVE INSPECTION AND SENSORS

Nondestructive Inspection of Fatigue Cracks

A. Fahr, SMPL, IAR, NRC

Airframe Structures

There is an on-going requirement for fast and reliable NDI methods to detect fatigue cracks located under installed fastener heads in aircraft structures. This applies to both thin skin fuselage and thicker wing structures. Several techniques have been developed abroad which have led to a variety of commercial instruments; however, no off-the-shelf NDI equipment has demonstrated the ability to reliably detect cracks within the shadow of the fastener head in an aircraft structure. This is especially true for the case of eddy current NDI of ferromagnetic fasteners.

Work at IAR/NRC has demonstrated the feasibility and high potential of using ultrasonic surface waves as a fast means of detecting small cracks, down to 0.010” in length, under and within the fastener head diameter [60]. In addition, work by IAR and DND/AVRS personnel in both eddy current [61] and immersion ultrasound [62] have shown similar results using relatively fast scanning techniques. Preliminary pulsed eddy current work has also shown promise for this application.

NDI reliability is an important issue particularly in life management of ageing aircraft. IAR/NRC performs significant research into structures and ageing airframe issues and NDI probability of detection (POD) data are needed. A key issue in the determination of NDI reliability is often the availability of appropriate specimens. Complex components or structures are usually very expensive, and withdrawal from service is therefore costly. Generation of fatigue cracks in the laboratory can also be difficult and expensive for complex parts, and may still not provide the same response as fatigue cracks from service exposure. Also, it is well known that the inspector in the hangar works in a very different environment than the laboratory, and even the simple fact that the inspector knows his work is being tested will change his performance.

In response to these issues, data from field inspections are being used at IAR/NRC to generate POD. This overcomes all of the difficulties mentioned in the preceeding paragraph but does create additional complications. In light of the promise of this method in obtaining reliability data, DND has sponsored IAR/NRC to continue research in this area in collaboration with the Air Vehicles Research Section (AVRS) of DND and Crosscurrents Research of Toronto. Many other organisations have recognised the potential benefits of using field data, and IAR/NRC is also a participant on two international committees that are directing relevant co-operative research efforts, through The Technical Co-operation Program (TTCP) and NATO’s Research and Technology Organisation (RTO) (formerly AGARD). IAR/NRC has begun the analysis of data from full-scale test articles and from DND fleet maintenance databases to evaluate their potential [63].

The major difficulties in using field inspection data are in the estimation of “miss” data: that is, there is no knowledge of a flaw until it is found. At this point, one must estimate the flaw size at previous inspections where the flaw was missed. This requires estimates of flaw growth and loading information, which is often not readily available. In some cases these problems can be overcome, as described in recent reports by IAR/NRC [64] and others. Work continues on the use of field inspection data; both in terms of data mining of existing maintenance databases for useful data, and in the methods of analysis of this data.

Engine Parts

A number of fatigue crack inspection reliability studies have been carried out at IAR/NRC with DND support as part of an effort to convert the maintenance of some critical components of certain aircraft engines (e.g J85CAN40) from a “safe life” approach to a “retirement for cause” approach. The results of these studies, which used real engine parts with service induced low cycle fatigue cracks, have been published in several papers and reports [65] to [71]. Currently, IAR/NRC is assisting DND in life management of the Nene X engines that are no longer supported by the original manufacturer. As a part of this activity, NDI probability of detection data are being generated for critical components of this engine in order to apply the retirement-for-cause approach to the parts that have reached their initial design life. The parts will be released into service for a given life interval based on a crack-free inspection. To allow a comparison between the response of a differential probe Eddy Current inspection to EDM surface notches and sharp fatigue cracks, QETE prepared reference standard cylinders with surface bore fatigue cracks generated using cyclic internal pressurisation (Figure 52). The EDM starter notches that were used to locate and promote the formation of surface cracks in the bore were machined away once cracks of the target depth were achieved, leaving only the fatigue cracks [72].

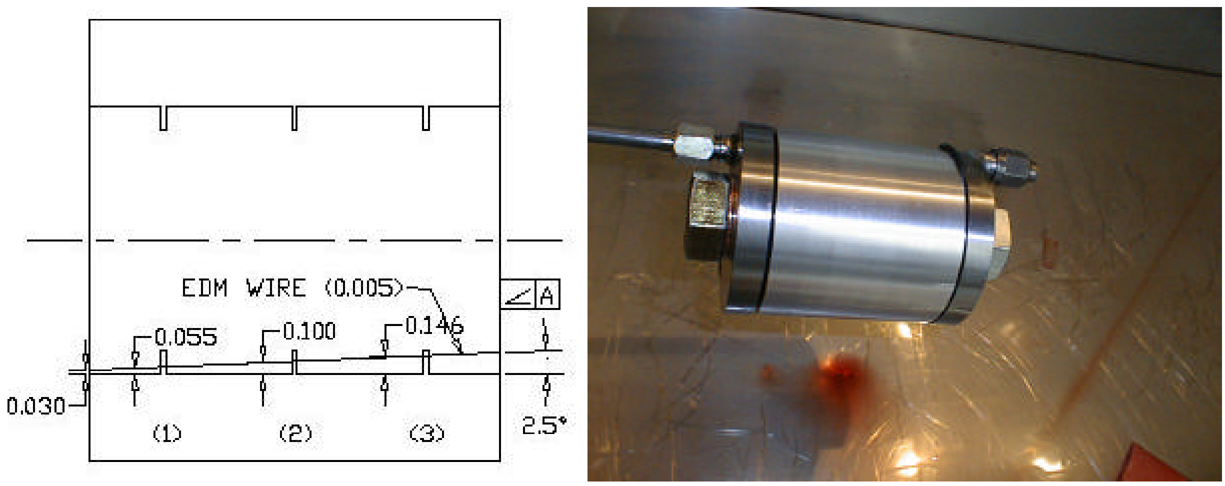


Figure 52. Location and depth of the EDM slots in internal machined fins (left) and cylinder undergoing cyclic pressurization in a hydraulic test stand to grow fatigue cracks.

Probability of Detection of Cracks in Location X19 of the CF188 Y470.5 Bulkhead

M. Yanishevsky, M. Sova, QETE

As part of the CF188 Shot Peening Certification program for the Y470.5 bulkhead, QETE has been evaluating the effectiveness of the in-service Eddy Current inspection technique used to detect cracking in location X19. Because of the radius in the inspection location and change in material thickness, it has been determined that the minimum reliably detectable crack size on pristine 7050 T7451 material exceeds the OEM specified value of 0.010" depth for failure by crack formation. As well, in most cases there is a multiple crack situation - up to 9 cracks have been found to develop independently at different points in the radius of interest on different planes. However, it has been determined that shot peening the X19 location further impedes inspection by altering the surface and its roughness which influence the lift-off characteristics of eddy current inspection.

To be able to find cracks less than 0.010" in depth, QETE developed an enhanced liquid penetrant inspection technique for use in the laboratory (Figure 53). Contrary to conventional LPI, a layer of LPI penetrant is applied by a penetrant soaked swab to the areas of interest and wiped gently, leaving a thin layer of penetrant. By applying ~50% of the maximum test load, and using ultraviolet light, and a video microscope camera system, cracks as small as 0.004" surface length (~0.002" depth) are being found and recorded as dark indications against a bright LPI penetrant background. While unloading the coupon, the penetrant that has entered the cracks squeezes out, corroborating the initial dark indications found with the coupon under load. Using this technique for measuring cracks, QETE is in the process of manufacturing location X19 type coupons, and is hoping to develop representative fatigue cracks with crack sizes in the ranges: "no cracks"; less that 0.010" depth; 0.010" to 0.050" depth; and 0.050" to "0.125 depth. The goal is to produce enough representative cracks and crack sizes, which will enable the development of better eddy current probes, optimize inspection techniques and allow the evaluation of the reliability (probability of detection) of candidate methods through a round-robin program [20], [21].

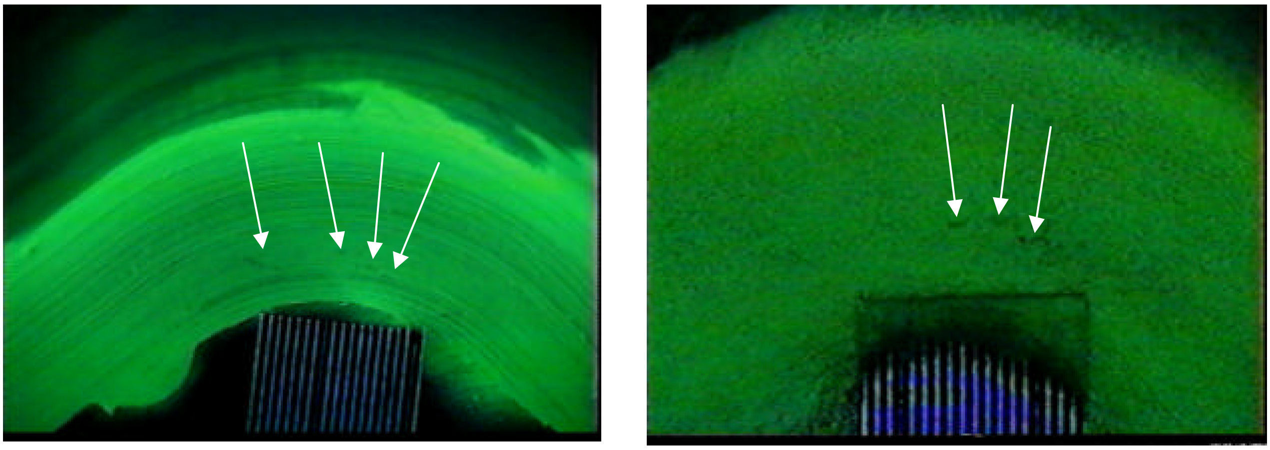


Figure 53. Examples of Surface Fatigue Cracks revealed by Enhanced LPI which were below the threshold of inspectability of conventional eddy current (surface as machined (right) and shot peened (left)).

The Role of Enhanced Visual Inspections in the New Strategy for Corrosion Management

J. P. Komorowski and D. S. Forsyth, SMPL, IAR, NRC

Many years of research into aircraft corrosion have not produced radical changes in corrosion maintenance. "Find-it and fix-it" is the order of the day for all civilian and military operators. The flurry of ageing aircraft activities in the late 1980's and early 1990's has produced revised Corrosion Prevention and Control Programs, but no change in the basic philosophy. When the Federal Aviation Administration (FAA) in the United States of America ceased funding corrosion related projects in the mid 1990's, few of the technological advances in nondestructive inspection technologies were transferred to the operators. The current corrosion maintenance philosophy reflected in aviation regulations and recommended practices does not stimulate progress in corrosion related technology.

A new corrosion management approach was proposed by Kinzie and Peeler [73]. This approach of predicting, planning, and managing corrosion stands in sharp contrast to the "find-it and fix-it" philosophy.

The IAR/NRC has pioneered work on the application of enhanced visual methods for corrosion detection in lap joints and the assessment of the impact of corrosion on lap joint structural integrity. The role of these enhanced visual methods in the new corrosion management was described by Komorowski and Forsyth in [74]. The enhanced visual methods – D Sight and Edge of Light™ were documented in previous Canadian ICAF National Reviews.

Most of the NDI technology needed to support the new philosophy of corrosion management is commercially available. Enhanced visual technologies will provide an important role in the system of non-destructive methods and post tear-down assessments providing the much needed corrosion metrics.

Acceptance of the new corrosion management approach will speed up further technology developments and the infrastructure will both increase aircraft safety and lower the economic burden related to corrosion.

Fiber Optic Sensors, Piezoelectric Sensors/Actuators, and Ultrasonic Sensors

Nezih Mrad, SMPL, IAR, NRC.

Progress has been made in three areas: fiber optic sensors, piezoelectric sensors/actuators, and ultrasonic sensors.

Bragg grating fiber optic sensors were evaluated for strain, pressure and temperature measurements for the purpose of composites process and patch bondline integrity monitoring. These sensors were shown to possess temperature measurement capabilities up to 350 ^{0}C. They also were shown to possess no pressure sensitivity, for pressure up to 200 psi. These sensors were also shown to monitor crack growth in addition to strain monitoring. Early investigation illustrated the fatigue life performance (up 2 million cycles) of these sensors which compared favourably to resistance strain gages.

Piezoelectric film sensors were also evaluated for crack growth monitoring and were shown to provide higher signal to noise ratio when compared to resistive strain gages. Based on modal analysis, an error of less than 1% was obtained between the two sensors types. The main advantage of the film sensors in that they require no power source. As well they can be used in both static and dynamic modes.

Piezoelectric ceramic actuators were manufactured and embedded into composite materials to develop a first generation smart structure. A technique of impedance measurement was adopted to monitor the structural integrity of the ceramic actuator during the smart structures manufacturing. A finite element analysis model was developed and earlier results illustrate low frequency agreement between the model and experimental results.

Early preliminary study on the use of in-situ ultrasonic process monitoring illustrates the capability of the sensing system for temperature measurement, detection of end of cure and thickness measurement. The system is still under evaluation and the use of high temperature piezoelectric film sensors is being investigated as an alternative due to the temperature limitation of the existing sensors.

All these sensors related research activities focus on the use of advanced sensors for process monitoring, patch repair bondline integrity monitoring, development of “smart” structures and patches and NDI. Progress made during the above mentioned investigations and evaluations will benefit several areas of aircraft structures including materials processing, NDI, structural health monitoring and vibration control [75] to [78].

DESIGN OPTIMISATION

Development of Genetic Algorithm (GA) for aerospace structural design optimization

G. Shi, SMPL, IAR, NRC

Multidisciplinary Design and Optimization (MDO) has recently gained wide acceptance in the aerospace industry because of the complexity of aerospace systems, which requires efficient co-ordination of disciplinary analysis capabilities and effective communication among geographically separated teams. The integrated design process will improve the product quality and reduce the design cycle time and cost. To develop and demonstrate IAR/NCR’s core competency in this new area, a research project on “Development of MDO Strategies for Aerospace System” was funded by the IAR/NRC New Initiatives program in the year of 2000.

As part of the study of MDO strategies, computational intelligence and soft computing techniques such as Artificial Neural Networks and Genetic Algorithms are identified as essential methods both for the integration of disciplines and for the optimisation needed within an individual discipline. The development of such techniques is thus necessary to enable an efficient performance of an MDO task. Through collaboration with Ryerson Polytechnic University, a Genetic Algorithm technique for aerospace structure design and optimization was developed. The technical highlights are listed below:

- A multiple constraint genetic algorithm (GA) was programmed to implement different applications in structural optimization, and a methodology for linking our Genetic Algorithm with existing finite element analysis program was developed to automatically iterate analysis and optimization cycle.

- To test the capabilities of the implemented GA-FEA structural optimization methodology, some structural design problems, for example, 10-bar tress structure and 18-bar truss structure, were defined and optimized. Results obtained proved the adequacy of the implemented structural design optimization method.

- A wing box design optimization was defined to show the capabilities of the developed GA technique. In the wing box optimization design the weight of the structure was defined as the optimization objective, subject to constraints imposed by structural and certification requirements. The FE model of the wing box was developed and a database of different cross-sections for some elements was implemented to reflect actual manufacturing variables. Upon a given wing-box geometry and loading conditions provided by a CFD analysis, two optimization cases were tested. The first case compared the GA methodology with traditional methods showing the potential benefits of the technique finding not only better solutions, but avoiding the hassles presented by traditional techniques when selecting the initial optimization points and obtaining the solutions accurate without any design space approximation. The second test was proposed to take advantage of the GA capabilities to handle continuous, integer and discrete variables at the same time. This allowed representing the wing-box main physical characteristics including skin thickness, spars and stringers cross-sectional areas and materials for all structural elements. Tests revealed good potential solutions for the wing-box optimization under given displacement, buckling and stress constraints, showing a feasible wing-box configuration with a 4% weight saving over real aircraft configurations.

The use of a genetic algorithm structural approach can be validated when the traditional methods cannot find adequate solutions due to the geometry of the design space, or in presence of complex relations between parameters linking multiple constraints and variable types. The proposed GA could find solutions when other methods fail; the main trade-off of the procedure lies in the time spent to find the solution.

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