# REVIEW OF CANADIAN AERONAUTICAL FATIGUE WORK 2005-2007

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## SUMMARY

This paper provides a review of Canadian work associated with fatigue of aeronautical materials and structures during the period 2005 - 2007. All aspects of structural technology are covered including full-scale tests, loads monitoring, fracture mechanics, composite materials and non-destructive inspection.

Organisation Abbreviations Used in Text:
AMTC - Aerospace Manufacturing Technology Centre
ATESS - Aerospace and Telecommunications Engineering Support Squadron (DND)
CF - Canadian Forces
DAES - Directorate of Aircraft Engineering and Support (DND)
DND - Department of National Defence
IAR - Institute for Aerospace Research
L-3 MAS - L-3 Communications (Canada) Military Aircraft Services (MAS)
NRC - National Research Council of Canada
QETE - Quality Engineering Test Establishment of DND
RAAF - Royal Australian Air Force
RMC - Royal Military College (DND)
SMPL - Structures and Materials Performance Laboratory
USAF – United States Air Force

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## LIST OF ACRONYMS FOR TECHNICAL TERMS

| | |
|---|---|
| ALEX | Aircraft Life Extension Program |
| ASIP | Aircraft Structural Integrity Program |
| BHEC | Bolt Hole Eddy Current |
| CF | Canadian Forces |
| DADT | Durability and Damage Tolerance |
| DIC | Digital Image Correlation |
| EDM | Electrical Discharge Machining |
| ELE | Estimated Life Expectancy |
| FE | Finite Element |
| FH | Flight Hours |
| FLEI | Fatigue Life Expanded Index |
| FLMP | Fatigue Life Management Program |
| FML | Fiber Metal Laminate |
| GIFTS | General Integrated Fatigue Tracking System |
| GUI | Graphical User Interface |
| HOLSIP | Holistic Structural Integrity Process |
| IAT | Individual Aircraft Tracking |
| IDS | Initial Discontinuity State |
| IFOSTP | International Follow-on Structural Test Program |
| IVD | Ion Vapour Deposition |
| LPB | Low Plasticity Burnishing |
| MSDRS | Maintenance Signal Data Recording Set |
| MOI | Magneto-Optic Inspection |
| NDI/NDT/NDE | Nondestructive Inspection/Testing/Evaluation |
| OEM | Original Equipment Manufacturer |
| OLM | Operational Loads Monitoring |
| PHM | Prognostic and Health Monitoring |
| PLF | Parametric Loads Formulation |
| POD | Probability of Detection |
| POF | Probability of Failure |
| RST | Residual Strength Test |
| SCC | Stress Corrosion Cracking |
| SEM | Scanning Electron Microscope |
| SESC | System Engineering Support Contract |
| SFH | Simulated Flying Hours |
| SIS | Structural Information System |
| SLAP | Service Life Assessment Program |
| SLMP | Structural Life Monitoring Program |

# INTRODUCTION

Canadian industry, universities and government agencies were solicited for information describing their fatigue technology related activities over the period 2005 to 2007. This review covers work performed or being performed by the following organizations:

Bombardier Aerospace

Department of National Defence (DND)

- Aerospace and Telecommunications Engineering Support Squadron (ATESS)
- Air Vehicles Research Section (AVRS)
- Canadian Forces (CF)
- Director General Air Equipment Technical Management (Fighters and Trainers) (DGAEPM (FT))
- Quality Engineering Test Establishment (QETE)
- Royal Military College (RMC)

L-3 Communications (Canada) Military Aircraft Services (MAS)

National Research Council of Canada

- Institute for Aerospace Research (NRC-IAR)

Celeris Aerospace Canada Inc.

Carleton University, Department of Aerospace and Mechanical Engineering

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

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## FULL-SCALE TESTING

**F/A-18 International Follow-on Test Program (IFOSTP)**
**Jean Dubuc, L-3 Communications - MAS (Canada)**

L-3 MAS and the National Research Council Institute for Aerospace Research (NRC-IAR) in Ottawa have teamed-up throughout the IFOSTP program to provide the CF and RAAF with full-scale certification of the center fuselage and wing. The residual strength test and the teardown of IFOSTP FT245 wing test article have been completed. Quantitative fractography performed the Quality Engineering Test Establishment (QETE) and the NRC-IAR in Ottawa is providing key inputs into the conduct of the ASIP and SRP programs.

As a complementary program, the NRC-IAR is also performing two component tests to substantiate the life of the wing in areas that were not sufficiently tested as part of IFOSTP. The first two component tests are addressing the trailing edge flap (FT169) and the aileron (FT193) back-up structure. L3 MAS has derived the test spectra from flight test data representative of the operational usage spectrum (designated 99LD, see ICAF 2005). The spectra in question include a very significant portion of dynamic loading and are comprised of about 900,000 lines and 350,000 lines per 325 hours of flight respectively. The two tests started in 2006 and should be completed in 2007 after 30,000 Spectrum Flight Hours (SFH), i.e. a scatter factor of 5.

**F/A-18 FT245 CF-188 Wing Fatigue Test – IAR/CF/Bombardier/L-3 MAS/RAAF**
R.S. Rutledge, NRC-IAR-SMPL

The Structures Laboratory of the NRC-IAR completed the wing fatigue test (Figure 1) to 18256 simulated flight hours by October 29th 2004. After completion of 18256 simulated flight hours of the IARPO3a spectrum, the CF18 wing in FT245 was inspected and residual strength tests were carried out and completed by October 28th 2005. The residual strength testing was required to statically test the wing specimen to load levels twenty percent higher than design limit loads. The Canadian Forces (CF) and Royal Australian Air Force (RAAF), through their airworthiness policies, have a requirement to test fatigued specimens to these levels to ensure the safe operation of the aircraft throughout the service life. To perform this additional testing the maximum loads that combine various interface loading actions, such as wing root bending moment or a combination of wing root bending and wing root torsion, were developed through a Gumble extrapolation of the IARPO3a 326 hour spectra to obtain one in 6000 hour occurrence load magnitudes throughout the envelope. The magnitude and combinations of loading that were applied in these tests were documented in the residual strength test report. Thirty-seven static tests in the test end configuration were applied following a planned end of fatigue testing inspection. Then three critical areas of the wing specimen were statically tested to 120% design limit load with induced damages. Following these load cases two ultimate load cases were applied. An ultimate wing root bending moment combined with an ultimate wing root shear case was applied as well as an ultimate aft fuselage bending moment case was applied. These additional tests were intended to aid the CF and RAAF in demonstrating the inconsequential nature of the cracked structures for airworthiness and help substantiate a fleet management approach that may employ a safety-by-inspection and replacement on condition philosophy. The RST report document provides a summary of the applied loads, measured strains, displacements and resultant data for each of the load cases. Post test examination of the structure has revealed numerous cracks. NRC Aerospace is in the process of documenting all the damages.

**Figure 1 FT-245 Wing Fatigue Test**

The wing test is part of the F/A-18 International Follow-On Structural Test Programme (IFOSTP) to determine and extend the life of the aircraft for the Canadian Forces (CF) and the Royal Australian Air Force (RAAF). The test has been described in earlier ICAF reviews. The test is intended to determine the safe-life of the primary structure, economic life of the inner and outer wing structure, obtain crack growth information to support management of the aircraft, validate repairs or modifications and obtain engineering data. In 2006-2007 the wing was disassembled and all damages were recorded in the structural information system database. Selected areas have been evaluated using quantitative fractography to obtain crack growth rate data. Over the duration of the IFOSTP NRC has provided numerous F/A-18 related documentation that includes: 12 journal publications, 42 published proceedings, 1 volume of a book, 223 internal publications, 3 trade magazine and 8 other reports.

**F/A-18 FT193/FT169 CF-188 Extended Fatigue Test of Dynamically Loaded Wing Attachments - IAR/CF/ L-3 MAS**

R.S. Rutledge, NRC-IAR-SMPL

FT-245 was the primary means of CF-18 wing structural certification. The test spectrum included dynamic loads caused by trailing edge and wing tip buffet, but the spectrum had to be severely truncated, particularly in terms of control surface attachment and wing tip loads, to complete the test in an appropriate time. Some areas of the wing may therefore be under-tested. In addition, the CF Lifing Policy has recently been revised such that areas subjected to dynamic loads must be tested to five times the required service life.

The extended wing testing follows the completion of the 18 256 simulated flight hours of the FT245 CF-18 wing test. The equivalent of 30 000 simulated flight hours of CF-18 usage (in accordance with IARPO3) has been applied to the outer wing in a test designated as test FT193 (Figure 2). The test targeted the outer wing tip, the wing-side aileron inboard and outboard half-hinges, and the wing-side aileron actuator attachment. This test was started in February of 2006 and was completed in March of 2007 with 29970 simulated flight hours applied by 17th March 2007. Twenty residual strength tests have been carried out.

For phase 2 of this project, the wing-side trailing edge flap outboard half-hinge and localized back-up structure is been certified in a test designated FT169 (Figure 3). FT169 started June 15th 2006 and currently has 25 200 simulated flight hours of the planned 30 000 for the outboard hinge area of the inner wing. Planned residual strength testing will follow.

Both tests utilize new control technology, SOFFT, developed by NRC under a collaborative research program with MTS Systems. These has have demonstrated an increased fatigue testing speed of over 5 relative to the FT245 wing test. In 14 months over 66,000,000 spectrum end levels were applied in FT193. In FT245 approximately 8,680,000 end levels were applied in four years.

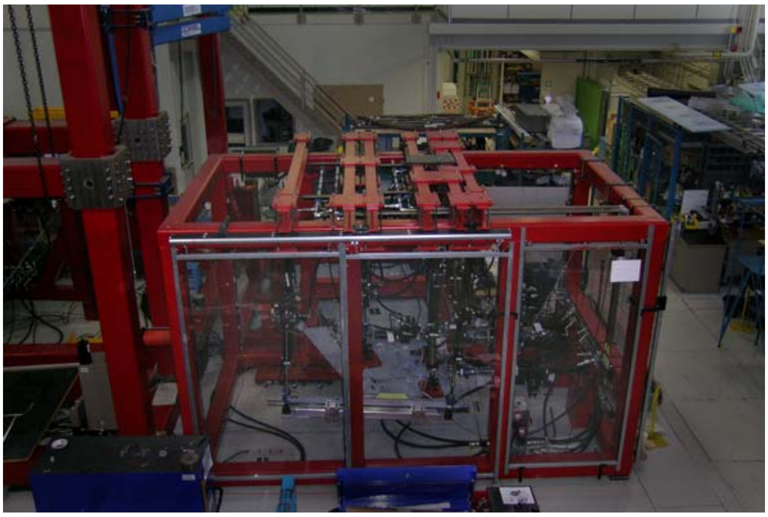

**Figure 2 FT-193 Extended Fatigue Test Set-up**

**Figure 3 FT-169 Extended Fatigue Test Set-up**

**F/A-18 FT312 CF-188 Wing-Fold Aft-Spar Shear-Tie -IAR/CF/ L-3 MAS**

R.S. Rutledge and D. Backman, NRC-IAR-SMPL

The extended wing testing follows the completion of the 18256 simulated flying hours of the FT245 CF-18 wing test. Due to recent revisions in the Canadian Forces Lifing Policy, components in the CF-18 subject to dynamic loading are now required to be certified to five lifetimes. Due to in-service component damage the Canadian Forces required a component test within the International Follow-On Structural Test Programme. Cracks found in the wing fold shear tie of several aircraft in the Canadian Forces (CF) CF-188 fleet prompted a full scale test to be undertaken in order to determine crack growth rates both before and after a proposed repair modification to this critical region. The primary test objective was to collect crack growth data at the wing-fold aft-spar shear-tie with severe blend geometry typical of a structural modification proposed for this area. The fatigue testing of this component utilized the FT245 stub wing (S/N A12-0312) shear-tie with loads developed by L3 Communications (MAS). NRC fatigued tested this critical area while monitoring the cracking automatically with an ultra sonic transducer tied into the control system to stop testing. NRC also measured the strains at peak and valley extremes automatically using digital image correlation. This test provided the time required to grow a 0.020-inch deep flaw under the IARPO3a spectra; incorporation of a typical depot level fleet modification; and test until failure with residual strength loads. Two quantitative fractography investigations were carried out. The first for a 0.020 inch deep crack generated pre modification. Then a post-test quantitative fractographic inspection of the crack generated in the L3-MAS ALEX66 blended geometry was carried out. Both quantitative fractography examinations have provided crack growth rate data.

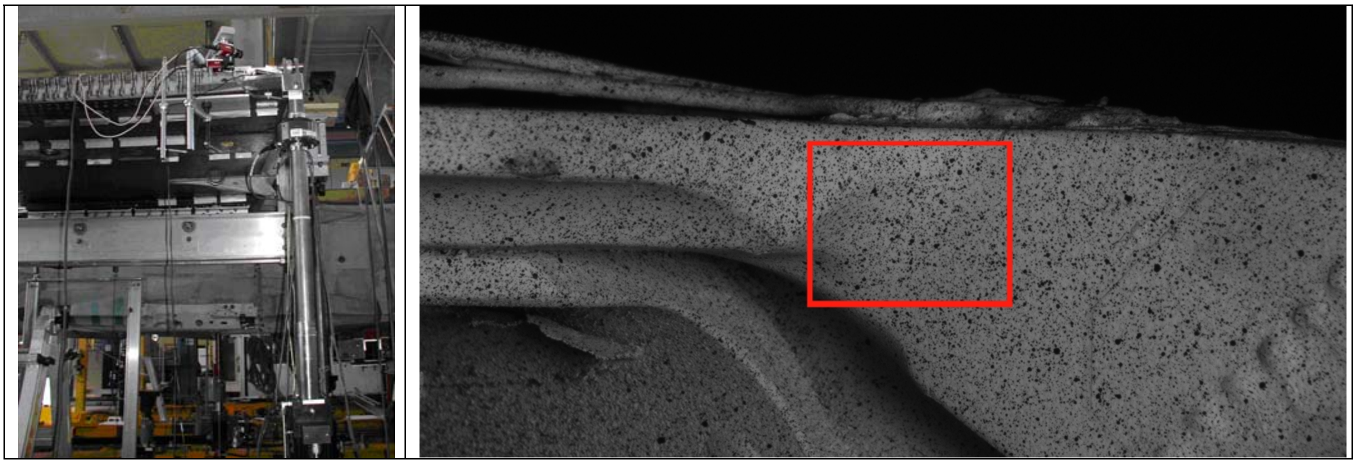

**Figure 4 FT-312 CF-18 Wing-Fold Aft-Spar Shear-Tie Test Set-up and Critical Area Post Blend**

Digital image correlation (DIC) was used to measure the surface strains in the stress concentration (Figure 4 and Figure 5).

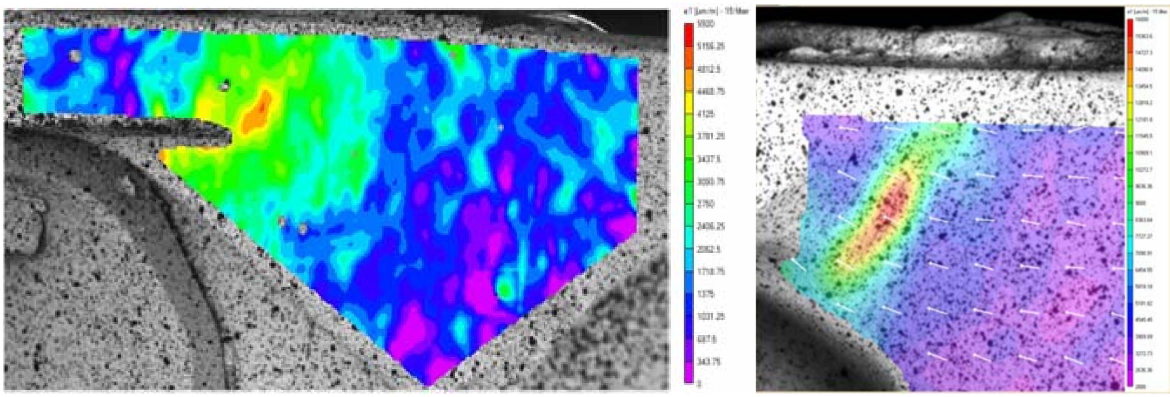

**Figure 5 FT-312 CF-18 Wing-Fold Aft-Spar Shear-Tie Principal Strains Measured Pre-Test and 11201 SFH Post-Modification**

After the aircraft life extension modification (ALEX 66.2) was carried out, the surface profile of the WFST post blend, as well as the strain distribution at a reduced load were both measured using DIC. Fatigue testing resumed; however, after several hundred load segments, testing was stopped to measure the strains in the critical area. The strain values in the critical area approached those seen pre-modification when a 0.015 inch (0.3742 mm) deep surface crack was found. As a result, NRC-IAR provided the surface profile of the WFST, measured using DIC, to both DND and L3-MAS for finite element analysis. Based on the L3-MAS FEM analysis, the load spectrum peak applied load was reduced to obtain surface strains consistent with the intended geometric blend. After the peak load reduction, fatigue testing of the ALEX 66.2 modification resumed and the area was found to have a long stable crack growth. It was determined that the repair modification was effective in extending the life of this component.

**F/A-18 FT865 CF-188 Aft Engine Mount Structure Fatigue Test -IAR/CF/ L-3 MAS**

R.S. Rutledge, NRC-IAR-SMPL

Failures have been documented on the aft engine hanger support assembly and cracks were found on both the original equipment manufacturer's test and on the Australian aft fuselage and empennage test, FT46. As well, several in-service cracks have been found and several holes have been found elongated during the incorporation of modification CD-050 at L3 MAS.

A review of the modification CD-050 revealed that on the CF fleet major lifing changes are required from the previous risk assessment carried out by L3 Com. The certification of the aft engine mount assembly requires a damage tolerance assessment. Therefore, DND requested that NRC-IAR carry out a fatigue test on the aft engine mount assembly of the 2nd transition structure (USN A/C 162865) that was previously used to react FT245 test loads as a reactions structure. The fleet certification approach was to test the mounts with induced damage representative of the worst possible crack size/configuration that could have been missed post CD-050 incorporation by L3 MAS. Damage induced in the structures of both mounts. A pre-test inspection found 16 damages in the attachment holes. Testing started on October $24^{th}$ 2006 and applied over five lifetimes of CF-18 usage based on L3 MAS defined loads by February $22^{nd}$ 2007 (Figure 6). Residual strength test loads were applied February $26^{th}$ 2007. The test specimen has been disassembled for teardown inspection. Post-test quantitative fractography has been planned but no data has been developed to provide crack growth data to date.

**Figure 6 F/A-18 FT865 CF-18 Aft Engine Mount Structure Fatigue Test Set-up and Starboard Side Attachment**

**F/A-18 FT333 CF-188 Trailing-edge Flap Outboard Lug Fatigue Test -IAR/CF/ L-3 MAS**

R.S. Rutledge, NRC-IAR-SMPL

The primary means of CF-18 wing structural certification was the FT245 wing test, conducted at the National Research Council under the International Follow on Structural Test Programme. The test specimen for this comprised the inner and outer wing boxes, with load inputs from the flight control surfaces, as well as a distributed loading over the wing. During the course of the testing, the trailing edge flap (TEF) was replaced, due to cracking in the aluminum leading edge structure. As a consequence of this, the current Canadian Forces TEF configuration (represented by TEF serial A16-0333) was only tested to 9782 simulated flight hours in FT245. The CF lifing policy requires fatigue testing of dynamically loaded components to five lifetimes. To provide data for the certification of the TEF outboard lug, the CF requested NRC to continue testing the TEF outboard lug, off-aircraft, to failure or a total of 30000 SFH, followed by a residual strength testing. The TEF was mounted in the Structures and Materials Performance Laboratory Multi-Axial Test Platform (Figure 7). Loads were applied directly to the TEF outer lug by three orthogonal actuators in the wing reference axes. The load spectrum was based on the TEF wing-side outer lug loads measured in block 39 of FT245.Full-scale Structural Loads as defined by an in situ calibration of the wing side lug with appropriate instrumentation. The continued fatigue testing started on December $12^{th}$ 2006 and five lifetimes of usage was completed on March $8^{th}$ 2007. Residual strength loads were applied March $12^{th}$ to $14^{th}$ 2007.

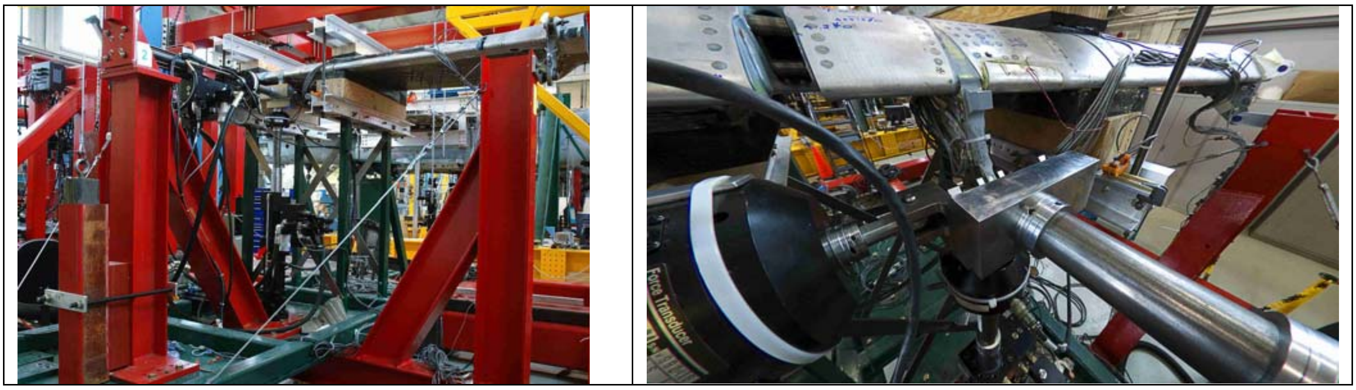

**Figure 7 F/A-18 FT333 CF-18 Trailing-edge Flap Outboard Lug Fatigue Test Set-up and Tri-Axial Lug Hinge Attachment**

**Fatigue Testing Technology**

R.S. Rutledge, NRC-IAR-SMPL and MTS Systems Corporation

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. As reported in previous ICAF reviews, modeling of a full-scale structural test system has been developed at NRC-IAR with the aim of improving the understanding of the complete structural test system. This has been used to investigate full scale testing typical problems as well as methods for increasing test speed. As reported in the last ICAF review, NRC-IAR and MTS Systems Corporation entered into a collaborative research program to develop a full-scale structural test system test bed at NRC-IAR. This was extended to continue collaborative research by demonstrating this technology on the extended testing of the CF-18 wing, FT-193. Due to the successful implementation and looming deadlines for full scale structural fatigue test completion NRC-IAR utilized this technology to carry out FT169. Both FT193 and FT169 are described elsewhere in this review and use MTS's AeroPro™ and NRC-IAR Aerospace's SOFFT.


**Figure 8 F/A-18 FT193 MTS's AeroPro™ and SOFFT Control System Set-up Adjacent to the Rig**

Both tests have been used to validate the method for increasing test speed that came out of the modeling work. By using information about the complete test system that can be obtained from a relatively simple structural response survey, test speed increases of more than a factor of 5 have been obtained under spectrum type loading (Figure 8). Details of the technology are proprietary to NRC-IAR.

## LOADS AND USAGE MONITORING

**CT114 Tutor Operational Loads Monitoring Program**

J. Dubuc, L-3 Communications - MAS (Canada)

The Tutor aircraft is used in the Snowbirds (air show) role in Moose Jaw and in the Utility role at AETE in Clod Lake (in replacement of the CT133 aircraft). Since April 2004, all twenty-six (26) aircraft are equipped with an Operational Loads Monitoring (OLM) system (i.e. four (4) flight parameters and four (4) strain gauges).

Twelve components/locations of the aircraft are tracked using the OLM data. On-going OLM data processing (using the General Integrated Fatigue Tracking System (GIFTS)) is done and results are presented on a monthly, semi-annual and annual basis to the fleet manager for aircraft rotation, component swapping and aircraft retirement decisions. Safety-by-Inspection results for the wing critical locations are presented in the annual report.

With the obsolescence of the data acquisition units (DAU) parts and with no more support provided by the OEM (Moog/Esprit Technology), a DAU replacement project was initiated by DND in 2006.

## FATIGUE LIFE PREDICTION AND ENHANCEMENT

**Future Airframe Lifing Methodologies**

M. Yanishevsky NRC-IAR-SMPL

For the last two years, the AVT (Air Vehicles Technology) Task Group AVT-125 of NATO (North Atlantic Treaty Organization) RTO (Research and Technology Organization) has been actively developing a technical report on Future Airframe Lifing Methodologies. Germany and the United States are the lead nations for this effort with contributions from Canada, France, Italy, The Netherlands, the United Kingdom and Partner for Peace countries Australia and Sweden. The intent of the report is to provide a guideline to facilitate future aircraft structural design certifications and force management/sustainment based on in-service lessons of previous and current practices, experiences and efforts in aerospace and other transportation and regulated industries. Best practices and research efforts that are underway are documented so that these can be incorporated into new designs and lifing approaches to reduce/avoid similar problems in the future.

Canada’s contributions to this document have included: the provision of a table of threats (intrinsic and extrinsic factors affecting performance and material selection as they affect Design and Certification of Metallic Structures; the Canadian experience with strain gauge and flight parameter based IAT monitoring, including information on future structural monitoring trends; description of Marine and Off-shore Industry fatigue methods; description of the Holisitc Life Modelling approach, which includes cycle dependant (fatigue) and time dependent (such as corrosion and other failure mechanisms); Canada’s approach to adapting new off the shelf aircraft procurements for Canadian roles; and the implications of fly-by-wire, adaptive structures, new pressure suits, sensors on force management and lifing strategies.

This document will provide guidance and direction to decision-makers, program managers, fleet operators (whose day to day decisions impact life, life cycle costs, readiness and availability), certifiers (best practices for achieving successful certification), sustainers (processes that establish life, costs, readiness and availability; monitoring of essential sensors/information), manufacturers (their efforts for supporting certification and sustainment). In addition, to help ensure military platforms meet their life, safety, reliability and availability objectives, Canada is participating in the organization of a follow-on RTO Symposium to be held in Fall 2008 in Canada, activity AVT-157 “Military Platform Ensured Availability”, to be held jointly between AVT-125 “Future Airframe Lifing Methodologies” and AVT-126 “Improving Military Gas Turbine Engine Reliability”.

**Coupon Tests for the Validation of the TEFOLZ Spectrum Truncation for the FT169/FT193 Component Tests**

M. Yanishevsky, NRC-IAR-SMPL

A multi-lab collaborative coupon test program (NRC, Royal Military College, Quality Engineering Test Establishment and Ecole Polytechnique) was developed by the Canadian Department of National Defence (DND) and L-3 Communications MAS to address five control points on the CF-188 aircraft sensitive to dynamic loading / truncation. The NRC contributed to the program by completing the experimental measurement of the fatigue lives for baseline and truncated spectra for the Trailing Edge Flap Outer Lug machined from Al 7050-T7452 hand forging. A coupon with a Ktn of 2.18 representing a medium to high stress concentration was used in order to have a high "Crack Initiation (CI)" to total life ratio according to CI89 calculations, with a notch radius typical of filet radii on the wing structure. Results of the test are being used by DND and L-3 Comm MAS to validate the truncation by comparing the experimentally measured lives with those calculated by analytical prediction and provide crack formation and crack growth data for assessing the predictive capabilities of CI89 and CG90 software tools under dynamic loading.

**Effect of dynamic loading (buffet) on fatigue scatter in aircraft aluminum 7050-T74511 for the CF-188**

D.L. DuQuesnay, Mechanical Engineering Department, RMC

This work examines the effect of dynamic buffet loads (small cycles) on the scatter in the fatigue life of aircraft aluminum. Current life cycle management of fighter airframes assumes, without engineering evidence, that buffet loads cause an increase in the scatter factor used in safe life calculations. Hence, the role of small cycles in spectra representative of the aft tail hanger position of the CF18 aircraft was examined in this study. The base load spectra with the dynamic content were filtered to remove specified amounts of dynamic damage as determined by the CI89 strain-based cumulative fatigue damage program. The effect of this filtering on the scatter in crack initiation life and total fatigue life of double edge notched fatigue coupons of 7075-T7451 aluminum alloy were examined. The results indicate that inclusion of the dynamic loading caused the distribution of the crack initiation life to become bimodal. Each mode could be described with a log-normal distribution, the standard deviation of which was lower than the standard deviation obtained for the filtered load spectra. There was no evidence that the standard deviation increased with small cycle content, once the true nature of the distribution was taken into account. For all of the spectra examined, the scatter in the crack initiation life was found to increase for specimens taken further from the centre of the block from which they were machined. There was also a slight increase in the mean fatigue life for specimens taken from outer sections of the block. In addition, the crack growth rate was related by a power law to the crack initiation life for all spectra independent of the small cycle content [1].

**The Role of Rivet Installation on the Crack Nucleation and Growth in Riveted Lap Splices**

C. Rans and P. Straznicky, Carleton University

As part of a larger research effort into improving the rivet joint design in the area of fatigue performance, a comprehensive study into the effects of rivet installation has been undertaken. The primary objective of this study is to improve our understanding of how both rivet type and the magnitude of the squeeze force used to install the rivet influence crack nucleation and propagation.

Finite element studies were carried out to investigate the formation of residual stresses due to rivet installation and observe the local stress state around a rivet in a loaded lap splice. Installation and subsequent loading of 2-row riveted lap joints containing both universal and countersunk rivets were simulated using the explicit code LS-DYNA. A combination of neutron diffraction and digital image correlation (DIC) experimental stress techniques were used to verify the finite element model. Results from the finite element model provided new insights into the local stress state around fatigue critical rivet locations. Through-thickness compression of the sheets was identified as a major contributing factor in the formation of residual stresses around the rivet hole [2]. The location and magnitude of peak secondary bending stresses was also found to be dependant on the geometry of the driven and manufactured rivet heads (Figure 9) [3]. Additional simulations investigated the effects of sheet material are ongoing.

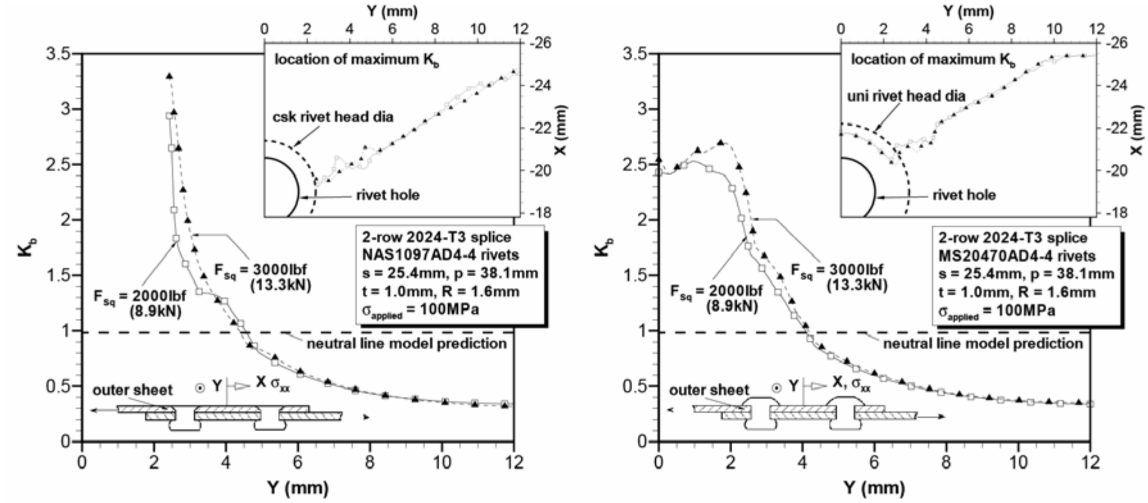


**Figure 9: Comparison of maximum secondary bending factor ($K_b$) magnitude and location for countersunk and universal rivet splices.**

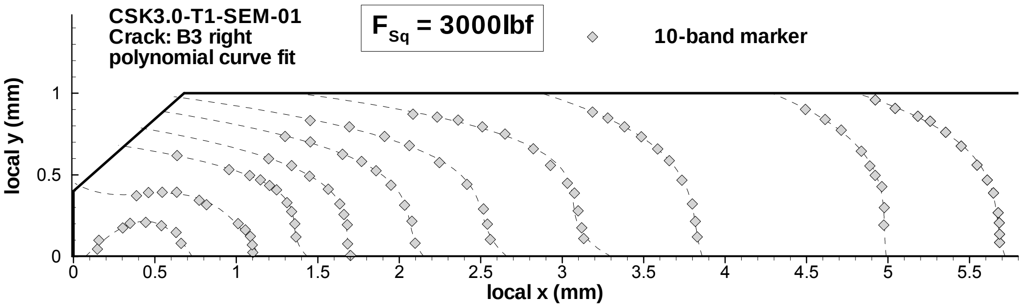


**Figure 10: Typical crack front plot resulting from crack reconstruction for a lap joint with high rivet squeeze force.**

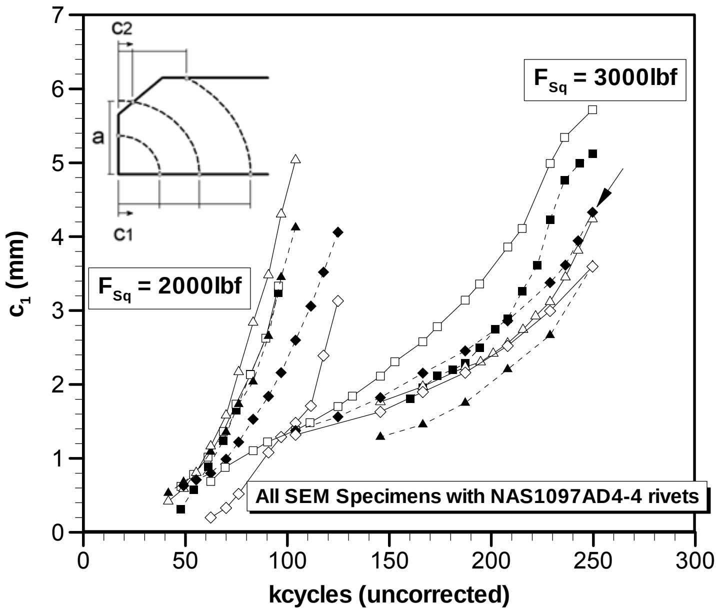


**Figure 11: Comparison of crack growth curves for countersunk rivet lap joints.**

In conjunction with the above finite element studies, a series of lap joint fatigue tests and post-test fractographic crack reconstructions using a scanning electron microscope in order to quantify the effects on fatigue performance. Simple 2-row riveted lap joint coupons were tested with a load spectrum which provides distinct fracture surface markings that aid in crack reconstruction (Figure 10). From the reconstruction, crack nucleation locations are determined and compared with likely locations identified from the finite element simulations. Crack growth curves (Figure 11) and the evolution of the crack front shape are compared for rivets installed with various squeeze forces, providing direct evidence of the effects of rivet installation on crack growth. Preliminary results indicate that in addition to slower crack growth rates, higher rivet squeeze forces result in highly elliptical crack front shapes that remain below the free joint surface for a longer period relative to the faying surface crack length.

**Alternative Countersinking Method for Thin Lap Splices**

A. Brown and P. Straznicky, Carleton University

This work was an extension of an earlier investigation by Rans and Straznicky [4] into the potential for dimpling thin GLARE laminates with the goal of extending crack nucleation life. The results of their study provided mixed results; GLARE splice coupons dimpled with conventional dimpling tools obtained fatigue lives similar to that of monolithic aluminum but open hole GLARE coupons showed promise for an increase in the material fatigue life. It was determined that damage to the prepreg plies in GLARE was responsible for the decrease in life. From that investigation, it was decided to explore the possibility of creating a modified dimple tool that did not damage the prepreg in GLARE. A parametric finite element study was created to observe how changes in tool geometry, process order, and loading altered the degree of damage in GLARE3-2/1-0.3. The result was a dimple tool set design that did not eliminate but did greatly reduce the damage incurred by the material during dimple forming [5]. The modified tools were also able to take advantage of residual stresses by inducing compressive residual stresses around the fastener hole.

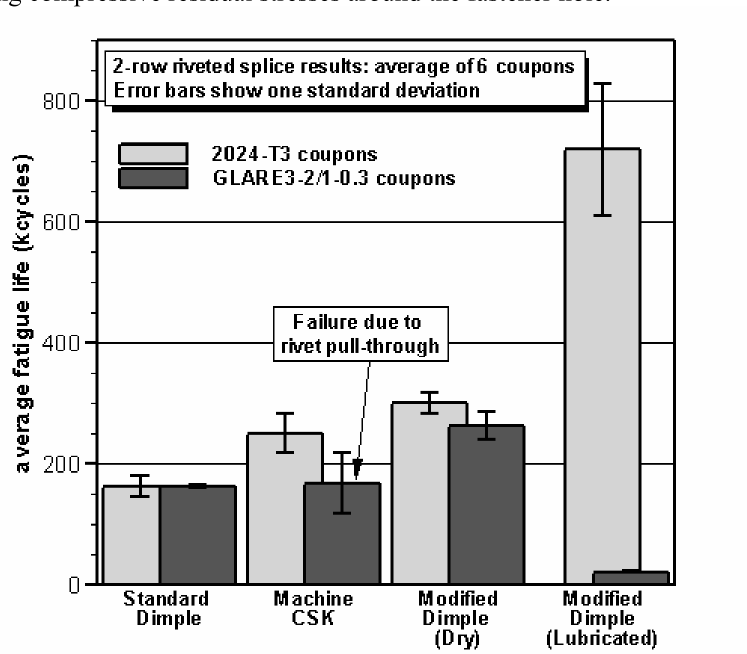


**Figure 12 Graph showing fatigue results from dimple coupon study.**

Results from coupon testing (Figure 12) were somewhat disappointing. The results showed that although an improvement in fatigue life was obtained over conventional dimpling tools, the modified dimpled GLARE coupons did not perform as well as monolithic aluminum coupons. The surprising result from this study was in the performance of modified dimple coupons that had a solid lubricant applied to the faying surface to reduce fretting damage. The fatigue life of the modified, lubricated GLARE coupons dropped drastically due to the increased faying surface slip and low material stiffness whereas the life of the modified, monolithic aluminum coupons rose sharply.

**Fatigue Enhancement Technologies**

M. Yanishevsky, NRC-IAR-SMPL and I. Mantegh NRC-IAR-AMTC

The NRC is collaborating with DND/DRDC to better understand material surface enhancement technologies and their applications to known and predictable aircraft fatigue problems. For many years the Canadian military have been under significant pressure to keep in service many of their aircraft fleets beyond their initial life expectations. To be able to achieve this, fatigue enhancements/repairs have been performed using shot peening to improve the fatigue life of life limiting geometric design details/stress concentrations, and cold expansion of fastener holes prone to fatigue cracking to improve their fatigue performance. Since it is difficult to quantify the residual stress profiles resulting from these technologies, the long term impact on structural performance, reliability, airworthiness and safety cannot be determined, and definitively ascertain the fatigue benefit/life improvements that could translate to direct cost benefits. The long-term intent of this program is to develop a guide for DND and its repair and overhaul third line subcontractors to determine appropriate use of these technologies, where these technologies can provide fatigue enhancement or where they can be considered as repairs to and beyond the original design expectation.

As part of this program, a joint L-3 Com MAS, SMPL, AMTC and QETE evaluation of various tools available for controlling and assessing shot peening was completed and a joint report issued. It was concluded that currently using a combination of X-Ray Diffraction (XRD) and Meandering Winding Magnetometer (MWM) technologies is the best available approach to meet the necessary requirements. The XRD would be used to establish/quantify the state of residual stress in appropriate reference standards, which would then enable the use of the MWM technology to perform quick process QC/QA of shot peened areas. As well, MWM would be able to conduct appropriate crack detection, again calibrated to appropriate reference standards prepared for specific crack geometries, crack sizes and other requirements. Detailed literature reviews were undertaken on the physical processes and analytical modeling of shot peening, hole cold expansion and low plasticity burnishing with the intent of developing better analytical tools, supported by case studies and measurements of residual stresses created by the fatigue enhancement techniques and laboratory tests assessing the impact of these surface treatments on fatigue performance.

In the case of shot peening, specimens representing the X19 location on the CF188 470.5 bulkhead manufactured and shot peened at L-3 Com MAS using robotic shot peening techniques were fatigued at NRC to various stages of damage and then sent to Positron for measurement / assessment. Initial indications are that the technique shows some promise for revealing the internal damage state of Al 7050-T7451 alloy so that with some further development, the technique may have use on actual aircraft structures to assess their state of damage throughout the life cycle for both the pristine and shot peened state. Numerical models simulating the shot peening process have been developed using LS Dyna 3D Finite Element Analysis code. The model assumes random shots in terms of time of impact and their location and considers the impact and energy loss between beads in the stream. The model, which will be verified by experimental results, will be used to investigate the effect of shot peening parameters and calculate the residual stress distribution through the thickness (Figure 13).

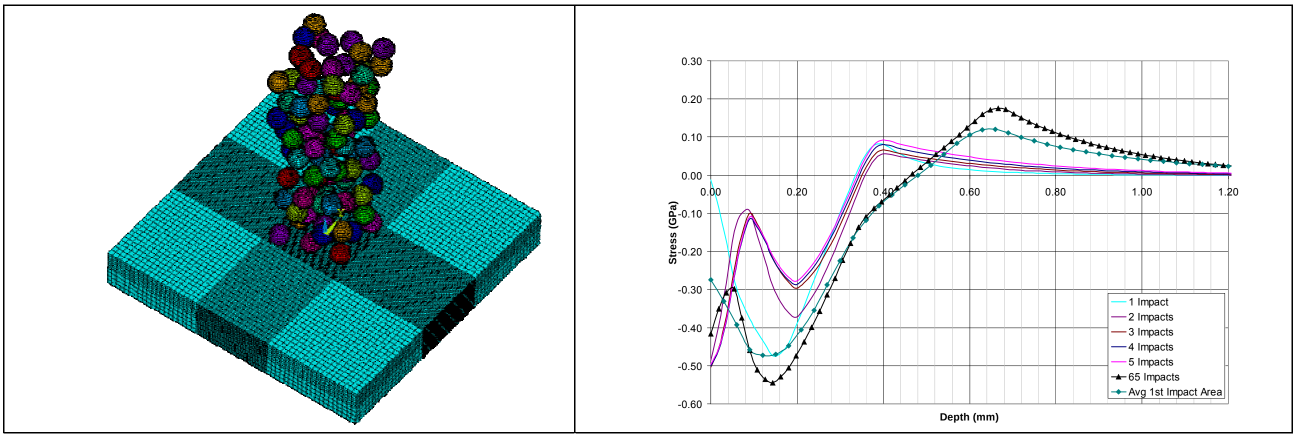


**Figure 13 3D dynamic FEA modeling is able to predict the influence of multiple shot impacts.**

As part of the study of hole cold expansion, a preliminary numerical study of fastener hole cold expansion using 3D finite element methods was carried out where several analytical models published in the open literature were recreated. and a theoretical model was developed that can be used to analytically predict the strains resulting from the hole cold expansion and riveting processes was developed with experimental verification using digital image correlation to measure the strain distribution, the details of which are presented elsewhere in this review. The results were then used to develop finite element prediction tools using the Patran Command Language (PCL) in order to study the residual stress profiles induced by the cold expansion process. With this computational tool, three-dimensional finite element parametric models with a variety of key parameters were automatically generated to address major issues associated with the optimization rate of the hole cold expansion so as to enhance the fatigue life of aircraft fleets (Figure 14). The cold expansion process was fully simulated in the models by taking into account dimensions of the actual cold expansion tools and realistic contact conditions.

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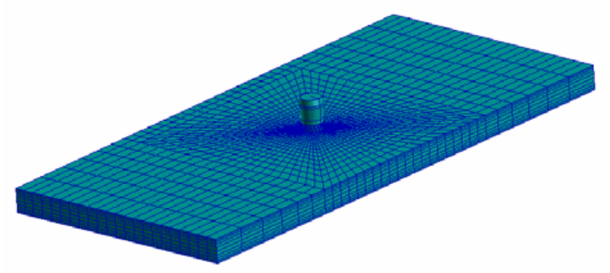
|  |  |  |
| Mandrel | Sleeve | Assembly |

**Figure 14 Parametric finite element models of all aspects of the hole expansion process have been developed and validated with experimental digital image correlation results.**

Using a design analysis matrix, a large number of numerical models with different parameters (elasto-plastic response of the material, proximity of the hole to the edge, sleeve thickness, friction, entrance versus the middle and exit side residual stress gradients, thickness of the sleeve, orientation of the split in the sleeve in relation to the low edge, etc) were generated and analyzed under several levels of cold expansion and externally applied tension loading (Figure 15 to Figure 17).

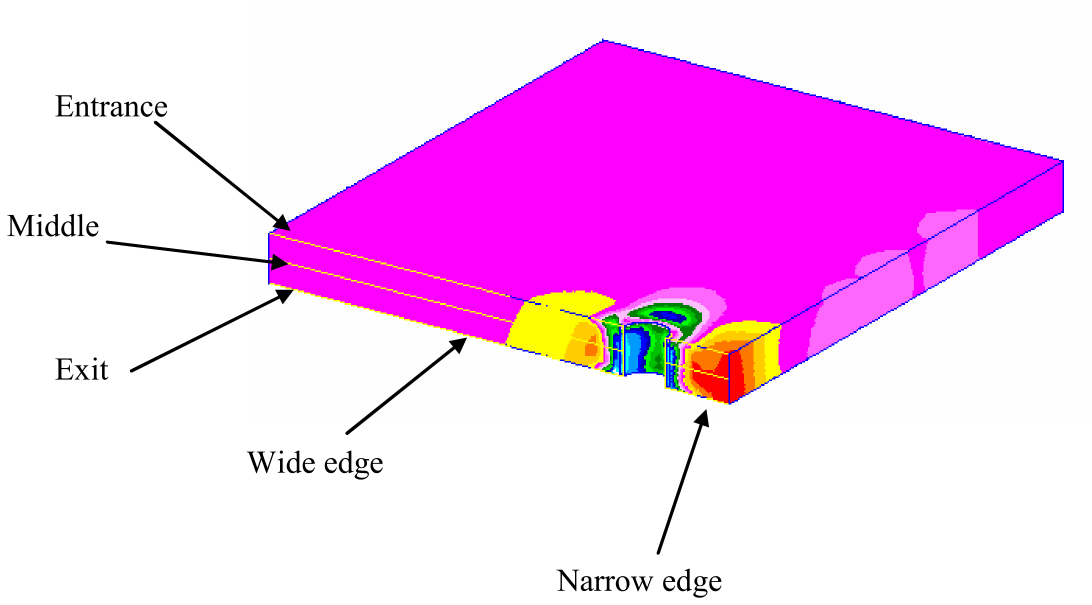
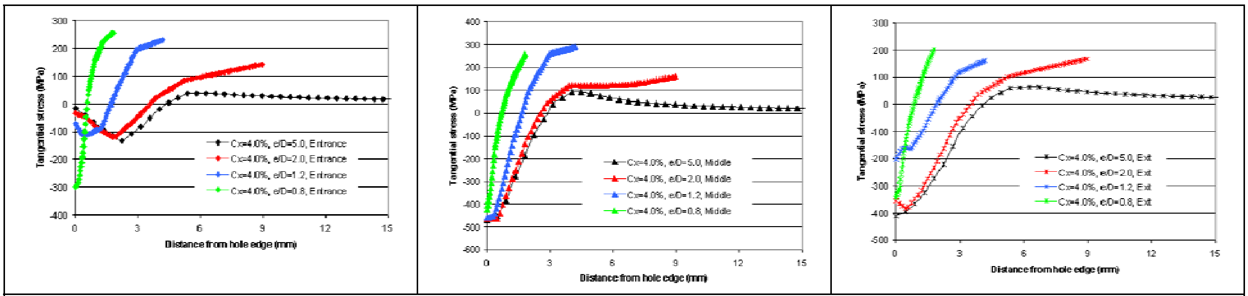
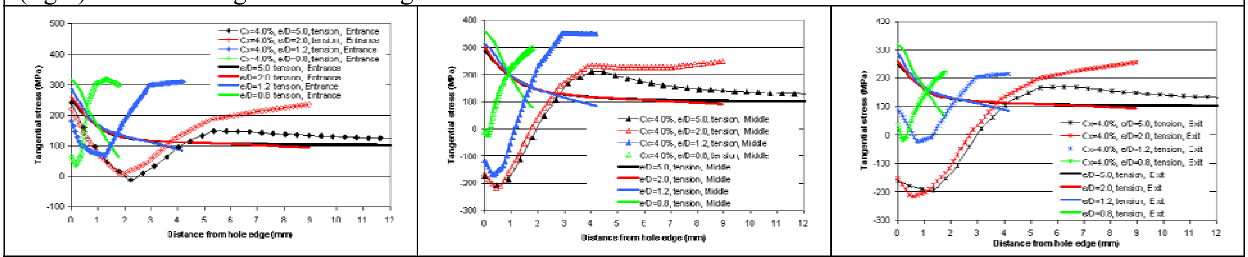



**Figure 15 showing the tangential stress path across the thickness of the parametric FEA model.**

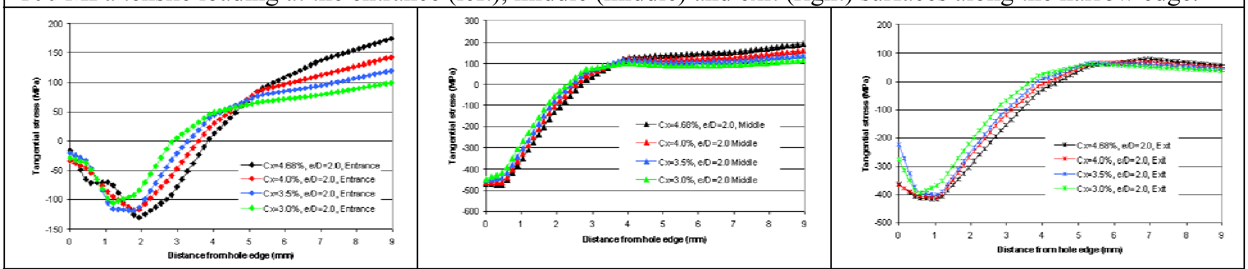
The developed FE models were validated by comparing with analytical solutions and test data measured by the digital image correlation system. The details of the digital image correlation measurements are described elsewhere in this review. The results of the parametric studies will be beneficial to engineers to better understand the residual stress state associated with cold expansion at the limits of applicability, which will potentially affect fatigue crack nucleation and growth at the perimeter of the cold expanded holes, in particular overcoming some of the previous difficulties quantifying the exact residual stress profiles and how they could impact long term structural performance, reliability, airworthiness and safety. A generic aircraft wing structure has been conceived and is being designed to evaluate competing solutions for critical areas subject to fatigue failure under low edge distance hole repair/enhancement scenarios, including use of hole cold expansion, bushings, interference fit and specialty fasteners [6, 7]



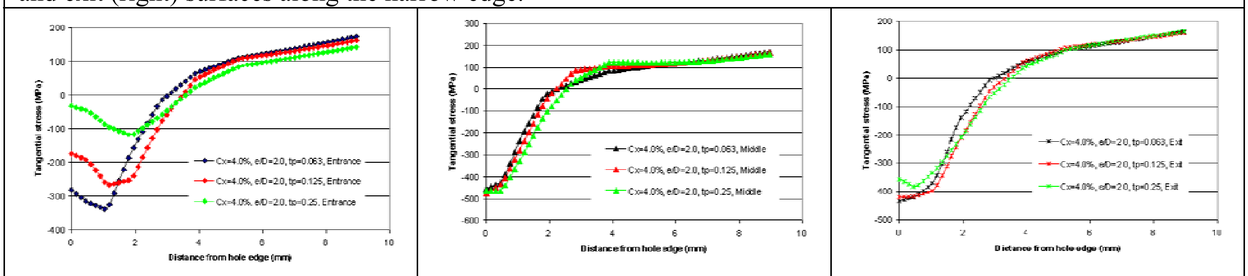

Comparison of residual stresses with different edge margins at the entrance (left), middle (middle) and exit (right) surfaces along the narrow edge.




Comparison of residual stresses with and without hole cold expansion with different edge margins with remote 100 MPa tensile loading at the entrance (left), middle (middle) and exit (right) surfaces along the narrow edge.




Comparison of residual stresses with different hole cold expansion levels at the entrance (left), middle (middle) and exit (right) surfaces along the narrow edge.




Comparison of residual stresses with different plate thicknesses at the entrance (left), middle (middle) and exit (right) surfaces along the narrow edge.

**Figure 16 FEA results comparing resulting residual stresses**

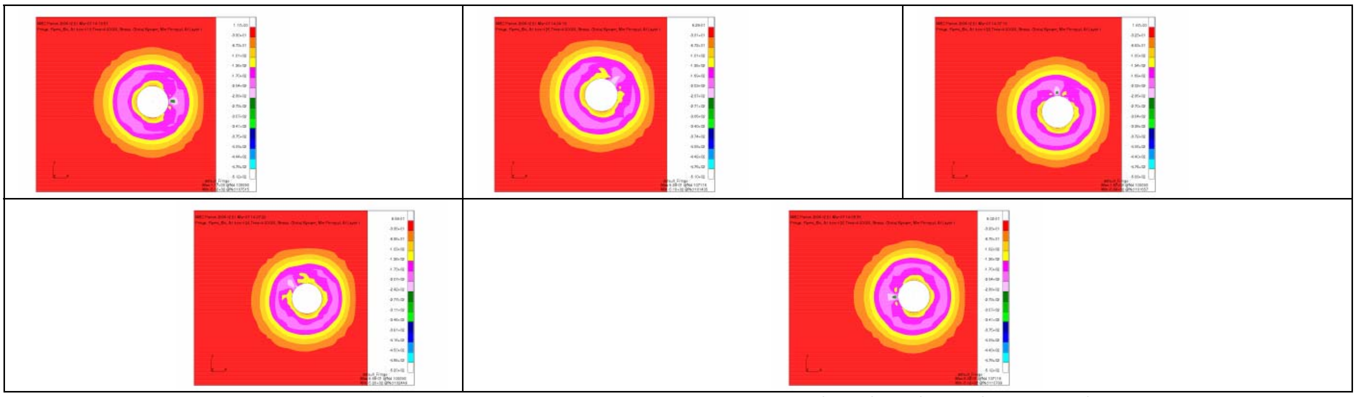


**Figure 17 FEA results of minimum principal residual stresses with 0°, 45°, 90°, 135° and 180° split orientations at the entrance surface.**

**Experimental and Theoretical Analysis of Hole Cold Expansion and Riveting in Fiber Metal Laminate (FML) Materials**

D. Backman, NRC-IAR-SMPL

The primary objectives of this research project were to:
1. Experimentally measure the strains (surface and through thickness) in fiber metal laminates (FML) due to both the cold expansion and riveting process and establish the effect of these processes, on structural performance.
2. Develop a theoretical model that can be used to analytically predict the strains resulting from the hole cold expansion and riveting processes.
3. Exploit the experimental results in order to provide recommendations for the design and production of FML for optimum performance (i.e. lowering life cycle cost by focusing on such issues as weight, durability and fatigue performance) with hole cold expansion and riveting.

A literature review was performed which examined the work of Nadai, Hsu and Forman, Budiansky and others who provided the ground work for creating a closed form solution of the cold expansion process. The approach taken by Ball in 1995 analytically examined the cold expansion process taking into account elastic-plastic unloading. The work performed by Zhang in 2005 built upon Ball's framework, incorporating modeling of cold expansion in finite plates. In both cases the material properties were modeled using a modified Ramberg-Osgood model which provides a relationship between stress and strain as follows:

$$\varepsilon = \begin{cases} \dfrac{\sigma}{\varepsilon} & \text{for } |\sigma| \le \sigma_{\mathrm{y}} \\[2ex] \dfrac{\sigma}{\varepsilon} \cdot \left| \dfrac{\sigma}{\sigma_y} \right|^{n-1} & \text{for } |\sigma| \ge \sigma_{\mathrm{y}} \end{cases} \tag{1}$$

where ε is the true strain, σ is the true stress, E is the elastic modulus, $\sigma_y$ is the initial yield stress and n is the strain hardening exponent. In the elastic region, the relationship between stresses and strains can be defined using the generalized Hookes law model. More complex, is the relationship between stresses and strains in the plastic zone, for which both Ball and Zhang rely on a model put forward by Hsu:

$$\varepsilon_r^{\ pl} = \left( \frac{1}{E_s} - \frac{1}{E} \right) \cdot \left( \sigma_r - \frac{R}{1+R} \cdot \sigma_\theta \right) \tag{2a}$$

$$\varepsilon_\theta^{\ pl} = \left( \frac{1}{E_s} - \frac{1}{E} \right) \cdot \left( \sigma_\theta - \frac{R}{1+R} \cdot \sigma_r \right) \tag{2b}$$

In the above equation, $E_s$ is the secant modulus at a point σ,ε on the stress strain curve and can also be obtained from the modified Ramberg-Osgoode model as:

$$\frac{1}{E_s} = \frac{\varepsilon}{\sigma} = \frac{1}{E} \cdot \left| \frac{\sigma}{\sigma_y} \right|^{\text{n-1}} \qquad (3)$$

The variable $R$ is defined as the ratio of in-plane transverse plastic strain to through thickness plastic strain in much the same way that the Poisson's ratio $v$ defines the ratio of lateral to transverse elastic strains. A value of $R$=1 would represent a purely isotropic material and an $R=0$ would represent anisotropic plastic behaviour. It is possible that this variable could be used to more closely tune the closed form solution to represent an orthotropic material such as GLARE. The derivation of the residual stress field was divided into two distinct parts. The first part derives a closed form expression for the stresses that result from the insertion of the oversized mandrel. This portion of the derivation would also be suitable for describing the stresses induced through the riveting process or any other process where an oversized mandrel or fastener is inserted into a hole. The approach Zhang and Ball used up to this point simulates the stress resulting from the insertion of an oversized fastener or rivet into a hole. For the purposes of making a comparison between experimental and theoretical results it was critical to have an explicit formulation for the strain values in both the elastic and plastic regimes. During the loading phase, the plastic strains can be calculated by using the following explicit expression for the radial and tangential strains:

$$\varepsilon_r = \frac{1}{E} \cdot \left( \frac{\sigma}{\sigma_{\mathrm{y}}} \right)^{\text{n-1}} \cdot \left[ \sigma_{\mathrm{r}} - \frac{R}{1+R} \cdot \sigma_\theta \right] \qquad \text{(4a)}$$

$$\varepsilon_\theta = \frac{1}{E} \cdot \left( \frac{\sigma}{\sigma_{\mathrm{y}}} \right)^{\text{n-1}} \cdot \left[ \sigma_\theta - \frac{R}{1+R} \cdot \sigma_r \right] \qquad \text{(4b)}$$

The closed form formulations first put forward by Ball and Zhang were combined and their theories were leveraged to produce a parametric model of the cold expansion process in MathCAD. The parametric model was used to model the FTI cold expansion process and could also be used to simulate the riveting process, as shown in Figure 18.

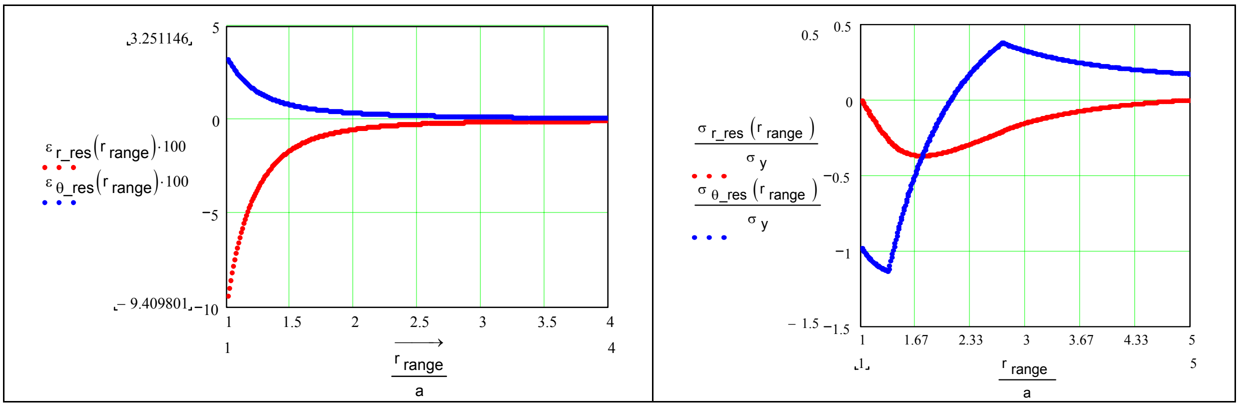



**Figure 18: Closed form solution for (left) residual strain and (right) residual stress after cold expansion**

GLARE 3-3/2 composed of three sheets of aluminium with a symmetrical arrangement glass fibers oriented in the 0/90 direction in between each sheet was manufactured at NRC-IAR for this program. The GLARE panel was inspected using both C-scan and thermography with qualitative findings shown in Figure 19.

A test fixture a three point support for the coupons was designed that was able to securely fix the coupons in place during riveting or cold expansion. As well, the fixture was able to accurately locate the coupons so that they could be removed and replaced with no change in position

Both the aluminium and GLARE coupons were cold expanded using tools provided by FTI Inc using qualified personnel and equipment.

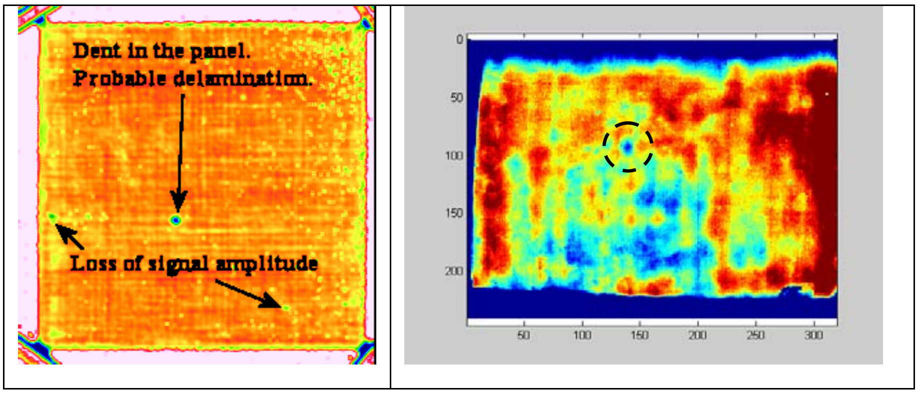



**Figure 19: NDI results (left) ultrasonic C-scan (right) thermography**

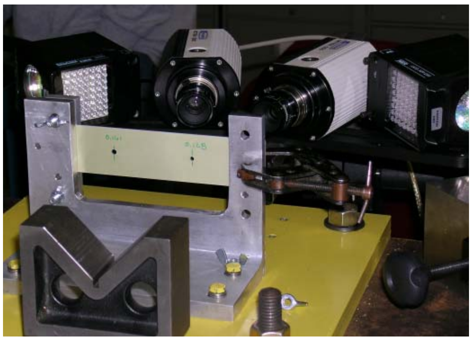


**Figure 20: Coupon test fixture with camera and lighting system**

**Figure 21: Cold expansion process for aluminum coupon**

Some sample results are shown in Figure 22 to demonstrate the difference between cold expansion in aluminum and GLARE sheets.

Although data collection is still ongoing some basic comparisons were performed between the experimental results, the analytical solutions and a three dimensional FEA simulation of the cold expansion process (Figure 23). The results show that the experimentally derived data validates both the analytical solution and the FEA model.

Riveting of GLARE and conventional aluminium was also conducted as part of the test program. The rivet was set in the coupon using the manual tool (Figure 24) and then removed from the fixture and riveted using a manually operated press.

A comparison of the strain field that results when riveting aluminum and GLARE is shown in Figure 25. Although the "butterfly" pattern that resulted from the cold expansion process is not evident in these coupons, there appears to be higher strains around the rivet in GLARE.

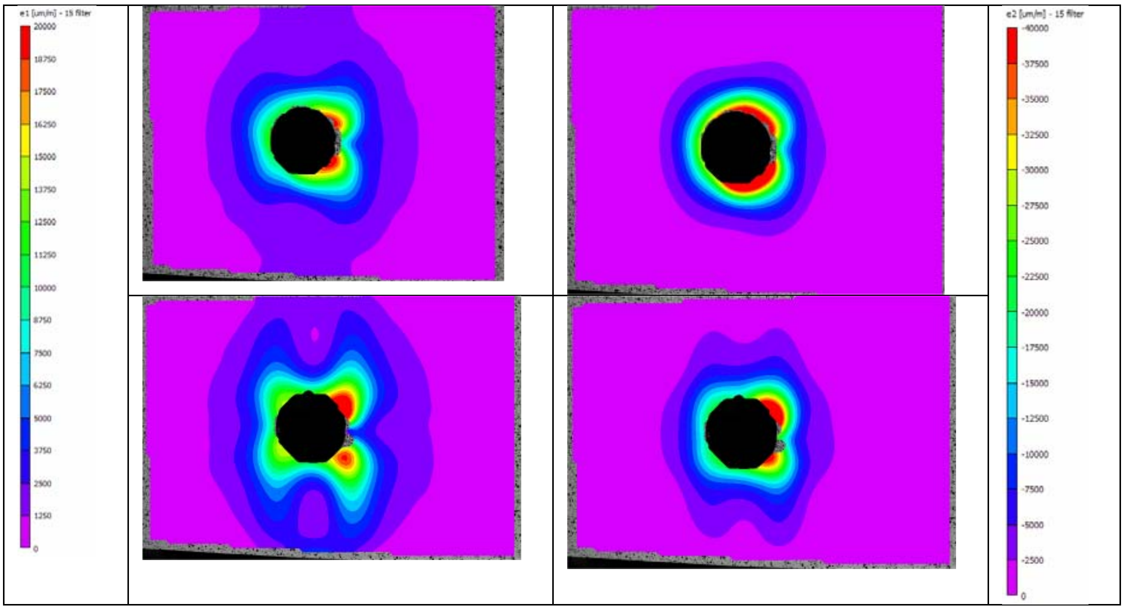


**Figure 22: Comparison of cold expansion in aluminum (top) and GLARE (bottom)**

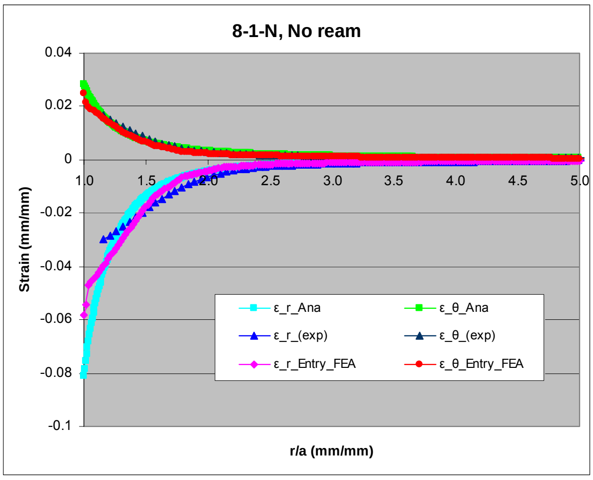



**Figure 23: Comparison between experimental, closed form and FEA results**

**Figure 24: Manual riveting process with a GLARE coupon**

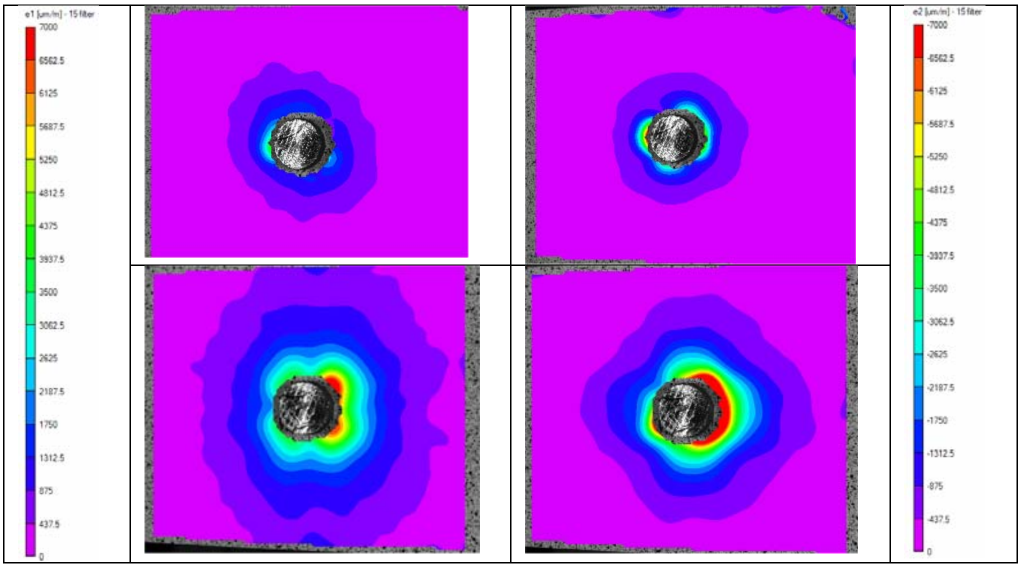



**Figure 25: Comparison of riveting in aluminum (top) and GLARE (bottom)**

**Surface modification technologies such as Low Plasticity Burnishing as superior alternate to shot peening**
F. Caza, Bombardier Aerospace

Low Plasticity Burnishing (LPB) is a rapid, inexpensive surface enhancement method performed on conventional CNC machine. It produces a deep layer of high compression with improved surface finish with minimal cold work. The benefits are improved resistance to high cycle fatigue and prevention stress corrosion cracking.

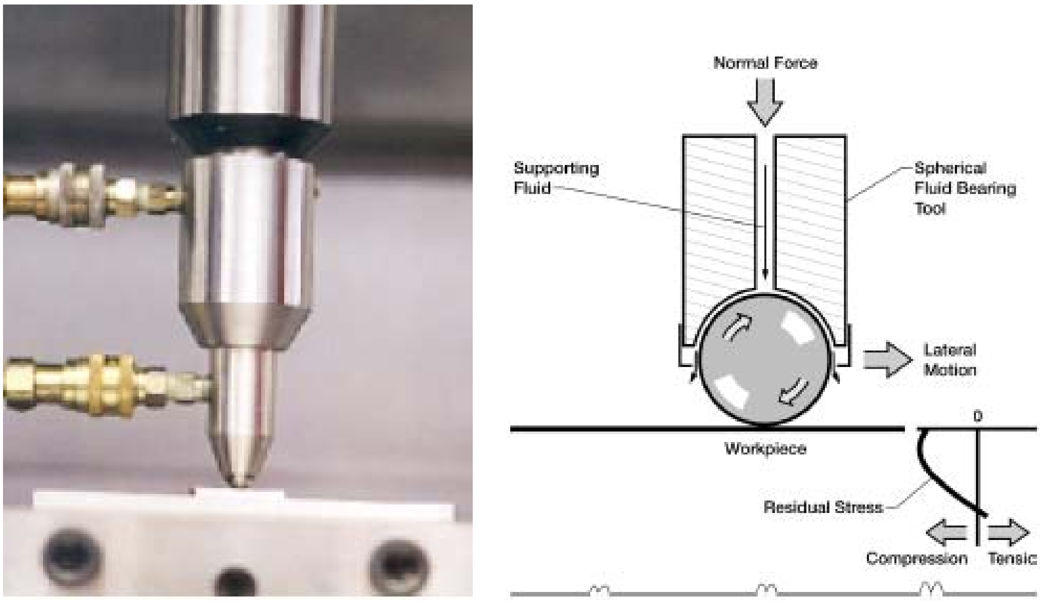



**Figure 26 Low plasticity burnishing.**

Fatigue, corrosion fatigue, X-ray diffraction residual stress profiles and corrosion testing (salt spray exposure) were performed to determine the influence of LBP on 7475-T7351 and 2024-T351 aluminium alloys, bare vs. clad. LPB was compared to shot peening.

Remaining work includes full review of test results, equipment, control parameters, cost analysis and manufacturing time comparison to determine if the benefits of LPB outweigh the current process (shot peening). If LPB is retained as a profitable process, the second phase of the project will be to establish optimum compressive layer profile and process parameters for different applications/thicknesses.

## FRACTURE MECHANICS AND CRACK PROPAGATION STUDIES

### Short Crack Model Development for HOLSIP framework

M. Liao, NRC-IAR-SMPL

A three-year collaborative DND-NRC project was started in 2005 to further develop the HOLSIP framework by concentrating on the short/small crack phase. Both experimental techniques and analytical methods will be developed to generate databases and carry out a short crack analysis, which will then be included in the HOLSIP framework. The material systems of investigated are 7050-T7542 forging and 7050-7451 plate. The major tasks includes, 1) Crack growth modeling in a residual stress field; 2) Micrographic analysis; 3) Short crack growth tests; 4) Post fracture examination; 5) Integration (test and model) with case studies. The project will be completed in 2008; this summary only provides some efforts on the short crack modeling in progress.

Before the short crack test data are generated for the 7050 aluminum alloys, some modeling work were carried out using the data from 2024-T351 aluminum alloys. A probabilistic short crack growth model was developed to estimate the fatigue life distribution for bare 2024-T351 coupons (Figure 27). Previous fractographic analysis has shown that the majority of cracks nucleate from the constituent particles present in bare 2024-T351 coupons. The existing fracture mechanics models have difficulty in 'growing' a small-particle-induced crack (i.e. infinite life), and cannot correlate the size of a crack-nucleating particle to the length of the fatigue life in cases where a small (large) particle results in a short (long) life. In this probabilistic model, particle width and height distributions were first used to account for the randomness of the particle size. Then another random variable, the stress intensity factor limit, $\Delta K_{IDS}$, for IDS/particle induced cracks, was introduced to account for the combined effect of microstructure features (e.g. grain size, grain orientation, and grain boundary) on short crack growth (Figure 28). A Microsoft Excel VBA (Visual Basic for Applications) program was developed and linked with the AFGROW COM (Component Object Model) server to estimate the life distribution using the Monte Carlo technique.

The comparison between analysis and test results indicated that by using two random variables, i.e., particle width and height, the model still could not simulate the so called "smaller (larger) particle, shorter (longer) life" cases, neither a good estimation of the fatigue life scatter. A fundamental cause is that varying the random particle size alone in a fracture mechanics model cannot simulate the scatter of short crack growth which is also affected by other microstructural features. By including the third random variable, $\Delta K_{IDS}$, for the IDS/particle induced crack, the developed model could provide a better estimation of the life scatter and distribution (Figure 29). More importantly, with the parameter $\Delta K_{IDS}$, the model was able to simulate the so called "smaller (larger) particle, shorter (longer) life" cases found in the tests.

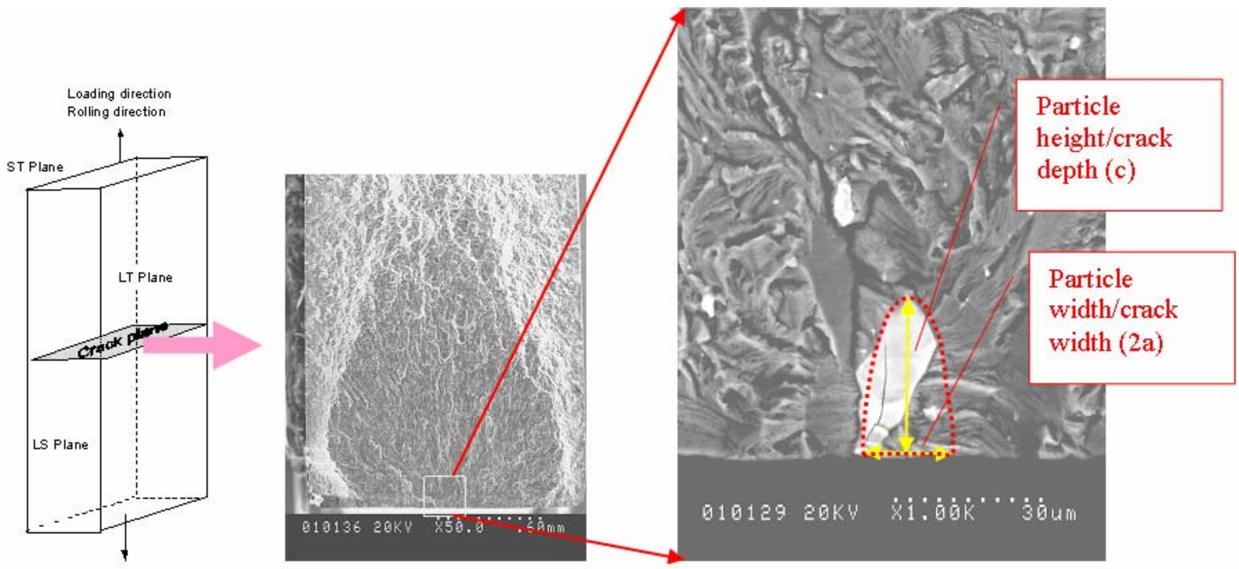



**Figure 27 Fracture surface and crack-nucleating particle based crack model.**

After the short crack tests are done, the above models will be examined for 7050 aluminum alloys, and then extended to a component case. In addition to the fracture mechanics based model, the dislocation theory based short crack model is also under investigation in this project. Details and an overview are expected to be presented in the next ICAF review.

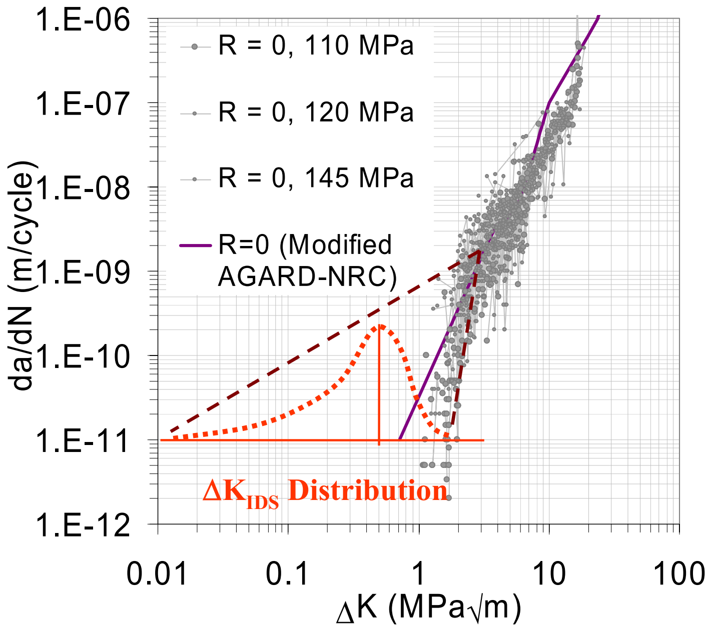



**Figure 28 Schematic of a $\Delta K_{IDS}$ distribution with the modified AGARD-NRC model and some AGARD data.**

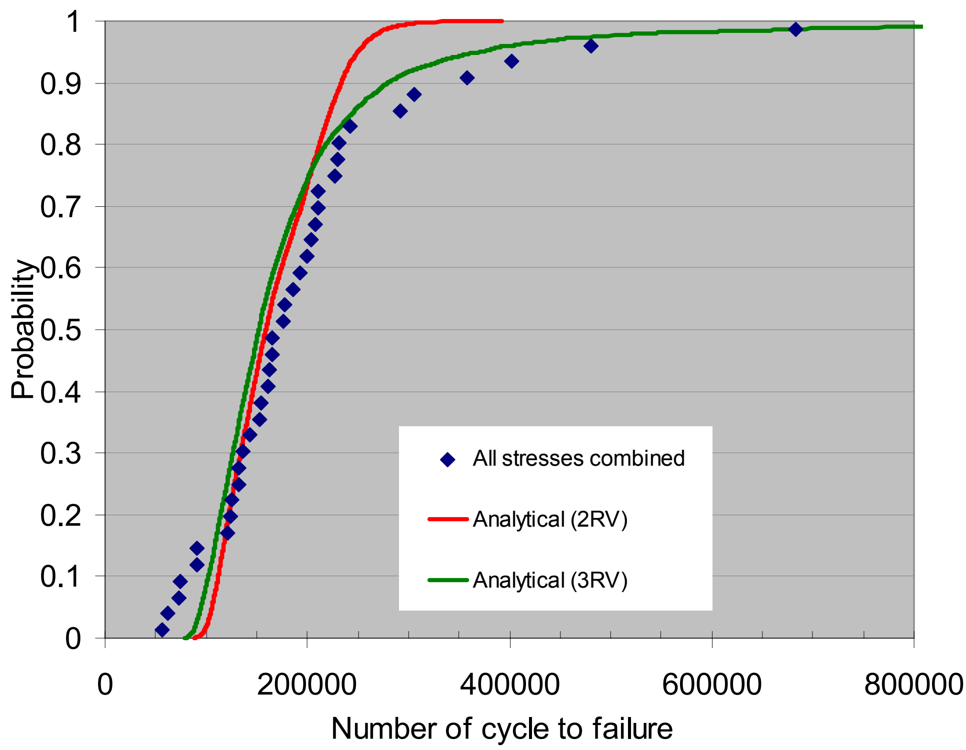



**Figure 29 Combined analytical and test results for all stress levels.**

**Fretting Fatigue of Single Lap Splices**

A. Brown and P. Straznicky, Carleton University

This fretting fatigue study is part of a larger research project on fatigue of riveted lap splices in conjunction with C.D. Rans and P.V. Straznicky. Little work to date has been performed on fretting in thin lap splices where fretting damage and crack nucleation occurs at the faying surface around rivet heads instead of at the rivet/sheet interface. This research will look at fretting in thin 2024-T3 aluminium with different common surface treatments (alclad, anodized). Both finite element simulation and coupon fatigue testing will be employed in the study of both simple Hertzian contact and single lap splices. With the assistance of the stresses obtained in the finite element simulations, predictive fatigue equations based on the Smith-Watson-Topper expression can be compared to experimental results. This modified SWT parameter has been successfully applied in other fretting cases [8,9] but is untested in thin lap splices.

Currently, a finite element model of a 2-rivet single lap splice has been developed using the commercial finite element package ABAQUS v6.6. Both rivet forming and splice loading was modeled with four different rivet squeeze forces. For the proposed predictive equation to be applied, the peaks in stress at the edge of contact, normally associated with fretting, must be obtained. To capture these peaks in stress, a very fine mesh is required. Since a fine mesh density makes a splice loading simulation prohibitively large, a sub-modeling process is being used. The coarse, global splice simulation is run to completion and then a sub-mesh is created in the area of interest such as the area surrounding a rivet. This sub-mesh can then be run as an independent simulation with displacements from nodes in the global model driving the nodes in the sub-model. This process is then repeated until a sufficiently refined mesh is obtained (Figure 30).

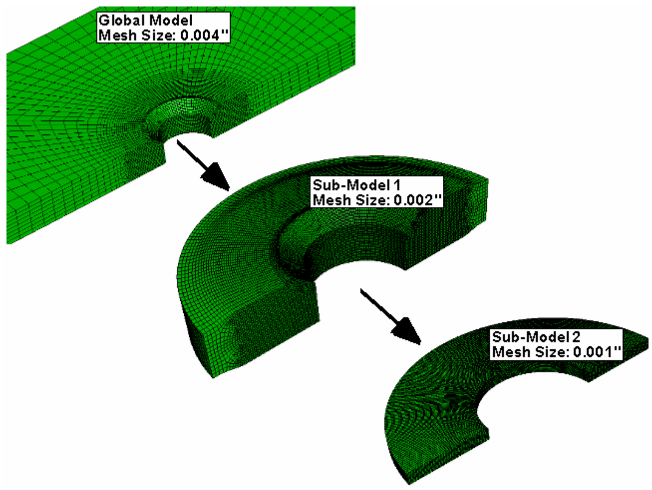



**Figure 30 Schematic of sub-modeling process.**

The splice simulation and testing will be complimented by simulation and testing of a simple cylinder on flat fretting contact. A fretting test rig has recently been completed along with Abaqus simulations of this simple set-up. The simple fretting set up allows for verification of contact results through comparison to analytical solutions for Hertzian contact.

**Fretting Fatigue Testing**

N. Bellinger, NRC-IAR-SMPL

The NRC carried out a number of tests associated with inducing fretting fatigue in various types of coupons as well as trying to detect fretting using various nondestructive inspection techniques. In order to carry out the required fretting fatigue tests a new hydraulic fretting fatigue rig was designed to apply normal loads to opposing fretting pads onto a fatigue specimen. The rig consisted of a load cell to continuously monitor the applied normal load, a stiffened support arm to prevent movement of the fretting pad on the load cell side of the rig and an actuator on the opposite side of the rig to the load cell to apply the required load, Figure 31. The rig was secured to a steel plate to increase its overall stiffness and prevent excessive movement during the loading cycle, which was also fastened to a MTS test frame.

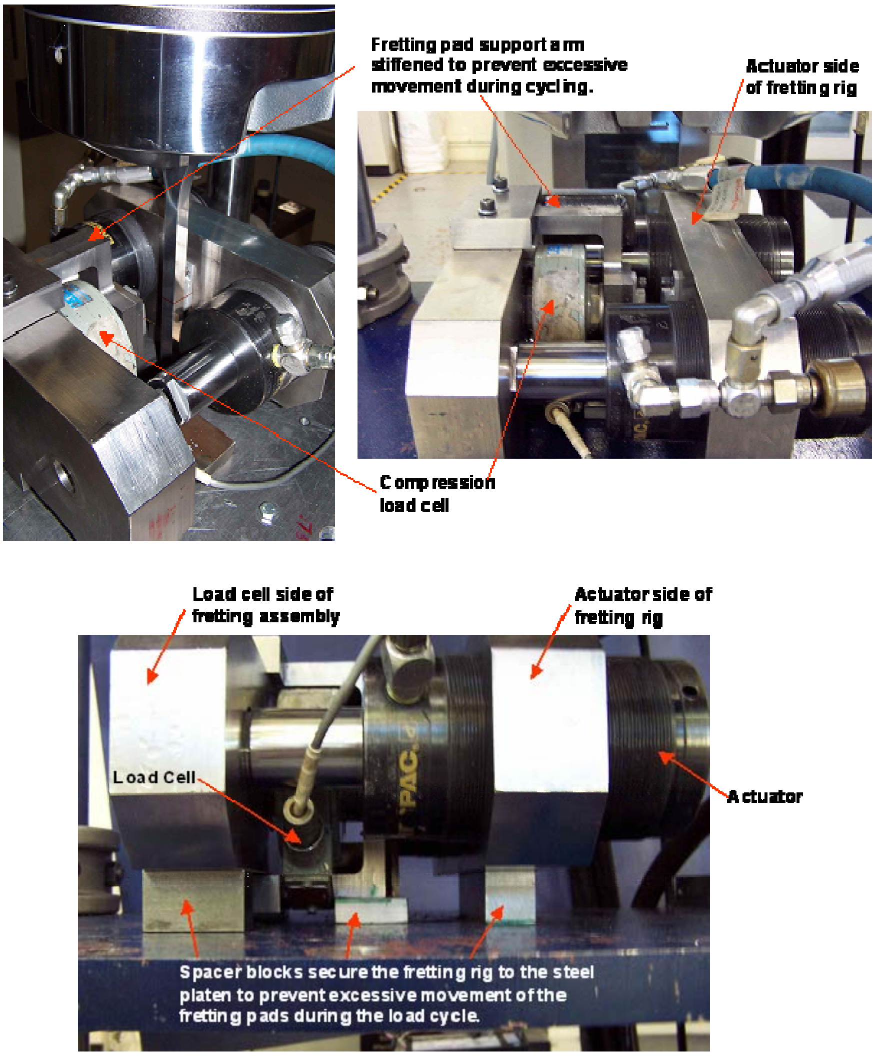



**Figure 31. Photographs of redesigned fretting rig.**

To be able to obtain fretting fatigue data that could be used to correlate with the analytical life prediction models representative of aircraft structures that were being developed by Analytical Predictions/Engineered Solutions (AP/ES) Inc., a building block test matrix was developed. In this matrix, both low and high Kt fretting fatigue tests were carried out as well as multi-fastener lap joint tests.  Since fretting degradation is strongly dependent on the type of material and Al 2024-T351 was widely used in fatigue and damage tolerant critical structures in aircraft structures, different thicknesses of this material system were used for all tests.  Also to try and eliminate, or reduce, the amount of fretting debris that would be present on the fracture surface, a nondestructive inspection procedure was developed to interrupt the tests at a predetermined crack size.  Four small ultrasonic transducers were mounted on a specimen (two on either side above and below the fretting pads) to continuously monitor the area where cracks were known to form, that is, at the edge of the fretting damage, see Figure 32, and trials determined that this setup was able to detect cracks that were approximately 0.02inch in depth.

The low $K_t$ test coupons were machined from as-received 2024-T351 rolled stock with of cross-section of 1.5-inch by 0.5-inch.  Each coupon had an approximately 6 inch long gage section with a 0.5-inch by 0.5-inch gage section.  Two different cylindrical fretting pads were used to carry out the tests; one had a 5-inch radius while the other one had a 9-inch radius.  All tests were conducted at 3 Hz in laboratory air.  During the tests the temperature varied between 68$^o$F and 72$^o$F while the relative humidity varied from 35% to 45%.  Each specimen was tested under constant amplitude loading with an R-value of 0.05.  The results of which are shown in Figure 33 along with the computed lives.  As can be seen from the results, the computed lives gave similar results to the experimental ones.

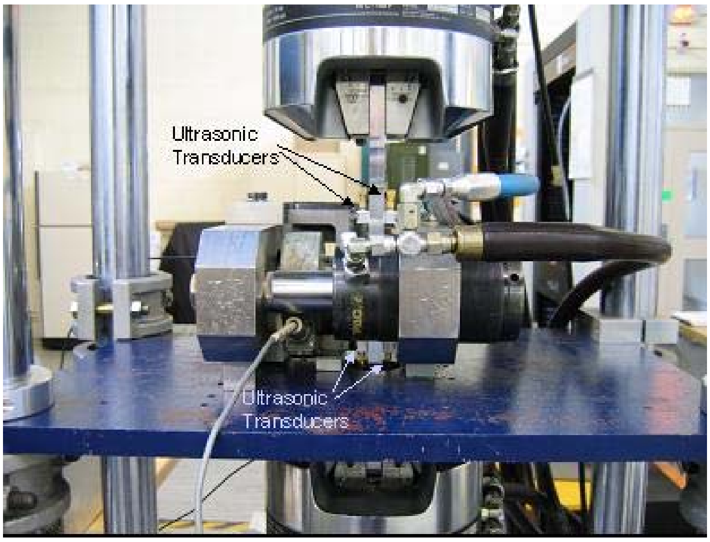



**Figure 32. Test setup showing test coupon containing ultrasonic transducers.**

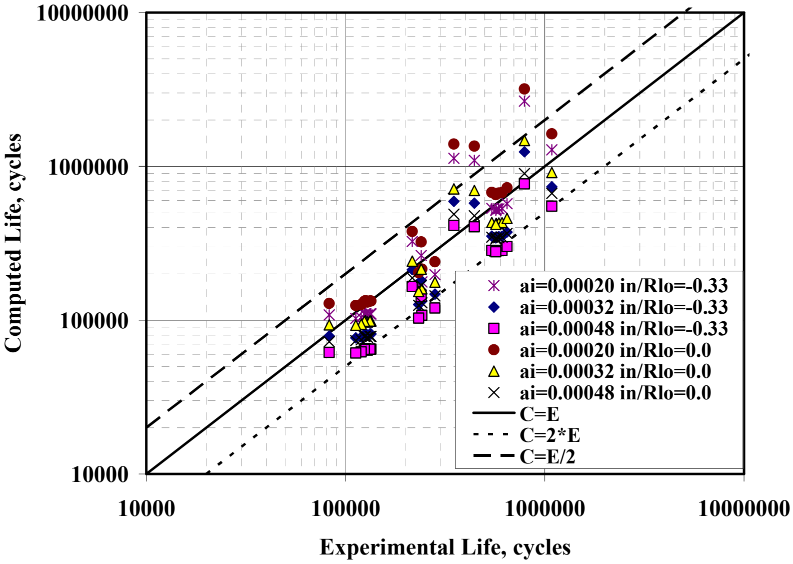



**Figure 33. Low $K_t$ Computed Lives as Function of Experimentally Measured Lives**

For the high $K_t$ tests, an open hole flat coupon was designed. For these tests an additional variable was added to determine the effect that a residual stress field would have on the fretting fatigue damage. To accomplish this, plugs with difference interference fit levels were placed into the hole of different coupons.

One way to look at the data is to plot the correlation Computed to Experimental (C/E) ratio against the normal cumulative distribution function (CDF), Figure 34. To interpret this figure, it should be noted that when the C/E ratio is less than 1.0, then the computed lives are under computing the test lives; if this ratio is greater than 1.0, then the computed life is over computing the lives. An 'ideal model' would essentially have a flat C/E ratio relationship, C/E=1.

As can be seen from Figure 33, the direct application of the Holistic Fretting Model formulation as used in the Low $K_t$ model did not correlate as well with the High $K_t$ fretting fatigue tests and thus some modeling differences are needed. The computations produced results that were longer than those observed in the tests, however review of some of the fracture surfaces indicated a potential that the primary failure mode may have been from scars produced from fastener installation in conjunction with fretting damage producing a failure mode slightly more severe than the scenarios from the Low $K_t$ experiments. Assessments using initial flaws of the size in proximity to those observed for the scars produced results closer to the test results. Thus these experiments may have a manufacturing damage condition that needs to be combined with the fretting damage inputs to meet the differences in failure modes.

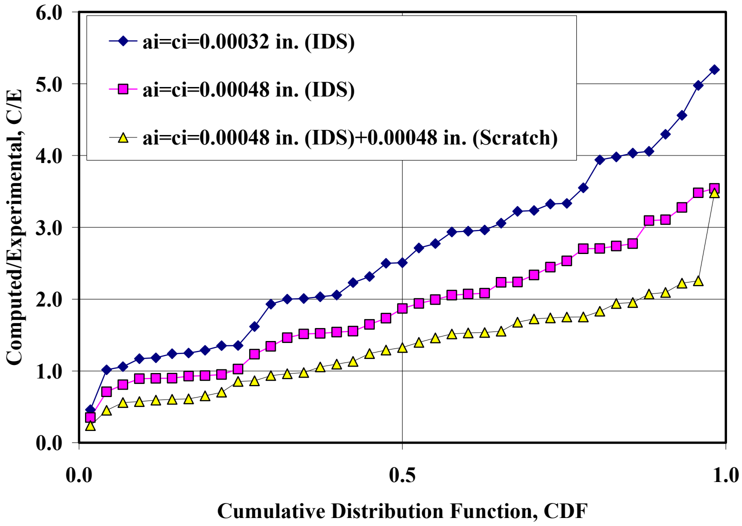



**Figure 34. High $K_t$ Computed/Experimental results**

The lap joint experiments consisted of 10 5x3 single shear lap joints made of two thin sheets that were cycled with constant amplitude stress spectra at R=0.02. Each lap joint was constructed from two 2024-T3 bare aluminum alloy 0.063 inch sheets. All joints were assembled with using a force controlled riveting process. Two rivet squeeze forces were used; 10 kN or 18 kN. Four of the specimens had a plastic sheet positioned between the two skins to substantially reduce the frictional load transfer between the faying surfaces while the other 6 specimens were assembled to produce typical aluminum to aluminum friction load transfer. No-load transfer fasteners were used on the lap joint ends to reduce bending effects on the edges to provide a consistent gage section within the specimen. A passive crack detection technique, shown in Figure 35, was also developed to stop the lap joint cycling before a crack became so large that a significant amount of fretting debris would contaminate the crack faces.

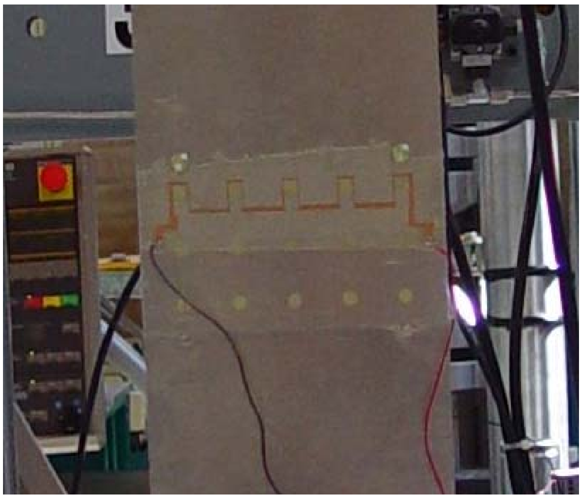


**Figure 35. Passive sensor for the early detection of cracking.**

The fatigue lives referred to in these experiments was the number of "cycles to a visible crack", which is defined by an indication of a crack. The Holistic Fretting Model correlated better with the experimentally measured lives on the lap specimen that were assembled to allow for natural fretting as shown in Figure 36.

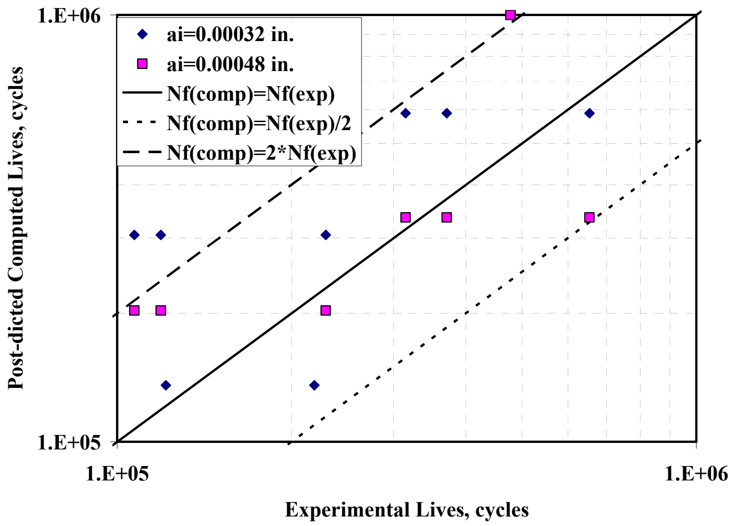



**Figure 36. Rivet Lap Joint Computed Lives as Function of Experimentally Measured Lives.**

**Recent NRC-IAR Fibre Metal Laminate Fatigue Research**

J. Laliberte, NRC-IAR-SMPL

Since 2004 NRC-IAR has undertaken an internally funded project to evaluate the effect of cure cycle modification on the fatigue behaviour of fibre metal laminates (FMLs). It is well understood that process induced residual stresses in FMLs can have a detrimental effect on fatigue life to measurable cracks [10, 11 and 12]. The mismatch in the coefficient of thermal expansion between the fibre reinforced prepreg and metallic layers induces tensile stress in the metal layers during cure. When the resin reaches its gelation point, these stresses are "locked into" the laminate and as the FML is cooled from its stress-free temperature back to room temperature the net stresses increase. By reducing these tensile stresses it is possible to improve the fatigue initiation life of FMLs [12, 13 and 14]. Previous work at NRC-IAR [15,16] and elsewhere [17] has shown that it is possible to modify residual stresses in composite laminates by changing the cure cycle of the resin. The present NRC-IAR project included work on cure cycle optimization, development of in-situ residual stress monitoring techniques during cure, detailed resin cure property assessments, improved fatigue crack growth modelling and extensive mechanical testing (static and fatigue). Fatigue testing was carried out on low $K_t$ and centre-crack tension (CCT) specimens. While the CCT testing is still ongoing, the low $K_t$ initiation (cycles to 2mm crack length) testing is almost complete. These results have shown that promising improvements in initiation life can be achieved simply by modifying the cure cycle to produce a lower stress-free temperature in the laminates (Figure 37). This is accomplished by adding additional steps into the cure cycle that follow the evolution of the $T_g$ in the resin and to ensure that resin gelation occurs at the lowest possible mismatch between the thermal stresses in the constituent layers while providing a full degree of cure.

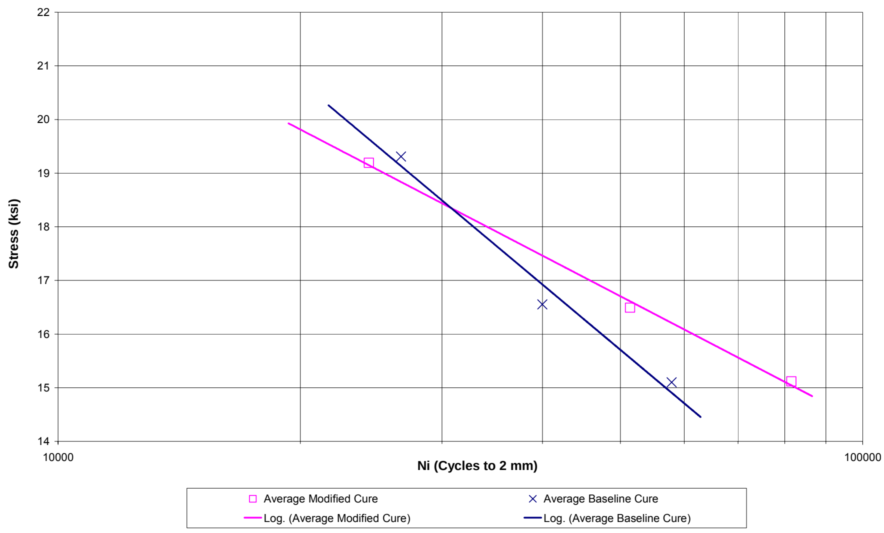



**Figure 37: Summary of low *Kt* fatigue test results for NRC-IAR baseline and modified cure cycle FML specimens.**

**Fatigue crack growth and crack closure measurements on thin sheet in aluminum and steel**

D.L. DuQuesnay, Mechanical Engineering Department, RMC

This project demonstrates a simple side-face strain gauge technique for the measurement of fatigue crack closure loads in thin plate materials. The sensitivity of measured closure loads in relation to strain-gauge position was investigated. Constant load amplitude and constant $\Delta K$ fatigue crack growth experiments were conducted on alclad and bare 2014 and 2024 series high strength aluminium alloys and on a mild steel using a standard SEN specimen configuration. The results showed that gauge positioning is a critical factor. In particular, a gauge placed close to the starter notch tip produced a compliance curve with an increase in slope rather than the expected decrease. This curve allowed a reproducible crack closure load to be readily measured. The closure loads obtained were consistent with those obtained from more conventional gauge positions. The technique should be useful for monitoring crack closure behaviour in aircraft skins [18].

**Characterization in DADT of titanium castings (vs. wrought alloys)**

F. Caza, Bombardier Aerospace

Fatigue and fracture toughness tests were performed to compare castings to wrought titanium (Figure 26). Axial Fatigue (120 specimens) tests were performed at ambient temperature at three different stress levels. The test coupons were tested at the following stress intensity factors of $K_T$ = 1.0, 1.5, 3.0. Stress ratios of +0.1 and +0.5 were evaluated.

C(T) specimens were used to determine fracture toughness used. Tests were performed at ambient temperature and at low temperature (-65$^{o}$F) per ASTM E399. R-Curve was determined for different thicknesses per ASTM E-561 (Figure 39). Test coupons were extracted from the midsection of the cast plate.

Fatigue crack growth testing was performed, at ambient temperature, in accordance with the requirements of ASTM E-647. The cyclic frequency was carried out under constant amplitude loading at 5 Hz. The test coupons were tested to failure at four different stress ratios: R = +0.1, R= +0.4, R= +0.8 and R= -0.5. ΔK range of 5.0 to 35.0 ksi(in)$^{1/2}$. A maximum stress level of 20 ksi was applied. Test coupons were extracted from the mid-section of the cast material.

**Figure 38 Titanium casting**

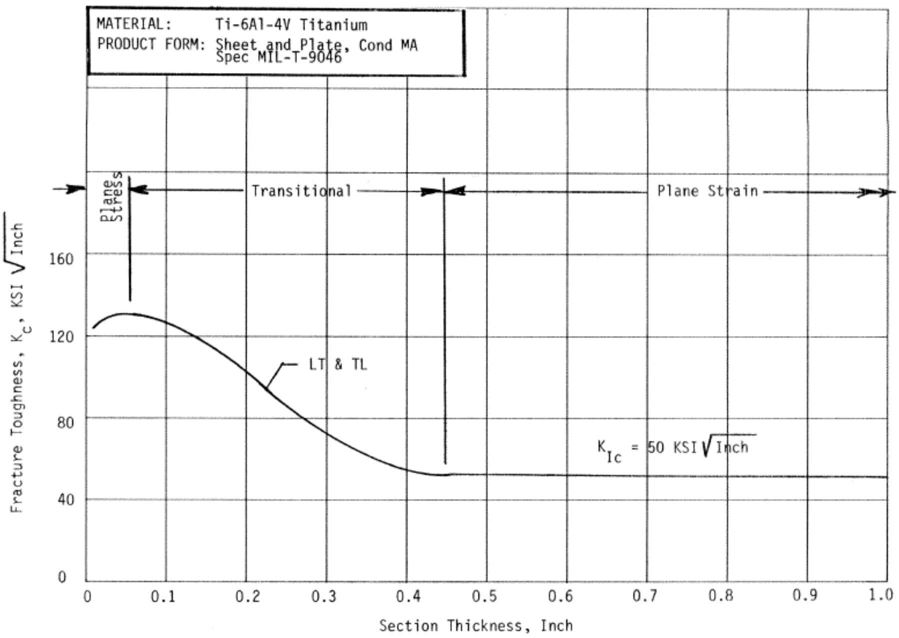



**Figure 39 Fracture toughness test results**

## FAILURE INVESTIGATIONS

**Fractographic Investigation and Determination of the Rate of Growth of Cracks from the F/A-18 IFOSTP Full Scale Fatigue Tests**

M. Roth, QETE, DND

The Quality Engineering Test Establishment (QETE) of the Department of National Defence (DND) is performing detailed fractographic investigations to determine the rate of growth of cracks found during the F/A-18 International Follow-On Structural Test Program (IFOSTP) full scale fatigue testing. The IFOSTP is a collaborative program to demonstrate the F/A-18 original design life of 6000 hours, where Canada has been testing the center fuselage and the wing, and Australia the rear fuselage and empennage. This was accomplished by repeatedly applying a spectrum of variable amplitude loads simulating 325 flight hours until reaching the desired life (i.e. 16636 simulated flight hours (SFH) for the fuselage and 18256 SFH for the wing). Knowledge of the crack growth rate is required to establish appropriate inspection intervals, maintenance actions, or modifications.

The F/A-18 IFOSTP spectrum was relatively short and repeated many times in the course of the test. Because of a relatively small advance of the crack front per block, ideally two consecutive spectrum blocks will leave similar striation patterns. When these repeating patterns can be identified, the distance between them will provide the crack growth per spectrum block. QETE has been using optical microscopy and scanning electron microscopy to identify the repeat patterns and record sequentially their coordinates starting as close to the origin as possible and travelling in as much a straight line as possible to the end of the crack. The crack depth is then plotted as a function of blocks or SFH to generate the crack growth curve.

The F/A-18 IFOSTP center fuselage test and wing test cracks from the following main areas have been investigated since ICAF 2005:

- Center fuselage: Y453 bulkhead [19], Y470 intake flange [20], Y488 bulkhead [21], Y497 former [22], dorsal deck [23], nacelle support [24], stringer [25], and upper floor [26]
- Inner wing: Front spar [27]
- Outer wing: Support rib [28]

The main findings were that:

- Cracking often nucleated very early in the component life at small surface pits, which were formed during the etching step preceding the deposition of the ion vapour deposited (IVD) aluminums corrosion preventative coating applied to the 7050 aluminium alloy of the F/A-18 structure.
- Crack propagation was initially slow and then increased markedly usually between 6000 and 10000 SFH.

## PROBABILISTIC AND RISK ANALYSIS METHODS

**Enhancement of Probabilistic Risk Assessment Methodology/Tool**

M. Liao, NRC-IAR-SMPL

With the support of a DRDC/DND – NRC-IAR collaborative project, NRC has continued to develop the methodologies and to enhance an in-house computer tool, ProDTA (Probabilistic Damage Tolerance Analysis), for aircraft structural risk assessment. Based on the HOLSIP (Holistic Structural Integrity Process) framework, ProDTA was designed to calculate the probability of failure (POF) of aircraft structures, by taking into account both fatigue damage and environmental related age degradation. Both fatigue and corrosion inputs are used by ProDTA, as shown in Figure 40.

In ProDTA, the coupling Monte Carlo simulation and probability integration techniques were further enhanced for accurately calculating the probability of failure [29]. A residual strength failure criterion was added in ProDTA to deal with component level failure risk analysis. Aiming to meet the increasingly needs on *Quantitative Risk Assessment* by the Canadian Forces (CF), a prototype GUI was developed for ProDTA and linked with MS Excel spreadsheet, as illustrated in Figure 41. Upon requested by the CF, a risk assessment was carried out for lower surface panel of CC-130 center wing, using ProDTA. All the input data were established based on the in-service damage findings and usage data from the Canadian Forces. The preliminary (Phase I) work showed that ProDTA gave satisfactory results when compared to OEM's analysis, which demonstrated that NRC is capable of doing the risk assessment for DND using our own code and inputs. In addition sensitivity studies were performed to show the significance of initial crack size distribution, maximum stress distribution, residual strength curve, crack growth curve, and probability of detection. The Phase II work is under discussion to include widespread fatigue damage into the risk analysis, and extend the analysis for more CC-130 locations, and/or other Canadian Forces aircraft.

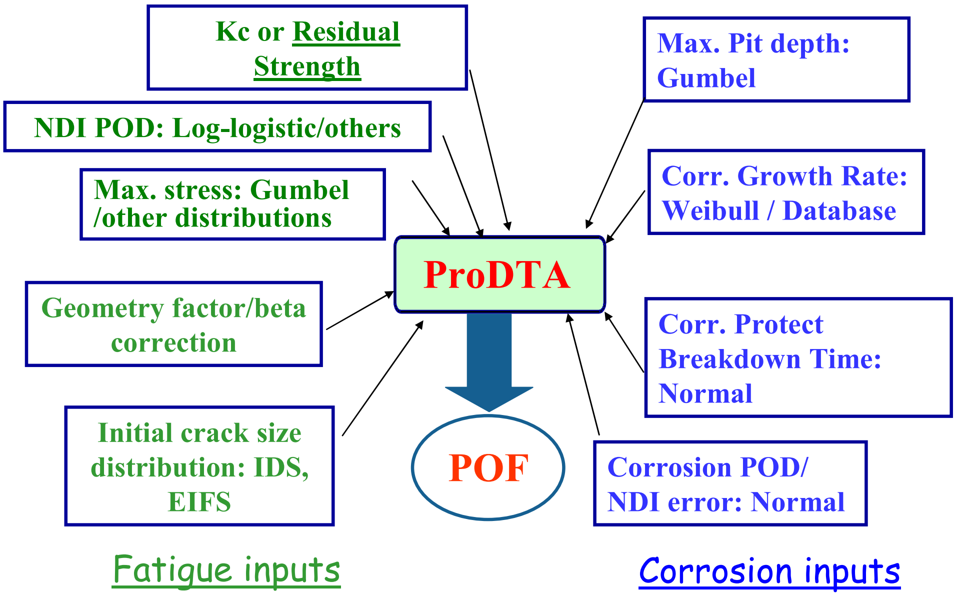



**Figure 40. Main inputs for the NRC in-house risk assessment program, ProDTA.**

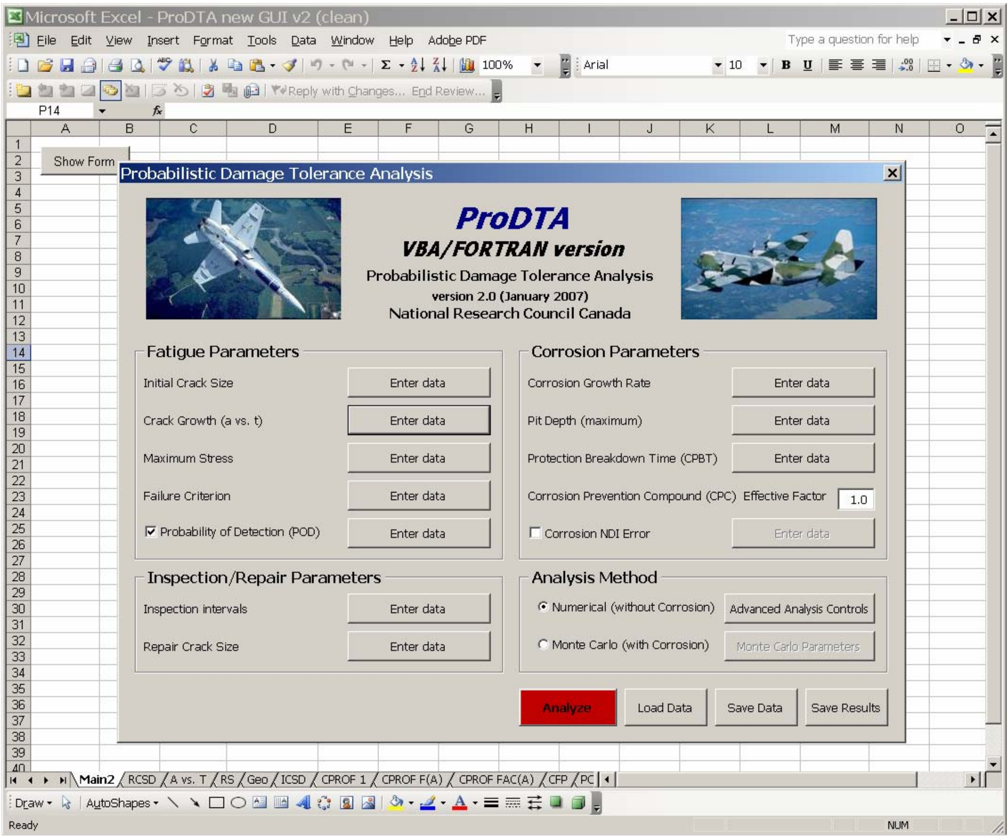



**Figure 41. A prototype GUI of ProDTA embedded in MS Excel spreadsheet.**

**Probabilistic Modeling of Fatigue Related Microstructural Parameters in Aluminum Alloys**

M. Liao, NRC-IAR-SMPL

The National Research Council Canada (NRC) and other organizations are developing the Holistic Structural Integrity Process (HOLSIP) to augment and enhance traditional safe-life and damage tolerance paradigms in both the design and sustainment stages. The HOLSIP-based lifing model uses the 'initial discontinuity state' (IDS) as a starting point, which is the intrinsic as-produced or as-manufactured state of a material. Examples of IDS include constituent particles, pores, and machining marks and scratches. In previous projects, the material IDS/particle data were measured from metallographic examination on polished unclad 2024-T3 sheets, and characterized by the area, height, and width of the particles. The fatigue subset data were obtained from the particles that nucleated the dominant fatigue cracks on the fracture surfaces.

A rigorous statistical procedure, which included probability plots, was employed to determine the best-fit distributions for the material IDS/particle data and fatigue subset data. Overall the 3P Lognormal distribution was found to be the best-fit distributions for the material IDS/particle size (area, width, and height) (Figure 42), and the weighed 3P Lognormal distribution was found to provide the best fit to the right tail of the distributions. The distribution parameters were obtained for the unclad 2024-T351 0.063", 0.16", and 0.5" sheets. Using the same statistical procedure, the distributions of the crack-nucleating particle sizes (the IDS fatigue subset) were also determined, as shown in Figure 43.

Statistically the material IDS distribution is the parent distribution of the IDS fatigue subsets. For fatigue life predictions, the fatigue subset is more critical than the material IDS distribution. But the fatigue subset is affected by many intrinsic and extrinsic factors, such as material state, manufacturing, stress level, geometry, and environment, etc. It is impractical to perform different tests and carry out measurements on each fracture surface to generate the fatigue subsets for all these conditions. Some studies were carried out to investigate the relationship between the material IDS distribution and its fatigue subsets.

Initially, an extreme value theory based model was investigated to correlate the overall particle distribution with its fatigue subsets. This model concluded that the critical density of crack-nucleating particles, for different thickness of 2024 sheets, was less than 0.5 % of the density of all particles. This effort also indicated that in addition to particle size, other microstructural features need to be considered for predicting the fatigue subsets of the material IDS/particle size distributions [30,31].

A new Monte Carlo simulation system was developed to estimate the IDS fatigue subsets based on the material IDS distribution [31,32]. Qualitative fatigue criteria were established based on the physical understanding of crack nucleation and short crack growth, including the effects of particle size, grain size, orientation, as well as stress levels (Figure 44). As shown in Figure 45, the preliminary predictions were promising and better than those from the extreme value theory based model, and agreed fairly well with experimental measurements. Further investigations are proposed to improve this new simulation method.

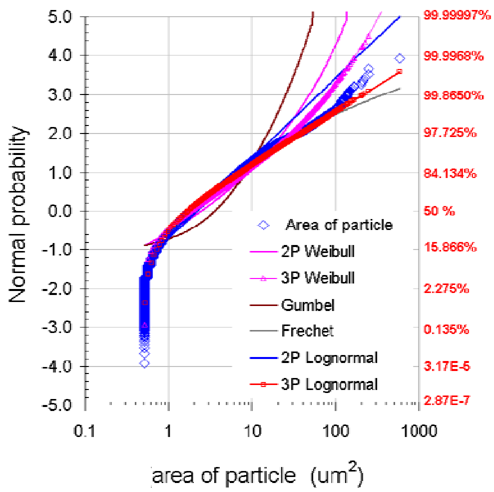



**Figure 42 Goodness-of-fit plot of distributions on Normal probability paper (e.g. unclad 2024-T351 0.063" sheet, ST plane)**

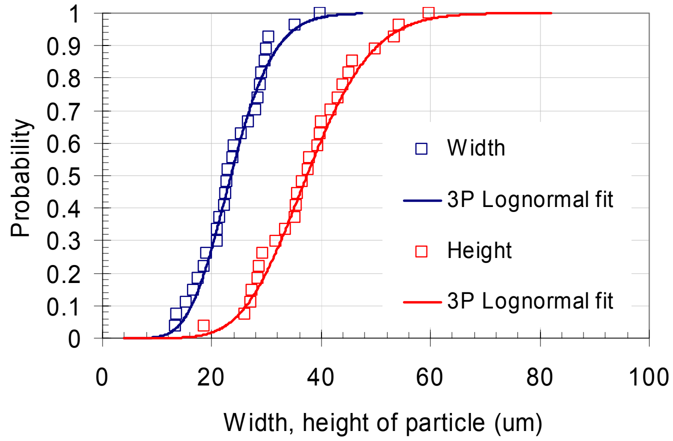



**Figure 43 IDS fatigue subsets: combined particle width & height distributions (unclad 2024-T3, 0.063, 0.16, 0.5 at 40, 44, 48 ksi, R0.05, R0.1)**

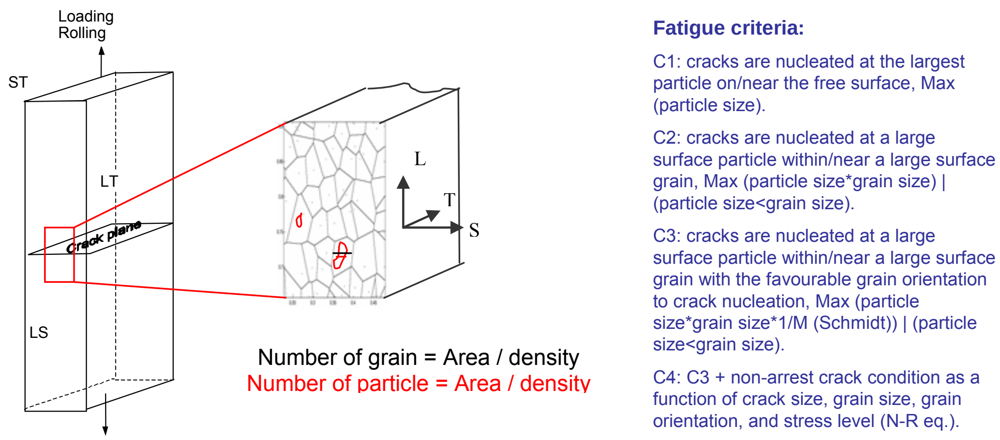



**Figure 44 Schematic particle and grain structure on fatigue critical area (LS plane)**

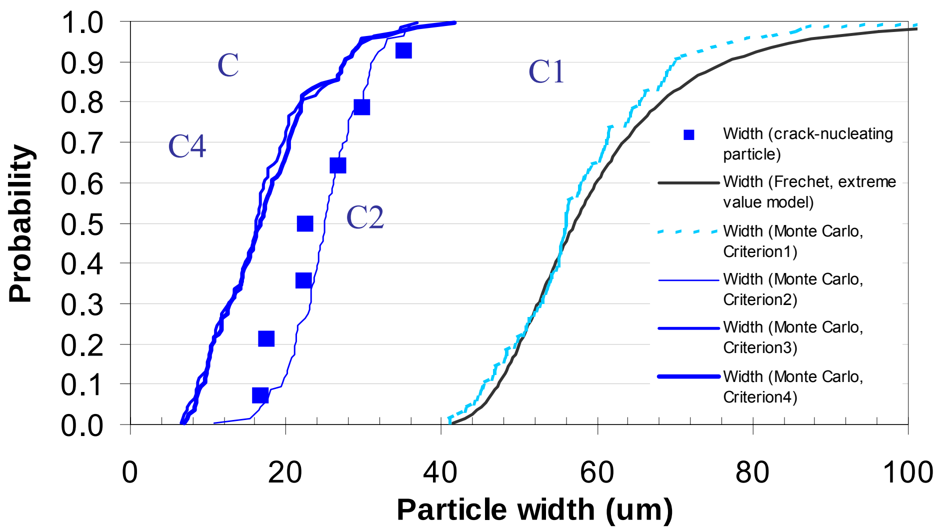



**Figure 45 IDS fatigue subsets prediction and test (e.g. 2024-T3/0.063"),$\sigma_{max}$44ksi, R0.1, similar results were obtained for 0.063", 0.16", and 0.5" at other stress levels.**

## AGING AIRCRAFT ISSUES

**Safe and Economic Management of Widespread Fatigue Damage (WFD) Using Prognostic/Diagnostic Health and Usage Monitoring**

S. Hall, Celeris Aerospace Canada Inc.

As the world aircraft fleet ages, there is increasing concern about the development of Widespread Fatigue Damage (WFD). This has resulted in regulatory agencies and Original Equipment Manufacturers (OEM) proposing and/or developing regulations and service bulletins to address WFD that from an inspection/maintenance perspective are onerous to apply, costly to implement and in some instances of questionable reliability.

While many of the concerns about WFD are well founded and should not be ignored, proposed/pending legislative structures for its management will seriously cripple vital aviation services in many parts of the world and have devastating economic and socio-economic consequences. While safety cannot and should not be compromised, the question arises as to whether there alternate, more economically viable, methods of addressing WFD?

This work advocates that a prognostic/diagnostic approach to Health and Usage Monitoring can provide at least an Equivalent Level of Safety to that intended by current WFD guidelines. Health and Usage Monitoring technology is combined with Comparative Vacuum Monitoring (CVM) technology to ascertain the actual loads to which an aircraft is being subjected while at the same time providing an early indication of the development of any WFD in critical areas. The data that is obtained is subject to ongoing analytical/experimental evaluation, such that the prognostic capabilities of the health and usage monitoring data combined with the diagnostic crack-detection capabilities of the CVM technology provide a viable, safe and economic approach to addressing WFD concerns [33].

**CF-18 Aircraft Structural Integrity Program (ASIP)**

J. Dubuc, L-3 Communications - MAS (Canada)

As part of the System Engineering Support Contract (SESC) contract, L-3 MAS conducts a full-fledged ASIP program on the CF-18 fleet on behalf of the CF. Most of the recent and current efforts are dedicated towards interpretation of the IFOSTP test series and fleet findings in order to define update the Structural Maintenance Program (SMP) of the aircraft, more specifically, the so-called ALEX Program (reported elsewhere in this report).

L3 MAS is currently conducting the definition and development of the third and final phase of the ALEX program, designated as CP3, using mainly full scale fatigue test results from the IFOSTP FT55 Center Fuselage, FT245 Wing and FT46 Aft Fuselage tests. Considering the CF baseline Operational Spectrum (BOS) and the use of higher scatter factors, ALEX represents a 50% to 100% extension in life over the original design, depending on areas of the aircraft. The main objective of these activities is to identify all structural deficiencies that could affect the CF fleet and define corrective measures that will allow the fleet to meet its targeted ELE (Estimated Life Expectancy). In this aim, IFOSTP results are complemented with Aircraft Sampling Inspections (ASI) performed on CF aircraft and similar data provide from the US Navy.

In the past, the flight control surfaces of the CF-18 had never been considered systematically as part of ASIP effort for CP1 and CP2. Specifically, the usage of these components and associated fatigue life had never been assessed for a Canadian usage and environment. Several of these components are now approaching the 6000 airframe hours for which they were originally designed and so there is an immediate requirement to perform an assessment on these components as part of the CP3 exercise, including the possible corrosion and degradation of bonded joints and honeycomb sandwich sub-assemblies. Again, ASI plays a major role in this program.

Given the scope and complexity of the ASIP program, L3 MAS has adopted a stage gate approach to this task, APEXE and has developed a series of unique engineering tools and processes:

Structural Information System (SIS)
Analytical Tools, namely the Spectrum Generation software *Specgen*
Risk Assessment process

See previous ICAF briefings from 2003 and 2005 for details.

A recent upgrade to *Specgen* is the *Hole Analysis* module. As is well known, the majority of fatigue cracks develop at fastener holes that are loaded both by thru-stresses, sometimes by-axial in nature, and bearing stresses which, in-turn are a function of various loading actions. *Hole analysis* is designed to manage these multiple sources of loading and the geometrical features of the joint and resolve them with appropriate stress concentration solutions to yield the Kt-sigma at any given point around the hole. This unique capability ensures that the actual hot-spot around the hole is found and enables the user to perform his work much more efficiently. Figure 46 shows some of the screens associated of the *Specgen Hole Analysis* module.

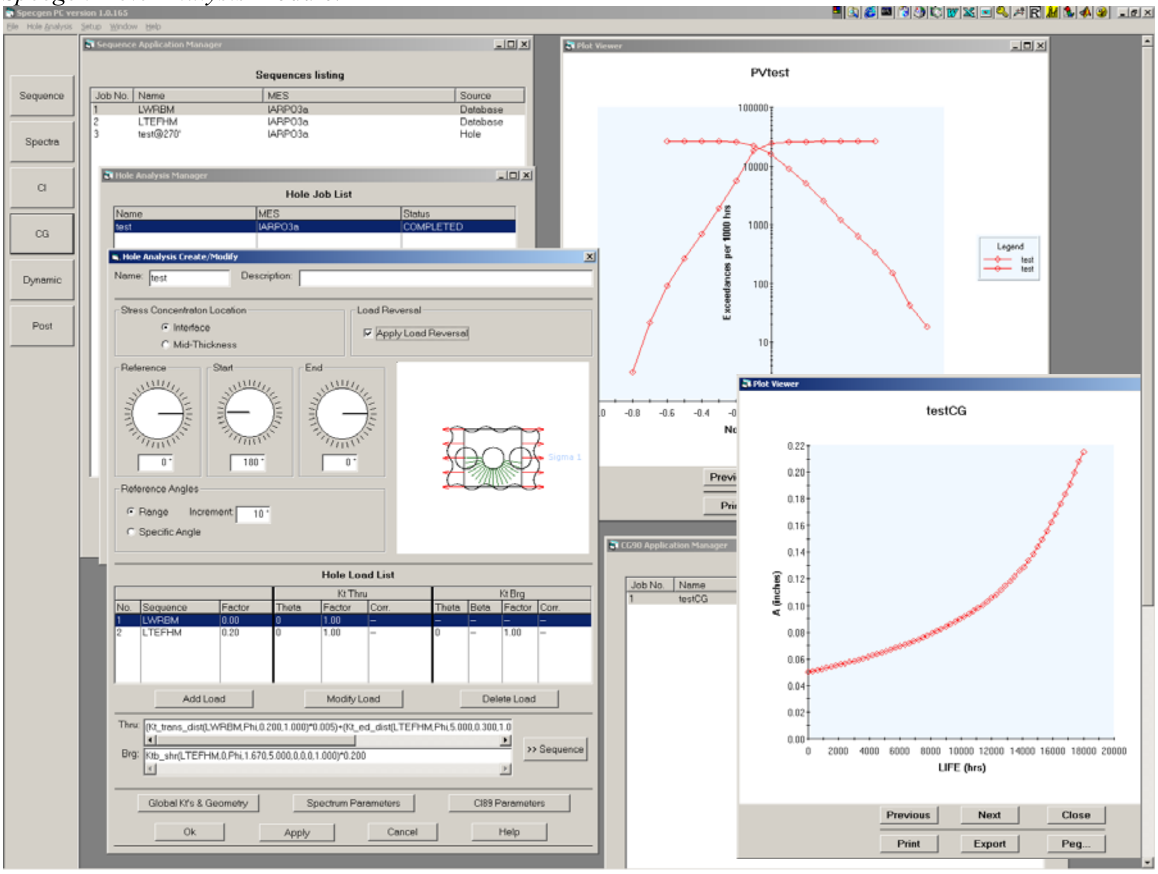


**Figure 46 Typical Screens from Specgen and Hole Analysis**

Another key aspect of the ASIP program is the Durability and Damage Tolerance (DADT) plan and associated fatigue and crack growth methodologies. Recently, work has been completed in order to validate and tune the crack initiation (CI) and crack growth (CG) tools used in the life assessment of the structural elements mainly for structures submitted to a high level of dynamic loading. Other methodologies developed include guidelines for DADT assessment of composite structures, creation of a corrosion control plan, non destructive techniques (NDT) related activities, creation of statistical tools for fleet management and others. To provide data input to the above methodologies, several coupon test programs were initiated. For example, a program assessing the effect of spectrum truncation on the fatigue life under several CF-18 interface load spectra was completed. Other coupon test programs include a Canadian program as well as participation to an international coupon initiative to assess the effect of surface renewal operations on extending the remaining fatigue life.

### CF-18 Aircraft Life Extension (ALEX) Program

J. Dubuc, L-3 Communications - MAS (Canada)

ALEX is a three-phase structural program aimed at progressively upgrading the fatigue resistance of the airframe and repairing existing cracks. As of the end of 2006, the ALEX CP2 program is well into its production phase at L-3 MAS depot level facility in Mirabel, Quebec. The CP2 package comprises over 70 modifications and inspections and encompasses various locations in the Center Fuselage, Wing and Aft Fuselage.

The majority of items under the ALEX program consist of structural modifications that involve renewal of holes (oversize) and surfaces (blend). Often, these modifications include the application of life improvement techniques such as cold working and shot peening. Some modifications are more involved and include stress reduction features such as reinforcing fittings or doublers. Analytical and experimental techniques are used to demonstrate qualification of each modification item against the aircraft specification.

Among the most significant modification items is the dorsal longeron program for which many different solutions had to be provided depending on the state of the damage at the different critical locations (Figure 47). First, field inspections were developed to identify the most critical aircraft in need of repairs. Preventative modifications are applied to aircraft without any significant damage. One of these modifications entails the removal of a relative thin layer of material (location A) and the combined use of cold working and interference fit fasteners (location D) and coupon testing is currently on-going to substantiate these innovative processes. The aircraft exhibiting severe cracking are being repaired by means of in-situ reprofiling of the longeron and installation of a repair strap.

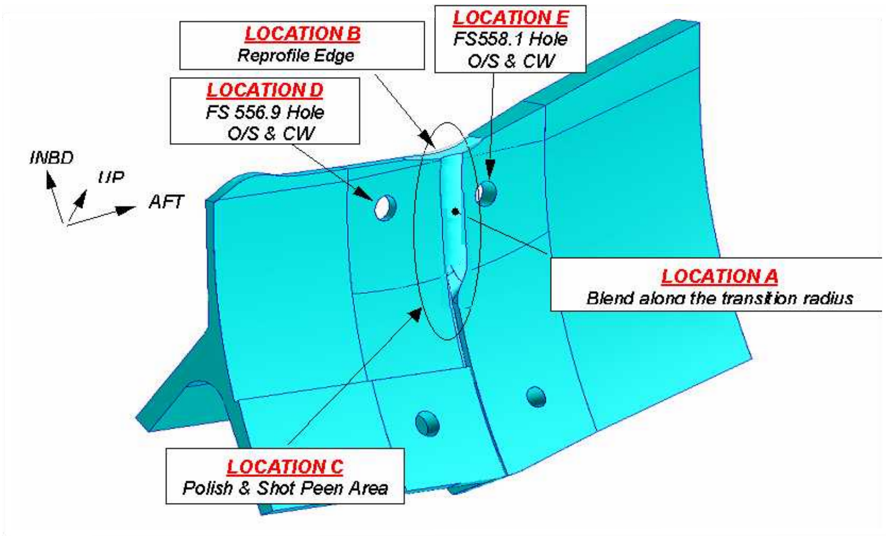



**Figure 47 Dorsal Longeron Critical Locations**

**Royal Australian Air Force F/A-18 Structural Refurbishment Program**

J. Dubuc, L-3 Communications - MAS (Canada)

L-3 MAS is currently under contract with the Royal Australian Air Force (RAAF) to perform all the NRE development and prototyping activities for the RAAF F/A-18 Structural Refurbishment Program (SRP), a program which is similar to ALEX but tailored to the constraints and requirements of the RAAF. The RAAF SRP is essentially a two-phase approach aimed at upgrading the fatigue limit of the airframe and repairing existing defects. Fleet-wide installations are conducted by the Hornet Industry Coalition (HIC), led by Boeing Australia.

Several of the structural solutions responding to RAAF fleet requirements are simple adaptations of CF ALEX modifications. Others are optimised redesigns that take full advantage of the fact that the RAAF has opted for the so-called Center Barrel Replacement (CBR) program to bring their aircraft to the targeted design life. In fact the RAAF is to perform CBR on all aircraft that are planned to fly up to a Fatigue Life Expanded Index (FLEI) of 1, i.e. 6000 hours of baseline usage.

On the wing, L-3 MAS has adapted an ALEX modification consisting of locally removing fatigue damaged material on the outer wing front spar (Figure 48 and Figure 49). The adaptation was to replace a 5-axis milling machine by a hand tool to achieve the desired reprofilling and thus maintain the fatigue improvement of the retrofit. There are several challenges to this task, related to the control of material removal due mainly to the component local geometry varying significantly from one aircraft to another. The team of technician and engineers finally came over that issue after identifying the critical dimensions and providing new measuring features to the tool kit.

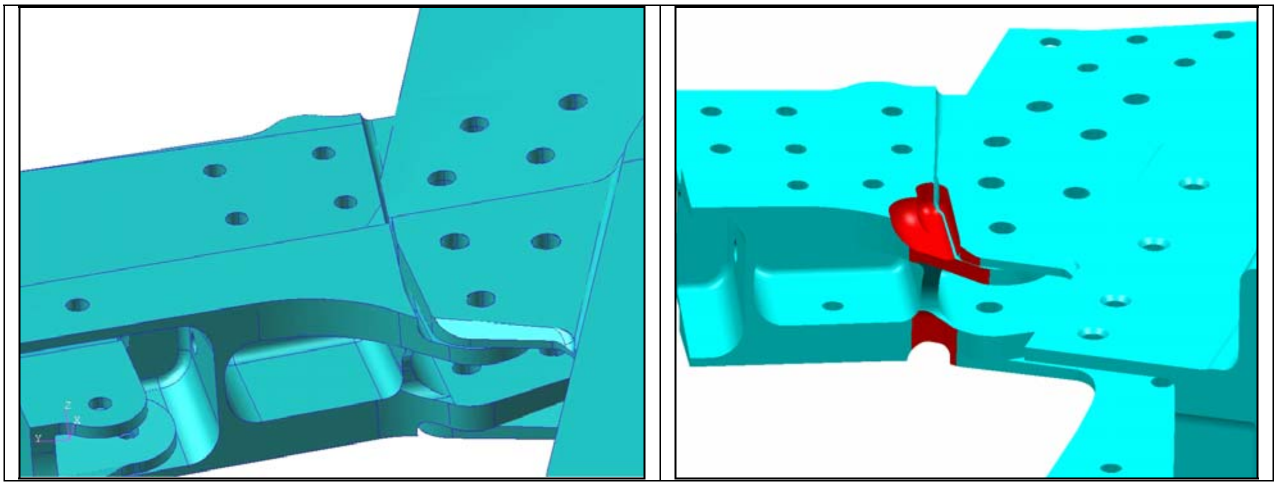


**Figure 48 Outer Wing Front Spar – Pre-Mod & Post-Mod**

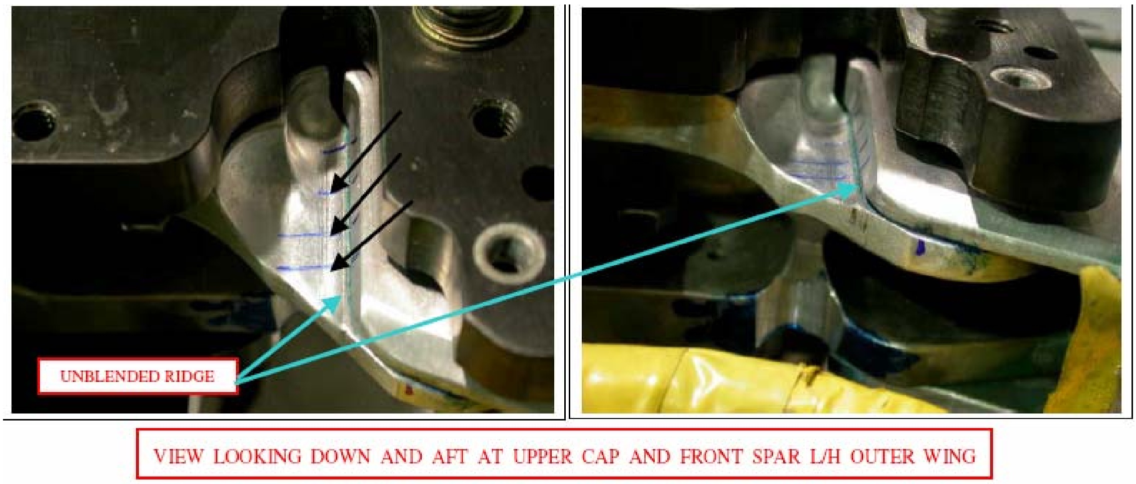



**Figure 49 Outer Wing Front Spar – Post-Mod**

**Articulated Robots for Structural Modifications**

J. Dubuc, L-3 Communications - MAS (Canada)

A robotic shot peening system has been developed at L-3 MAS in order to provide optimum peening conditions in hard to reach locations (ref. ICAF 2003). More recent Research and Development (R&D) activities at L-3 MAS have focused mainly on expanding the capabilities of the robotic system to perform blending and surface renewal on areas already submitted to fatigue damage (ref. ICAF 2005).

A robotic material removal modification is currently under development on the aft shear tie of the inner wings (Figure 50). A deburring ball nose cutter is used to make blend in confined areas made with curved surfaces. A custom robotic trajectory generation software was developed based on a matrix of probing points defined in Catia. The application is then capable of removing a constant layer of material based on the existing part. The robot can remove a thin layer of material within the required tolerance of 0.003” to 0.006”.


**Figure 50** Robotic Material Removal Modification on the Inner Wing


**Figure 51** Robotic Material Removal Modification on the Inboard Leading Edge Flap

Another R&D project involves a stronger robot to replicate a material removal application currently done by a 5-axis CNC machine (refer to figure 6). A milling approach using 6 different cutters is used with higher RPM to minimise vibrations of the robot and achieve blends on the Inboard Leading Edge Flaps within the tolerance of +/-0.0025". A metrology arm was used to validate the dimensions and to calibrate the robotic system. Once calibrated, we can rely on the repeatability of the robot to make precise blends. The robotic blending process was tested on representative parts made in plaster material and on real part made in 7050 aluminium material. A user-friendly interface guides the operator during the entire modification.

This robotic system costs less than a 5-axis CNC machine. The productivity is improved because of the automation. It requires less machining skills and experience to make the modification.

**CF-18 Fatigue Life Management Program (FLMP)**

J. Dubuc, L-3 Communications - MAS (Canada)

Aircraft fatigue management is a key requirement of the structural integrity policy implemented by the Canadian Forces (CF) in order to ensure continued safety of flights and operational readiness of its fleets. The Fatigue Life Management Program (FLMP) encompasses all the elements needed to fulfil this requirement for the CF-18 fleet. This includes the on-aircraft Maintenance Signal Data Recording Set (MSDRS) and Structural Life Monitoring Program (SLMP) from which, flight-by-flight derived, strain sensor based, fatigue life indices are computed. See previous ICAF summaries for further details.

Major improvements have been made to SLMP in the last year, most notably to the gauge calibration and fill-in methodologies:

The strain-based approach is dependent on proper calibration of the Wing Root (WR) gauge as the strain readings are known to drift with time due to local effects at wing-fuselage assembly. Calibration flights for this purpose are performed at set intervals and following any aircraft maintenance action on the WR gauge. However, this approach is not efficient as it was shown that the gauge can drift significantly between calibration flights and hence leading to inaccuracies in the fatigue lives calculated.

Major efforts in the last year have been made to develop a methodology to eliminate the requirements for calibration flights and to continuously determine the gauge drift factor for the “next” flight flown. Improvements have also been made to the accuracy of the predicted gauge strain response itself, using available flight test programs with known calibrated loads. The Parametric Loads Formulation (PLF) sets of equations (based solely on recorded flight parameters) used to estimate the Wing Root strains/drift and also to fill-in the data when the gauge is defective were refined & updated to ensure more accurate calibration of the gauge readings/predicted strain response.

In the past the missing data portion, either due to faulty tapes or tapes not installed on aircraft, was filled in using the aircraft cumulative fatigue rate. Accuracy of this methodology cannot be properly measured. Hence, the missing data fill-in technique was updated to use the average rate of the actual missing missions instead. This was possible since all sorties are entered electronically in a database and a reconciliation can thus be made to the recorded flights coming from the MSDRS of the aircraft.

The results of these improvements to the CF-18 fatigue tracking system have lead to a major reprocessing of all fleet data in order to ensure to have the most accurate picture of the current fatigue life of the fleet.

**C130 Center Wing teardown**

M. Bunn, ATESS/NDTC

The ATESS Nondestructive Testing Center was tasked to provide a feasibility report and teardown plan. The plan identifies the costs and timelines associated with a complete in-situ inspection, teardown, post-teardown inspection and report for a CC-130E Hercules center wing. It is important to note that the E-model Hercules aircraft are fitted with H-model center wings. The center wing teardown and consequent inspection results will aid DAES' engineering staff in determining the risk of structural failure by widespread fatigue damage (WFD) as well as potentially extend the economic life of the CC-130 E-Model fleet to the year 2010 and beyond. A further benefit will be targeting areas of potential failures that may arise in the CF's H-model Hercules aircraft in the future. The NDT inspections carried out by ATESS will identify life-limiting factors such as severity of corrosion, fatigue-related defects, excessive wear, etc.

This report will determine the requirements for the cleaning, stripping, complete disassembling of the wing, the NDT investigation and report of findings in both hard and electronic copies. Included in this report will be the investigation and applicability of new NDT technologies as well as validation of current center wing NDT inspection techniques and methods.

The level of effort to accomplish this task will be determined by the funds allocated, time constraints (personnel, project), availability of support personnel/equipment and the priority of other taskings taken on by ATESS.

The scope of the Project Tasking Directive was used as a partial guideline in producing this report. The inspection and teardown of the wing is a large project, therefore, a logical and methodical process has been developed to make the tasking manageable. This methodology has been developed by the requirements of future component testing of the center wing balanced with concurrent inspections, teardown activities and the application of new NDT inspection technologies. Comparing the similarity of this work to that which ATESS successfully carried out on the SLAP (P3/CP140) wing and it's comprehensive report has lead to a direction that follows a similar scope, content and style.

A 4-phased approach will be used to accomplish this project:

Phase I – Initial Visual Inspections;
Phase II – In-situ NDT Inspections;
Phase III – Disassembly; and
Phase IV – Post Teardown Inspections, Fractograhic Analysis and Final Report.

At a time closer to the actual wing change-out the sectioning of the center wing will be determined. Below are 3 options from which a selection can be made:

Option # 1 - crate and ship the center wing to ATESS as one unit;
Option # 2 - make cuts chordwise at the outer cutlines, crate and ship to ATESS as three units; or
Option # 3 - make cuts chordwise at all four cutlines, crate and ship to ATESS as five units. (Recommended)

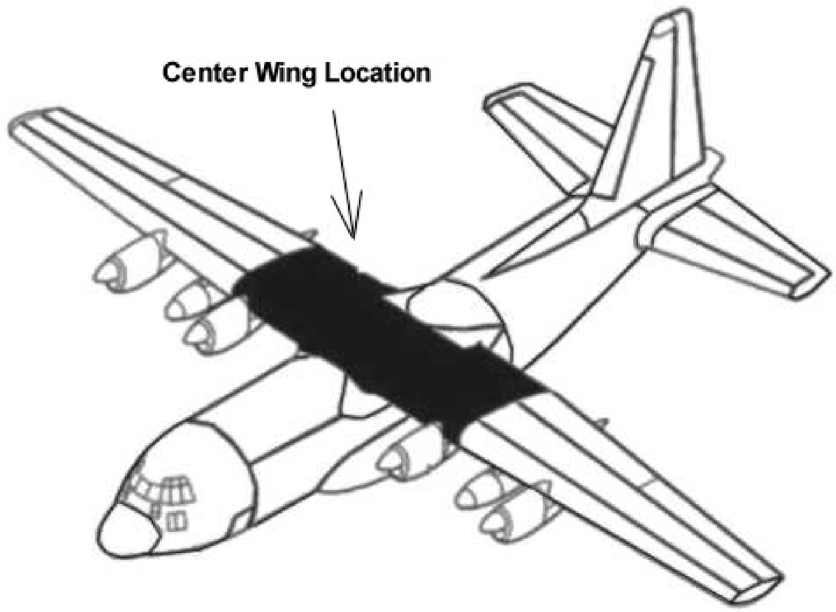



**Figure 52 C 130 Center wing**

In each of the above options Herc Solutions will be responsible for center wing removal, crating and shipping of the wing section(s) to ATESS. The cutting of the center wing into sections will be carried out at the following Center Wing Stations (CWS):

CWS 133.0 Left*;
CWS 49.0 Left;
CWS 49.0 Right; and
CWS 133.0 Right*.

*NOTE -The shortened stringers located close to the lower access door at WS 133 L/R are not to be cut through during initial wing sectioning. The ends of these stringers shall have fasteners removed and the wing planks will be gently pried from the intact stringers. These particular stringers are to remain intact and have a potential risk of cracking or being cracked.

The primary reason for sectioning the wing in this manner is work manageability. Having the wing in separate pieces allows concurrent teardown and inspection activities to take place when the center wing is at SPAR-Trenton. As well, this approach will free up wing sections for the application of new NDT technologies. The location of the cutlines were determined through telecons with Mr. Terence Cheung at DAES 4-2-3C1 to ensure specific areas of concern in the primary center wing structures were kept relatively intact for future testing. For example, the Lower Front Spar between CWS 140 and CWS 210.

It is important to note that when cutting the wing at the recommended cut lines that no primary structural holes be cut through. The wing as shown in Figure 52 and Figure 53 is to be crated and shipped to ATESS.

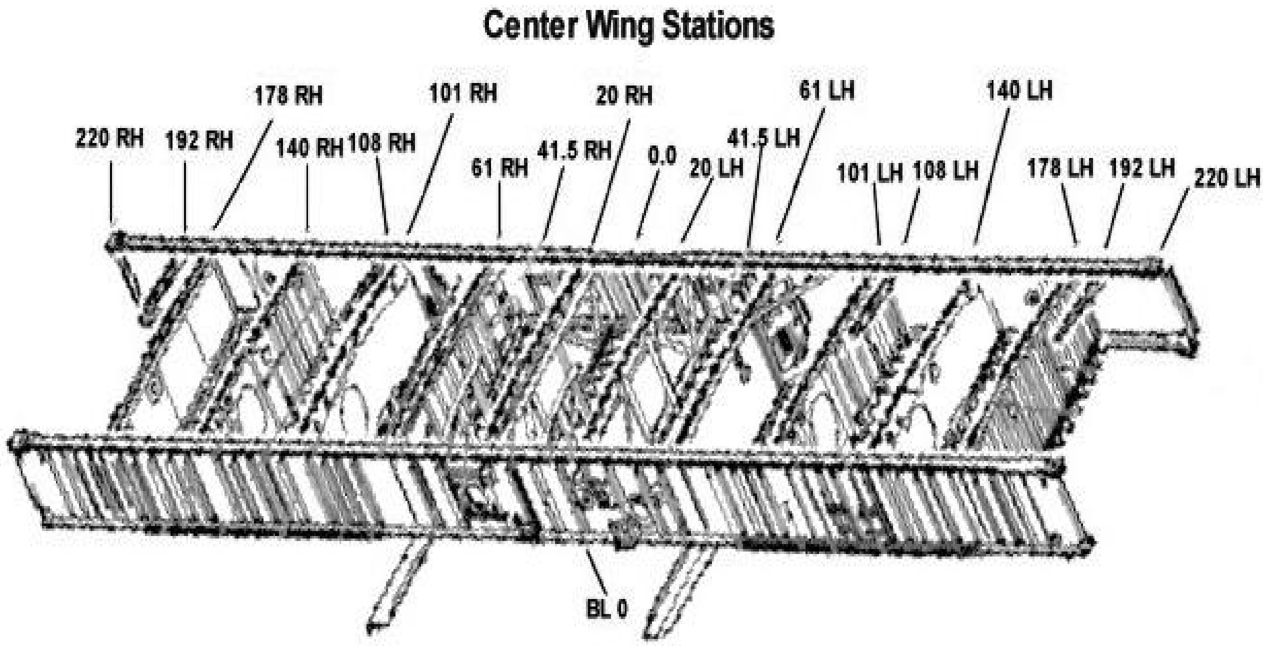



**Figure 53 Center Wing stations.**

**NRC Support to P-3C Orion / CP140 Aurora Service Life Assessment Program (SLAP)**

M. Yanishevsky, NRC-IAR-SMPL

The testing for the project to support the collaborative program between the United States Navy (USN), the Canadian Forces (CF), the Royal Australian Air Force (RAAF) and the Royal Netherlands Navy (RNLN) on the P-3C Orion / CP140 Aurora Service Life Assessment Program was completed in 2006. A test article representing the lower front spar/web/lower wing skin at Wing Station WS167 was tested to determine the potential benefit that cold expansion of fatigue critical holes could provide for the Canadian fleet of Aurora aircraft (Figure 54).

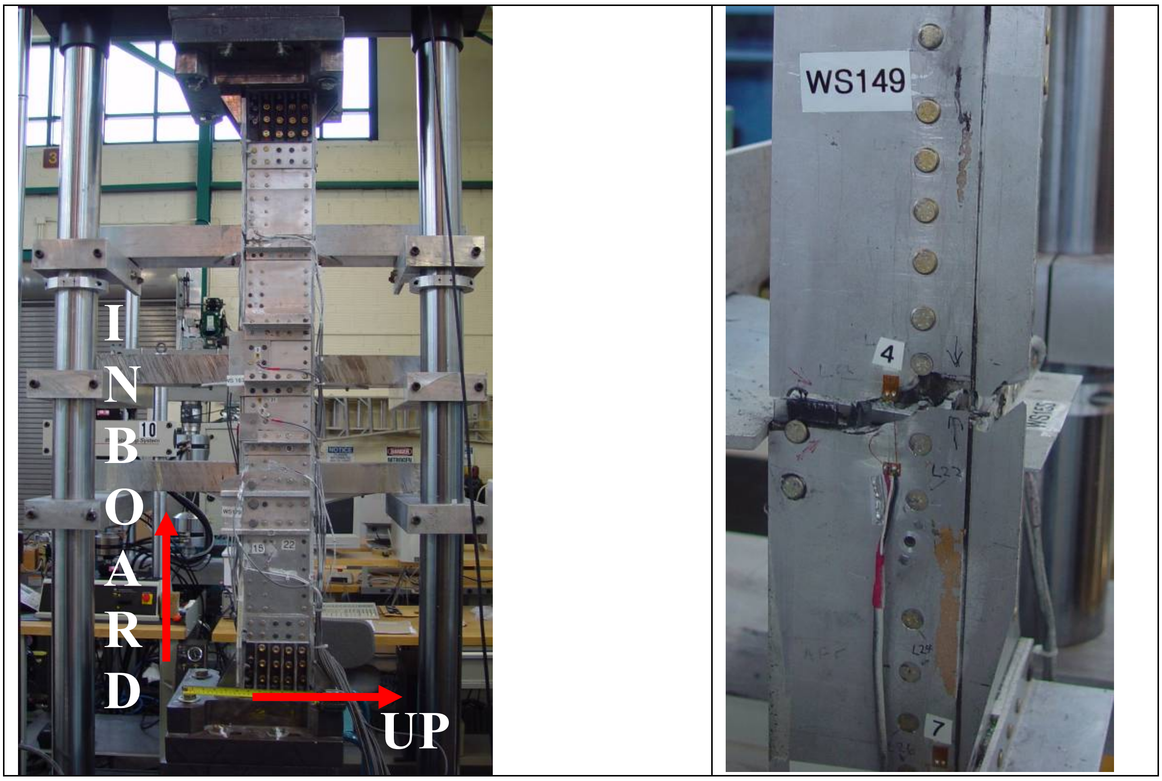



**Figure 54 The WS167 test article: aft view (left) and the failure in the lower wing skin (right).**

The test article was pre-fatigued using a representative spectrum to15,800 Simulated Flight Hours (SFH) to create a damage state that would be representative of when hole cold expansion of critical fastener holes could be accomplished fleetwide. Two web hole cracks were found that were subsequently repaired; the first by cold expanding the hole to the first oversize, and the second, being of substantially larger size required cold expansion of the hole with special bushing installation to enable joining of the affected structures using the original nominal size fastener. The test article was then spectrum fatigue tested to catastrophic failure. The two web cracks that were repaired and cold expanded, did not redevelop. First crack indications were found using eddy current and ultrasonic nondestructive testing techniques after 45,000 SFH of additional spectrum fatigue testing following repair (total Simulated Flight Hours on the component = 60,800 (45,000 + 15,800)). A total of seven other cracks (four from web holes and 3 from lower wing skin holes), some ending up quite large, developed and were found prior to catastrophic failure of the component at 62,500 SFH following repair (total hours on the component = 78,300 (62,500 + 15,800)) at a final failure load of 59,711 lbf (N.B. max spectrum load was 63,440 lbf). Post test teardown inspection of the component using bolt hole eddy current probes revealed several previously undetected cracks - 8 additional in the web, 6 additional in the lower wing plank cracks and 3 additional in the spar cap for a total of 24 small size cracks, missed by surface eddy current and ultrasonic inspections. Compared with the original test which developed its first cracks at 18,200 SFH and failed catastrophically at 44,500 SFH with 50 cracks found at teardown this test demonstrated that cold hole expansion was an effective fatigue enhancement strategy for the CP140 Aurora aircraft to help maintain structural integrity and safety of flight for this fleet until new wings become available starting in 2013 (Figure 55).

Additional tests were conducted to assess the performance of Fatigue Technology Inc (FTI) Forcetec rivetless nut plates to repair the main hole on coupons with large edge margins which determined that Forcetec rivetless nut plates are an excellent repair scheme for fatigue sensitive standard riveted Dome Nut design. Life improvements in excess of 15 times were demonstrated; failures occurred in the base material well away from the design detail under test (Figure 56 and Figure 57).

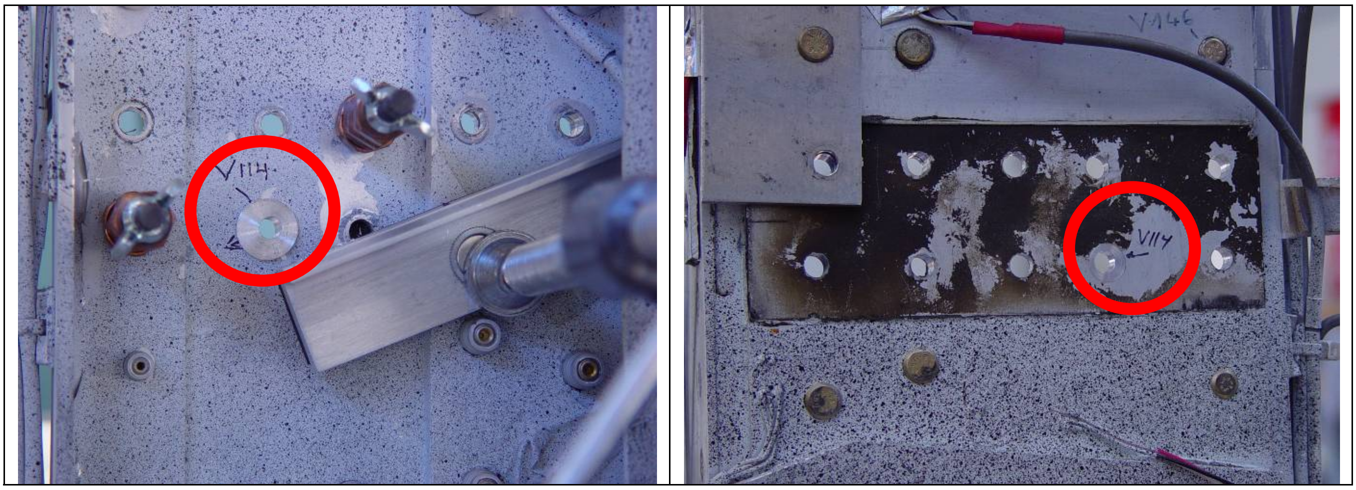



**Figure 55 Repair of hole V114 by offset oversizing followed by hole cold expansion and shrink fit bushing repair (no new cracks redeveloped in this location).**

**Figure 56 Large edge margin coupons were assess the fatigue performance of dome nut and satellite rivet holes, versus a pre-cracked fastener hole with cold expanded FTI Forcetec collar.**

**Figure 57 Two Forcetec Rivetless Nut plates that were successfully tested in the WS167 component spectrum fatigue test.**

**Evaluation of 2014-T6 aluminum alloy wing spars form retired service exposed and unexposed CT-133 airframes.**

D.L. DuQuesnay, Mechanical Engineering Department, RMC

The tensile and fatigue properties of 2014-T6 aluminum alloy taken from extruded wing spars of a retired CT-133 airframe that was in service since the 1950s were measured in the laboratory [34]. The properties of newer spars made of 2014-T6 for this airframe that were not service-exposed were also measured for comparison. Tests were performed on coupons machined from thick (50mm) to thin (5mm) sections of the spars. In general, although there was an influence of extrusion thickness on the measured properties, the mechanical and fatigue properties exceeded Military Handbook 5G specifications for 2014-T6. However, some significant differences were noted between the tensile properties of service-exposed material and the unexposed material. Fracture toughness tests revealed pre-cracking patterns that showed a pronounced influence of residual stresses on the development of asymmetric crack fronts in very thick extrusions. The fatigue crack growth rates in the Paris region measured on service-exposed material at several stress ratios (R=0.01 to R=0.63) were generally higher than for the unexposed material although threshold stress intensity factors were about the same for both. The material properties were used to predict the fatigue behaviour of a service-exposed spar section subjected to a simulated flight-loading spectrum in the laboratory.

**The Development of Bonded Repair Guidelines for Stress Corrosion Cracking Using Residual Stress Measurement Techniques**

M. Yanishevsky, NRC-IAR-SMPL

Proto Manufacturing Ltd, Martec Ltd and NRC are participating in a collaborative program to develop bonded repair guidelines for stress corrosion cracking (SCC) using residual stress measurement techniques. This program was jointly funded by the USAF/AFMC/AFRL and the Proto contractor team. The state of candidate structural components on C-141 Starlifter aircraft were assessed using X-ray residual stress (Figure 58 and Figure 59), hardness and conductivity measurements in the laboratory and at several USAF Air Force Bases known to be susceptible to SCC. Parametric studies of the component were carried out by Martec using finite element analysis in order to determine whether the thermal strain mismatch during bonding and the constraints from the surrounding structure would generate compressive residual stresses under the patch that could be used to further alleviate the SCC problem in the web / flange transition area. A Pilot Test Program was conducted by NRC to determine whether SCC growth would continue once the corrosive environment was removed. This program was aimed at establishing whether there were grounds to pursue extended research on a cleaning process to decouple the synergy of load and environment. Round Al 7075-T6511 SCC were loaded to two levels of pre-stress and then exposed to an alternate immersion tank containing a 3.5% NaCl solution for two days. After exposure and drying, break load residual strength tests were conducted and the results showed an increase in strength when compared to similar data from a previous test program. Fractographic analysis with SEM determined that the progression of SCC appeared to have been arrested by the cleaning, drying and subsequent environment isolation processes. Proto Manufacturing assessed several additional aircraft components (CC130 sloping longerons, C141 FS 998 frame hubs and web/flange transition areas) from the NRC-IAR Specimen Library, which have historically been known to exhibit SCC susceptibility so that they could characterize and map the residual stresses in the vulnerable areas. The NRC also manufactured several four-point bend specimens from extruded material and aircraft components for Proto to experimentally establish X-ray Elastic Constants for a more accurate determination of the residual stress. By taking into account that the tensile residual stresses measured on the SCC coupons are higher in the centre of the extrusion, the observed reduction in applied tensile strength for this area was expected. The work continued with a highly detailed metallographic examination of grain flow due to forging of the FS 998 component was conducted to determine prime locations with end grain exposure which would have a high propensity for the development of SCC cracks. It was determined that following forging, several sections underwent heavy machining which contained such exposed end grains. This information was used in addition to the X-ray Diffraction residual stress measurements to design an appropriate composite patch repair for the area. To reduce the potentially detrimental effects of post curing tensile residual stresses in the surrounding areas of the bonded patch, the concept of using a second patch made of glass fibre reinforced epoxy was conceived to create a physical barrier to environment to preclude possible development of new SCC thermally induced tensile residual stress areas near the bonded patch repair site. This project is in its last stages with two guidelines being developed, the first for X-ray Diffraction measurement for SCC scenarios and the second for developing bonded repair schemes to restore strength and stiffness without creating further deleterious effects of the repair on future long term structural performance. Efforts are being made by the Proto team to find appropriate commercial and military follow-on work to apply and demonstrate the newly established residual stress measurement, analytical patch design and patch bonding process development capabilities to address this incipient problem on environmentally susceptible aircraft components [35,36,37,38].

Figure 58  Measuring residual stress on a C-141 998 frame pocket with XRD.

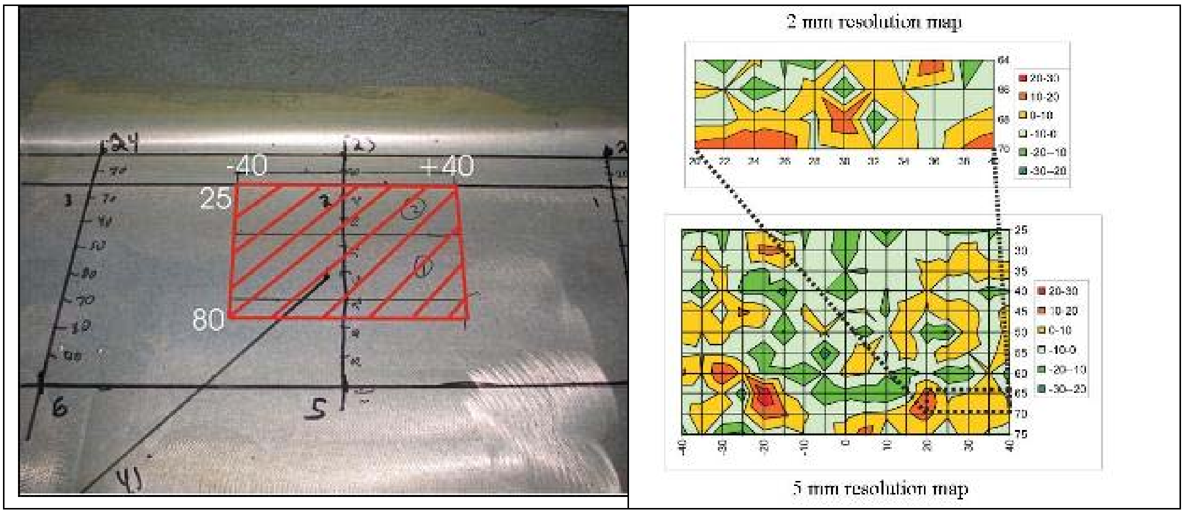



Figure 59  Residual-stress map of C-141 998 Frame pocket.

**Corrosion-Fatigue behaviour of aluminum, HSLA steel and stainless steel under periodic overload and variable amplitude loading in simulated seawater**
D.L. DuQuesnay, Mechanical Engineering Department, RMC

The corrosion-fatigue behaviour of 7075-T6 Aluminum Alloy and high strength low alloy steel (CSA G40.16M grade 400), and a ASTM 316 L stainless steel subjected to periodic overloads was examined.  The aluminum alloy is typically used in aerospace structural components such as the wing spars of aircraft.

Axial fatigue specimens were subjected to a loading spectrum that consisted of a periodic overload of yield magnitude followed by a number of smaller cycles: two hundred cycles in aluminum and fifty cycles in steel.  The specimens were fatigue tested while they were fully immersed in an aerated and re-circulated 3.5%-wt NaCl simulated seawater solution.  A separate set of fatigue tests was also performed on the aluminum in a salt-fog environment.

The results for the corrosion-fatigue testing were compared to data obtained for the same overload spectrum applied in laboratory air.  A damage analysis showed that the presence of the corrosive environment accelerated the damage accumulation rate to a greater extent than that observed in air.  There was a drastic reduction in the fatigue strength of the material when it was simultaneously subjected to overloads and corrosive environment when compared to cycling in air.

The reduced fatigue-life of these materials was due to a combination of effects: premature crack initiation due to pitting corrosion on the surface of the specimens; anodic dissolution at the crack tip; and hydrogen embrittlement. For practical purposes, the endurance-limit of both of the materials disappears under these conditions.

The fatigue behaviour of a high strength micro-alloyed carbon steel and an ASTM 316 stainless steel were evaluated in air and in simulated seawater environments. Axial dog bone specimens were subjected to both constant amplitude fully reversed fatigue loading as well as periodic overloading spectra to evaluate and compare the fatigue behaviour of the two materials under simulated service conditions. It was observed that the interaction between the overloads and the corrosion accelerated the fatigue damage rate compared to that observed for either corrosion alone or overloads alone for both materials, but the effect was severe for the micro-alloyed steel and only slight for the stainless steel. In the presence of overloads and saltwater, all small cycles in the loading spectrum appear to contribute to fatigue damage in the micro-alloyed steel, i.e., there is no observed fatigue limit under these conditions. Examination of the fatigued specimens generally showed that cracks initiated from corrosion pits in the micro-alloyed steel; a damage analysis and supplementary experiments reveal that the occurrence of pitting alone does not account for the observed increase in damage accumulation rates and that the fatigue mechanisms include both accelerated initiation from pits and accelerated crack growth rates due to corrosion-stress interactions. Effective stress-life curves were generated for estimating the fatigue life of components subjected to variable amplitude loading conditions in a marine environment. [39, 40]

## JOINING TECHNIQUES

**Fatigue behaviour of aircraft Al-epoxy-Al lap joints: effects of cladding layer, tapered ends, patch stiffness, and spectrum loading**

D.L. DuQuesnay, Mechanical Engineering Department, RMC

The fatigue behaviour of double lap shear specimens and centre notched specimens of clad 2024-T351 aluminum alloy patched with adhesively bonded aluminum patches was investigated. The specimens and patches were nominally 3 mm thick. The fatigue tests were performed under constant amplitude load control at a stress ratio of R=0. The specimen configurations and loading pattern were intended to simulate a repair of skin material in an aircraft fuselage. The maximum stress in the substrate was kept below the unnotched fatigue limit of the material. It was observed that the FM73 adhesive joint/patch could initiate cracks in the cladding layer that propagated to cause substrate failure even at relatively low stresses. The results were similar to those observed previously in single lap shear specimens of the same materials. The substrate failures were observed to occur both at the edge of the patches and at the repaired site, which in this case was a circular hole in the substrate. The effect of removing the cladding layer on fatigue performance was investigated. Adhesive failure was the predominant failure mode at high stress levels, whereas the clad substrate failure was predominant at low stress levels. It was found that the removal of the cladding layer prior to adhesive bonding was always effective in preventing substrate failure. [41]

Corrosion damage, battle damage and foreign object damage in aircraft are often repaired using bonded patch repairs with or without mechanical fasteners. Adhesive bonding technology has improved such that bonded doublers are also used to stiffen and strengthen primary and secondary structure for life extension. The success of these patches depends to a great extent on the stress distributions at the adhesive interfaces. Debonding of the patches is typically assumed to be the failure mode, however substrate fatigue failures initiating in the cladding layer have been reported to occur on some laboratory coupon tests. This study examines the effect of the stress distribution at the edge of the patch on the fatigue behaviour of adhesively bonded double strap joints.

The fatigue life and failure mode of double strap joints constructed from bare and clad 2024-T3 aircraft aluminum alloy were examined experimentally and analytically. Substrate failures were found to occur at lower stress levels, while the failure mode at higher stresses was debonding of the patches by adhesive failure. Transitional failures, which were essentially adhesive failures with cladding pull-out observed on the bonded clad substrate specimen surface, occurred at stress levels between the substrate and adhesive failures. The stress/strain state at the edge of the patches was varied by changing the patch thickness, the patch modulus, and by tapering the edges of the patches. The stress and strain states in the adhesive were modelled using a non-linear finite element analysis. The analytical models revealed power law relationships between fatigue life and the peak principal strain for an adhesive failure mode, and between fatigue life and the nominal axial stress in the substrate for a substrate failure mode. At a given stress level in the specimens, the analytical fatigue life, that is the shorter of the adhesive and substrate failure modes, was in good agreement with the experimental data. This evaluation technique provides a means of determining the fatigue life and failure mode of clad aluminum bonded specimens, so that the substrate failure mode, as well as the adhesive failure mode, can be considered during patch design. [42]

The effect of variable amplitude loading on the fatigue life and failure mode of bonded double strap (DS) joints was investigated. The joints were made of 2024−T3 aluminum in either the clad or unclad (bare) condition and were bonded with FM73 adhesive. It was found that the introduction of periodic overload cycles increased the damage caused by subsequent smaller cycles. The failure mode for the bare specimens was always adhesive failure. Under constant amplitude loading, the clad specimens failed by fully adhesive failure at higher stress levels, by substrate failures at stress levels nearer to the fatigue limit and by transitional failures (mainly adhesive with damage caused to the cladding layer) at intermediate stress levels. The presence of overload cycles shifted the failure mode of the clad specimens towards fully adhesive failure. Effective-stress-range versus fatigue-life curves were developed for the clad and bare specimens using periodic overload spectra. The effective stress range vs. failure life curves were used in conjunction with a linear cumulative damage summation to predict the failure lives of double strap specimens subjected to two variable amplitude aircraft load spectra. The predicted and observed fatigue lives were in good agreement. [43]

## GAS TURBINE MATERIALS AND STRUCTURES

**T56 Series III Engine Turbine Rotor LCF Life Update**

W. Beres, NRC-IAR-SMPL

The work on T56 Series III gas turbine engine life update was initiated in 2001 and completed in 2006. The main objective of this effort was to re-evaluate low cycle fatigue crack initiation and crack propagation lives for the T56 series III turbine rotor. This project was carried out as an collaborative effort of Rolls-Royce Corporation, Indianapolis, USA; NRC-IAR, Canada; CSIR and SAAF South Africa; DSTO and RAAF, Australia; Department of National Defence, Canada; US Navy and US Air Force. The main contribution of the NRC-IAR to the project was devising and performing heated tests of two turbine spacers and two turbine discs in the NRC-IAR's spin rig facility. As of the end of 2006 spinning of both components were completed. The spin test results were applied to verify the crack initiation and crack propagation lives for these two T56 components.

**Dwell Fatigue**

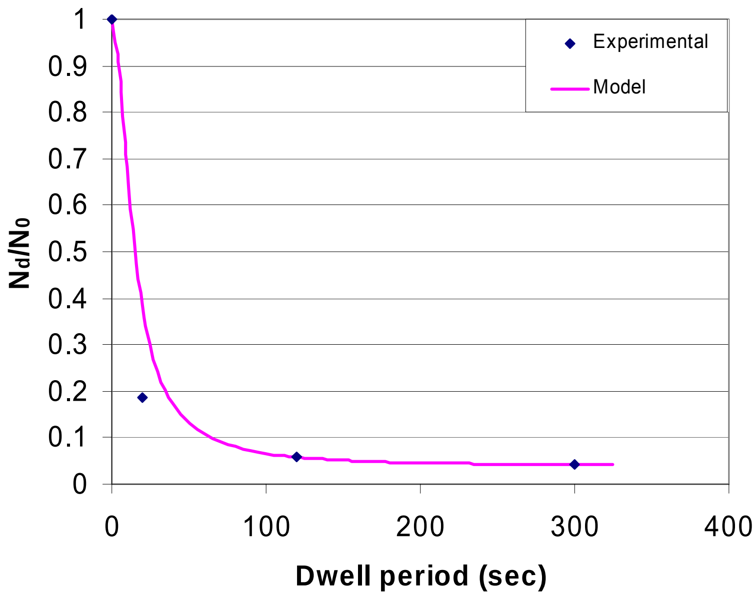
X.J. Wu, NRC-IAR-SMPL

Work has been conducted at NRC-IAR-SMPL on modeling cracks in terms of Zener-Stroh-Koehler (ZSK) dislocation pile-ups in anisotropic materials, with an aim to describe the damage generated in titanium alloys under dwell-fatigue conditions at near ambient temperatures, which has been a problem since early 1970s, first found in IMI 685, and later in almost every newer version of near $\alpha$ titanium alloys such as IMI 834 and Ti 6242. The model being developed considers the kinetics of the dislocation pile-up as a competition between dislocation glide and climb, which leads to an evolutionary equation of work hardening. The fatigue-dwell interaction is depicted as the coalescence of micro-ZSK cracks with the dominant propagating fatigue crack, resulting in a total damage accumulation rate, as

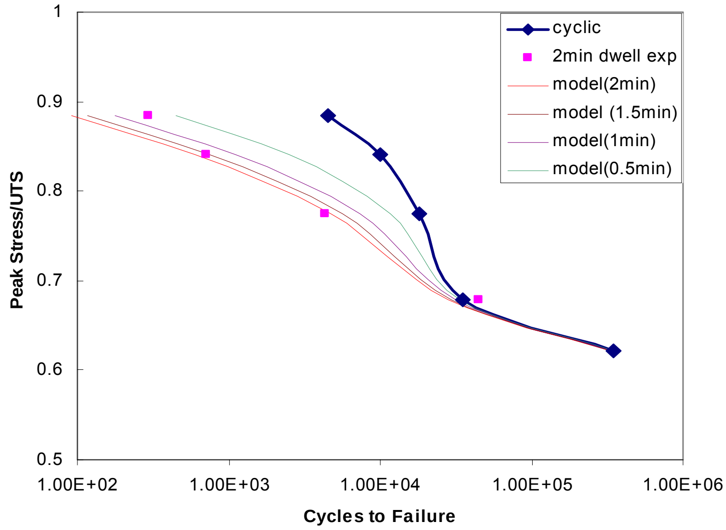
$$\frac{da}{dN} = \left(1 + \frac{l}{\lambda}\right)\left(\frac{da}{dN}\right)_f \qquad (1)$$

where $l$ is the ZSK crack length and $\lambda$ is the $\beta$ grain size.

Figure 60a illustrates the dwell sensitivity in terms of the dwell time, while Figure 60b shows the S-N behavior affected by dwell. The model description compares favorably with the experimental data published in the literature [44, 45.




(a)




(b)

**Figure 60  Comparison of the model with experimental data in the literature.**

## NONDESTRUCTIVE INSPECTION AND SENSORS

### Infrared Thermography

M. Genest and A. Fahr, NRC-IAR-SMPL

Fretting fatigue

Various NDE methods were evaluated for their potential to detect fretting damage on the faying surfaces of intact specimens representing aircraft fuselage joint construction. Thermal imaging was used to monitor these coupons while they were fatigued to various cycles in order to attempt to understand the fretting process. In all specimens, fretting damage was observed, first on the rivet/bore interface and then on the faying surfaces (Figure 61). Damage accumulation in terms of area was very rapid at low cycle counts, continuing to increase but at a much reduced rate throughout the specimen life. It was found that the measured temperature difference between the "hot spots" and the surroundings correlated to the area of fretting damage (Figure 62) providing indication that fretting is partially responsible for the local increase in temperature. [46]

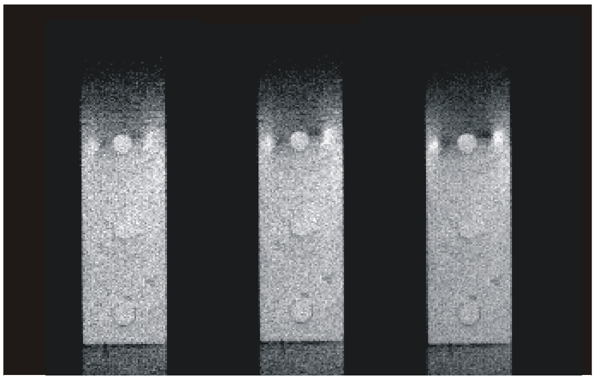


**Figure 61. Raw thermal images of specimen #10 during loading at 143,000, 145,000, and 147,000 cycles (from left to right). The crack tips are visible as bright spots (higher temperatures) on either side of the top rivets.**

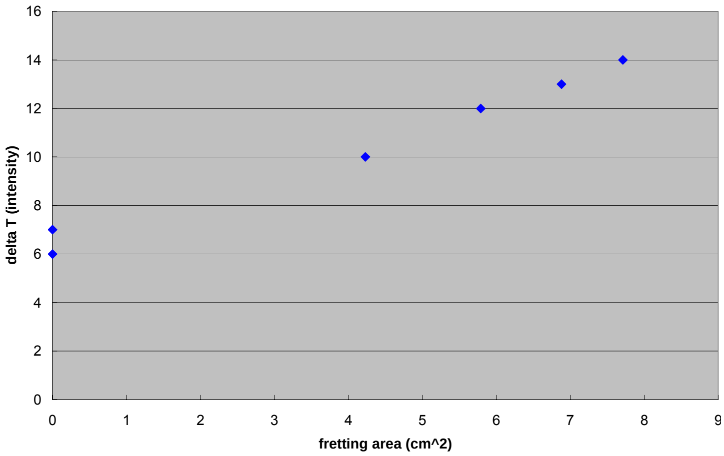
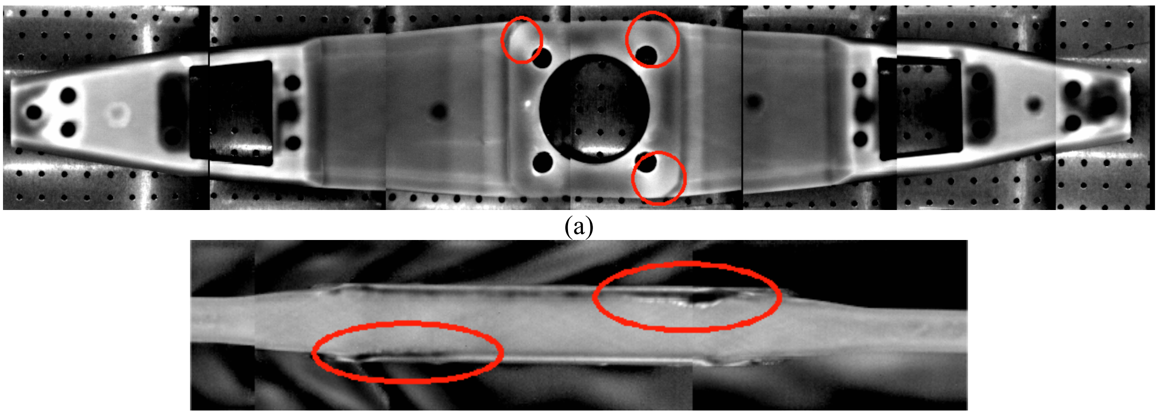



**Figure 62. A plot of the difference in apparent temperature (arbitrary units) between the detected hot spots and the surrounding area as a function of the fretting damage measured after teardown.**

Composite delamination and cracking

The Canadian Forces CH149 helicopter tail rotor half hubs are made of fibre-reinforced polymer composites, mainly carbon fibres and glass fibres, as well as metallic fittings and bushings. Vibration-induced delamination and cracking are posing safety problems resulting in premature retirement or repairs. Pulsed thermography was used along with other NDE methods to detect delamination and fibre cracks. The results, verified with optical techniques indicated the potential of this method for fast and economical inspection of the helicopters in the field [47].

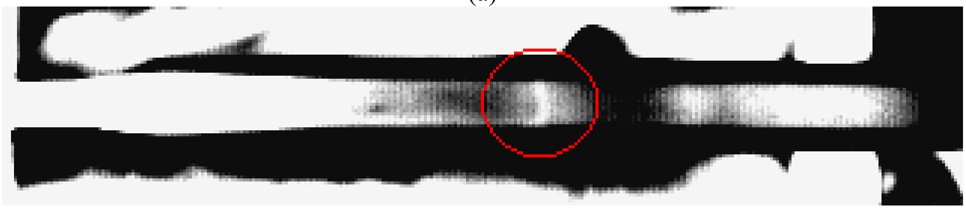


(a)

(b)

**Figure 63. Pulsed thermography inspection results (a) top view and (b) side view showing delamination**

(a)

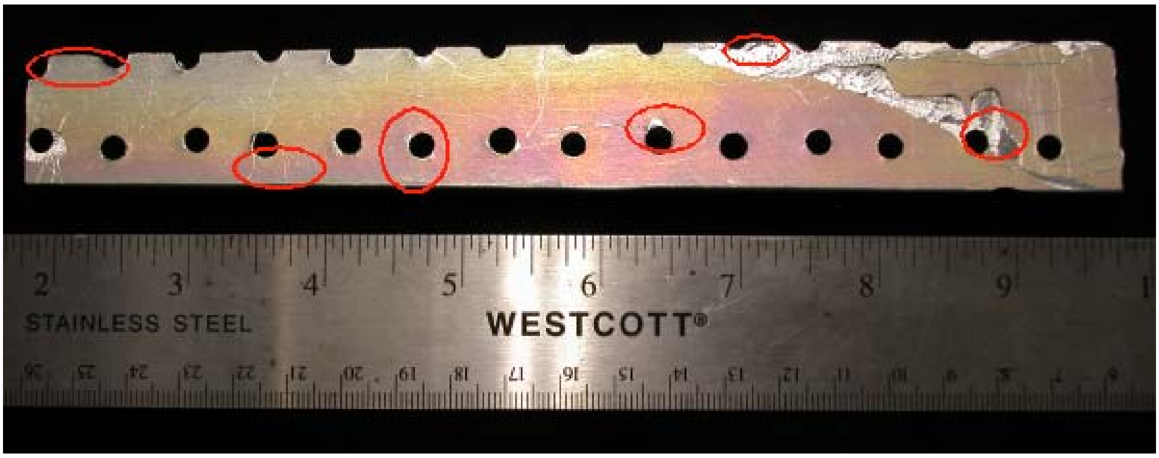


(b)

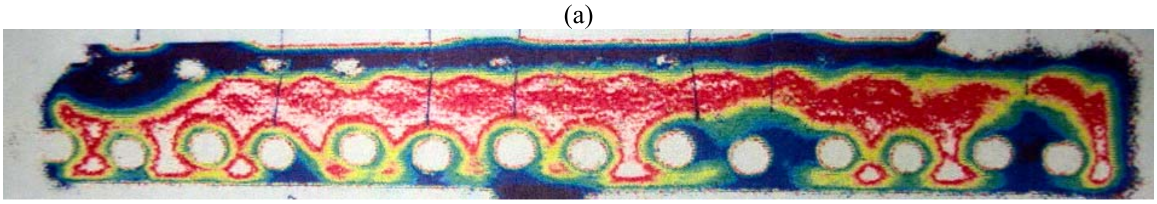
**Figure 64. A photograph of a tail rotor half-hub side view (a). Pulsed thermography inspection results showing crack in carbon fibre frame (b).**

<u>Fatigue and stress corrosion cracking</u>

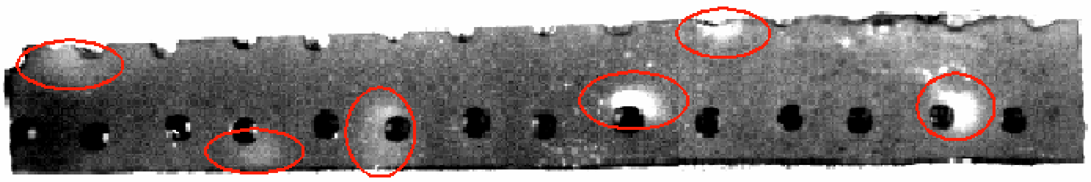
Stringer parts of a CP140 horizontal stabilizer are made of 7075-T6511 aluminium alloy extrusion. In this study infrared thermography was used along with ultrasonic C-scan technique to detect stress corrosion cracking (SCC). The results, shown in Figure 65, indicated that while ultrasonic C-scanning provides convoluted image of the total damage, pulsed thermography can easily detect SCC located less than 5mm deep from the inspection surface [48].




(a)



(b)

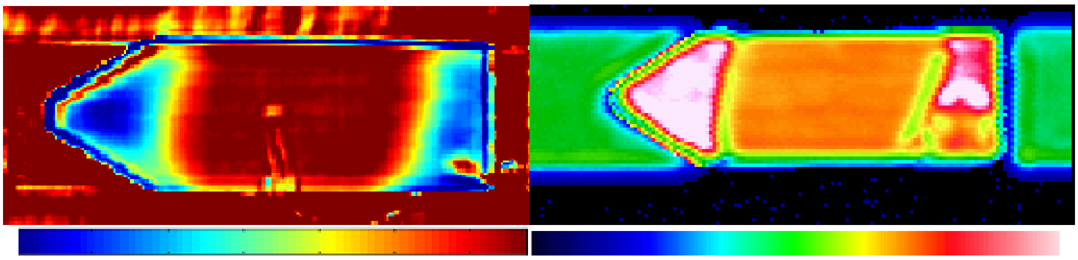


(c)

**Figure 65 A photograph of the lower stringer– aft flange containing stress corrosion cracks (a). Ultrasonic C-scan image of the above part (b). Thermal image of the above part obtained by pulsed phase thermography showing stress cracking sites (c).**

<u>Fatigue-induced disbond</u>

Poorly-manufactured bonded patch repairs could fail under fatigue loads resulting in disbonded areas between the patch and the repaired structure. Pulsed thermography was used to detect the onset of disbonding of the tip of a patched specimen that was later verified using ultrasonic pulse-echo C-scanning [49, 50]. The thermography identified disbonding much earlier than the ultrasonic C-scan indicating the potential of this method for detection of the early stages of debonding or kissing bonds.



(a) (b)

**Figure 66. Pulsed infrared thermal image of disband (a). Ultrasonic C-scan image of the same disbond**

**Electromagentic and Ultrasonic Techniques**

C. Mandache and A. Fahr, NRC-IAR-SMPL

Recent fatigue-related research work includes development of inspection methods for second and third layer cracks in aluminum lap joint structures of aircraft fuselage and thick-section wings. Occurrence of fatigue cracks in the fastener holes of aircraft structures is a common problem both in commercial and military aircraft. In some military aircraft, individual fasteners may be highly loaded such that the critical crack length can be less than the countersink width. Thus, a potentially critical fatigue crack can be completely hidden by the fastener head. In this study, the potential of ultrasonic Rayleigh waves and electromagnetic methods for the detection of cracks under fasteners was investigated on laboratory-made samples simulating two-layer fuselage lap joint structures. The ultrasonic Rayleight waves generated with a piezoelectric immersion transducer and the pulsed eddy currents were induced using a coil exited with a broadband electromagnetic pulse. The Rayleigh wave method proved to be very sensitive to first layer cracks and provided measurable indications from hidden cracks as small as 0.3mm [51]. However, the ultrasonic waves could not penetrate to the second layer due to the air gap between the two layers to detect second layer cracks. The electromagnetic methods such as pulsed eddy current technique was not as sensitive as the Rayleigh waves but detected cracks in both the first and the second or third layers (Figure 67 Figure 68) when they exceeded beyond the fastener heads with similar sensitivity for all the three layers [52].

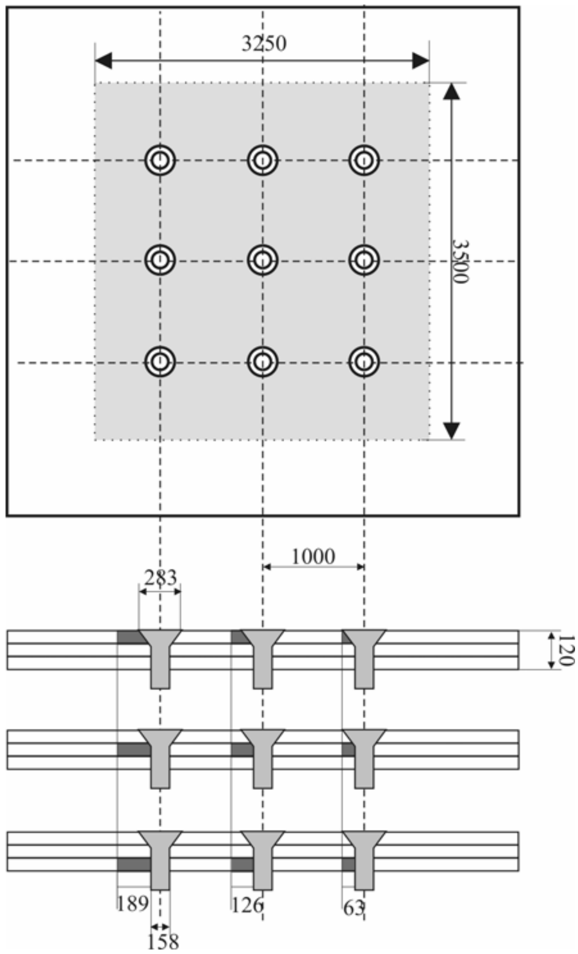



**Figure 67 Schematic diagram of the three-layer lap joint specimen containing simulated cracks (all dimensions are given in thousands of an inch, 1 inch=25.4 mm). Shaded area represents the scanned surface.**

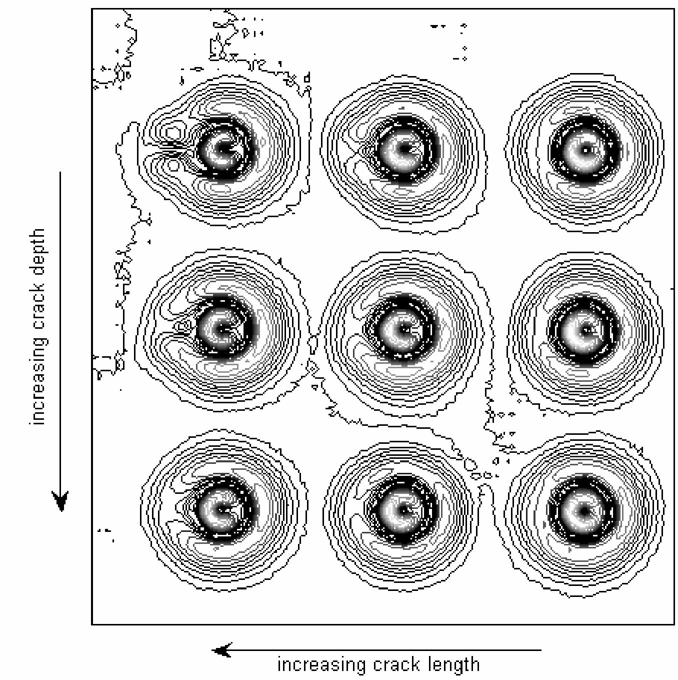



**Figure 68 C-scan contour plot of the overall surface (tangential) magnetic field. Open contours indicate disruption in the magnetic field due to cracks.**

**Magneto-Optic/Eddy Current-Imaging (MOI)**

F. Caza, Bombardier Aerospace

MOI is an Eddy-Current technique inducing a magnetic field in materials that can be readily visible on a video image. It allows for faster inspection of skin/stringer assemblies (60% faster than traditional eddy current). Complete area can be covered in one image (no mapping is required). Inspection head is shown in Figure 69 and a sample resulting image in Figure 70.

**Figure 69 Magneto-optic inspection head.**

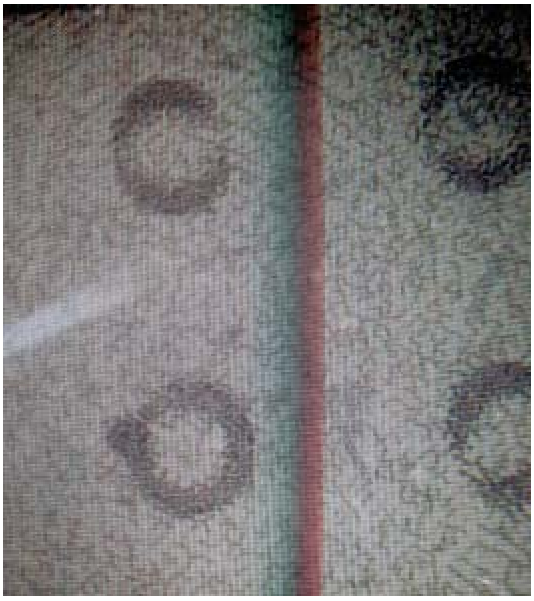


**Figure 70 MOI image of 4 rivet holes inspected.**

Bombardier assessment confirmed that:

- MOI technology is a highly sensitive and reliable method to detect small surface fatigue cracks.
- Tolerate paint thickness up to 0.0135-inch without loss of inspection sensitivity.
- MOI images provides easy go/no go decision making.
- Paint stripping is not required.
- Real-time inspection.
- Very rapid scanning.
- Minimal operator training.
- Lightweight and portable.

- Significant time savings i.e. 1/10th the time of conventional manual probe scanning.

Equipment is now used to monitor Bombardier test aircraft and decision has been made to implement MOI in service.

**NDI technique for 2nd Layer wing splice**

M. Bunn, ATESS/NDTC

CF have been working on a 2nd Layer wing splice NDI technique for CP-140 (P3) and since January 2005 significant progress has been made in pinpointing inspection methods for more detailed investigation. In June 2005 CF held a NDT working group meeting at ATESS to discuss inspection options. Representatives from USN and Norway attended and participated.

Several NDI Technologies were investigated by the CF: Eddy current methods: FastScan system, ring probes, linear field probes, linear array probes, sliding probes (A-scan and C-scan), pulsed eddy current (PEC), and bolt hole eddy current (BHEC); and Ultrasound: shear wave through backside of riser (inside wing), and phased array through outer skin (directed beam).

FastScan System was retained as it is currently in use in the commercial airline industry. System is off-the-shelf and fully developed.

In order to develop wing splice NDI technique trial inspections of CP-140 (P-3) were performed in January 2006 (Figure 71). Improvements to technique made by switching to dual frequency inspection. Finalized technique procedure includes defined standard set-up procedures performed on second layer defect of 0.115” at 5 O’clock. Inspected were 1/2 and 2/3 Splice from WS 80-160 and WS220-350 of wing with paint (P-3C) and with paint removed (P-3A).

**Figure 71 Trial Inspection of P-3A Wing**

On P-3A wing 8 defects were identified. Defects were confirmed using Bolt Hole Eddy Current (BHEC) for 7 of the defects. Defects appeared to be minor amounts of corrosion. Wing is undergoing teardown at a contractor facility. Inspection of the P-3C wing confirmed that paint must be removed to facilitate inspection technique.

A total of 1,389 holes was inspected with PHASEC 2200 equipment. 294 holes contained defects greater than 0.115” (the set-up flaw size). Total of missed flaws in the detectable range was 12. Total of 43 false calls were made. This translates to 96% of the detectable flaws identified with a false call rate of 11%.

Some inspection limitations were noted: the technique was unable to detect EDM Notches at the 6 and 12 O'clock position, based on test pieces used, false indication was given for a 3/16 fastener with a fastener spacing of 0.400" center to center. It was also found that edge effects may interfere with inspection, fasteners may need to be pulled.

Technique is very user-dependant therefore it is recommended that specific training be provided to inspectors. Training will require at least four test pieces for practice and evaluation. Two test pieces for technician practice and training and two test pieces to qualify the technician. Each test piece should have at least 25 fasteners and contain a combination of holes with defects and defect free holes. The defects (corner notches) in the test piece should range from approximately 0.080" to 0.250".

CF has inspected an aircraft in service. Inspection of the 1/2 and 2/3 Splices from WS 80-160 & WS 220-350 took approximately 16 NDI hours with 2 inspectors required. One defect identified and confirmed with BHEC. Defect found at WS 321.5 Panel 1 at the 1/2 Splice. It is approximately 0.060" at the 6 O'clock position. During removal of defect, second crack at the 9 O'clock position identified.

**NDE Reliability**

A. Fahr and M. Khan, NRC-IAR-SMPL

Damage tolerance assessments of aerospace structures require detailed knowledge of the defects in that structure. Currently, it is assumed that 0.050" is the smallest detectable crack size in fastener holes using BHEC inspections. The Canadian Air Force is currently investigating the assumed defect size through a comprehensive POD study of specimens reflective of box wing structures of the C130 and CP140 (P3). This study, being carried out at IAR [53], involves POD development for lab-grown cracks, EDM notches and real (natural) cracks in real structures. Concurrent to the POD study, eddy current modeling is being validated as a means to predict inspection results. The modeling variables include: crack length, crack depth, lift-off, frequency, probe tilt, material conductivity. The modeling is expected to significantly reduce the time and financial burden of empirical POD studies that involve the MIL-HDBK-1823 methodology. The POD data generated along with the modeling will help to establish new POD with change of the aircraft structure or any other conditions that affect inspection results in a timely and cost-effective manner.

**NDE for PHM**

C. Mandache, M.Genest and A. Fahr, NRC-IAR-SMPL

As a part of a broader activity to develop advanced sensor technologies for prognostic and health monitoring (PHM), air-coupled ultrasonic sensors are being investigated for bearing fault detection while eddy current sensors are being studied for crack detection in turbine blades during engine operation. The air-coupled ultrasound uses frequency changes of air-born sound waves to identify damage and the eddy current technique uses the crack-induced blade elongation and changes in the air gap between the blade and the engine casing to identify damage. These activities are at the early laboratory stages and yet to be tried on real engine environment.

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