REVIEW OF CANADIAN AERONAUTICAL FATIGUE WORK 2007-2009

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SUMMARY

This paper provides a review of Canadian work associated with fatigue of aeronautical materials and structures during the period 2007 - 2009. 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
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
HOLSIP	Holistic Structural Integrity Process
IAT	Individual Aircraft Tracking
IDS	Initial Discontinuity State
IFOSTP	International Follow-on Structural Test Program
IVD	Ion Vapour Deposition
MSDRS	Maintenance Signal Data Recording Set
NDI/NDT/NDE	Nondestructive Inspection/Testing/Evaluation
OEM	Original Equipment Manufacturer
PLF	Parametric Loads Formulation
POD	Probability of Detection
RST	Residual Strength Test
SCC	Stress Corrosion Cracking
SEM	Scanning Electron Microscope
SESC	System Engineering Support Contract
SFH	Simulated Flying Hours
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 2007 to 2009. 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 (DGAEPM)

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

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

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

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

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FULL-SCALE TESTING

F/A-18 FT245 CF-188 Wing Fatigue Test – IAR/CF/Bombardier/L-3 MAS/RAAF

R.S. Rutledge, NRC-IAR-SMPL

The Structures and Materials Performance Laboratory of the NRC-IAR completed the wing fatigue test to three repeat load lifetimes (RLL) of simulated flight hours (SFH) by October 29th 2004. After completion of the three RLL of the IARPO3a spectrum, the CF188 wing in FT245 was inspected and residual strength tests to load levels twenty percent higher than design limit loads (DLL) were carried out and completed by October 28th 2005; the residual strength summary was provided in the 2005-2007 review. In 2007 after successful application of these static loads, an investigation was carried out to determine what structural damages were present at test end. After removing the wing from the rig, X-ray inspections were taken. All the upper surface fasteners were then removed from the specimen and the upper doors and skin detached. A detailed inspection was then carried out to identify significant damage locations for further fractographic examinations. While these inspections were carried out, the fasteners connecting the lower skin, spars and ribs were removed systematically to break down the specimen assembly into its constituent parts (Figure 1 and Figure 2). The parts were then given a detailed inspection using appropriate non-destructive testing inspection techniques. All parts were tagged with bar codes and non-destructive testing technicians entered all part results in the structural information teardown database. The non-damaged parts were identified and damaged parts were written up in notices of structural deficiency (NSDs).

Figure 1. FT245 Inner Wing Fatigue Test Specimen at Disassembly

There were 33 parts on the inner wing found containing damage at teardown. Multiple damages were found on many of these. A total of 201 NSDs were issued on inner wing parts. These NSDs were categorized into fracture critical, durability critical and secondary critical parts. As a result of the inner wing teardown, 90 NSDs were issued for 10 fracture critical parts of the inner wing, 107 NSDs were issued for 19 durability critical parts of the inner wing, and four NSDs were issued for four secondary critical parts of the inner wing.

Figure 2. FT245 Outer Wing Test Specimen and Wing Fold Transmission at Teardown

As a result of the outer wing teardown, 35 NSDs were issued for 16 parts of the outer wing. These NSDs were also categorized into fracture critical, durability critical and secondary critical parts. As a result of the outer wing teardown, 18 NSDs were issued for 6 fracture critical parts of the outer wing, 10 NSDs were issued for 8 durability critical parts of the outer wing, and 7 NSDs were issued for three tertiary and secondary critical parts of the outer wing.

At the same time as the fatigue test article was dismantled, an in-service right hand inner wing of aircraft 188720, S/N A12-0161, was systematically dismantled and parts were categorized in order of priority before they were inspected. An inspection package was followed to ensure each part was inspected to the required scope. In preparation for documenting the inspection results, a teardown database was developed to cater to the specific needs of the teardown. Sixty five components were examined and 53 notices of structural deficiency were raised and added to the structural information database of the CF188 during the teardown activities for the 720 in-service wing.

For both teardowns the components that showed positive damage results were documented through notices of structural deficiency and reviewed to determine the need for qualitative fractographic analysis. Fifty fractography reports were issued for the damages found on FT245 and most of these were quantitative fractography reports. However, in several instances repeat patterns within the fracture surface could not be identified. For these damages qualitative fractography was carried out. On the inner and outer wing of aircraft 188720 eighteen qualitative fractography reports were produced.
At the end of the evaluation process, the inner and outer wing parts were tagged, catalogued and stored for the remaining service life of the fleet as future examination may be required to justify fleet management decisions.

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

The FT245 wing full scale fatigue test included dynamic loads caused by trailing edge and wing tip buffet on the CF-188, however, the spectrum had to be severely truncated, particularly in terms of control surface attachment and wing tip loads, in order to complete the test in an appropriate time frame. Therefore, some areas of the wing were under-tested. In addition, the Canadian Forces Lifing Policy was revised for this aircraft such that areas subjected to dynamic loads must be tested to five times the required service life. As a result the Canadian Forces decided to perform additional component testing to ensure that no failures to the lug attachments and wing tip would occur in five lifetimes of testing using the IARPO3 spectrum. They also included testing of some limited back-up structure in the region of the hinges by applying representative wing bending, shear and torque in these areas in addition to the attachment loads. The National Research Council Canada was tasked to carry out this testing. There were two phases of testing, since separate testing of the inner and outer wings was required. Phase 1, designated as test FT193 (Figure 3), tested the outer wing tip, the wing-side aileron outboard half-hinge, wing-side aileron inboard half-hinge and wing-side aileron actuator attachment, for which the fatigue testing was completed in 2007. Phase 2, designated, FT169 (Figure 4), tested the wing-side trailing edge flap outboard half-hinge and localized back-up structure and was carried out in parallel, for which the fatigue testing was also completed in 2007. After fatigue testing, residual strength testing to 120% design limit loads was conducted. Once testing was completed, the test specimens were inspected, notices of structural deficiency issued and damaged areas were excised for fractographic examination. Eleven quantitative fractography reports were completed in Canada for FT193 damaged locations and three for damage locations in FT169. Both tests provided crack growth data estimates for these evaluated locations.

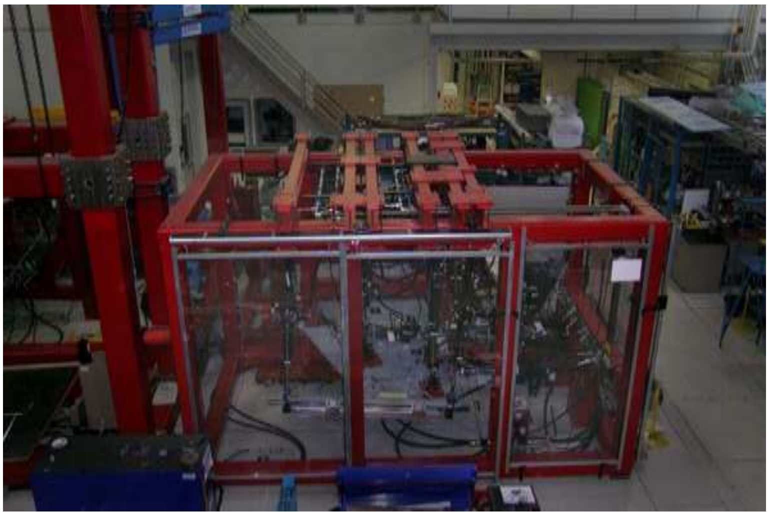


Figure 3. FT-193 Extended Fatigue Test Set-up

Figure 4. FT-169 Extended Fatigue Test Set-up

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

R.S. Rutledge, NRC-IAR-SMPL

As reported previously in the 2005-2007 ICAF summary, the objective of FT333 was to determine the fatigue life of the trailing edge flap (TEF) outboard lug (Figure 5), up to a maximum of 5 repeated load lifetimes (RLLs) of simulated flight hours (SFH) of IARPO3 spectrum loading. At the start of FT333, this component had undergone 1.6 RLLs of SFH of testing on FT245, implying an additional 3.4 RLL's of SFH testing was required. Following completion of the fatigue loading, an RST to 120 % of design limit load (DLL) was completed. The test specimen was retained from FT245. The outboard hinge was modified to the thicker aluminum configuration that was implemented for the fleet. This TEF was first installed on FT245 at 3 074 SFH and removed at 10 758 SFH due to cracking of the aluminum leading edge skins. A further period of testing from 12 714 SFH to 14 812 SFH followed. At that time additional cracking of leading edge skins and repairs could not be tolerated within the fatigue test schedule, at which point this TEF was finally removed and a different configuration TEF with a titanium hinge was installed on FT245.

FT333 fatigue testing the CF-188 TEF outer lug demonstrated this lug had a total fatigue life of at least five repeated load lifetimes under FT245 loading, and did not fail under subsequent RST loading to 120% design limit loads. The test specimens were inspected, notices of structural deficiency issued and damaged areas were excised for fractographic examination. Four quantitative fractography reports were completed in Canada for FT333 damaged locations to obtain crack growth data estimates for these.

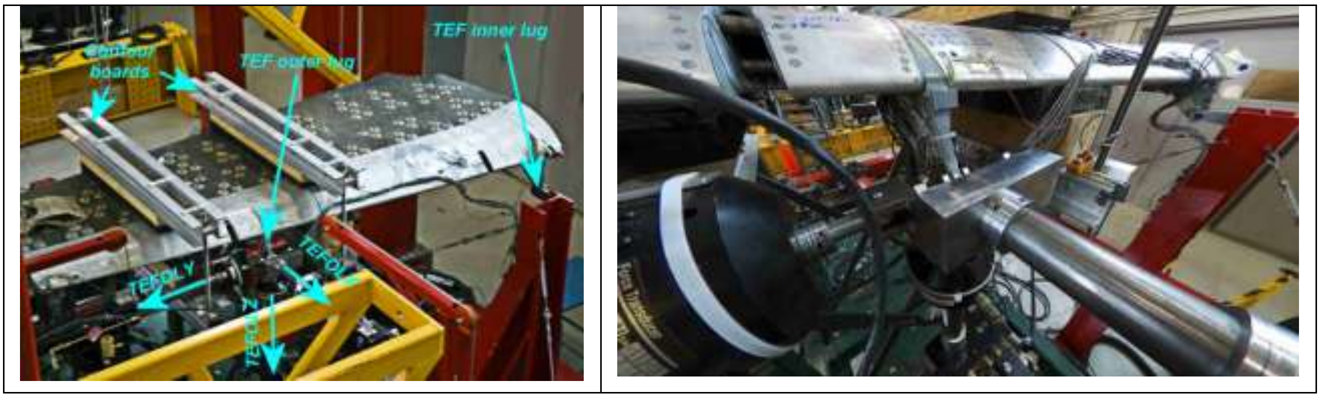


Figure 5. F/A-18 FT333 CF-18 Trailing-edge Flap Outboard Lug Fatigue Test Set-up and Tri-Axial Lug Hinge Attachment

FATIGUE LIFE PREDICTION AND ENHANCEMENT

CF-18 Aircraft Structural Integrity Program (ASIP)

J. Dubuc, L-3 Communication 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 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. L-3 MAS is currently conducting the definition and development of the third and final phase of the ALEX program, designated as CP3. 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.

One of the key features of the ALEX program is the use of in-situ robotic system to enhance the fatigue life of critical areas of the structure by machining and/or shot peening. This process is used to enhance the accuracy and consistency of the rework in situations where manual operations are difficult. See ICAF Canadian National Reviews from 2003 to 2007 for details.

Several structural methodologies have been developed in support of the CF-18 ASIP Program. Recently, efforts have focused on the following areas.

Hole Improvement Methodology:
In the framework of aircraft structure repair and overhaul, techniques for salvaging fastener holes often involve material removal to eliminate damages such as cracks or elongations. After this removal, life extension processes are needed to

restore or even increase the original fatigue life of the hole. Life extension techniques typically used are split sleeve cold working, interference fit installation of fasteners and in some cases, installation of cold shrink fit bushings.

An experimental program has been developed to study the potential improvements of various repair techniques, mainly combining split sleeve cold working with interference based methods.

7050-T7451 aluminum test specimens subjected to various hole improvement techniques have been tested under a typical CF-18 fighter spectrum at high stress levels.

Results have shown that combining cold working with the installation of interference fit fasteners did not provide any improvement when compared to the configuration with interference fit fasteners alone. Cold shrink fit bushings (with an interference fit fastener installed) were shown to provide a life improvement factor of more than 3 when compared to open holes. Furthermore, the bushing material (CRES vs. aluminum) was shown to have little effect.

Surface Renewal:
It is known that renewal of etched surfaces (IVD/ anodize) may improve the fatigue life of a structure. There is also strong evidence that blending regions where small fatigue cracks are present or are expected has a significant impact on the remaining fatigue life. A methodology to account for the benefit of surface renewal is required and must account for the amount of fatigue damage cumulated prior to the surface renewal.

Several repairs and modifications designed and incorporated on the CF-18 fleet include some type of surface renewal procedures. These renewal operations have a beneficial effect on the fatigue life of a component for two main reasons:

i. Surface renewal allows removing the surface material which is the most subject to fatigue damage due to previous in-service loading, and
ii. Surface renewal allows improving the surface finish, particularly by removing the detrimental effect of pre-IVD (Ion Vapour Deposition) etching or anodizing.

The most common surface renewal procedures are:

a) Light polish: the surface is polished to remove a potentially detrimental surface finish as pre-IVD etching, or anodizing. This polish, typically of 0.003”, may be completed alone or may be followed by shot-peening;

b) Light blend: the surface is blended usually at high stress locations where fatigue cracking could occur during the service life of the aircraft. Typically, this light blend is completed after an NDT inspection has shown that the surface is crack free. The blend depth will be equal to the detectability threshold of the NDT technique used, to ensure that the remaining surface is free of any potential cracks; and

c) Deeper blends will be completed when cracks are detected. The location of the crack is blended until the crack is not detected anymore by NDT techniques, then a confidence cut equal to the detectability threshold of the NDT technique used is added.

In a previous coupon program it was shown that a light polish before shot peening provides a significant life improvement compared to shot-peening alone. There is also strong evidence that blending regions where small fatigue cracks are present or are expected has a significant impact on fatigue life. However, there is no systematic method to account for the life improvement due to the surface renewal operations when predicting the remaining fatigue life of the structure. Certification of repairs involving surface renewal or removal operations is generally based on specific coupon tests, which can yield to an expensive and time-consuming repair development process.

To address the above issues, a joint coupon project was undertaken. The main objectives of the project are:

i. To build experimental data showing the effect of surface renewal/removal operations on the fatigue life of a structural component. This data is to provide input to allow building a methodology to address surface renewal during fatigue analyses;
ii. To identify the depth of surface renewal required to reset the fatigue life for a given level of expended life; and
iii. To minimize the need for separate coupon programs for future surface renewal modifications

To provide useful data for such a methodology, an international joint coupon program was completed with Finland, Australia, USA and Switzerland.

Preliminary results tend to show that a light polish of 0.1 mm on all faces of the coupon can restore the full fatigue life if completed prior to 50% of the coupon crack initiation life. Results have also shown that on notched coupons, a blend deep enough to remove cracks detected using Eddy-Current inspection, can restore the full coupon fatigue life even if coupon was cycled to 70% of its life to failure. However, a light polish on the adjacent faces completed concurrently with the blend was seen to be required to ensure such a fatigue life reset.

Dynamic Spectrum Methodology:

The general approach used to predict sequences for CF-18 fatigue analysis is done by combining the two primary source of loading: the manoeuvre (or steady state) and the dynamic (or buffet) loading. Each loading is predicted separately and combined together afterward (Figure 6). Dynamic loading is considered for aft fuselage and empennage locations and some wing and control surface locations as well.

It has been demonstrated through flight testing and wind tunnel testing that dynamic loading seen on the aft fuselage and empennage structure is mainly driven by Angles of Attack (AoA) and Incompressible Dynamic Pressure (Q). As a result, the dynamic loading is typically characterized using these two parameters. For simplifications, the dynamic loading has traditionally been considered stationary, or Gaussian, within a given AoA and Q range. Thus dynamic response can be studied by dividing the flight envelope into pre-defined AoA-Q bins or regions. These bins are relatively small sections of the flight envelope defined by a range of AoA and a range of Q. Then flight test data is collected and divided amongst the bins in time continuous segments to construct a Dynamic Database (DDB). The AoA-Q binning approach can be understood visually by a matrix with dynamic data populating each cell or bin (Figure 7).
There exists a trade-off between AOA-Q refinement and reliability/accuracy of each bin. Thus a balance between the amounts of data required versus the bin refinement (size) must be chosen. Hence, an alternative approach to refining bins (or reducing the scatter in each bin) was considered to improve the dynamic characterization on the CF-18 aircraft. This approach consists of fitting a smooth continuous surface through the binned dynamic flight test signals (Figure 8). The resulting smooth surface will represent the average energy content and allow a means to produce and predict dynamic cycles for any given AoA and Q values, including those with sparse data.

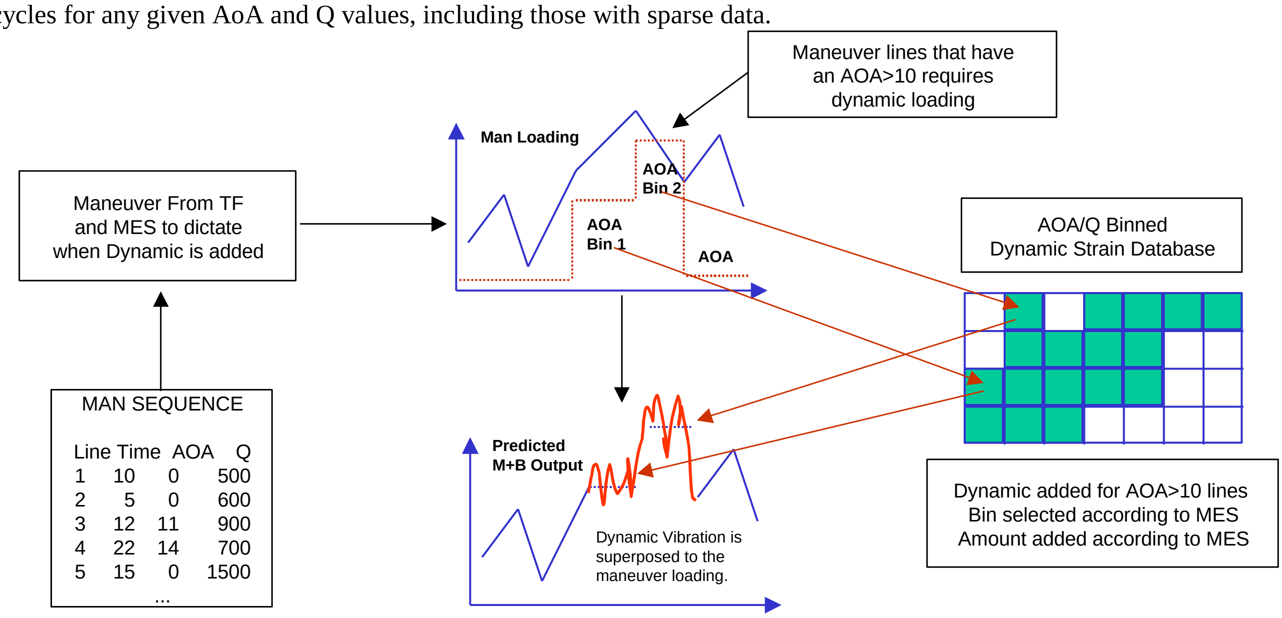


Figure 6. Dynamic Spectrum Creation Overview

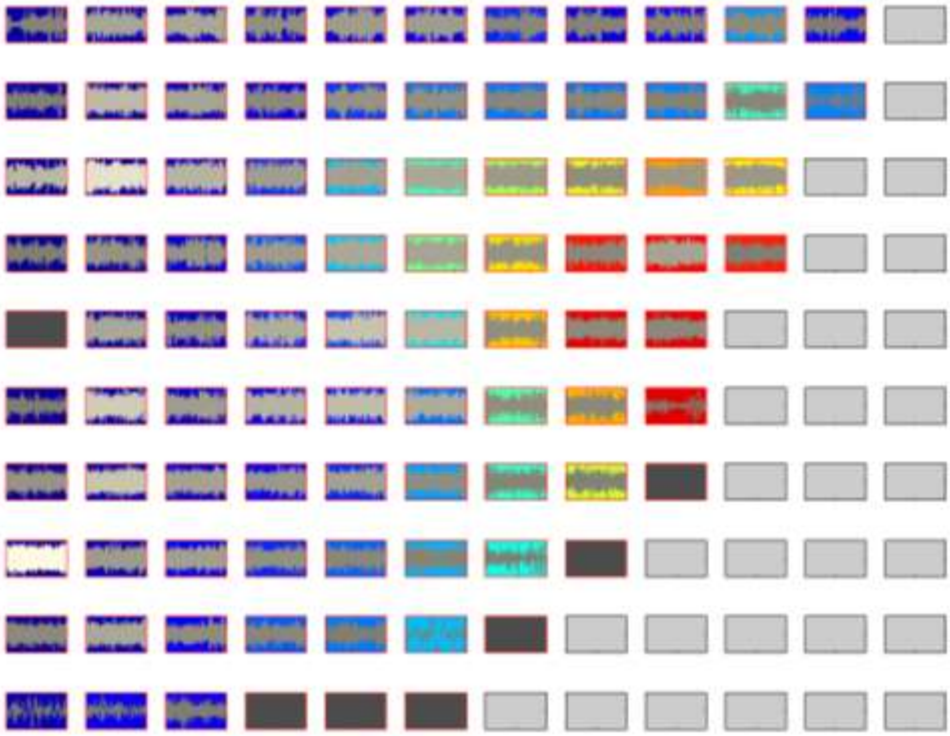


Figure 7. Dynamic Database (DDB) Graphical Example.

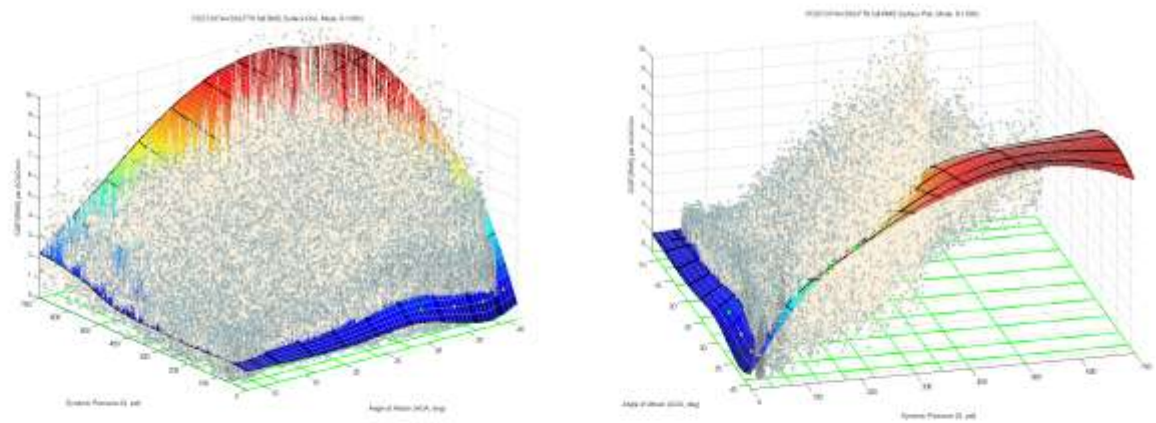


Figure 8. Dynamic Signal Surface Smoothing Approach.

Tracking Factor & Dynamic Scatter Factor:
The CF-18 Basis of Certification (BoC) ensures safety of flight through application of uncertainty factors to represent material and structural response variations. Unfactored service lives are divided by the Test Factor which is the product of several uncertainty factors, as follows:

Test Factor = LF * SF * TF

Where: LF is the load uncertainty factor (rarely applied), SF is the material (and dynamics) scatter factor and TF is the tracking factor. The BoC prescribes ceiling values of 5.0 for Test Factor and 1.5 for Tracking Factor.

For manoeuvre only loading that is properly tracked (fatigue damage characterized for individual aircraft), the adopted approach yields Test Factors in the neighbourhood of 2.5. Past approaches required the use of judgement to determine the level of tracking. For locations not considered to be tracked, the maximum TF was prescribed, giving Test Factors around 3.75 for manoeuvre only loading (Figure 9). For locations driven by significant dynamic loads, the maximum Test Factor of 5.0 was often taken. To avoid unnecessary maintenance and associated penalties in aircraft availability, methodologies were required to derive applicable TF and SF for cases with intermediary levels of tracking and dynamic content.

In the case of tracking, a methodology was developed that relates the TF to the representativeness of the available tracking indices (ex. wing root bending or aft fuselage splice bending) for the location being evaluated through driving load correlation analysis. In the case of dynamic load/response, two distinct studies were performed: one based on analysis of flight test data from multiple sources for many structural locations and the second based on a statistical analysis of in-service failures at similar locations. In both cases, the proportion of the fatigue damage caused by high-cycle, dynamic turning points serves as the reference to establish the combined material and dynamic scatter factor.

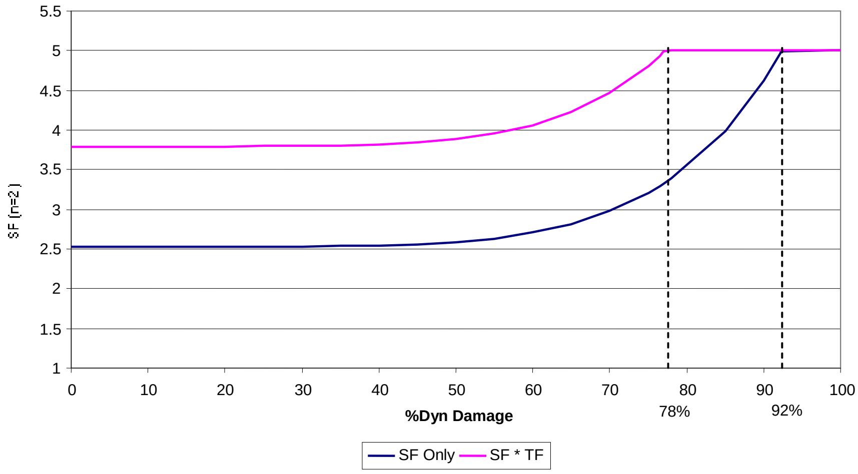


Figure 9. Effect of Combining SF and TF (TF set at 1.5)

CF-18 Fatigue Life Management Program

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.

In response to a fleet airworthiness requirement circa 2004, an index to measure the fatigue life expended (FLE) for the upper inboard longerons at the aft fuselage splice (AFT FLEI) was developed. The load estimation for this index relies on the aircraft vertical accelerations (Nz) and strain input at the roots of the horizontal stabilators to estimate the fuselage up-bending at the splice. This index furthers reduction of unnecessary maintenance actions on these longerons, because without it, they would be compared to the wing root bending index (WR FLEI) with a TF of 1.4. As it stands, a TF of only 1.0 is required when using AFT FLEI, and analysis of fleet data shows that CF-18 aircraft are accumulating AFT FLEI at lower rates than the WR FLEI, allowing even further maintenance reductions.

More recently, an airworthiness issue was identified at the fuselage connection points (stubs) of the 20% canted vertical tails. Figure 10 shows how these tails are excited by the strong vortex flow emanating from the Leading Edge Extension (LEX) at high angles of attack. It was therefore important to characterize the damage coming from these dynamic loads, found to be a function of both the AoA and incompressible dynamic pressure (Q) when developing fatigue usage indices at the stubs. While the AoA-Q usage time (table) was already written in available input files, several functionalities needed to be developed in the CF tracking software. To start, the AoA-Q table needed to be read in and have its format converted. Next, manoeuvre and dynamic load estimations needed to be carried out for each flight through use of parametric load formulations (PLF) and dynamic databases, respectively. The PLF is applied to the flight parameters provided at all strain and Nz turning points to estimate the manoeuvre loads. Dynamic loads are found through combination of the AoA-Q usage time and dynamic databases containing varying dynamic cycles (frequency, amplitude) depending on the values of AoA and Q. Then, missing and bad data fill-in routines needed to be developed (note that very little bad data exists as the approach is 100% parametric). This index has been used to select fleet leaders for a sampling inspection and to help extend maintenance intervals at the stubs.

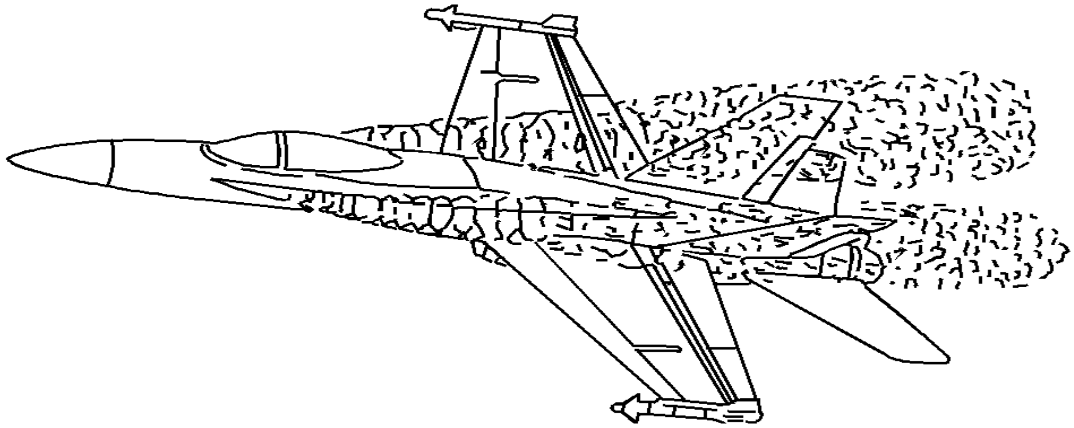


Figure 10. CF-18 LEX Vortex Formation

SDRS Validation Project

M. Tourond, DND-AEPM

Background:

The Structural Life Assessment Program (SLAP) involving the Royal Australian Air Force (RAAF), the United States Navy (USN), Lockheed-Martin Aerospace (LMA), and the Canadian Forces (CF), consisted of a full scale fatigue test (FSFT) of a P-3 Orion airframe under a USN 85-percentile spectrum. The SLAP has generated many serious defects which have had a direct impact on the airworthiness of the CP140/A aircraft. Of note are the findings in the region of the inboard nacelle where the front spar (web and cap) and lower planks have been found with extensive damage.
In the early 1990’s the CF adopted AN/ASH-37, the current Systems & Electronics Inc (SEI) Structural Data Recording System (SDRS), for its Individual Aircraft Tracking (IAT) within the CP140 Aircraft Structural Integrity Program (ASIP). The AN/ASH-37 was used to monitor and record structural load data generated during flight. It consisted of a 20 channel solid state recording system designed for airborne environments and operated under microprocessor control. It had a removable 2 MB memory unit and a multi-axis accelerometer. Three strain sensors mounted in the wings (WS92, WS147 and WS223) and one in the tail (HSS50) supplied the structural strain data.

The CF, jointly with the USN, developed a suite of post-processing tools to allow the SDRS recorded data to be used as the primary IAT information set. However, without a validation of the SDRS readings, DND was not able to effectively use and benefit from the new software and over 56,000 hrs of usage data. This effectively reduced the validity of the inspection cycle for the structure’s critical points, thus affecting the CP140’s airworthiness. In order to ensure the immediate airworthiness of the structure, the CF has been forced to undertake intrusive sampling inspections as well as substantially augment the current Third Line Inspection and Repair (TLIR) phase at the depot-level maintenance facility, IMP Aerospace. Currently the CP140 Estimated Life Expectancy (ELE) is based on the results of the SLAP FSFT which was run with the USN 85th percentile fatigue spectrum. A better understanding of the differences between the USN spectrum and the CF spectrum will reduce this need and allow the CF to determine inspection thresholds and repeat inspection intervals based on actual CF usage rather than parametric Mission Profile data.

The SDRS Validation Study was initiated by the Directorate of Technical Airworthiness and Engineering Support (DTAES) and Sponsored by the Directorate of Aerospace Equipment and Program Management – Maritime (DAEPM(M)) located in Ottawa, ON. Testing and Evaluation was carried out at the Aerospace Engineering and Test Establishment in 4 Wing Cold Lake, AB.

Objective:

The objective of the study was to:

a. Verify the accuracy of the data collected by the CP140 SDRS by recording data from the Flight Test Instrumentation (FTI) installed next to CP140 SDRS instrumentation; and

b. Measure the strain to establish Stress/Load Ratio (SLR) at key structural points using FTI installed throughout the aircraft (see Figure 1).

Test Method:

The Aerospace Engineering Test Establishment (AETE) with the aid of various AMS staff at 14 Wing Greenwood installed a total of 105 sensors (see Figure 11 for a limited view) on the aircraft at 14 AMS, Greenwood. The strain gauges for the wings were installed symmetrically (mirrored) to ensure asymmetrical manoeuvres were captured and to ensure redundancy in case of a faulty reading or an unserviceable gauge. Appendix 1 of Annex A to AETE Report 2006-017 defines the FTI in further detail. Ground testing (fuel load) was conducted following FTI installation to calibrate the FTI system and derive fuel load offsets during flight. Lastly, AETE conducted flight testing of the CP140 through a diverse spectrum of test points within the aircraft's operational envelope.

Data Reduction and Analysis:

The SDRS data files were all downloaded successfully from the onboard system and were post processed into a format compatible with analysis tools at IMP Aerospace. The engineers at IMP were then able to further process and analyze the data set to ensure each flight contained sensible and usable data. The results for the 4 SDRS sensors and corresponding FTI sensors were processed first to satisfy the first objective. The outcome showed a reasonably good correlation between the onboard SDRS and FTI outputs, thus validating the current SDRS.

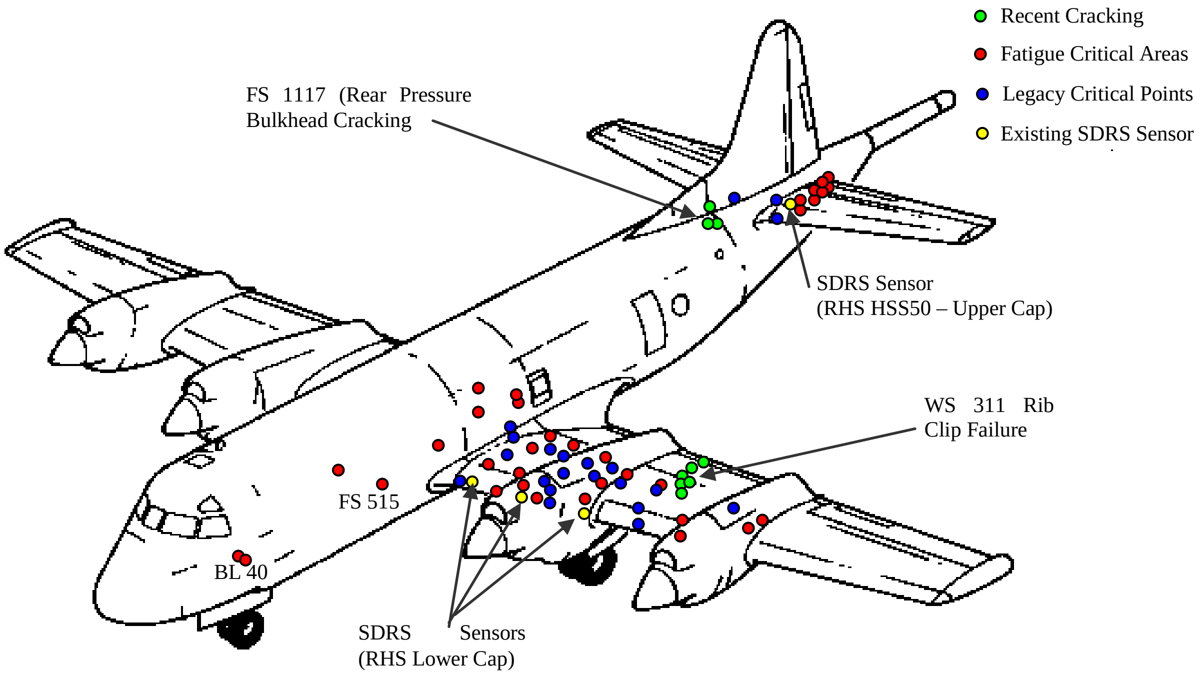


Figure 11. Schematic of installed FTI locations

Future Analyses and Safe Operating Life (SOL):

In the next few years, in satisfaction of the second objective, IMP will continue validation analysis of the remaining critical structural locations (i.e., CF Structural Significant Items (SSI), Fuselage Station 1117, and Wing Station 311) in much the same fashion as the SDRS gauges.

Following validation of the SDRS and remaining critical locations, it will be necessary to develop Stress to Load transfer functions for each location. This information will be imported into the Individual Aircraft Tracking software (C-IAT), which uses recorded flight data to track fatigue and crack growth damage specific to the aircraft usage and specific to the structural configuration of each tracked location.

The Fleet Average Reference Spectrum (FARS) will be developed based on the vast set of previously recorded SDRS data (56,000+ hrs). This will be used as both a comparative tool to the USN 85th spectrum and as a predictive tool for Fatigue and Crack Growth Analyses. The FARS along with the associated transfer functions will be used to reprocess the SSI locations prior to their integration into the C-IAT. These updated tools will then allow the Safe Operating Life (SOL) of the fleet to be re-evaluated.

Maritime Helicopter Program ASIP Service

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

L-3 MAS is currently under contract with Sikorsky International Operations Inc. to provide an Aircraft Structural Integrity Program (ASIP) Service for the Maritime Helicopter Program (MHP). L-3 MAS will design and develop a Service and related enabling systems to execute ASIP activities on the Canadian Forces H-92 fleet for the next 20 years. The purposes of the ASIP Service are:

- Establish, evaluate and substantiate the structural integrity of MH structure, engines and dynamic components,
- Provide continual assessment of the in-service integrity by collecting and utilizing the operational usage and structural condition data,
- Identify structural integrity related technical problems and investigate assigned technical problems.

Two enabling systems (ES) will be part of the MH ASIP Service to execute designated ASIP activities during the in-service support timeframe. The first ES is called the Structurally Significant Items (SSI) database which will allow collecting and storing the SSI information from the Original Equipment Manufacturer (OEM) design data to the in-service support structural condition data. The second ES is called the Usage Monitoring Tool (UMT). It allows us to collect, analyze and interpret the operational usage data as part of the helicopter usage monitoring and actual usage comparison with the design baseline. Periodic reports and recommended corrective actions will be provided to the MHP management office.

Fatigue Enhancement Technologies

M. Yanishevsky NRC-IAR-SMPL

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 such as cold expansion of fastener holes prone to fatigue cracking to improve their fatigue performance have been performed.

Since it is difficult to quantify the residual stress profiles resulting from hole cold expansion, 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 this technology, where it can provide fatigue enhancement or where it could be considered as a repair to enable service beyond the original design expectation.

Work on this program was reported on previously in the ICAF 2007 Canadian National Review. During the last two year period, efforts have been focused on evaluating the performance of coupons representing typical scenarios found in aircraft structures: No Load Transfer (NLT) situations, where non-structural brackets support fuel lines or electrical wire bundles); Medium Load Transfer (MLT) situations, representing joints with doublers, stiffeners and ribs); and High Load Transfer (HLT) situations, representing the attachment of major load carrying members). These are shown schematically in Figure 12.

Aluminium alloy 7075-T6 1/4 inch (6.35 mm) thick plate was evaluated with doublers made of 7075-T73 1/8 inch (3.175 mm) plate. The coupons had 1/4 inch (6.35 mm) holes machined and at varying edge distances ranging from e/D=2.0, 1.5, 1.2, 1.0 and 0.8. The coupons were assembled with sealant using Hi-lok HL50 interference fit fasteners; the holes being as-machined or cold expanded using FTI proprietary split sleeve 8-O-N-1 CxSS tooling. Though several coupons without hole cold expansion did fail in fatigue, some did exhibit significantly higher fatigue lives just by cracks nucleating by fretting mechanisms at distances away from the “fatigue limiting” holes. Indications are that “well assembled” joints that transfer load by friction / clamp do not fail by fatigue due to pin loading; instead they tend to develop fretting fatigue cracks away from the hole where these clamp up forces are high. Joints with hole cold expanded holes tended to fail by the development of fretting fatigue cracks, again at a distance away from the holes.

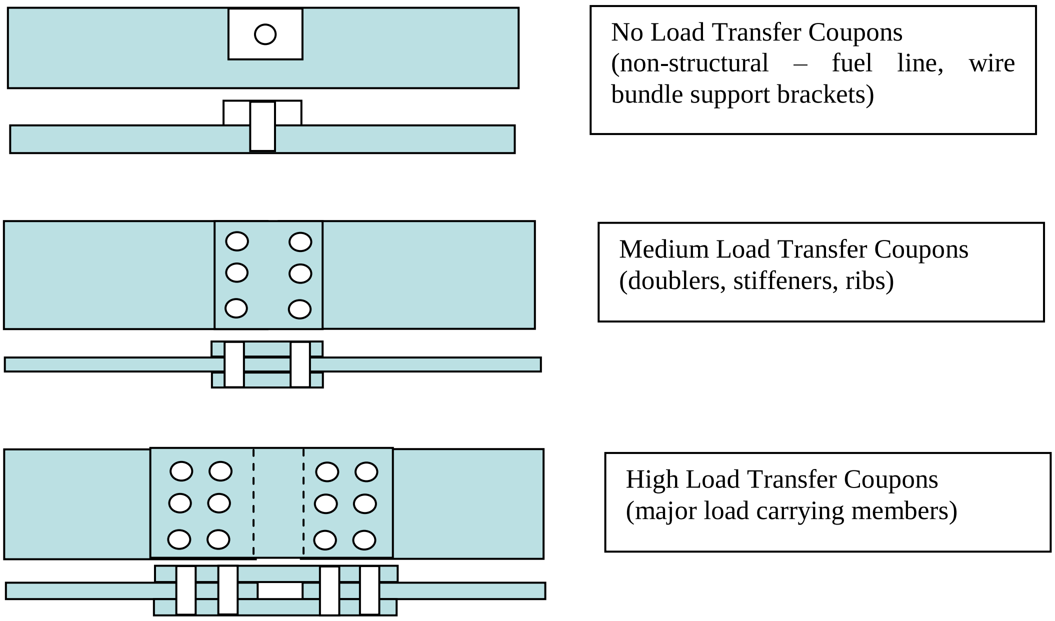


Figure 12. Schematics of No Load, Medium Load and High Load Transfer Joint Coupons.

The Hi-lok HL50 nominal 1/4 inch (6.35 mm) fastener and nut interference fit process was modeled and evaluated for several e/D scenarios as shown in Figure 13. Hole interferences of 0, 0.001, 0.0035 and 0.005 inches (0, 0.0254, 0.889 and 0.127 mm) were examined and it was determined analytically that of the scenarios evaluated, the 0.0035 inch (0.889 mm) interference provided the optimum assembly conditions.

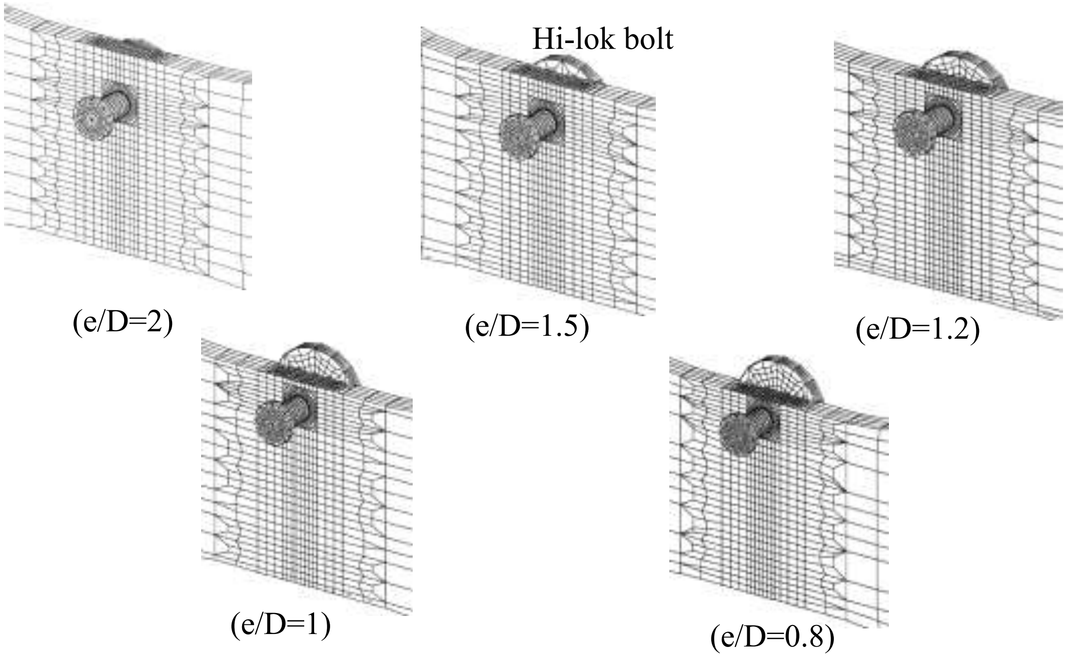


Figure 13. Schematics of the No Load Transfer Interference Fit FE Models.

Proto Manufacturing Ltd X-Ray Diffraction (XRD) elastic residual stress measurements were used to quantify the residual stresses resident in the 7075-T6 1/4 inch (6.35 mm) rolled plate as well as the residual stresses caused by FTI 8-0-N-1 hole cold expansion tooling on both the entrance and exit sides, as shown in Figure 14 and Figure 15. Similar measurements were taken using Digital Image Correlation (DIC), which was able to calculate the residual elasto-plastic strains, shown in Figure 16. Both XRD and DIC were used to quantify the effect of cold expanding four close proximity holes with Hi-lok interference fasteners with 0.0035 inches (0.889 mm) interference. An example of a DIC image is provided in Figure 17.

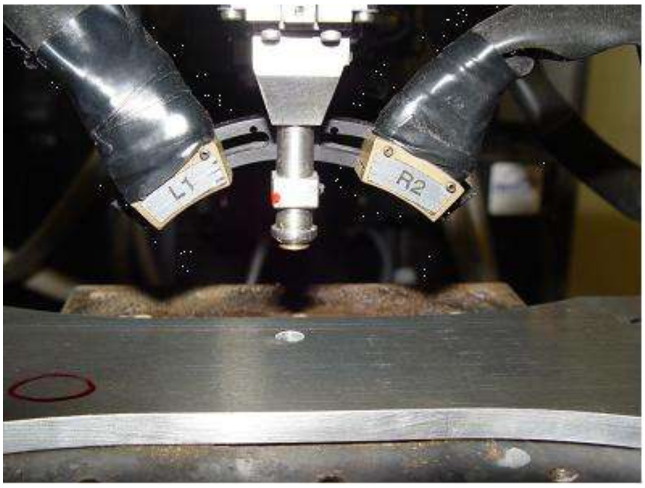


Figure 14. The No Load Transfer Coupons with Cold Expanded Hole being Evaluated for Residual Stresses using X-Ray Diffraction Technology (Proto Manufacturing Ltd.).

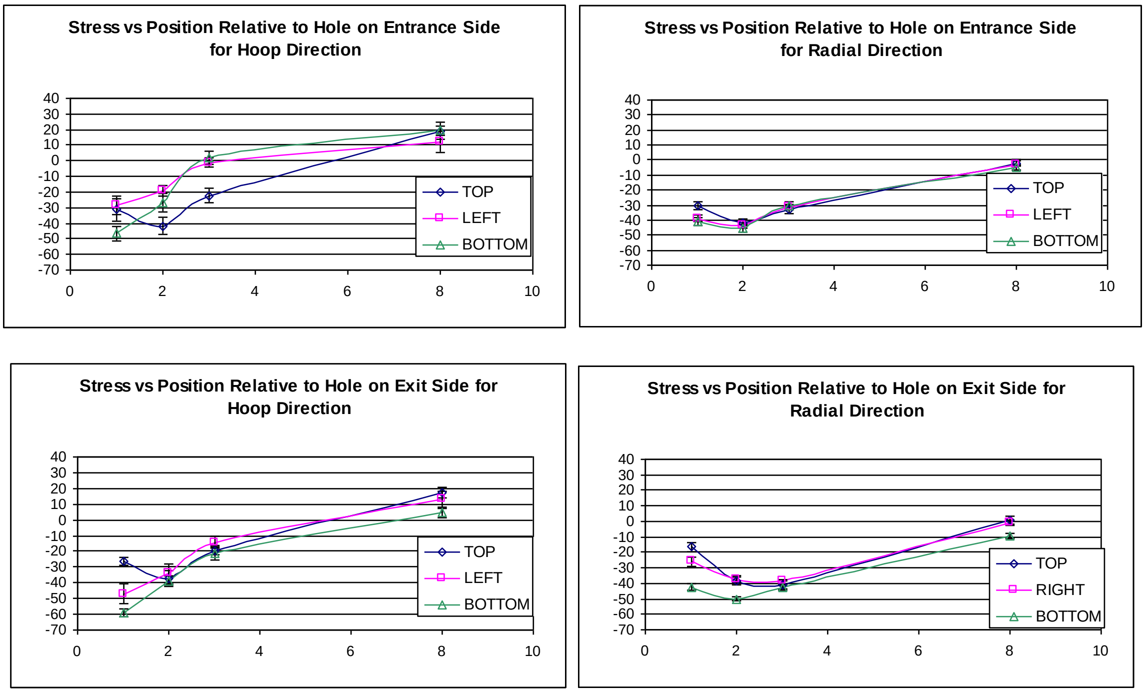


Figure 15. X-Ray Diffraction Measurements Taken on No Load Transfer Coupons to Measure the Residual Stresses Created by Hole Cold Expansion for e/D=2.0 Holes.

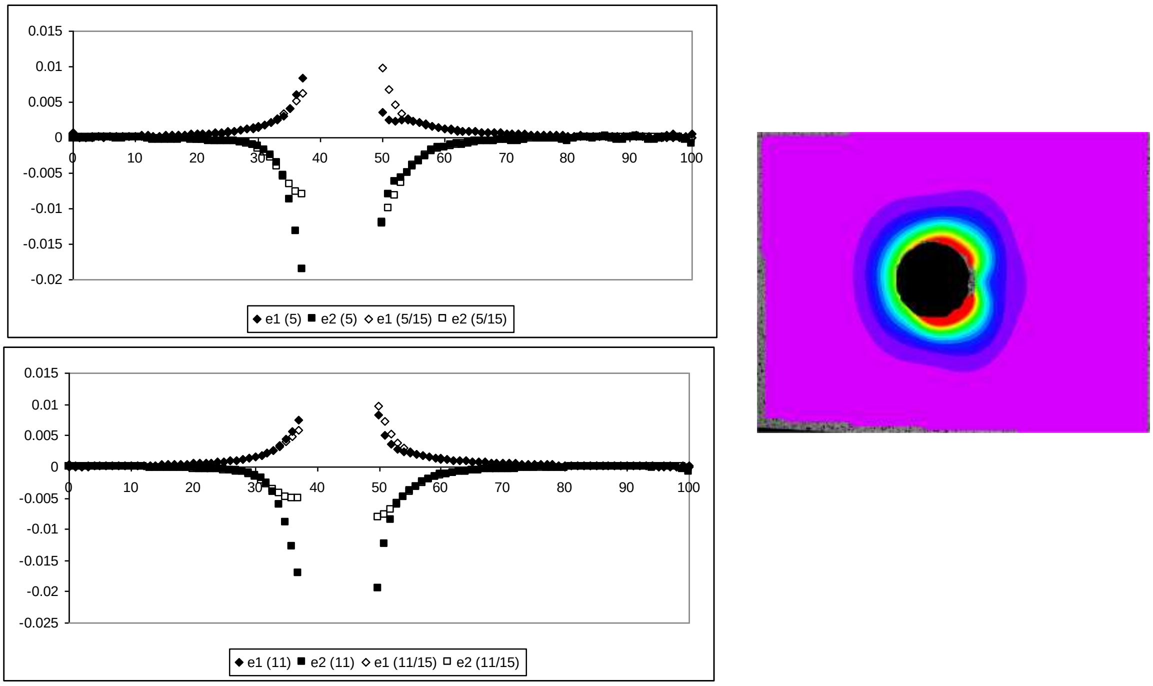


Figure 16. Digital Image Correlation Residual Strain Measurements Caused by Cold Expanding Holes - the Effect of the CxSS Split in the Sleeve is Evident in the Right Hand Image.

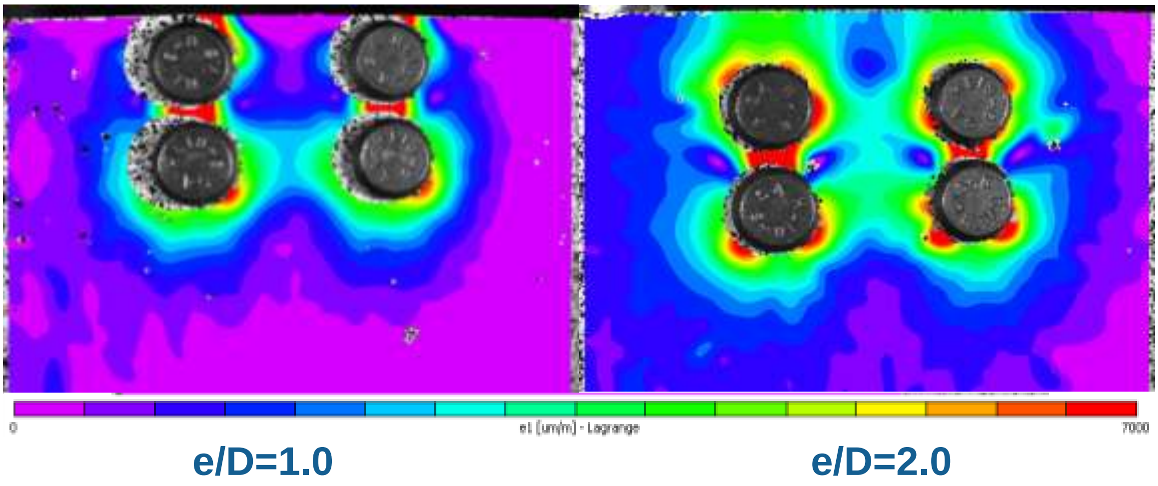


Figure 17. Digital Image Correlation Strain Measurements Caused by Cold Expanding Four Holes and Inserting Hi-Lok HL50 Interference Fit Fasteners.

Finite element (FE) modeling tools were developed using the Patran Command Language (PCL) in order to study the residual stress profiles induced by the FTI cold expansion process. Simulations representing 8-0-N-1 tooling for single holes at a variety of e/D margins were conducted, the results providing an understanding of the cold expansion process including differences between the entrance, mid-plane and exit side residual stress characteristics (these were reported previously in the ICAF 2007 Canadian National Review). Simulations of up to four holes were also conducted. These simplified models used uniform displacement scenarios.

A comparison between DIC and FE modeling results demonstrated that there was little effect when holes were spaced at 4 hole diameters from each other; however, significant effects were seen when the hole spacings were reduced to 2 hole diameters. The DIC measurements were found to be significantly lower near the hole edge; this is primarily due to the fact that the DIC technique is not able to measure strains in close proximity to free edges, an example of which is shown in Figure 18.

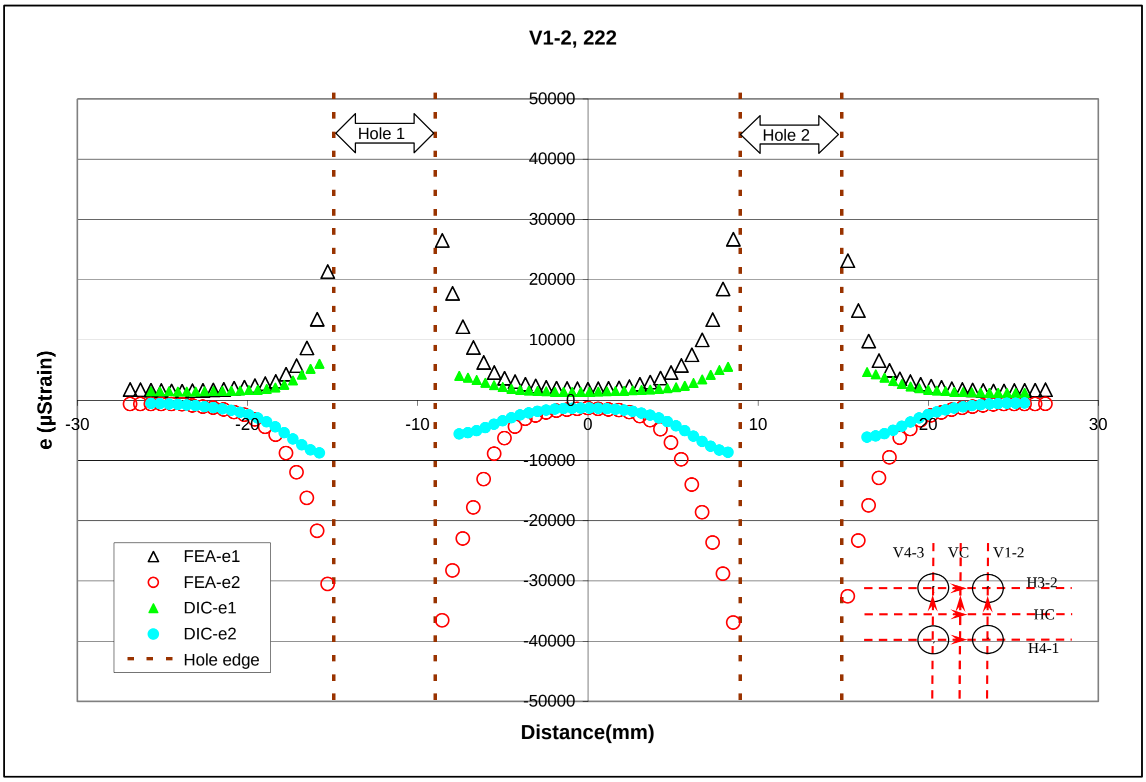
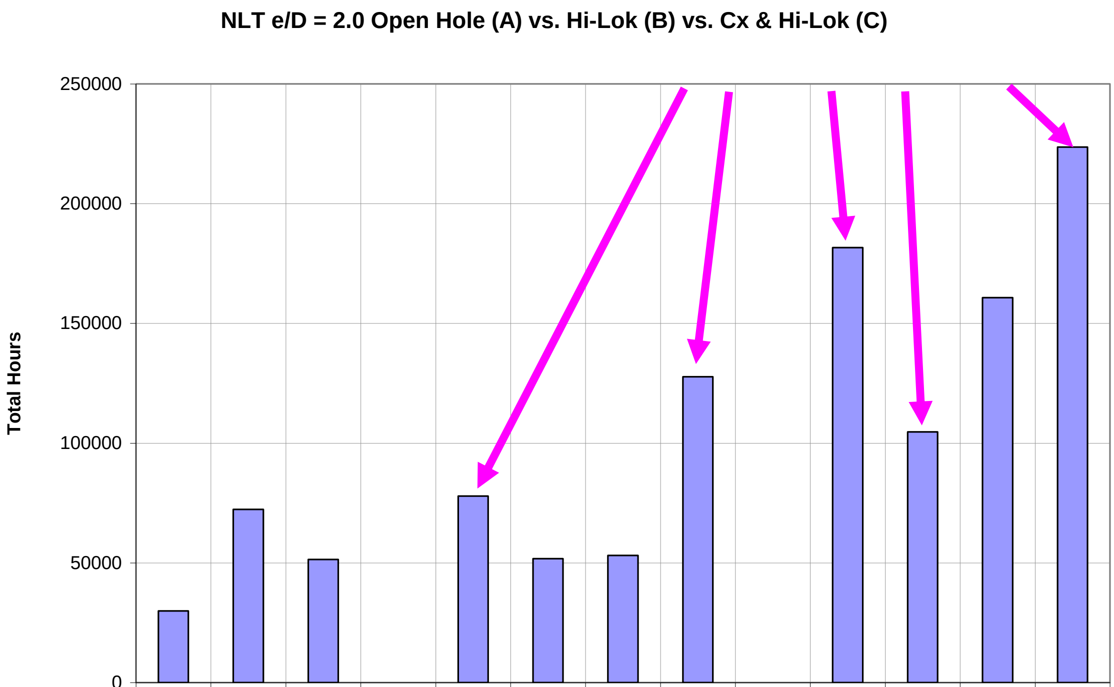


Figure 18. Comparison of DIC and FE Results for Four Hole Coupons with Close Proximity Holes.

Experimental testing is progressing using a Canadian Forces spectrum representative of CP140 Aurora (P-3 Orion) usage. An example of the effect of hole condition on No Load Coupons for e/D=2.0 is shown in Figure 19.



	A01	A02	A03		B02	B03	B04	B05		C01	C02	C03	C04
Total Hours	29945	72293	51402		77871	51739	53096	127715		181615	104713	160725	223641

Figure 19. Comparison of e/D=2.0 NLT Coupon Results: (A) Open Hole; (B) Hole Filled with Interference Hi-lok; and (C) Cx'd Hole with Interference Fit Hi-lok. (Arrows are pointing to data points where test failures occurred away from the hole design detail.)

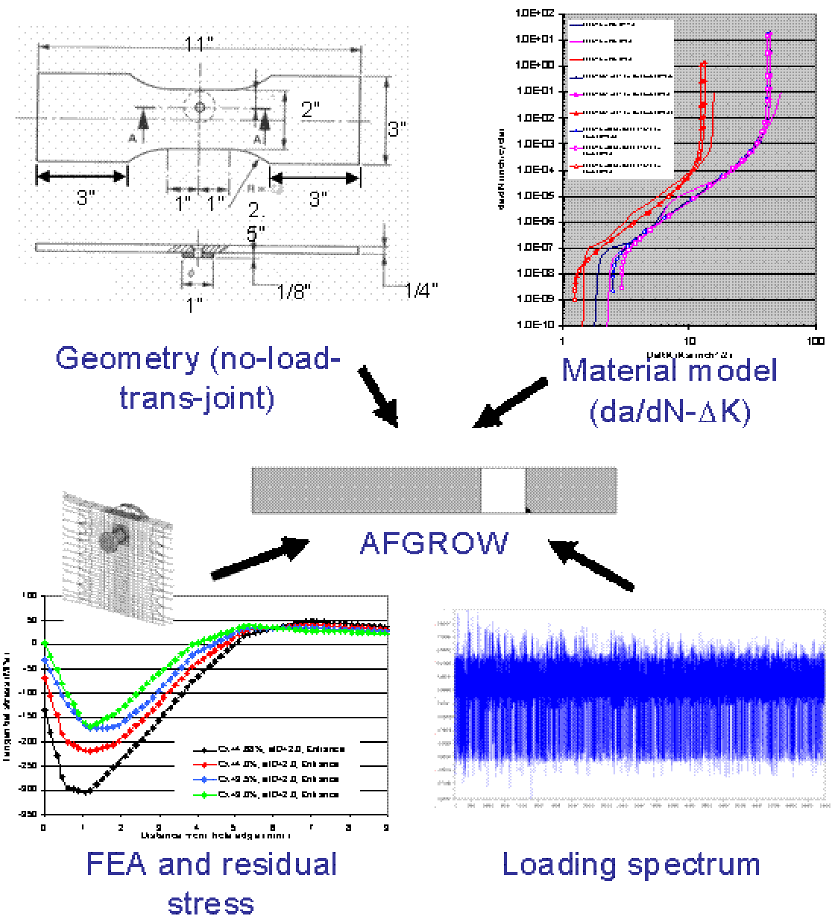


Figure 20. Schematic Depicting the Fracture Mechanics and FE Based Lifing Process.

An analytical approach for predicting the performance of holes that have both experienced hole cold expansion with interference fit fasteners is being developed to forecast the performance of holes and joints processed using FTI CxSS technology. A schematic of the approach is described in Figure 20.

The results of these 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.

Fatigue Life Enhancement of Fibre Metal Laminate Materials as a Result Of Hole Cold Expansion

D. Backman NRC-IAR-SMPL

A fatigue life enhancement project has been ongoing that has been looking at measuring the residual strain fields in FML after cold expansion as well as looking at the improvements in fatigue life as a result of cold expansion.

In the first phase of this program digital image correlation methods were used to measure the residual strains in FML after cold expansion and compare them to the residual strains in monolithic aluminum [1,2,3]. Figure 21 shows the difference in residual strains between aluminum and FML during cold expansion while Figure 22 shows the difference in residual strain as a result of riveting.

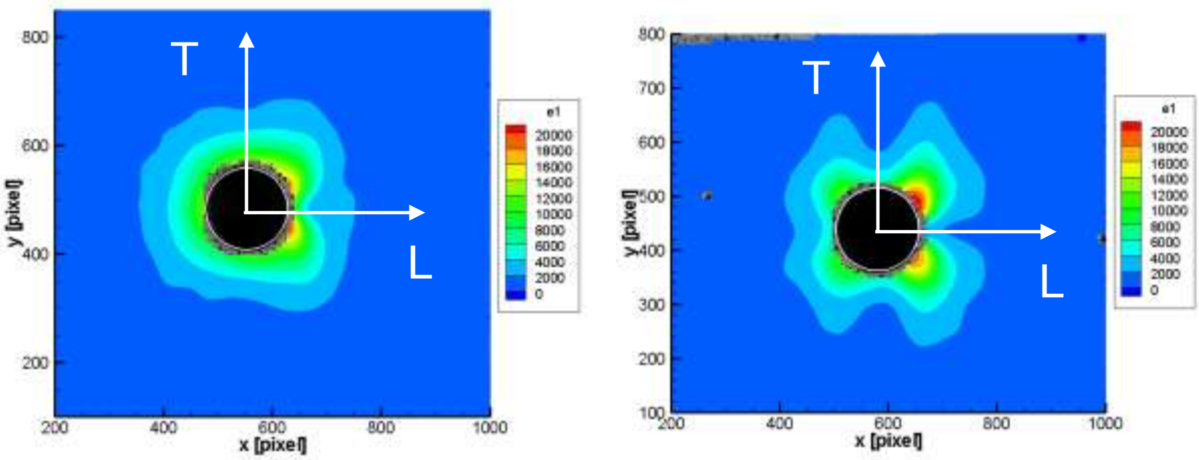


Figure 21. First principal strains 2024-T3 aluminum (left) and FML 3-3/2 (right)

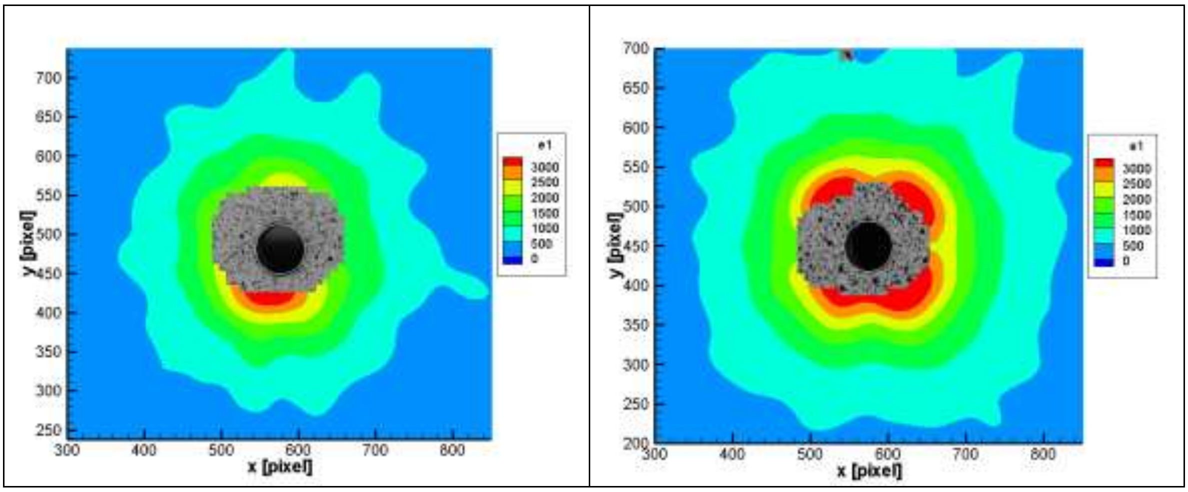
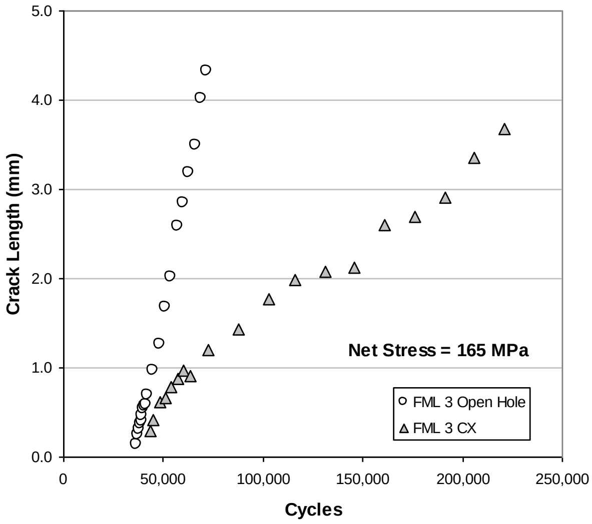


Figure 22. First principal strains due to riveting in (left) aluminum and (right) FML.

The next phase of the program investigated the fatigue life of FML after the cold expansion process. The initial hypothesis is that cold expansion would retard the growth of short cracks, one of the inherent problems with this material. The results so far (Figure 23) show that although cold expansion is extremely effective at extending the fatigue life of FML, it does very little to retard the early formation of these short cracks when compared to a standard open hole coupon [4,5].



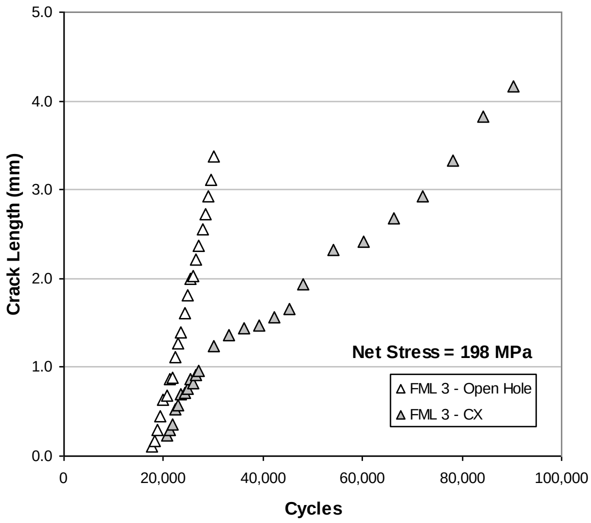


Figure 23. Crack growth curves coupons with an open hole and a cold-expanded hole (CX) at applied stress165 MPa and 198 MPa

FRACTURE MECHANICS AND CRACK PROPAGATION STUDIES

Multiscale modeling of cracks in polycrystal metals

G. Shi, K. Chen, M. Liao, G. Renaud and N. Bellinger, SMPL-IAR-NRC

The primary objective of this project is to develop a coupled atomic-meso-macroscopic modeling approach to study the basic mechanisms of crack nucleation and growth in aircraft components.

At the atomic-scale level, *ab initio* density functional theoretical calculations for shear and tensile behavior of aluminum were carried out. Some specific coincidence site lattice (CSL) types of grain boundaries (GB) were constructed, and the calculated decohesion and sliding strengths for these grain boundaries will be used to assess crack nucleation at these specific grain boundaries. Furthermore, the generalized stacking fault energies (intrinsic, extrinsic, stable and unstable stacking fault energies) were calculated, in which the results can be used to assess the dislocation nucleation at the crack tip. In addition, cohesive model represented by a relationship of stress vs. strain for Al along the $\sum 5$ grain boundary was also calculated. This calculated cohesive model will be used as an initial input for the next meso-scopic level modeling on crack propagation in aerospace components. The cohesive model in Al-based alloys on other CSL symmetrical tile grain boundaries is being conducted.

Large-scale paralleled (MPI) molecular dynamics simulations on crack nucleation along grain boundary in Al-based alloys (Al and Al-Mg alloys) were conducted. The inter-grain crack nucleation at CSL grain boundaries was investigated. The research focuses on the interactions between different types of dislocations (edge and screw types) and grain boundaries (GB), based on which an empirical formula on crack nucleation could be proposed. In addition, a DSC (Displacement Shift Complete) type of dislocation model for CLS grain boundary was also generated. This includes a network of GB dislocations created by a small deviation from a perfect CSL orientation and an edge dislocation on a DSC lattice at GB. In addition, multiple intra-grain edge or screw dislocations away from a GB were also generated. These three scenarios cover defects caused by intrinsic GB mismatch, GB sliding and plastic deformation of a grain, in respected order. A physics understanding will be extracted for each scenario and acquire appropriate simulation data accordingly.

To convert Orientation Imaging Microscopy (OIM) data files into Patran/Nastran Finite Element (FE) models, a Patran Command Language (PCL) modeling procedure was written. This technique can be used to create highly detailed 2D meso-scale models based on measured data, therefore representative of real grain layout and orientation.

To build up microstructural geometry models using Voronoi techniques, the MicroSimu 2D software and 3D grain structure model were purchased. The preliminary procedures were established for building the microstructural models for typical aircraft aluminum plates, in association with the OIM analysis results on grain size and orientation.

Based on the grain structure geometry model, a PCL program was written to randomly-generate 2D grain structure FE models. Because this approach is not based on physical measurements, it allows for parametric studies based on grain size, grain shape, and grain orientation distributions.

In order to use the cohesive zone model for the study of crack nucleation and propagation, techniques for the implementation of the cohesive elements in grain boundaries were developed. A PCL program was developed for the generation of cohesive models, which includes the automatic insertion of cohesive elements between grain boundaries, assignment of grain orientation based on random angles, properties of cohesive model and criteria for crack nucleation and evolution.

Preliminary studies on inter-granular crack nucleation and evolution using two-dimensional finite element cohesive models which were generated by the Voronoi tessellation method and PCL program were carried out at meso-scale level. Stable convergence procedures were investigated and identified. Parametric studies on the material properties in grains and criteria of crack nucleation and evolution on grain boundaries were conducted.

Experimental techniques/results on micro-crack, designing of test coupon and test plan were reviewed for generating crack nucleation/short crack data. Fatigue coupon (smooth) was designed and 10 coupons were manufactured. Metallurgic polishing was completed for the specimens, and NRC paint-on crack sensors were installed.

A laboratory technical report (LTR) on dislocation theory based short crack and crack nucleation model for 2024-T351 was published[6]. A preliminary investigation on Tanaka and Mura dislocation theory based nucleation model was carried out to determine the fatigue crack nucleation life of 2024-T351 AA, and some key microstrucural parameters for further multi-scale modeling efforts were identified. Also, a paper on dislocation theory based short crack growth modeling was published and presented in the conference ICHMM08[7].

A collaborative project with McGill University on “Effects of Microtexture and Grain Boundary on Fatigue Crack Nucleation and Growth in Aircraft Aluminum Alloys – Phase II” was conducted. A number of 7050 samples were supplied to McGill University for the OIM analysis. Two progress reports on grain size distribution, grain aspect ratio measurement was completed on 7050-T7452 forging, and microstructural data are produced for modeling purpose. Preliminary OIM results on 7050-T7452 fatigue samples were achieved on the grain size and misorientation angles near the crack nucleation area.

An in-kind research collaboration was initiated with University des Saarlandes (Germany) on “Effect of grain boundary on crack nucleation and small crack growth”. One aluminum sample and one fractured sample were selected and shipped to this university for OIM analysis. The preliminary Orientation Gradient Map (OGM) results were first obtained on the sample away from fatigue crack, which showed no plastic deformation. The next step will move on to the crack nucleation region.

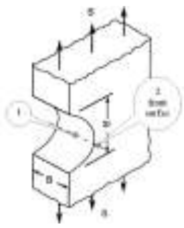
Short Crack Model Development for Primary Airframe Materials

M. Liao, NRC-IAR-SMPL

A three-year (2005-2008) collaborative DND-NRC project was carried out to further develop the HOLSIP (holistic structural integrity process) framework by concentrating on the short/small crack phase. Both experimental techniques and analytical methods were developed to generate databases and carry out a short crack analysis, which has been included in the HOLSIP framework [8],[9]. 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 major achievements are highlighted as follows.

A material characterization was completed on 7050-T7452 forging (CF-18 primary material) to determine the material IDS/particle, pore size, grain size and orientation distribution. The material library is established [10],[11], [12]. In total, over 62 single edge notched tension (SENT) coupons have been tested under 5 stress ratios and 6 stress levels constant amplitude loading, inserted with marker band loading sequences [8]. A brief residual stress survey was carried out and indicated that varied residual stresses were presented in the test coupons (Figure 25) [13]. Maker band readings successfully recovered most short crack growth data (Figure 24). Fractographic analysis was carried out to measure the crack-nucleating IDS (initial discontinuity states) features, i.e., particles and pores (Figure 25). The marker band crack growth data was used to develop the short crack growth rate database and to help determining accurate stress intensity (K) solutions for short cracks in association with three-dimensional (3D) FE modeling (StressCheck) [8]. Finally, the short-long crack growth da/dN-ΔK curves were developed for the 7050-T7452 forging, as well as the 7050-T7451 plate, in associated with the physically measured crack-nucleating IDS/particles/ pore. The short-long crack models were first verified using NRC and DSTO coupon tests, under constant loading spectrums, and then applied for a CF-18 component life assessment.

Besides of the fracture mechanics based model, the Tanaka-Mura dislocation model for short crack growth was also investigated for 2024-T351 in this project, which showed the promising results on simulating microstructural (e.g. grain boundary blocking) effects on short crack growth (Figure 27) [14].



XRD results

Location	Residual stress (ksi)
1 (center)	Means (-5, +3), Errors (±0, ±4)
2 (side)	Means (-8, +2), Errors (±0, ±3)

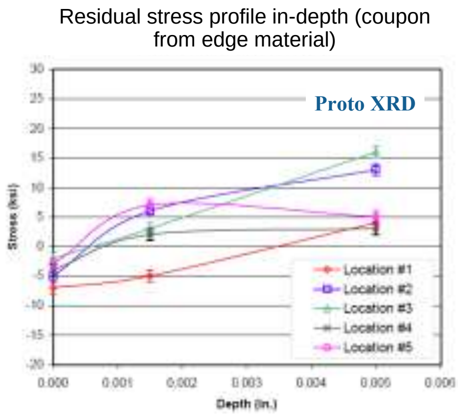


Figure 24. Residual stress measurements on the 7050-T7452 (forging) coupons.

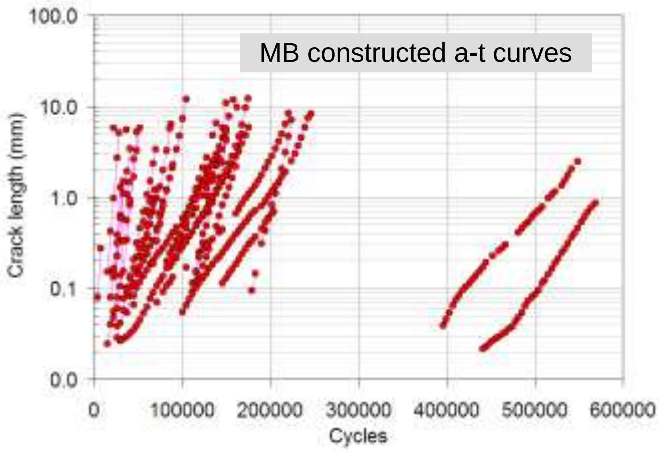


Figure 25. Crack growth data reconstructed from marker band reading.

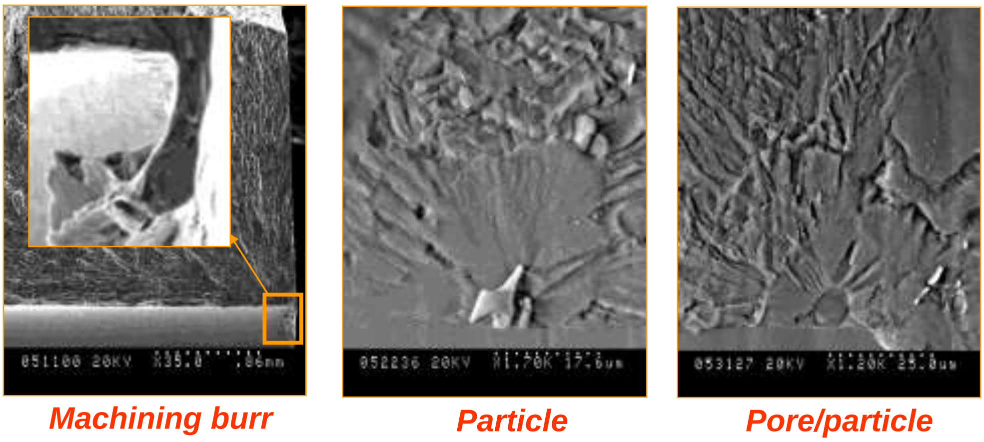
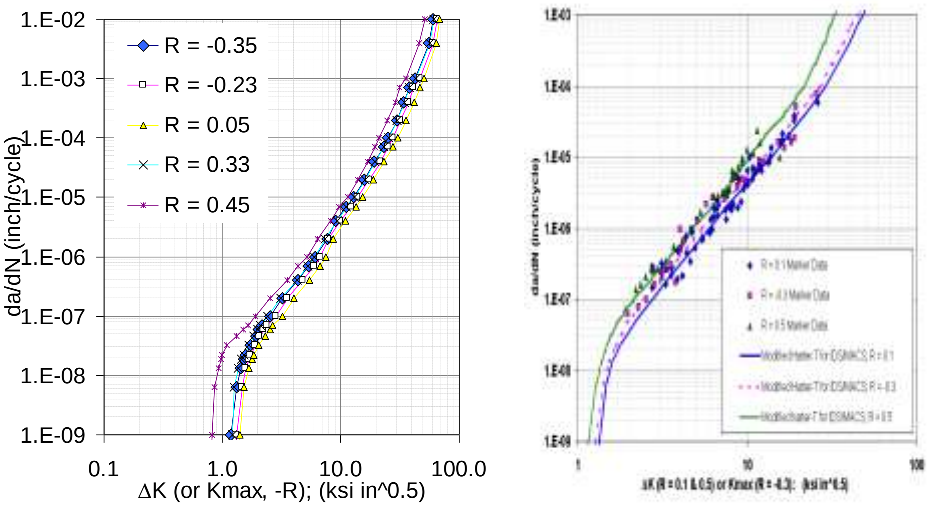


Figure 26. Post–fracture analysis to identify and measure crack-nucleating features.



7050-T7452 (forging) 7050-T7451 (plate)

Figure 27. Short-long crack growth da/dN-ΔK curves for 7050 forging and plate.

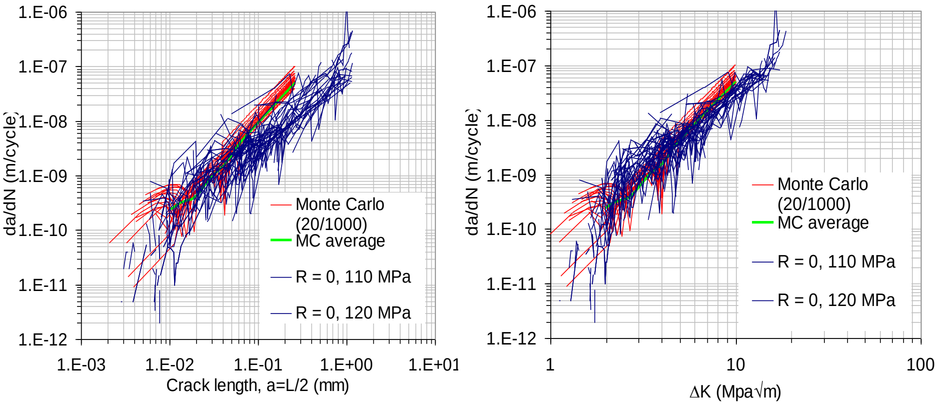


Figure 28. The relationships between da/dN and a , da/dN and ΔK for 2024-T351, dislocation modeling (Monte Carlo) vs. test (AGARD 1982) results.

Short Crack Model Development for Helicopter Structural Life Assessment

M. Liao, NRC-IAR-SMPL

It is being realized that the traditional safe-life (SL) approach is overly conservative and very costly for helicopter structural life management. In addition, cracking and/or environmental related damage (e.g., corrosion) in helicopter structures often challenge the SL philosophy. The conventional damage-tolerance (DT) approach has been shown to be very difficult (almost impossible) to apply to helicopter structures, mainly due to: 1) the initial flaw assumptions of the DT approach are improper (too large) for most helicopter fatigue critical locations; 2) the main failure mechanism for helicopter components is high cycle fatigue (HCF) with significantly long lives of crack nucleation and short/small crack growth which are not addressed by the DT approach. Presently the flaw-tolerance (FT) approach is being used by some manufacturers to address regulatory agencies' requirements on the damage tolerant capability in helicopter design, but this approach has similar inherent issues as both the SL and DT approaches (with high scatter factors and improper initial flaw assumptions). A recent Helicopter Damage Tolerance Round Robin Challenge program, organised by Cranfield University, concluded that the large scatter in short crack growth rate data and the lack of accurate three-dimensional stress intensity factor solutions result in inaccurate life prediction and incorrect analytical crack growth curves (shape). Overall, none of these approaches have been able to address the reliability/risk issues of helicopter structures, in a quantitative manner.

The HOLSIP (holistic structural integrity process) framework and the physics-based models contained within are applicable to both fixed and rotary wing aircraft. Some HOLSIP models have been verified against experimental results and fleet case studies for aluminum components. In the proposed project, both test databases and short/small crack (near threshold region) models will be developed. In association with other projects on helicopter usage and quantitative risk assessment, the HOLISP framework will be extended to provide a better/accurate life assessment for helicopter structures.

A three-year collaborative DND-NRC project was begun in 2008 to further develop Short Crack Model Development for Helicopter Structural Life Assessment. The objective of this 3-year project is to further develop the physics-based short/small crack models for better life assessment of helicopter structures. Both experiments and analytical modeling will be carried out on the New Maritime Helicopter structural materials to generate the necessary databases, and then be integrated in the HOLSIP framework for helicopter structural life assessment. Details and an overview are expected to be presented in the next ICAF review.

Interaction of Fatigue with Time-Dependent Dwell/Creep Damage

X. Wu, NRC-IAR-SMPL

The requirements of gas turbine engine components operating at higher and higher temperatures drives the failure mode into a creep-fatigue regime. IAR/NRC has conducted a systematic study of modeling the interaction of fatigue with time-dependent dwell/creep damage [15,16]. A nonlinear creep/dwell-interaction model is derived based on the mechanism of fatigue crack nucleation and propagation in coalescence with creep/dwell damages (cavities or wedge cracks) along its path, which leads to the total damage accumulation rate as given by

$$\frac{da}{dN} = \left(1 + \frac{l_c + l_z}{\lambda}\right)\left\{\left(\frac{da}{dN}\right)_f + \left(\frac{da}{dN}\right)_{env}\right\} \tag{1}$$

where $(da/dN)_f$ is the pure fatigue crack growth rate, $(da/dN)_{env}$ is the environment-assisted crack growth rate, l_c/l_z is the size of distributed creep/dwell damage such as cavity and wedge cracks, and λ is the average spacing between them.

In the above formulation, dwell damage at low temperatures are envisaged as wedge cracks in the form of dislocation pile-ups, and creep damage in polycrystalline materials at high temperatures are produced by grain boundary sliding. Therefore, Eq. (1) covers the creep/dwell-fatigue phenomena over a broad temperature range of engineering concern. In particular, the model has been used to explain dwell fatigue of titanium alloys and high temperature creep-fatigue interactions in Ni-base superalloys under various loading cycles with stress/strain holds at the peak loads. Figure 29 shows the dwell-time effect on fatigue of IMI 834. Figure 30 shows the effect of creep on cyclic life in comparison with the baseline fatigue life and the description of the model.

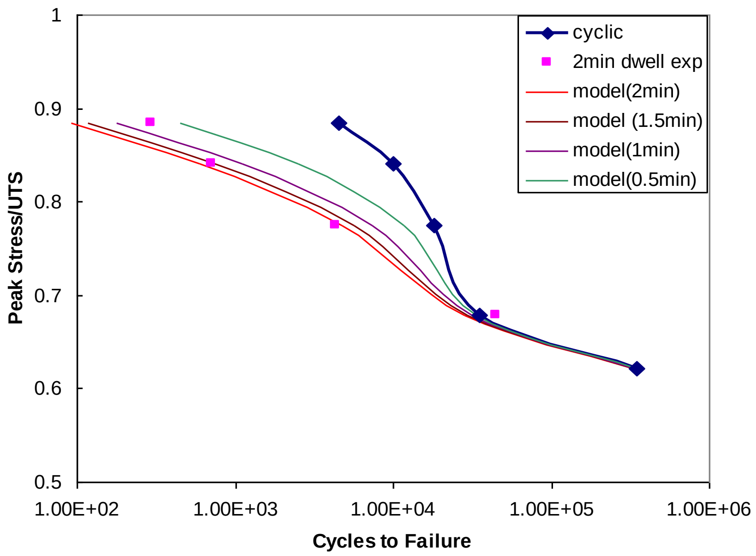


Figure 29. Comparison of the model with experimental data on IMI 834.

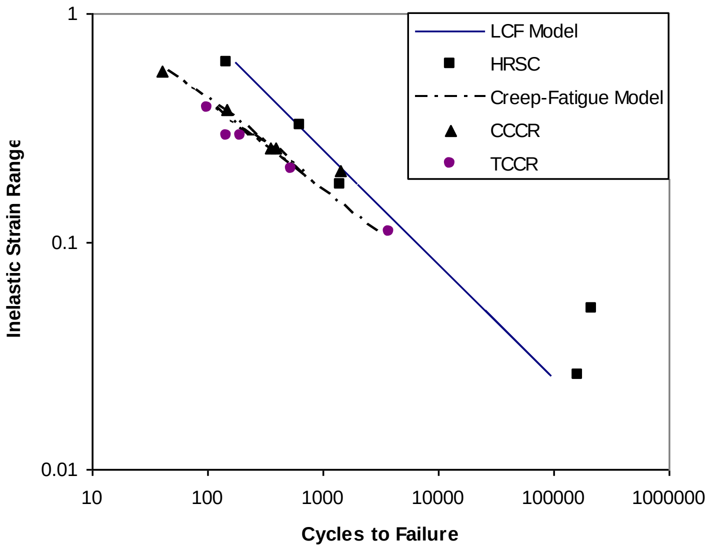


Figure 30. Comparison of the model with experimental data for Rene 80 at 871oC, where HRSC signifies high rate strain controlled cyclic test, CCCR signifies compressive cyclic creep rupture test, and TCCR signifies tensile cyclic creep rupture test.

PROBABILISTIC AND RISK ANALYSIS METHODS

Advanced Risk Assessment Methodologies for Aircraft Structures Containing MSD (multi-site fatigue damage)/MED (multiple element damage)

M. Liao, NRC-IAR-SMPL

A risk based management approach/tool has been adopted by many military air fleets. In the past few years, the Canadian Forces (CF) have been introducing and revising a Record of Airworthiness Risk Management (RARM) process to manage technical and operational airworthiness for all CF aircraft. Today, the RARM has become the single most critical decision making tool in the CF air fleets. In RARM, both *qualitative* (defining hazard probability as ‘frequent’, ‘remote’, ‘extreme improbable', etc.) and *quantitative* (defining hazard probability as '10^{-3}', '10^{-5}', '10^{-8}', etc., per flight hour) risks are defined for all CF aircraft platforms, including unmanned air vehicles (UAVs) and helicopters. When there are sufficient data available, a *quantitative* risk assessment (RA) can be performed to substantiate the assignment of a risk index number. When a *qualitative* RA indicates a high/medium risk, a detailed *quantitative* RA is often demanded to support better decision-making. However, some challenges remain for a *quantitative* RA, such as how to establish methodologies to get sufficient/meaningful input data, particularly for a complicated damage scenario such as multiple site fatigue damage (MSD)/multiple element fatigue damage (MED).

Upon request by the Department of National Defence (DND) of Canada, NRC was tasked to develop advanced methods for carrying out risk assessment of build-up structures. The lower surface panel of the CC-130 centre wing, shown in Figure 31, was selected as a case study for this work. It is known that the MSD and MED related structural failure of the centre wing caused a catastrophic accident on a C-130A in Walker, California (1997).

Advanced methodologies were developed for carrying out durability and damage tolerance analysis and *quantitative* RA for built-up structures with MSD/MED [17],[18]. The methodologies included 1) a beta library for MSD/MED cracks in which most beta solutions were verified with StressCheck FE and closed form solutions (Figure 32), for both the standard and MSD crack scenarios (Figure 33); 2) crack growth analysis with MSD/MED interactions (Figure 33); 3) MSD/MED residual strength analysis; and 4) MSD/MED crack growth Monte Carlo simulation. All these advanced methods were implemented in a crack growth software, CGCC130MSD, developed by NRC. To carry out the MSD/MED risk analysis, an EIFS distribution was developed based on in-service inspection data [19][20]. Finally, an NRC risk analysis code, ProDTA, was revised to process the crack size distribution from the MSD crack growth program in order to calculate the PoF for build-up structures. In this wing panel case study, the PoF for the MSD scenario is significantly higher than the PoF for the standard crack scenario, especially when the structure is near the end of its service life (Figure 35). Future work includes the use of large-scale structure test results to validate the NRC MSD/MED analysis tools, in collaboration with other organizations.

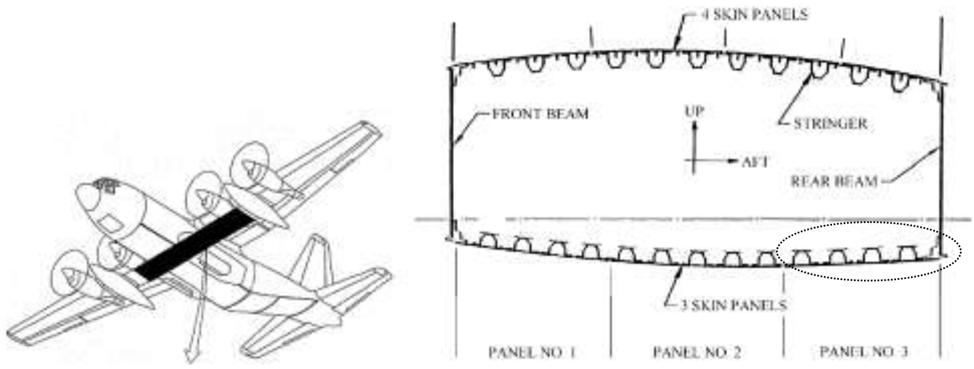


Figure 31. CC-130 centre wing box, and the lower surface panel (circled, panel No. 3) studied

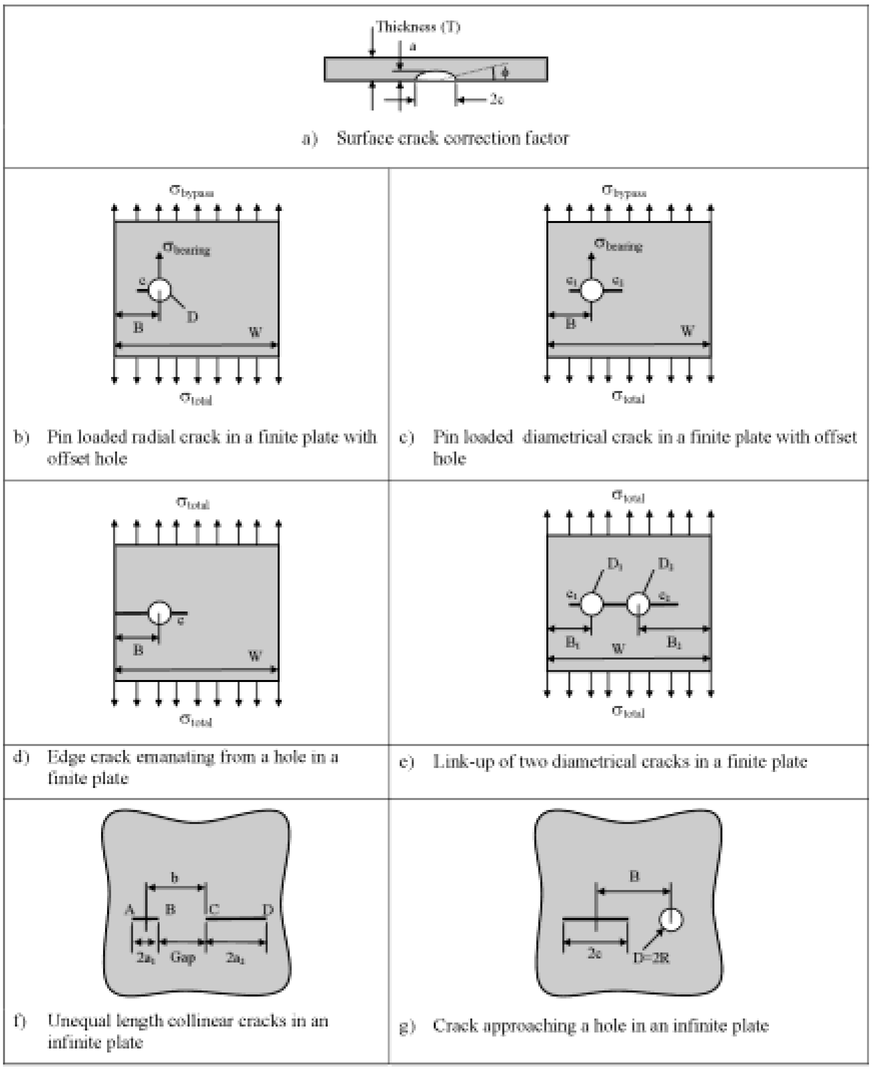


Figure 32. Schematic representation of the β-factors used to solve MSD problem by the principle of superposition

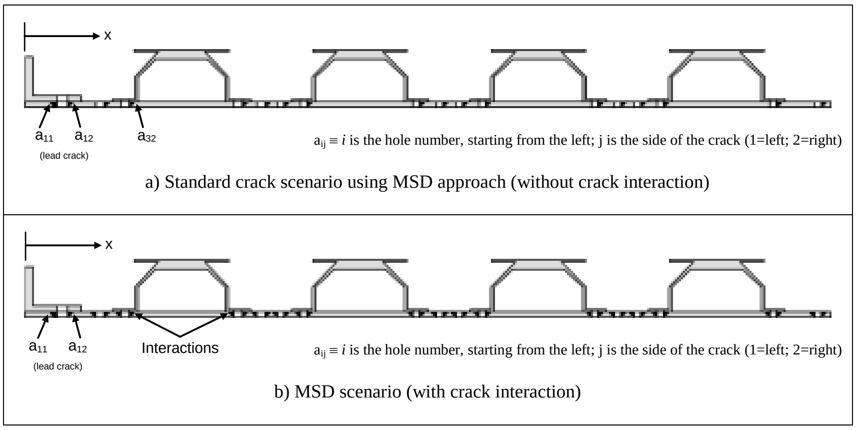


Figure 33 Standard crack scenario versus MSD scenario

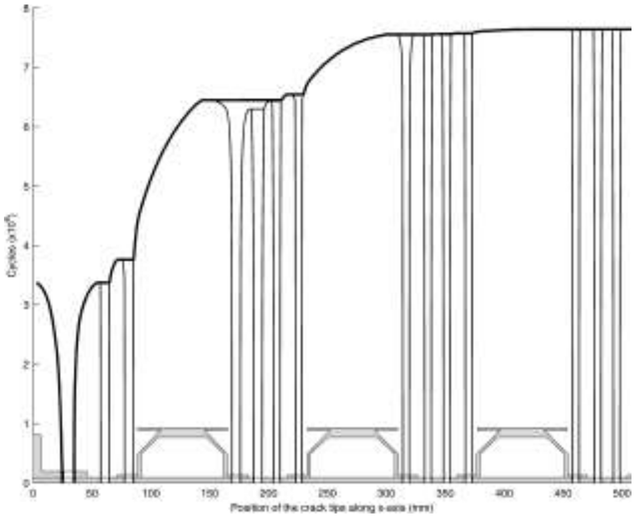


Figure 34. Crack growth curves for the MSD scenario (all cracks)

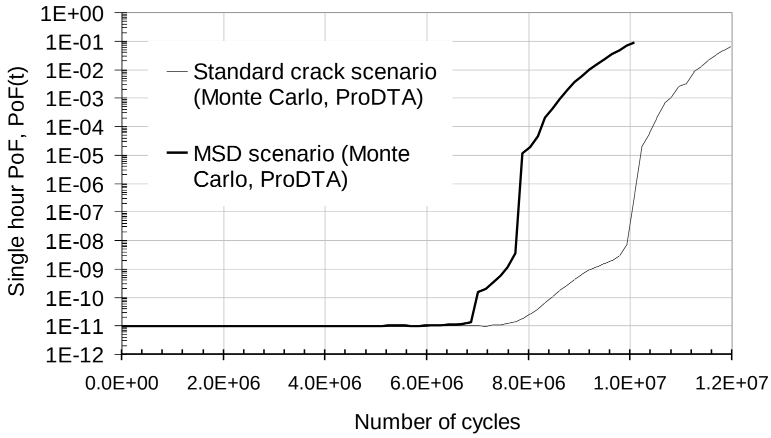


Figure 35. Single flight hour PoF results for the standard crack and the MSD scenarios

AGING AIRCRAFT ISSUES

Damage Tolerance Analysis of the North American SNJ-6 Wing Lower Attachment Angle

R.S. Rutledge, NRC-IAR-SMPL

The National Transportation Safety Board (NTSB) Office of Research and Engineering Materials Laboratory Division provided a report for an accident of a North American SNJ-6 (AT-6F) airplane, registration number N453WA, which occurred in Kissimmee, Florida, on May 9th 2005. This report examined parts of the inboard end of the right wing that mate to the right side of the centre-wing carry-through structure. A large fatigue crack was found in the failed right hand lower inboard wing attach angle, shown in Figure 36 and Figure 37. Based on this investigation, the emergency airworthiness directive (AD) 2005-12-51 and the special airworthiness information bulletin CE-05-72 were issued by the Federal Aviation Administration (FAA) that required inspection of the inboard and outboard upper and lower wing attachment angles of both wings for fatigue cracks. While the repeat inspection interval for the upper wing attachment angle was increased to one thousand (1000) hours, the lower angle currently has to be inspected every two hundred (200) hours. The National Research Council Canada Institute for Aerospace Research (NRC-IAR) was tasked by the North American Trainer Association (NATA) to carry out a damage tolerance analysis (DTA) for the critical lower inboard attachment angle component in order to determine a repeat inspection interval. NRC provided recommendations on a suitable repeat inspection interval for different usage and non-destructive inspection (NDI) capabilities. To support the damage tolerance analysis work, NRC-IAR carried out flight and laboratory testing. Crack growth predictions were conducted using two different MIL-A-008866B spectra (basic and advanced trainer class spectra), three initial crack sizes (0.60, 0.40, and 0.25 inch) [15.24, 10.16 and 6.35 mm], and four spectrum clipping values (5.67g, 5.0g, 4.5g, and 4.0g). A total of 24 crack growth scenarios were analysed. A critical input to the analysis was the aspect ratio of surface crack length to crack depth, which was determined from fractographic images of the failed component. The crack growth and residual strength evaluations estimated the remaining time to failure in a broad range from 1572 hours (advanced trainer; max. g-level of 5.67g; initial crack size of 0.60 inch [15.24 mm]) to 21832 hours (basic trainer; max g-level of 4.0g; initial crack size of 0.25 inch [6.35 mm]). The repeat inspection intervals were calculated using a combined scatter factor of 8 which included environmental effects and other unknowns such as the effect of the material forming process (plate versus extrusion). The DTA results indicated that the spectrum selection and the initial damage size assumed for the inspection methods/procedures had a significant effect on the calculated repeat inspection interval. Therefore, it was recommended that operators monitor the aircraft usage/loading and establish a reliable NDI detectable crack size in order to determine a cost-effective repeat inspection interval.

Figure 36 NRC-IAR Harvard Mark IV

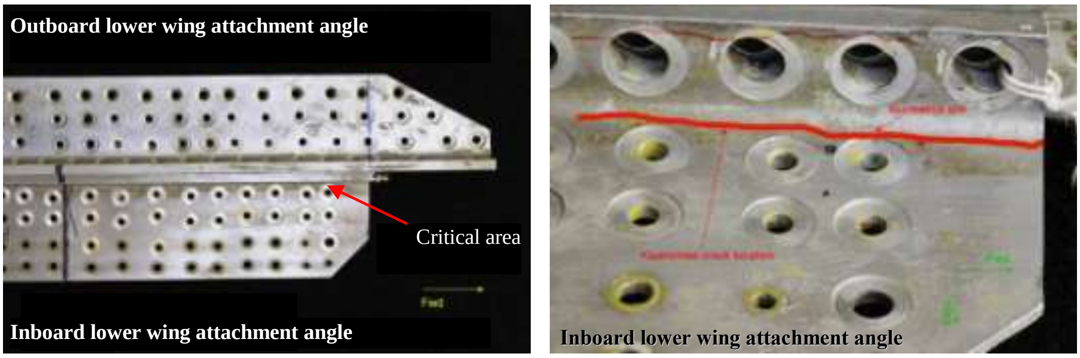


Figure 37 Lower Wing Lower Attachment Angle

JOINING TECHNIQUES

Fatigue Performance of Composite Joints

G. Li NRC-IAR-SMPL

The NRC is initiating the study of fatigue performance of composite joints to better understand fatigue life improvement caused by bonded/bolted assembly method. A single-strap butt joint configuration was selected for this study. This type of joint could be one of the possible configurations used to joint pre-cured fuselage barrel sections. To assess the opportunity for using this type of joint in aircraft structures, fatigue tests were carried out.

The joint size used for the experiments was determined according to the variation in the bonded overlap edge forces, such as bending moment and shear force. To avoid coupling in composite panels, laminates in symmetrical and balanced stacking sequences were used to make joints. Three combinations between adherend and doubler were adopted, they were (i) Case 1 joints made using laminate in $[45/-45/0/90]_{2s}$ layup sequence for both adherend and doubler; (ii) Case 2 joints using $[45/-45/0/90]_{2s}$ laminates for adherends and $[45/-45/0/90]_{3s}$ for doubler; and (iii) Case 3 joints $[0/90/45/-45]_{2s}$ for both adherend and doubler. Assembly methods included; (i) bonded only, (ii) combination of 2-row bolted and bonded, and (iii) 4-row bolted and bonded together. Carbon fibre laminas of CYCOM 5276-1 T40-800 145/35 0.125" wide tapes, made by Cytec Engineered Materials, were laid up using an automated fibre placement (AFP) machine. Film adhesive FM300 and blind bolts, also called Composi-LokII fasteners, made by Monogram Aerospace Fasteners, were used to assemble these joints. The cyclic test parameters were σ_{max}=100MPa, R=0.02, and *f*=10 Hz.

Fatigue results showed that the 4-row bolted/bonded composite butt joints were significantly strong. Compared to the purely bonded joints, the improvement in fatigue life was approximately 174% for 2-row bolted/bonded and 3818% for 4-row bolted/bonded joints.

Friction Stir Welding Demonstrator

L. Kok, Bombardier Aerospace

As with any novel technology, before any commitment to production is made, it is prudent to have in hand technical knowledge of the candidate process to be exploited. In that vein, Bombardier has undertaken a collaborative FSW project to develop the expertise to exploit the process in-house. A multi-year project leveraged between various Bombardier sites, and industrial collaborators has led to a joint development of capability to industrialize FSW lap joints on a regional aircraft fuselage.

Using DOE techniques, a pin tool that achieved adequate weld strength was developed. Further iterations on this basic pin tool focused on improved weld morphology and fatigue strength, and finally tool wear. As shown in the Figure 38, coupon and detail specimen tests confirmed that the weld had the desired fatigue life required for the application. This gave inputes to advance to the next level of component tests.

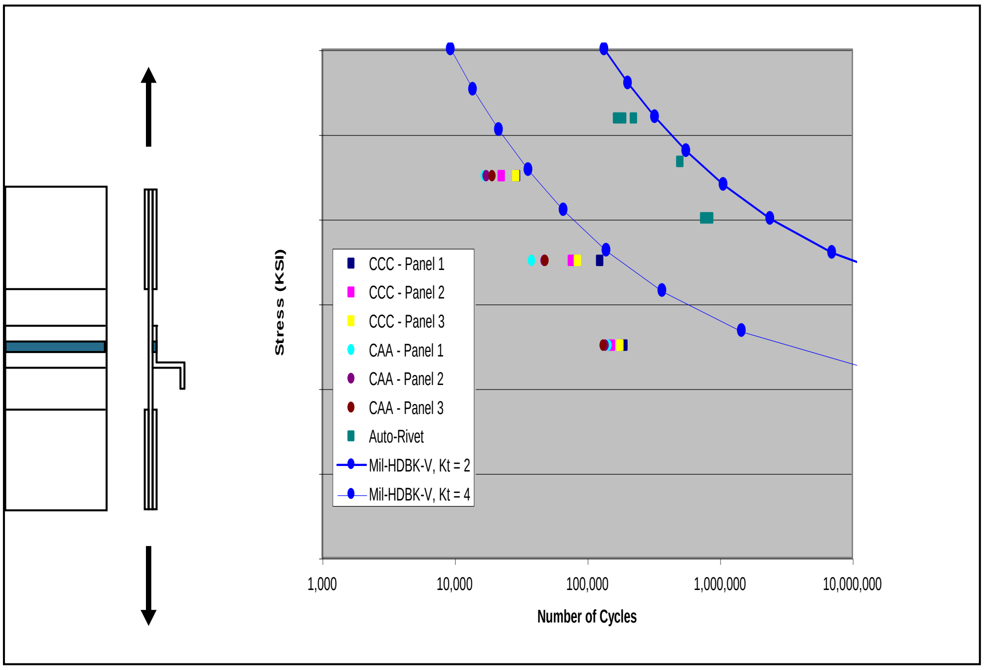


Figure 38. Coupon design and specimen test results.

A series of component tests were designed to not only explore the fatigue life of the weld; but also residual strength and crack initiation. The stiffened panel residual strength test exhibit failure much the same as seen in monolith IS machined structures as compared to bonded or riveted skin stringer panels (Figure 39).

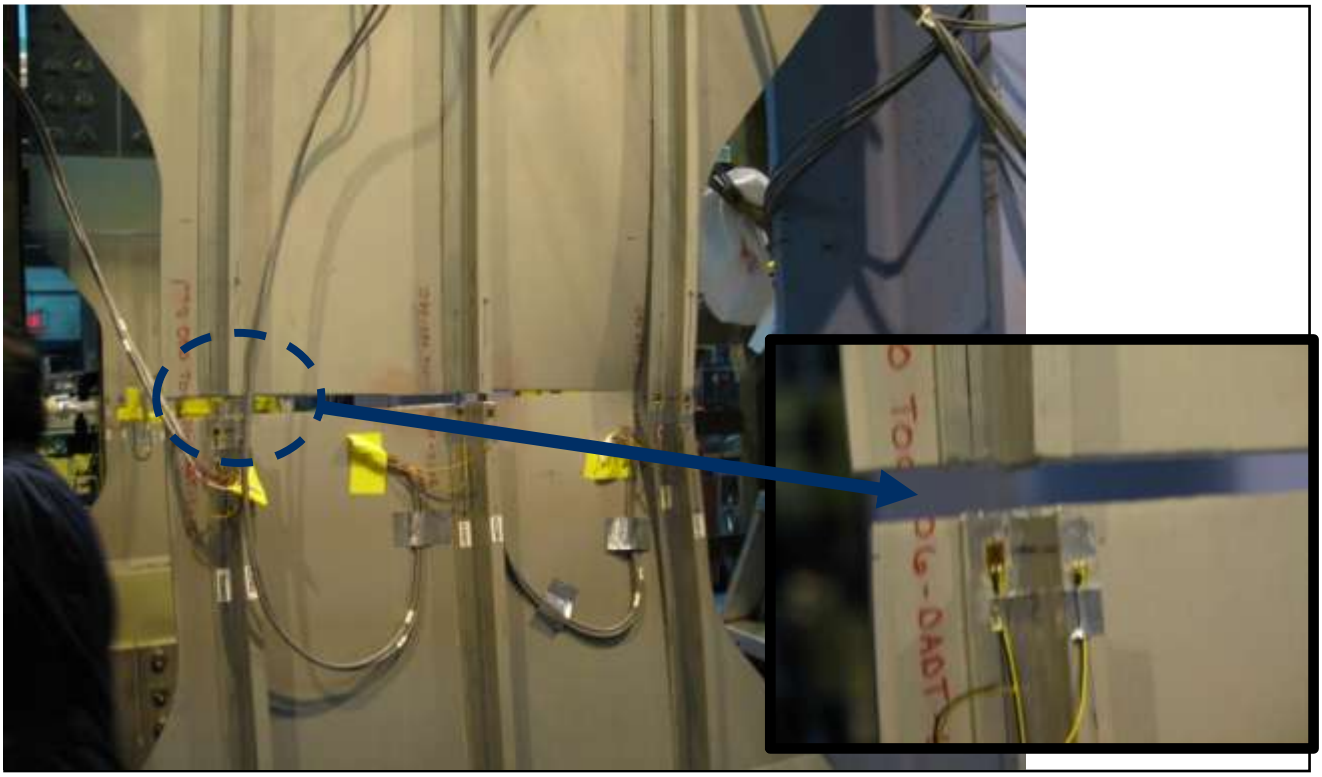


Figure 39. FSW panel failure.

As a final phase of the fatigue and damage tolerance assessment of FSW lap joint, full scale component fuselage panels were manufactured (Figure 40) and subjected to full scale fatigue and damage tolerance assessment. The panels as designed have successfully completed a lifetime of regional aircraft fuselage spectrum loading. Currently the second lifetime with damage assessment is being conducted on the air tank facility capable of torque, bending and pressure loading. Residual strength testing is set to follow.

Figure 40. Fuselage panel with FSW stringers.

From the encouraging results to date it is fully expected that a decision to commit FSW lap welds in the near future can be made with little technical risk.

NONDESTRUCTIVE INSPECTION AND SENSORS

A Novel Crack Sensor for Composite and Metallic Structures

M. Martinez, NRC-IAR-SMPL

NRC-IAR has developed a Structural Health Monitoring (SHM) System to detect damage on small and large scale airframe structures. The SHM System developed by NRC-IAR is called Surface Mountable Crack Sensor (SMCS). This system consists of a three-layer insulating and conductive paint system. The area of concern is prepared in the same manner that would be used for the placement of a strain gauge. Due to the nature of the sensor, the shape and geometry is customizable to fit the needs of the region being monitored. The conductive nature of the SMCS allows for the system to be interrogated using low voltage signals and minimal power. A wired interrogator that questions the integrity of the sensor has been developed with the intent of providing ease of use for an operator in the field. An installation kit has also been manufactured and delivered to Canada's Department of National Defense.

The Surface Mountable Crack Sensor (SMCS) was designed to overcome some of the drawbacks of traditional crack gauges. The SMCS consists of two to three paint-on layers; the first layer is a white ceramic non-conductive paste while the second layer is a conductive paint placed on the surface of the non-conductive paste. The first layer has two purposes; to act as an insulator when the sensor is placed on metallic conductive structures and to guarantee that the sensor breaks up, due to the passage of a surface crack. The breaking of this first layer minimizes the risk of the circuit re-closing as the crack opens and closes. The second layer is a conductive layer, which can be sandwiched between two insulating layers for protection. The use of acrylic epoxy has also been used to protect the conductive layer. The shape of the sensor is painted on the surface of the structure as required. As a crack passes through the conductive paint it causes the circuit to break, thus interrupting a signal and indicating the occurrence of the crack. Figure 41 shows the SMCS on both a metallic and composite structure [21]. Laboratory experiments have demonstrated the ability of the sensor to detect crack growth under 3000 microstrain. Further research and development is being carried out for strain fields greater than 3000 microstrain.

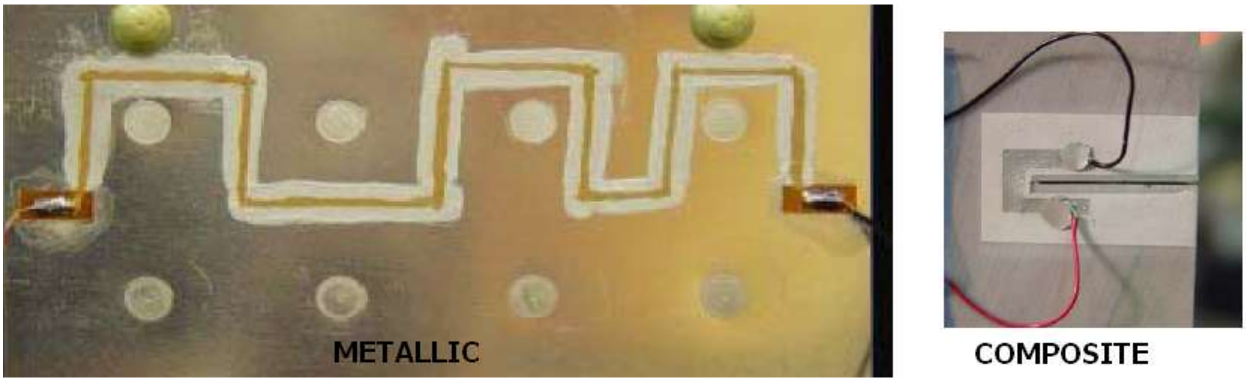


Figure 41. Conductive Paint Sensor on metallic (left) and composite (right) coupons

An application kit has been produced that contains all the necessary materials and instruments required to perform structural health monitoring of an area likely to crack or that is already cracked. Figure 42, shows the case containing the consumables and equipment.

Figure 42: SMCS Kit

The SMCS is monitored by means of an interrogation unit. This unit continuously monitors up to 4 channels at a time. The unit is battery operated and thus does not require any external power. Both active and passive monitoring has been investigated with this sensor. In the case of cyclic fatigue loading the use of an active monitoring (constant monitoring vs. spot checks) is necessary in order to avoid obtaining false readings. The internal logic of the interrogator turns an LED the first time the circuit is opened and keeps the LED turned on until the operator presses the clear button indicated in Figure 43.

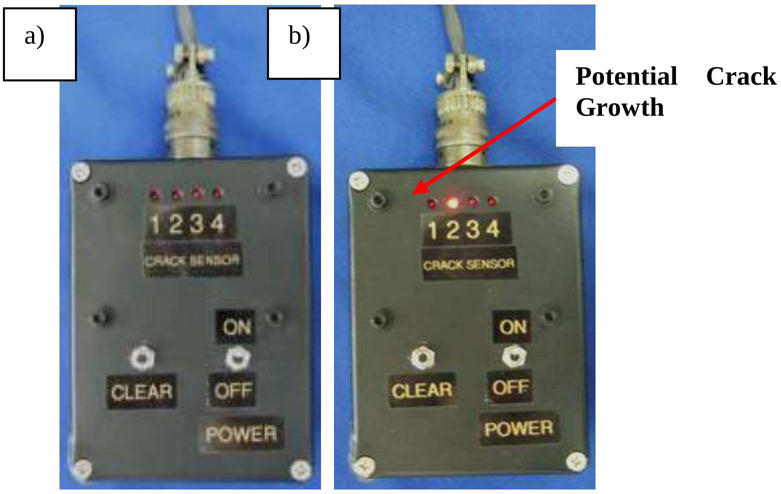


Figure 43: Before Test Interrogation Procedure

Several tests with private companies have demonstrated that this technology is able to detect cracks inside reverberation chambers, where other SHM technologies have failed to withstand the operating environments.

Validation and Demonstration of fatigue crack detection using CVM technology

J. Pinsonnault, Bombardier Aerospace

The Comparative Vacuum Monitoring (CVM) technology was tested in Bombardier Aerospace lab to validate the fatigue crack detection on metallic coupons. Two CVM systems were tested, the laboratory equipment and the field equipment. The CVM sensors tested were made of silicon and Teflon. They were surface mounted on test coupons having different geometries (Figure 44), materials and surface finishes. The materials tested were Aluminum 2024-T3 and 7475-T351, Steel 4340, Titanium Ti-6AL-4V and Aluminum-Lithium 2199-T8. The surface finishes tested are no finish, primer and topcoat.

Figure 44. Three coupon geometries.

The laboratory equipment was tested on 95 coupons having a thickness from 0.040 to 0.800 inch and no false positive or false negative indication reported. 90% probability of detection with a confidence of 95% was generally obtained before a crack reached 0.014 inch under the beginning of the sensing gallery.

The field equipment was tested on the 66 coupons having a thickness equal or inferior to 0.150 inch and one false indication was observed due to a connector malfunction during the validation testing. 90% probability of detection with a confidence of 95% was generally obtained before a crack reached 0.100 inch under the beginning of the sensing gallery.

42 sensors were also installed at 18 different locations on a complete aircraft fatigue test. During the period of the test, no false negative indication was reported with the laboratory equipment or the field equipment.

Demonstration of Distributed Strain Sensing Using Optical Fibers

The capability of distributed strain sensing using a optic fiber technology was demonstrated on a complete aircraft fatigue test. The selected technology has the capability to measure strain at the centimeter resolution up to 70 meters. The optical fibers were installed at 7 locations. These locations were selected to provide strain distribution curves for stress investigations. Overall, 67 sensing segments were successfully installed and 800 sensing points were interrogated. The optical fibers showed results which are in good agreement with conventional strain gauges and finite element model. Figure 45 gives an example of strain distribution of one fiber for 3 load levels.

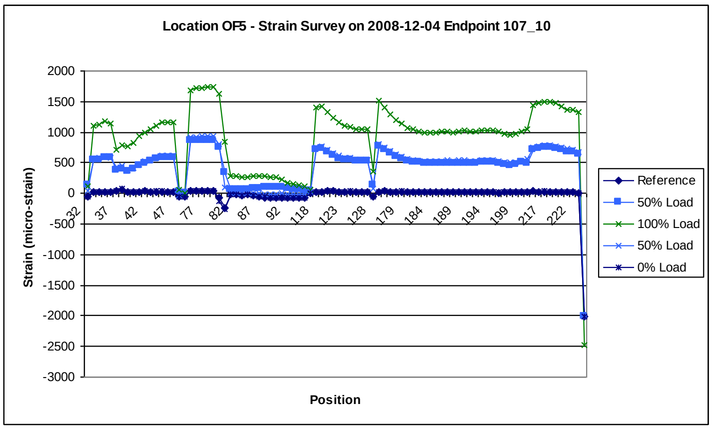


Figure 45. Strain distribution of one fiber for 3 load levels.

Generic Bolt Hole Eddy Current Probability of Detection Study

Y. Caron, DTAES-DND

The ATESS Generic Bolt Hole Eddy Current (GBHEC) Probability of Detection Study was a comprehensive effort spanning numerous years and was successfully completed in 2008. The intent of this project was to reassess the current POD information ($a_{90/95}$ value) for Bolt Hole Eddy Current (BHEC) and to investigate POD modeling to allow portability of the data particularly to the CC130 and CP140 wing box structures.

The GBHEC study used layered coupons with 3/16 inch fastener holes designed to represent the wing box structures of the C130 and CP140. Coupons contained both EDM notches and lab grown cracks that were oriented in various configurations to allow for comparison of EDM indications to crack indications. Data was collected using NDT technicians across Canada in which over 30000 data points were collected. Data was analyzed using various methods to estimate the POD; however, the draft MIL HDBK 1823 software was the primary analysis tool used to present the data.

The results of the reassessment of the current $a_{90/95}$ value varied depending on flaw location as four locations were investigated: top skin corner, top skin mid-bore, faying surface corner, and bottom skin corner. Flaw size distributions also varied which inhibited direct comparison between flaw locations. The overall result was that the $a_{90/95}$ value that resulted for all four configurations was much less than the original value; however, this value by itself does not provide sufficient information to be properly applied in risk analysis.

The results of the empirical study were broken down into three groups: lab grown cracks in new material, EDM notches in new material, and EDM notches in in-service material. From this data, a reliability assessment of cracks in real structure was then estimated in which POD modeling was used. POD modeling represents an inexpensive and timelier alternative to costly experimental POD studies in which it has the potential to partially substitute and complement experimental POD data and allow for the portability of POD information (Figure 46).

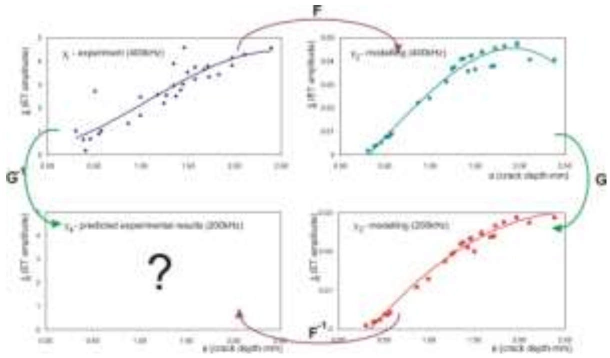


Figure 46. Example of how POD modeling is used to predict POD curves.

POD Modeling was evaluated using transfer functions (TX) from existing data as well as using a Model Assisted (MA) approach. A combination of both is also a valid approach in applying a generic set of data to a specific structure. (next section of Canadian Review), also see [22].

Numerical Modelling as a Cost-Reduction Tool for Probability of Detection of Bolt-Hole Eddy Current Testing

C. Mandache, A. Fahr, M. Khan, and M. Yanishevsky

Probability of detection (PoD) studies are broadly used to determine the reliability of specific non-destructive inspection procedures, as well as to provide data for damage tolerance life estimations. They require inspections on a large set of test pieces, a fact that makes these statistical assessments time- and cost-consuming. Numerical simulations could be used as a cost-effective alternative to empirical investigations, in order to predict the inspection outputs as functions of the input characteristics related to the test piece, transducer and instrument settings, and, subsequently, to partially substitute and/or complement inspection data in POD analysis.

This study has been initiated at the request of Canadian Forces and driven by the aging aircraft fleets need for maintenance according to damage tolerance approaches [23]. The NDT method employed was eddy current, while the specimens were representative of fastener bolt-hole wing areas of Lockheed Martin CC-130, Hercules and CP-140, Aurora aircraft.

Physics-based models are regularly employed in NDT to predict the transducer response for a given inspection situation and are used as tools to optimize the inspection, develop new probes, understand the involved phenomena, etc. A boundary element-based numerical modelling software [24] was employed to predict the eddy current signal responses when varying one of the following eight inspection parameters: (i) probe characteristics, such as frequency, lift-off, tilt, off-centre scanning; (ii) crack size, including length, depth and simultaneous variation of both length and depth; and (iii) test piece properties, such as electrical conductivity. After all the input parameters were set, the individual inspection situation was relatively fast (around 2 minutes on a Dual Core 2.4 GHz processor and a computer having 2 GB of RAM).

The main goal of the model-assisted approach is to predict the experimental results due to a change in one or more testing parameters, without actually performing the experiment. An example exercise is used here as a capability demonstrator of numerical models in predicting eddy current signals, and ultimately estimating the PoD. In this case, the driving frequency of the probe is lowered from 400 kHz to 200 kHz, without actually performing the experiments. This consideration has a practical interest, since lowering the driving frequency allows for better penetration of the eddy currents in the material and, consequently, the method's ability to detect deeper cracks.

The defect's length and depth were obtained by replica measurements and used as input parameters in the eddy current simulations. The vertical signal amplitude for the 400 kHz excitation frequency when inspecting for mid-bore cracks was used as the basis of the study. Although it provided a defect indication, the right-censored data (*i.e.* saturated eddy current signals) was not taken into account in these simulations, since it provides no relationship to the crack geometry. For the same crack dimensions, the eddy current signals were simulated for 400 kHz and 200 kHz driving frequencies. The transfer function approach establishes relationships between the signal amplitude variation with the crack depth for the two frequencies, and between the experiment and simulation. This situation is depicted schematically in Figure 47.

In order to predict the eddy current signal responses at 200 kHz, y_4, the inspection results should be first modelled for the two cases – 200 and 400 kHz (y_3 and y_2 respectively), while maintaining all other testing characteristics as inputs to the simulation code. The relation between the experimental, y_1, and modelled, y_2, signals of interest at 400 kHz could be established through a function, $\boldsymbol{F}$, while the relationship between the simulated signals, y_2 and y_3, could be described by a function $\boldsymbol{G}$. The eddy current response for the 200 kHz driving frequency situation could then be found through the inverse application of either function.

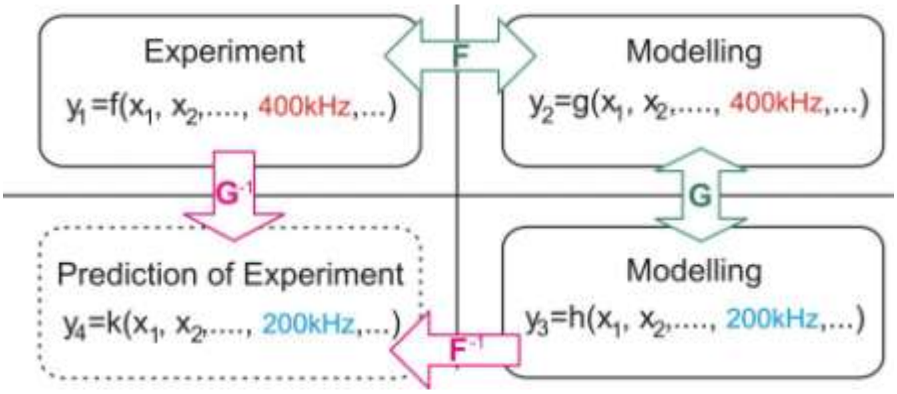


Figure 47. Transfer function approach for predicting eddy current results at 200 kHz excitation frequency.

The experimental eddy current amplitude considered here, *a-hat*, was obtained as the average recording of five different inspectors. The functions F and G are found based on the relationships between the fitting equations describing the variation of the signal amplitude, *a-hat*, versus the crack depth, *a*; therefore, the predicted experimental eddy current responses for the 200 kHz could be found through either F^{-1}(y3) or G^{-1}(y1). These are shown in Figure 48, along with the original experimentally collected data. As evident, the two results, predicted either by F or G, are very similar, validating the selected approach.

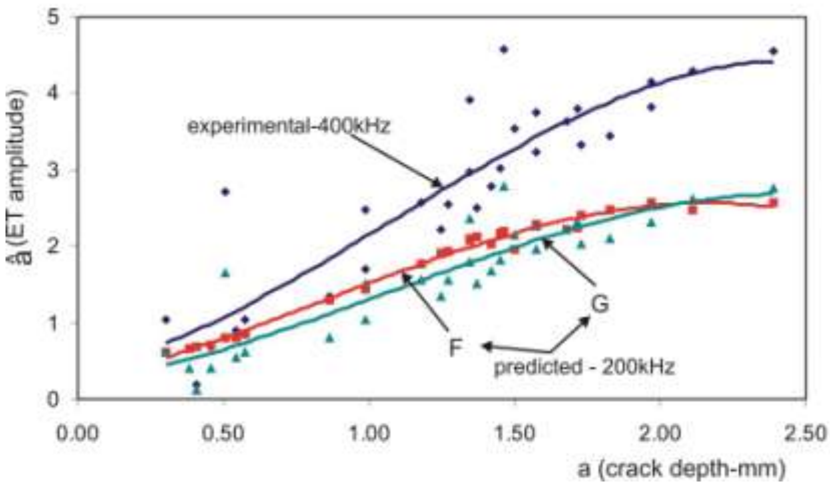


Figure 48. Actual eddy current inspection results at 400 kHz and predicted ones at 200 kHz, obtained through transfer functions and numerical simulations.

In order to validate the modeling predictions, bolt-hole eddy current inspections at a driving frequency of 200 kHz were performed by using the same inspection procedure as for the 400 kHz case. Only eleven cracked specimens were used for this experimental validation, and distributed in the mid-size range of depths, most critical for PoD analysis. These results add extra credibility to the modeling and transfer function approach. The average of the predicted eddy current amplitudes through F and G, and the experimental results for 200 kHz driving frequency, along with the original experimental data at 400 kHz, are shown in Figure 49.

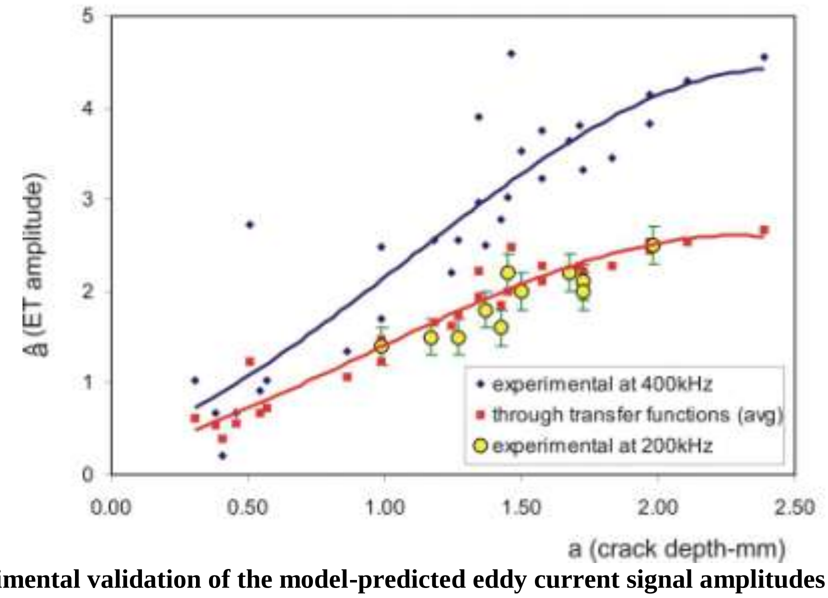


Figure 49. Experimental validation of the model-predicted eddy current signal amplitudes for 200 kHz driving frequency.

With the numerically predicted data an actual PoD curve was plotted based on the modelled results and transfer function approach, as shown in Figure 50. This was done by using statistical analysis based on the R programming language [25]. Although eddy current instrument gain and noise considerations were not taken into account, the $a_{90/95}$ value for the experimental study was found to be 0.643 mm, while on this case study, for 200 kHz driving frequency, the $a_{90/95}$ value is predicted to be 0.431 mm, based on the data obtained through modeling and transfer functions. This result indicates that lower testing frequency would improve the bolt-hole eddy current 90/95 flaw size.

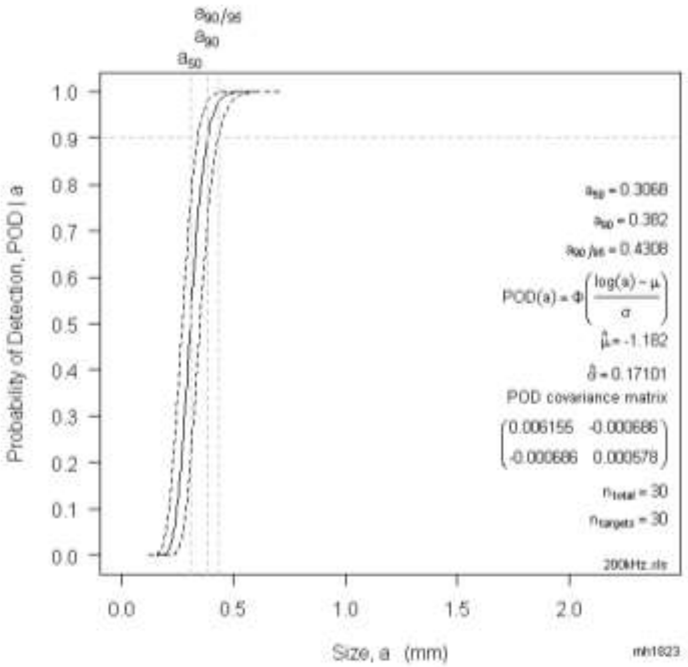


Figure 50. PoD curve obtained based on the approach described above.

The use of modeling in NDT provides tremendous information regarding the best inspection parameters to be selected in a specific testing situation. Model-assisted PoD aims at establishing transfer function relationships between NDT signals from real and simulated (either physically or numerically) discontinuities. This could extend the range of applications of a single study to a set of similar situations by changing the input parameters. The ultimate goal of this work was to demonstrate how physics-based simulations have the potential to substitute and complement experimental data in PoD studies, reducing costs, effort, and resources, while increasing platform availability.

The case study included here represents a hybrid scheme, combining the transfer function approach and the physics-based numerical models to illustrate the way by which the already collected NDT data could be used for predicting the feature of interest dependence on the defect size, *a-hat* versus *a*, by changing one of the testing settings, in this situation – the driving frequency.

Gathering appropriate inspection data for PoD analysis, either from laboratory or field testing, is a tremendous effort. Maintaining databases with existing data, and complementing or adapting this data for new situations though numerical modeling represent the faster, more economical solution for future PoD studies.

NDI Technique Detectable Limts

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

Non-destructive inspection (NDI) techniques detectable limits are useful on the CF-18 program for two main reasons. Detectable anomaly sizes are used for confidence cut depths when performing surface renewal or crack chasing and, as initial flaw size for damage tolerance analyses (DTA).
An investigation is conducted to determine the detectable limits for the most common NDI techniques used on the CF-18. The data collected was based on literature review and on a statistical analysis of CF-18 in-service inspection data.
Based on the above investigation, a detectable crack size of approximately 0.020” was determined for Eddy-current inspection of fastener holes. Figure 51 shows an example of the in-service data collected.

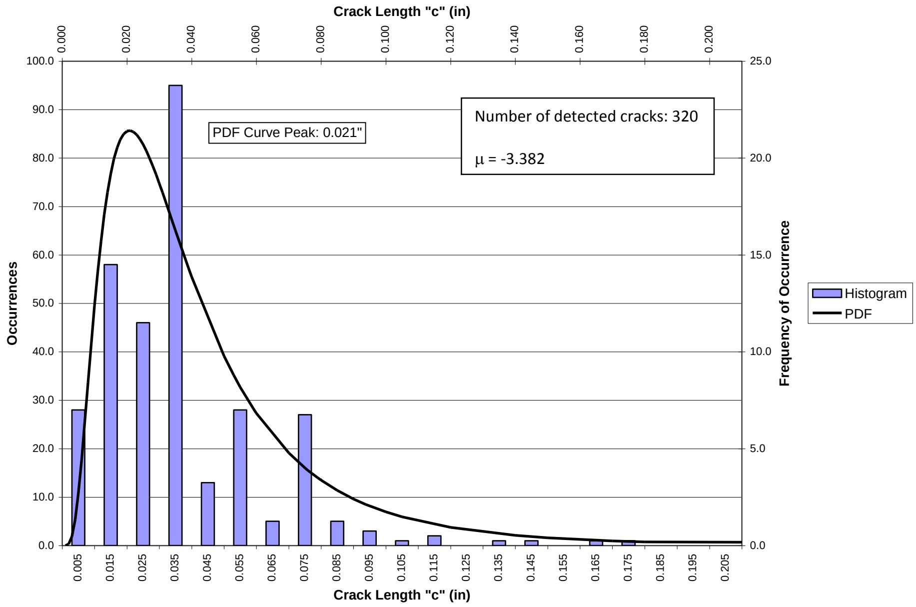


Figure 51. Example of in-service inspection data collected for Eddy-current inspections

ACKNOWLEDGEMENTS

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REFERENCES

1 Backman, D., Liao, M., Crichlow, L., Yanishevsky, M. and Patterson, E.A. - Michigan State University, "The Use of Digital Image Correlation in a Parametric Study on the Effect of Edge Distance and Coupon Thickness on Residual Strains and Fatigue Life after Cold Expansion", Journal of Strain Analysis, 4/21/2008.

2 Backman, D. and Patterson, E. - Michigan State University, "Measuring Residual Strains in Aluminium and Fiber Metal Laminate Materials during Cold Expansion and Riveting - SEM XI International Congress & Exposition on Experimental and Applied Mechanics From 6/2/2008 To 6/5/2008, Orlando/Florida.

3 Backman, D., Crichlow, L., and Yanishevsky, M., "Theoretical and Experimental Analysis of Hole Cold Expansion in 2024-T3 Aluminum", LTR-SMPL-2008-0048, 2/22/2008.

4 Backman, D. and Patterson, E. - Michigan State University., "Fatigue Life Enhancement of Fiber Metal Laminate Materials as a Result of Hole Cold Expansion", SEM XII International Congress & Exposition on Experimental and Applied Mechanics From 6/1/2009 To 6/3/2009, Albuquerque, New Mexico.

5 Backman, D. and Patterson, E. - Michigan State University., "Fatigue Life Enhancement of Fiber Metal Laminate Materials as a Result Of Hole Cold Expansion", Society for Experimental Mechanics Annual Conference & Exposition on Experimental and Applied Mechanics From 6/1/2009 To 6/4/2009, Albuquerque, New Mexico/USA.

6 Liao, M., “Dislocation Theory Based Modeling on Short Crack Growth and Crack Nucleation,” NRC-IAR LTR-SMPL-2008-0008, 2008 (for both 46-QJ0-56 and 46SS-J401).

7 Liao, M., “Dislocation Theory Based Modeling for Short/Small Crack Growth in Aircraft Aluminum Alloys,” Proceedings of the Second International Conference on Heterogeneous Material Mechanics,” Huang Shan, China, June 3-8, 2008

8 Liao, M., Benak, T., Renaud, G., Marincak, T., Yanishevsky, M., Bellinger, N.C., Mills, T., Prost-Domasky, S., Honeycutt, K., “Development of Short/Small Crack Model for Airframe Material: 7050 Aluminum Alloys”, The proceedings of the Aging Aircraft Conference, Phoenix, AZ, April 21-24, 2008

9 Liao, M., Renaud, G., and Bellinger, N.C., “Probabilistic Modeling on Short Crack Growth in Airframe Aluminum Alloys”, Journal of Aircraft 2008, vol.45 no.4 (1105-1111).

10 Liao, M. “Probabilistic modeling of fatigue related microstructural parameters in aluminum alloys”, Journal of Engineering Fracture Mechanics 76 (2009) 668–680.

11 Liao, M., Chisholm, K., and Mahendran, M. “Statistical Analysis of Constituent Particles and Pores in 7050-T7452 Aluminum Alloys”, Journal of ASTM International, Vol. 6, No. 2, Paper ID JAI101558, Available online at www.astm.org.

12 Merati, A., Gallant, M, and Liao, M., “A study of fatigue crack nucleation in 7050-T7452 hand forged aluminum”, COM 2008 Proceedings, Winnipeg, Manitoba, Canada, August 2008.

13 Liao, M., and Benak, T, “Residual Stress Measurements on 7050-T7452 Forging Coupon Specimens”, LTR-SMPL-2007-0109.

14 Liao, M. “Dislocation Theory Based Modelling on Short Crack Growth and Crack Nucleation”, NRC report, LTR-SMPL-2008-0008.

15 X. J. Wu, P. Au, “Deformation kinetics during dwell fatigue”, Materials Science and Technology 23/12 (2007), pp. 1446-1449.

16 Wu, X.J., “A model of nonlinear fatigue-creep (dwell) interactions”, Journal of Engineering for Gas Turbines and Powers 131 (2009), p. 032101.

17 Liao, M., Bombardier, Y., Renaud, G., Bellinger, N.C., Cheung, T., “Development of Advanced Risk Assessment Methodologies for Aircraft Structures Containing MSD/MED”, The 25th Symposium of the International Committee on Aeronautical Fatigue (ICAF 2009), Rotterdam, Netherlands, 2009.

18 Bombardier, Y., Liao, M., Renaud, G., “Fracture Mechanics Modelling of Multiple Site Damage Scenarios”, 12th International Conference on Fracture, Ottawa, Canada, 2009.

19 Liao, M., Renaud, G., Bombardier, Y., Bellinger, N.C., “Development of Initial Crack Size Distribution for Risk Assessment of Aircraft Structures”, Proceedings of the RTO-MP-AVT-157 Military Platform Ensured Availability Symposium, Montreal, Canada.

20 Liao, M., Cheung, T., Renaud, G., Bellinger, N.C., Caron, Y., Bombardier, Y., “Quantitative Risk Assessment for Aircraft Structures”, The proceedings of the Aging Aircraft Conference, Phoenix, AZ, April 21-24, 2008.

21 Martinez, M., Li, G., Marincak, T., Benak, T. and Oudovikine, A., “Crack Detection on Composite and Metallic Aerospace Structures”, 4th European Workshop on Structural Health Monitoring, CPR-SMPL-2007-0117, From 7/2/2008 To 7/4/2008, Krakow, Poland.

22 Forsyth, D., Estimating the Effect of Noise on PoD from the GBHET PoD Study, Feb 2008

23 Fahr, A., Khan, M., Mandache, C., Yanishevsky, M., Brothers, M., LTR-SMPL-2007-0138: Generic Bolt Hole Eddy Current Testing Probability of Detection, internal NRC-IAR report, 2007.
24 Mandache, C., Khan, M., Yanishevsky, M., Fahr, A., Forsyth, D., LTR-SMPL-2007-0121: Generic Bolt Hole Eddy Current Testing Probability of Detection – Numerical Modelling Handbook, internal NRC-IAR report, 2007.
25 Annis, C., http://www.statisticalengineering.com/, retrieved on September 29, 2008.