REVIEW OF CANADIAN AERONAUTICAL FATIGUE WORK 2009-2011

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Summary

This paper provides a review of Canadian work associated with fatigue of aeronautical materials and structures during the period 2009 - 2011. All aspects of structural technology are covered including full-scale tests, loads monitoring, fracture mechanics, composite materials and non-destructive inspection.

Organization Abbreviations Used in Text:
BA – Bombardier Aerospace
BHTCL - Bell Helicopter Textron Canada Limited
BHTI - Bell Helicopter Textron Inc.
CF - Canadian Forces
DAES - Directorate of Aircraft Engineering and Support (DND)
DRDC – Defence Research and Development Canada (DND)
DSTO - Defence Science and Technology Organization (Australia)
DTAES - Directorate of Technical Airworthiness and Engineering Support (DND)
DAEPM(M) - Directorate of Aerospace Equipment and Program Management – Maritime
DND - Department of National Defence
GE – General Electric
IAR - Institute for Aerospace Research
L-3 MAS - L-3 Communications (Canada) Military Aircraft Services (MAS)
LMA - Lockheed-Martin Aerospace
NRC - National Research Council of Canada
RAAF - Royal Australian Air Force
RMC - Royal Military College of Canada (DND)
SMPL - Structures and Materials Performance Laboratory
USN - United States Navy

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List of Acronyms for the Technical Terms

ALEX	Aircraft Life Extension Program
ASI	Aircraft Sampling Inspections
ASIP	Aircraft Structural Integrity Program
BOS	Baseline Operational Spectrum
CFH	Component Flight Hours
CVM	Comparative Vacuum Monitoring
DIC	Digital Image Correlation
DADT	Durability and Damage Tolerance
DM	Damage monitoring
EDM	Electrical Discharge Machining
ELE	Estimated Life Expectancy
FARS	Fleet Averaged Reference Spectrum
FC	Flight Cycles
FCS	Flight Control Surfaces
FE/FEA/FEM	Finite Element/Finite Element Analysis/Finite Element Model
FH	Flight Hours
FLEI	Fatigue Life Expanded Index
FSFT	Full Scale Fatigue Test
FSW	Friction Stir Welding
FUI	Fatigue Usage Indices
FTI	Flight Test Instrumentation
HOLSIP	Holistic Structural Integrity Process
IAT	Individual Aircraft Tracking
IDAC	Integrated Data Acquisition System
LOV	Limit of Validity
MES	Master Event Spectra
MSD/MED	Multi-site Fatigue Damage/ Multiple Element Damage
NDI/NDT/NDE	Nondestructive Inspection/Testing/Evaluation
NSR	Non-Standard Repair
OEM	Original Equipment Manufacturer
OM	Operational Monitoring
PCL	Patran Command Language
PEC	Pulsed Eddy Current
PZL	Plastic Zone Link-up
POD	Probability of Detection
RARM	Record of Airworthiness Record Management
SBI	Safety-By-Inspection
SDRS	Structural Data Recording System
SFPOF	Single Flight Probability of Failure
SSI	Structurally Significant Items

SG	Strain Gauge
SLAP	Service Life Assessment Program
SHM	Structural Health Monitoring
SLMP	Structural Life Monitoring Program
SMP	Structural Maintenance Program
SsCx	Split Sleeve Cold Expansion
TF	Transfer Functions
TrF	Tracking Factor

Introduction

Canadian industry, universities and government agencies were solicited for information describing their fatigue technology related activities over the period 2009 to 2011. This review covers work performed or being performed by the following organizations:

Bombardier Aerospace
Bell Helicopter Textron Canada Limited
Department of National Defence (DND)

- Defence Research and Development Canada (DRDC) - Air Vehicles Research Section (AVRS)
- Canadian Forces (CF)
- Director General Air Equipment Technical Management (DGAEPM)

I.M.P. Group Ltd.
Kelowna Flightcraft Ltd.
L-3 Communications (Canada) Military Aircraft Services (MAS)
National Research Council of Canada

- Institute for Aerospace Research (NRC-IAR)

University of Waterloo

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 and Component Testing

H-STAB Component test

Jean Dubuc, L-3 Communications (Canada) Military Aircraft Services (MAS)

Historically, the Flight Control Surfaces (FCS) of the CF-18 had not been considered systematically as part of the Aircraft Structural Integrity Program (ASIP) effort. Certification basis was demonstrated by Original Equipment Manufacturer (OEM) for 6000 Component Flight Hours (CFH). The usage of these components and associated fatigue life had not yet been assessed for a Canadian usage and environment. However, significant work was recently performed and it clearly showed that the fleet was quickly approaching the OEM certification limit of 6000 CFH for many FCS.

Supportability for the fleet until Estimated Life Expectancy (ELE) is a concern, and potential development of sparing and procurement strategies becomes a requirement. Fleet logistic plan shows that, to meet fleet ELE, half of the HSTAB fleet needs to reach 8,000 CFH.

From a fatigue point of view, analyses and testing will be required for the 8,000 CFH target usage. From a static point of view, analysis and testing will be required to certify an aged (and possibly damaged) HSTAB. Currently, a static component test is planned at the National Research Council Canada in order to test the residual strength of the H-Stab with several sizes of disbonds between the HSTAB skin and honeycomb core. This test will help characterize and quantify the static strength and damage tolerance of an aged HSTAB. It will also present an opportunity to gather/develop NDI techniques in order to support CF Safety-By-Inspection (SBI) management strategy.

CRJ200 Fleet Leader Program - Complete Airframe Fatigue Test (CAFT)

N. Préfontaine, Steve Dupasquier, Bombardier Aerospace, Commercial Aircraft

A comprehensive durability and damage test program has been developed for the Bombardier CRJ Regional Jet family. Much of the program was described in the previous Canadian National Reviews.

Since the previous update, the CRJ200 Complete Airframe Fatigue Test had completed a total of 137,000 cabin pressure cycles and 120,000 flight load cycles. In early 2010, the airframe has been used as a test rig for the fatigue testing of the forward engine mount yoke fittings. A total of 320,000 engine flight cycles was cumulated to publish a safe life of 80,000 FC.

In December 2010, an ultimate cabin pressure test was performed to validate the structural modifications. Through the testing phases, the critical areas, structural modifications and repairs were instrumented with over 700 strain gauges (589 axials & 147 rosettes). Emerging Structural Health Monitoring (SHM) devices were evaluated on the airframe.

The Teardown Inspection was launched in early 2011. The airframe will be partly dismantled and thoroughly inspected for presence of cracking. The repairs for service and structural modifications will be validated.

Durability and Damage Tolerance Testing of Composite Fuselage Barrel

R. Khotari, Bombardier Aerospace, Core Engineering

As Bombardier is moving aggressively into composite technologies for various large aircraft components, it was necessary to develop technologies related to full-scale manufacturing, analysis and testing. One such effort was a collaborative project between Bombardier, Bell Helicopter, NRC and Composite

Atlantic. A sandwich composite test barrel was manufactured as part of this project using primarily unidirectional tape material from Cytec. The test barrel shown on the test rig is included in Figure 1.

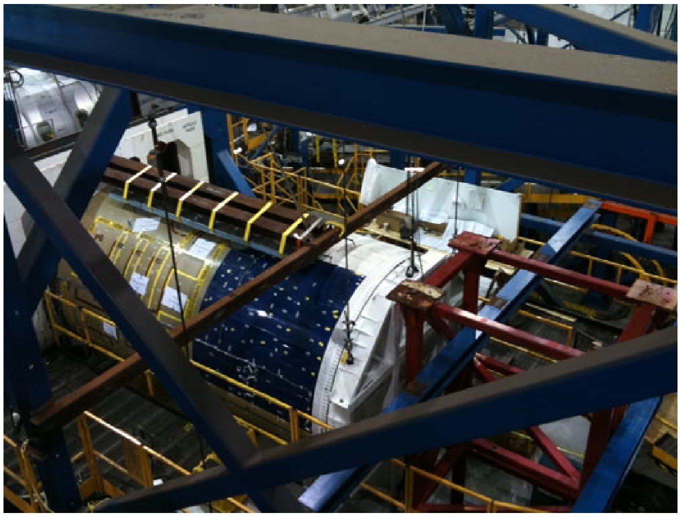


Figure 1. Composite fuselage barrel on the test rig

The test barrel was tested under fatigue and static loading after introduction of multiple impact damages. The impact damages on the fuselage simulated typical in-service and production damages. The DADT testing enabled better understanding of damage growth, SG correlation, FEA modeling techniques and repairs. The various challenges unique to sandwich constructions were studied during test program.

The damage map on the sandwich construction is included in Figure 2.

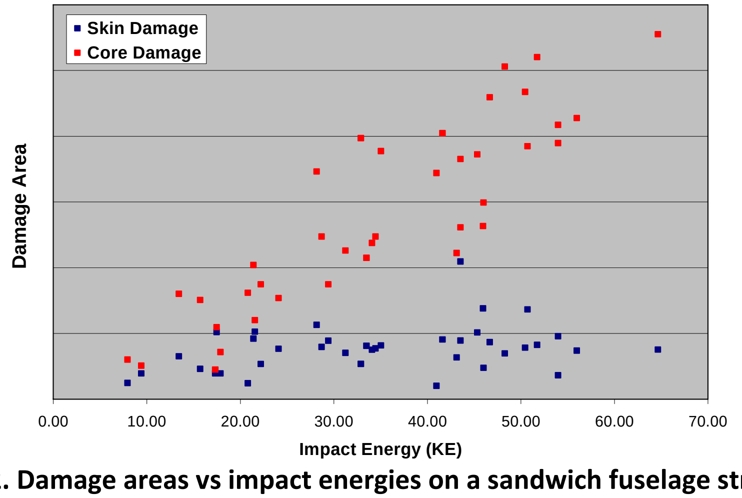


Figure 2. Damage areas vs impact energies on a sandwich fuselage structure

The example of a strain change recorded at certain locations during the fatigue cycling is included in Figure 3.

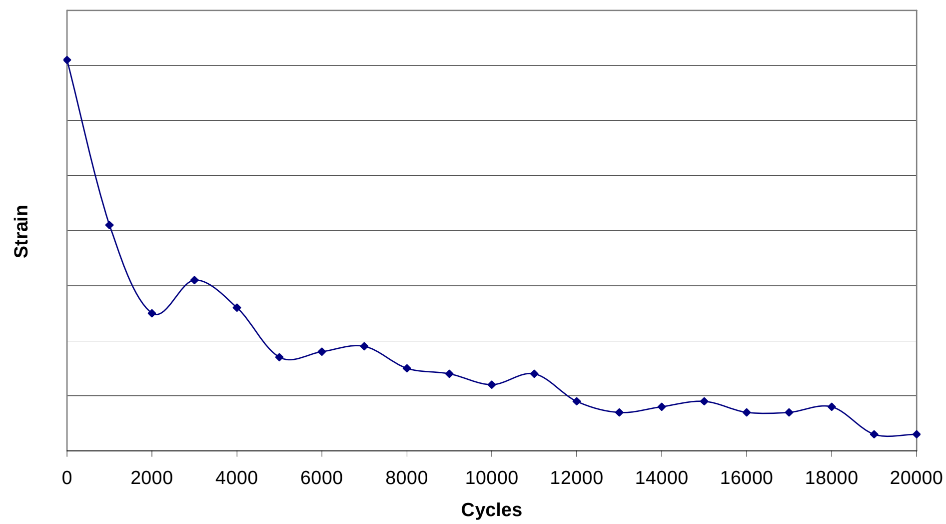


Figure 3. Example of a strain gauge monitoring during fatigue cycling

The FEA correlation allowed better understanding of the results and improvement of FEA model. This knowledge was used for further structural optimization resulting in more efficient structure.

F/A-18 CF188 Static Residual Strength Testing of Modified Engine Mounts

R.S. Rutledge and C.A. Beltempo, National Research Council Canada

The F/A-CF188 engine thrust mounts secure the gas turbine engine to the airframe by means of a clamp arrangement which react the thrust loads. The heavyweight outboard mount is comprised of a swing link, which pivots between fuselage frames to allow lateral displacement of the power plant. Thrust loads are reacted by the forward face of the mount abutting a stop in the fuselage structure whilst a protrusion on the trunnion engages a slot in the trunnion mount to provide thrust and drag reaction. During service the CF 188 engine thrust mounts have been found to wear more than expected, and there are limited spares available for replacement. In an attempt to alleviate this problem, a repair to the thrust bearing race was proposed by Aerospace Welding of Blainville, Québec. This repair comprises of over-sizing the hole and returning it to nominal size by plasma spraying. NRC Aerospace was tasked by DND (DTAES) to statically test samples of repaired engine mounts, modified to the maximum repair limits, to specifically selected operational envelope loads to ensure that the material behaves linearly during limit loading, and does not fail under ultimate loading.

The residual strength tests on the maximum repair limit configuration engine mounts were located in the Multi-Axial Test Platform at SMPL, with the mount located on a pillar, and vertical and horizontal loads applied by means of a lever system, see Figure 4.

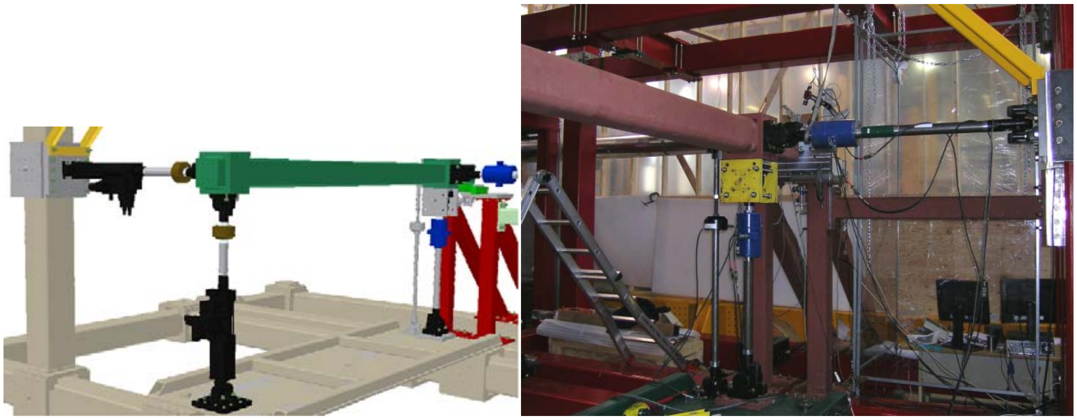


Figure 4. Engine mounts residual strength test fixture schematic and test setup

The test was controlled by an MTS Aero-90 Test Control System, and data was recorded using the Integrated Data Acquisition System (IDAC). Test loads to ultimate plus ten percent were provided by L-3 MAS. The additional ten percent was applied to account for thermal expansion stress effects. Additional temperature correction factors at ultimate loads were included as knockdowns for material strength, in lieu of testing the mount at the true service temperature. The testing of the most extreme, post-repair, outer heavyweight engine mount survived five load cases at ultimate plus loads without any obvious signs of yielding or dimensional change. The CF 188 most extreme post-repair inner lightweight engine mount survived six load cases at ultimate plus temperature correction plus at least 7% design limit loads without any obvious signs of yielding or dimensional change.

CSeries Pre-Production Fuselage Barrel Test

A. Charbonneau, Bombardier Aerospace, Commercial Aircraft

A pre-production fuselage barrel was manufactured and is currently being tested, Figure 5, as part of BA effort in characterization of Aluminum Lithium (Al-Li) alloy and as an up-front validation of design concept. The objectives of this test are:

- Develop and validate best design practices and manufacturing parameters for Al-Li Fuselage
- Validate the design and analysis methodologies.
- Correlate stress/strain prediction from FE model with strain gauges data.
- Demonstrate structural integrity of the major joint concepts.
- Provide early identification of potential fatigue crack sites.
- Validate typical repairs and allowable damages.
- Develop and validate NDI inspection techniques for primary structural components and typical repair.
- Assess 2 bay crack, propagation and residual strength
- Validation of design principles against Wide Spread Fatigue Damage, hence ensure anticipated operating limit.

Figure 5. Images of Al-Li fuselage barrel and test rig

To achieve these objectives, a fuselage barrel representative of the production design was manufactured by the fuselage supplier. The test specimen was heavily instrumented with the plan of applying the following loading conditions:

- Static testing at 50% of design limit load for a few loading conditions
- Durability testing phase (Phase 1) with the application of fatigue spectrum loading for two lifetimes of 60000 flight cycles
- Damage Tolerance testing phase (Phase 2) with the application of fatigue spectrum loading for one lifetimes of 60000 flight cycles with artificial damages
- Residual strength tests with the application of residual strength loads with artificial damage of critical dimension

The static testing was done in December 2009 and the fatigue cycling started in February 2010. The Damage Tolerance phase (Phase 2) was completed on 17 January 2011 with no major finding. Artificial damage and further instrumentation are being incorporated with a target to restart the test in March 2011. Typical structural repairs and allowable damage were incorporated into the specimen during Phase 1.

In conclusion, testing will simulate three aircraft design lives through 180,000 test cycles, see Figure 6. An extra step to ensure the aircraft will meet schedule and customer expectations

Figure 6. Image of fuselage barrel test rig

Bell Helicopter Major Airframe Composite Components Certification

Bell Helicopter Textron Canada Limited

As part of the certification of a new helicopter design, full scale fatigue testing of major airframe composite components, Figure 7, are being conducted by Bell Helicopter Textron Canada Limited (BHTCL), in collaboration with the full scale test facility of Bell Helicopter Textron Inc. (BHTI). Components tested are a horizontal stabilizer, Figure 8, a vertical fin, an aft fuselage and a tailboom. All tests were conducted in an environmental chamber in order to account for operational environmental conditions. The test specimens included simulated manufacturing defects and impact damage as well as typical field repairs. Fatigue and residual strength tests have been successfully completed.

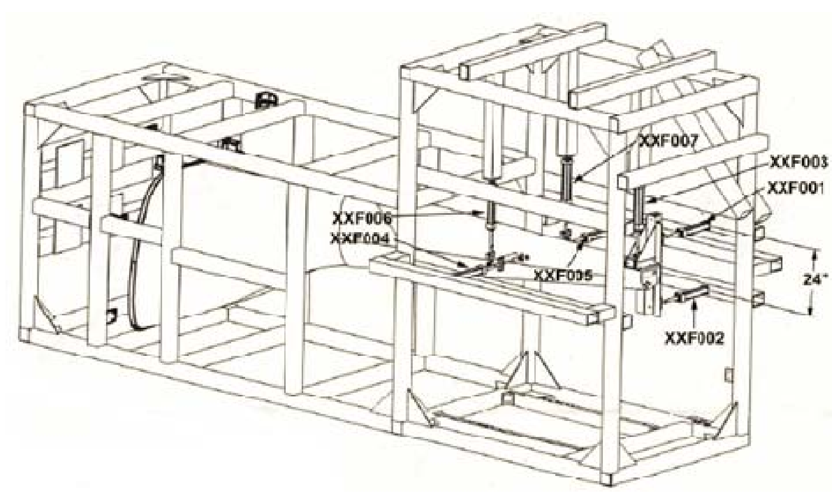


Figure 7. Tail Boom & Aft Fuselage Test Fixture Schematic

Figure 8. Horizontal stabilizer test fixture

Bell Helicopter Major Airframe Metallic Components Certification

Bell Helicopter Textron Canada Limited

As part of the certification of a new helicopter design, full scale fatigue testing of major airframe metallic components have been conducted by Bell Helicopter Textron Canada Limited (BHTCL), in collaboration with the full scale test facility of Bell Helicopter Textron Inc. (BHTI). Components are roof structure, main rotor control actuator support sub-structure, Figure 9, and a horizontal stabilizer spar tube. The actuator support fittings and the spar tube were tested in the high cycle fatigue regime. Both tests have been successfully completed. The roof beams test is conducted in the low cycle fatigue regime and is still in progress.

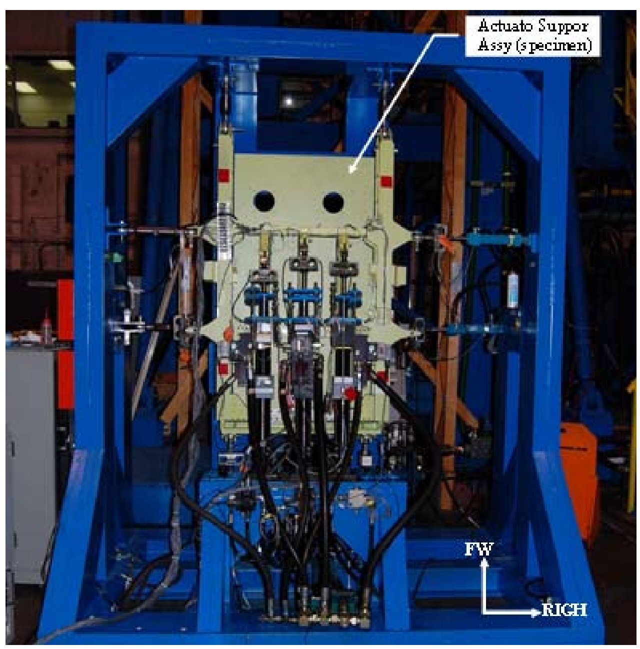


Figure 9. Main rotor control actuator support sub-structure test fixture

Fatigue Life Prediction and Enhancement

Simulation of Intergranular Crack Nucleation and Evolution in Polycrystal Metals on Mesoscale

G. Shi and G. Renaud, National Research Council Canada

A major challenge in life cycle prediction and management of aircraft structural components is the lack of information on the crack nucleation and short crack propagation stages. However, for many airframe materials, the life of an aircraft component could be almost completely exhausted within these two phases. Therefore, studies on smaller-scale levels, such as the microscopic scale or the mesoscale, are strongly needed to better understand the physical nature of the crack nucleation and propagation.

The National Research Council Canada is carrying out research on the development of a simulation method for the study of the mechanisms of intergranular crack nucleation and propagation in polycrystal metals on the mesoscale. Microstructural geometry models were built randomly using Voronoi techniques. Based on these grain structure geometry models, two-dimensional grain structure finite element models were created using a Patran Command Language (PCL) program which can statistically assign grain material properties and orientation angle in each grain to represent the heterogeneity of the grain structure. Techniques for the implementation of the cohesive elements between grain boundaries were developed in PCL for the generation of two-dimensional cohesive models, which included the automatic insertion of cohesive elements between grain boundaries, calculation of grain boundary misorientations based on random angles in each grain, properties of the cohesive model, and criteria for crack nucleation and evolution.

Simulations on intergranular crack nucleation and evolution along grain boundaries using two-dimensional finite element cohesive models were carried out on the meso-scale level. The cohesive zone model used in the study was composed of a bi-linear traction-separation relationship, a stress threshold for crack nucleation and an energy based damage evolution law. Several aspects that affect the crack nucleation and propagation were studied, which included random grain geometries, grain boundary misorientations, grain boundary peak strength, grain boundary fracture energy, grain property, and grain plasticity.

The simulations demonstrated that the cohesive model is a useful and efficient modeling tool for the study of the intergranular crack nucleation and evolution on the mesoscale level. The simulation results showed that the factors studied have large impacts on intergranular crack nucleation and evolution based on the current model capabilities and conditions. For instance, grain boundary cohesive properties, such as the grain boundary peak strength and the grain boundary fracture energy, directly affect the intergranular crack nucleation and evolution. The lower the grain boundary strength, the earlier the cracks nucleate. The lower the grain boundary fracture energy, the earlier and faster the cracks propagate after nucleation occurred. Different grain geometries and grain boundary misorietations result in different crack nucleation and evolution patterns. Moreover, the grain material properties and plasticity also have an influence on the crack nucleation and evolution along grain boundaries.

Advanced Bonding for Aircraft Fatigue Enhancement

P.R. Underhill, D.L. DuQuesnay and H. Britt, Royal Military College of Canada

During the initial phase of this project, a number of important advances have been made. Work with ternary silane mixtures has shown that wedge test crack lengths closely approaching zero crack growth can be achieved. Using Bis (trimethoxysilyl) ethane (BTMSE), the ternary silane mixtures outperform the sol-gel method when FM73 was the adhesive. Unfortunately, it seems that it will be necessary to replace BTMSE with its ethoxy analog, bis (triethoxysilyl) ethane, (BTESE). This compound is very poorly soluble in water and hydrolyzes much more slowly than its methoxy counterpart. Approaches to alleviate these issues are currently under investigation including approaches suggested by the sol-gel method.

A gloss meter has been shown to give consistent readings when used with the grit-blasting. It can be used to indicate when over-blasting has occurred, which produces excessive voiding in FM73. However, it has also been shown that a modified form of the grit-blasting gun can produce very uniform treatments with no chance of over-blasting. The gloss meter's best application may be in qualifying such gun improvements. The applicability of the gloss meter to abraded surfaces is much more questionable as these surfaces produce much more variability in gloss indicative of the poor control on the abrading process. The gloss meter may be useful in developing abrasion techniques that are much more reproducible.

The simplest alternative to grit-blasting is abrasion. The application of UV cleaning to abraded surfaces has shown promise and is under further investigation. Application time has been shown to be a very important parameter when applying the sol-gel application to abraded surfaces. This was an unexpected result based on work with grit-blast surfaces, which are relatively insensitive to application time. Work is currently underway to see how manipulating this parameter can bring us close to what can be achieved with grit-blasting. Work is also underway to see if the same effect is present when silane pre-treatments are used.

AE9696 outperformed FM73, especially when used with the sol-gel method. There are issues surrounding achieving bond line uniformity. It is believed that these are now under control and that more repeatable results can now be obtained.

Fatigue Assessment of Aluminum Alloys in Aging Aircraft

D.L. DuQuesnay and P.R. Underhill, Royal Military College of Canada

The principal objective of this project in its final year was to determine what aspects of the load spectrum affected the fatigue scatter. Two aircraft spectra were examined, a CF-188 fighter spectrum and a CC-130 transport spectrum. The alloys examined were a service exposed 7075- T6 Hercules wing skin. Since skin material is commonly made from the more damage tolerant 2024-T3 alloy, the transport spectrum was also examined on specimens machined from a rolled sheet of 1/8" thick 2024-T351 alloy. Furthermore, since some of the most fatigue sensitive details in aging aircraft are fastener details, a study on the effect of cold expansion on the standard deviation of the fatigue life distribution for 0.25 nominal fastener holes was also undertaken. For the cold expanded holes, the specimens were machined from as-received 1/8" thick sheets of 7075-T651 alloy.

Transport spectra gave lower scatter (lower standard deviations of fatigue life distributions) than fighter spectra because of the compressive peaks and compressive ground cycles present in transport spectra. Tension dominated spectra therefore have higher scatter than spectra with compressive peaks and with compressive cycles. For a wide variety of loading spectra, the standard deviation of the log-life distribution is generally less than 0.1 for a variety of 7xxx aluminum alloys and 2024 aluminum alloy. The current practice of using a standard deviation of 0.1 for aluminum structural details is generally appropriate and may be conservative. However, there are conditions where a higher standard deviation (and hence scatter factor) may be required, such as in cold worked fastener holes. Very high scatter was observed in cold expanded fastener holes in a 7075-T6 alloy under constant amplitude loading, although the scatter for similar cold expanded holes under a fighter spectrum was in line with unworked holes.

CP140 Implementation of Measurement-Based Spectra

Modecai Oore, I.M.P. Group Ltd.

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 85th percentile spectrum. The SLAP discovered many fatigue limiting areas which have had a direct impact on the airworthiness of the CP140/A fleet. Of note are the findings in the region of the inboard nacelle where the front spar (web and cap) and lower planks were found to have extensive fatigue 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 gauges mounted in the wings (WS92, WS147 and WS223) and one in the tail (HSS50) supplied the structural strain data.

Currently the CP140 Estimated Life Expectancy (ELE) is based on the results of the SLAP Full-scale fatigue test (FSFT) which was run with the USN 85th percentile spectrum. Fatigue and crack growth lives for structurally significant items (SSI) on the CP140/A fleet are currently determined using spectra based on parametric Mission Profile data specific to the CF. This analytical tool has been found to produce spectra that are overly conservative resulting in unnecessarily short crack growth life predictions. These short life predictions instigated the development of spectra based on recorded CF flight data (SDRS data).

In January 2009, the SDRS validation report was released as the culmination of considerable effort to validate the SDRS, which included in-flight tests of an instrumented aircraft. 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. The outcome of the study was a fully validated SDRS that could be used to develop measurement-based spectra.

Three follow-on steps were required to make use of the SDRS data:

a. Develop a Fleet Averaged Reference Spectrum (FARS) for each of the four SDRS strain gauge locations that represent the average use of the CP140/A fleet.

b. Determine Transfer Functions (TF) to transfer the FARS from each of the SDRS strain gauge locations to other locations of interest (such as SSI) on the aircraft.

c. Calibrate the crack growth software FASTRAN to the FARS.

FARS Development:

The FARS development made use of over 60,000 hours of flight data and was conducted in two phases: a usage study and determination of the block size. The usage study compared the SDRS data from 1996 to 2008 to identify any trends and/or shifts in usage of the CP140 fleet over time. This was used to determine the 'baseline' dataset from which the FARS was developed. The block size determination narrowed the baseline dataset down to a block size of 10,000 hours so that it could be stored and processed more readily, while still maintaining representation of the fleet usage. Once the FARS was built for each of the four strain gauges, a study to determine sensitivity to aircraft order and if there is a need for a randomization procedure was conducted using FASTRAN. The study showed little impact on life due to changes in aircraft order or randomization.

TF Development:

To make use of the FARS at locations other than where the SDRS gauges were installed, TF were required to transfer the spectra. The first step in TF development was to ensure that good linear correlation existed between the SDRS gauges and the areas on the aircraft where the analyses would be performed. A survey using all of the wing strain gauges from the FSFT proved that excellent correlation exists for the majority of the wing structure. For fatigue life predictions, TF that consisted of stress factors from the SDRS gauge location to the location of interest were found to give good results. These TF are developed using detailed Finite Element Models (FEM) of the aircraft.

Crack growth was found to be much more sensitive to the TF value and full linear TF were developed that included slope and offset components. The TF for crack growth are developed using the Flight Test Instrumentation (FTI) strain gauge data from the SDRS validation flight test. Data from the SDRS gauges is plotted against the FTI data from the other strain gauges installed on the aircraft and linear regression is performed to obtain a TF. To obtain TF to locations between the FTI gauges, stress factors obtained using detail FEM are used to modify the slope of the TF equation and triangulation is used to determine a modified offset. Validation of the methodology is still underway although preliminary results are promising.

FASTRAN Calibration:

FASTRAN is currently calibrated to coupon test data developed using analytical spectra representative of the USN usage. A re-calibration effort has been initiated using analytical spectra representative of Foreign Military Sales spectra, including two Canadian Forces (CF) spectra. The re-calibration coupon testing spectra were derived using the SLAP analytical spectra development tools; therefore, they cannot be considered representative of the SDRS FARS. As such, IMP developed a plan to conduct additional testing to calibrate FASTRAN to the CP140 SDRS FARS.

Several of the tests have been completed; including tests using the FARS for each of the three wing strain gauges. However, several additional coupon tests were planned to confirm the applicability of the calibration rates to different spectra and crack growth configurations. Only one of these validation tests has been completed to date. A preliminary FASTRAN calibration has been carried out using the data currently available; however, validation is required before the calibrated FASTRAN can be used for activities impacting the airworthiness of the fleet, i.e. Damage Tolerance Analyses for SSI processing, Individual Aircraft Tracking (IAT) and Non-Standard Repair (NSR) analyses.

Upon completion of the FASTRAN calibration and the TF derivation methodology, it will be possible to re-process all of the currently defined SSI in the fleet. The FARS-based spectra will also be used with new SSI and NSR. Initial results and experience have shown life prediction increases of two to three times over the SLAP spectra generation tools which will have a significant impact on determining the Safe Operating Life of the aircraft in the fleet.

A Method for Efficient Determination of Inelastic Plastic and Creep Stresses and Strains near Notches and Cracks

Grzegorz Glinka, University of Waterloo

A method for efficient determination of plastic and creep strains in notched and cracks bodies has been developed. The method is based on the discovery of the independence in localized plastic or creep zones of the strain energy density on the material stress-strain curve. It means that the strain energy density in the plastic or creep zone around the notch or crack tip can be determined from relatively easy linear elastic boundary solution. The advantage in using the method lies in the fact that local inelastic strains and stresses can be determined based on the linear elastic stress data without the necessity of solving the complete elastic-plastic boundary problem. Therefore the method is computationally very efficient, sufficiently accurate and particularly suitable for lengthy incremental fatigue and creep stress-strain analyses of engineering components and structures subjected to cyclic loading histories and high temperatures. The method has been developed for a general 3-D stress-field.

CF-18 Aircraft Structural Integrity Program (ASIP) and Life Extension Program (ALEX)

Jean Dubuc, L-3 Communication (Canada) Military Aircraft Services (MAS)

As part of the 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, Aircraft Sampling Inspections (ASI) and fleet findings in order to define and update the Structural Maintenance Program (SMP) of the aircraft, more specifically, the so-called ALEX Program. Considering the CF Baseline Operational Spectrum (BOS) and increased scatter factors, ALEX represents a 50% extension in life over the original design. L-3 MAS is currently conducting the definition and development of the third and final phase of the ALEX program, designated as CP3. To that end, over 250 fatigue and damage tolerance analyses have been conducted so far since 2006. In many instances, life-limited areas cannot be substantiated using safe-life principles and they are assessed against the airworthiness and logistic risk analysis methodologies adopted by the CF in order to decide on the mitigation strategy or, in some instances, acceptance of the risk. This subject is covered in a separate paper by Mr Yves Beauvais. At completion, CP3 is expected to include approximately 120 maintenance packages to be implemented at around 80% of the service life of the aircraft, starting on fleet leaders in 2012. Whereas the first two phases of ALEX entailed mostly modifications aimed at meeting the safe-life criteria, CP3 is primarily a safety-by-inspection program comprised of one-time or recurring inspections and on-condition repairs. This apparent change in philosophy from safe-life to safety-by-inspection results from the application of the CF logistic risk process where these two options are assessed, on a case-by-case basis, in terms of their cost and downtime impact, accounting for anticipated failure rates.

Throughout the ASIP Program, L-3 MAS has developed several durability and damage tolerance methodologies, as reported previously in ICAF reviews. One of the efforts that has been completed recently is the development of a methodology to account for surface renewal. This subject is covered in a separate paper by Mr Zahi Hajjar. A few other specific sub-projects within the CF-18 ASIP program are presented below.

Hole Improvement

Jean Dubuc, L-3 Communication (Canada) Military Aircraft Services (MAS)

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.

L-3 MAS reported its initial efforts in that area at ICAF 2009. Since then, a second phase of 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.

The second phase of testing has focused on configurations with low edge margin holes (e/D of 1.6, 1.3 and 1.0). It was seen that the life improvement provided by interference fit fasteners remained the same with the various edge margins. Combining split sleeve cold working with interference fit fastener installations has always provided lower lives than with interference fit fasteners alone. This effect was more pronounced at higher e/D ratios and decreased at edge margin of 1.0 where both configurations gave the same life.

Hole Cold Expansion – Low Edge Margin Interim Guideline

M. Yanishevsky, D. Backman, G. Shi, G. Li, and M. Liao, National Research Council Canada

NRC has recently completed a multi-year study of the effect of Split Sleeve Cold Expansion (SsCx) of holes in situations with low edge margins. A guideline was developed for the Canadian Department of National Defence and its Maintenance Repair and Overhaul contractors, that cites: the generally accepted practices for fastener holes, including minimum recommended edge margins and requirements for satisfactory bolt bearing and bolt shear tear-out; discusses factors that affect joint performance such as secondary bending, clearance fit holes; discusses under what loading scenarios / considerations for which SsCx is particularly effective; precautions vis-à-vis indiscriminate use of hole cold expansion without prior evaluation; effect of geometric changes in the vicinity of the holes to be cold expanded; material concerns (grain orientation and elongation effects); residual stress distributions created by hole cold expansion (through the thickness gradients and effects of sleeve orientation); origins of cracks in SsCx treated and untreated open holes and use and effect of interference fit fasteners in conjunction with cold expanded holes; complications arising from unanticipated new failure mechanisms, such as fretting occurring at doubler edges and cracks forming on free surfaces and growing in such a way as to avoid growing into holes surrounded by compressive residual stresses, and the impact these have on inspection; comparisons of X-ray diffraction (XRD) [surface elastic stresses] and Digital Image Correlation (DIC) [both elastic and plastic surface strain] measurement techniques for measuring compressive residual stresses / strains that can be used to aid modeling for stress-based, See Figure 10, strain-based and Holistic Structural Integrity Process (HOLSIP) based fatigue life predictions; coupon level experimental experience with spectrum fatigue testing of SsCx treated and untreated holes with interference fit Hi-Lok fasteners in three types of coupons representing “symmetrical” structural joints, Figure 11, NO LOAD TRANSFER JOINTS (simulating “non-structural” joints, such as brackets, supports for conduits, fuel lines, wire bundles, etc.) MEDIUM LOAD TRANSFER JOINTS (simulating symmetrical doubler, stiffener and rib joints), and HIGH LOAD TRANSFER JOINTS (simulating symmetrical splices in major load carrying members) consisting of AA 7075-T6 mains with AA7075-T73 doublers; component detail level experimental experience with AA7075-T6 material and SsCx as well as opportune times to undertake hole cold expansion to support Life Extension are discussed.

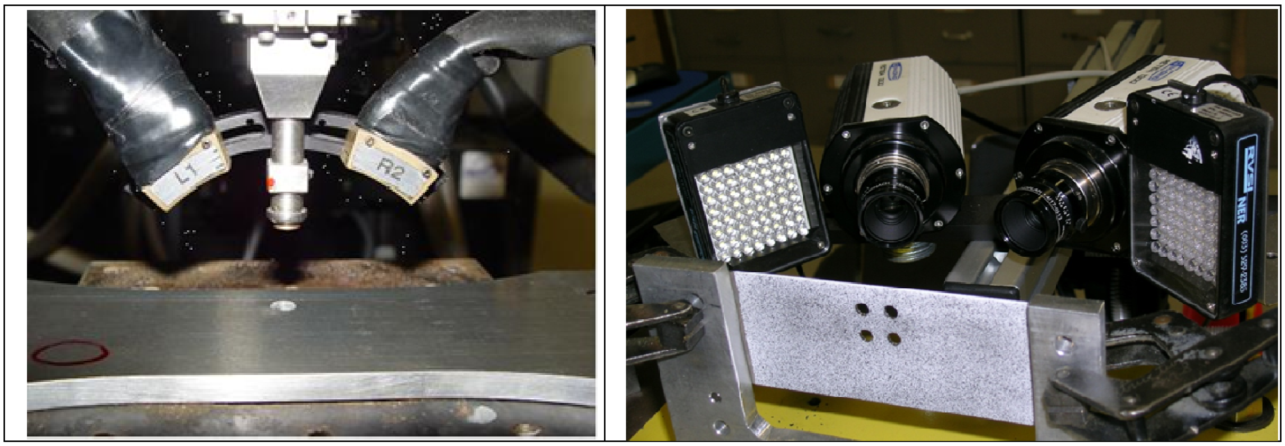
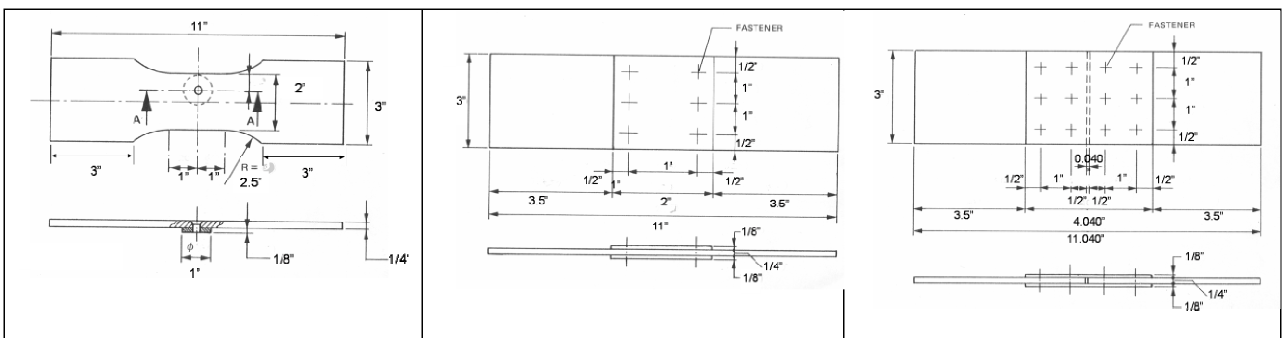


Figure 10. X-Ray Diffraction (left) and Digital Image Correlation (right) measurement of residual stresses



No Load Transfer	Medium Load Transfer	High Load Transfer

Figure 11. Schematics of the three symmetric load transfer joints investigated

Probabilistic and Risk Analysis Methods

Advanced Damage Tolerance and Risk Analysis Methods/Tools for Aircraft Structures Containing MSD (multi-site fatigue damage)/MED (multiple element damage)

Min Liao, National Research Council Canada

In 2009-2011, NRC continued developing the advanced damage tolerance and risk analysis methodologies and tools for aircraft structures containing MSD/MED, with the financial support from the Department of National Defence (DND) of Canada. The major achievements are summarized below;

1) Enhanced NRC β-factor library with new/revised solutions: including unequal diametrical cracks in an infinite plate [1]; edge cracks growing through a hole with various boundary conditions [2]; unequal diametrical cracks at an offset open loaded hole[3], continuing damage modelling [4]; and revised Poe's β-factor for a stiffened panel with fastener flexibility and arbitrary located stringers [5] (Figure 12);
2) NRC generic FE-based β–factor tool: to calculate β-factors representing configurations with load transfers to adjacent structures, non-planar crack paths, complex boundary conditions, and irregular 3D geometries. Three parametric models were developed, as shown in Figure 13 [6][7][8]. These models were packaged as MS-Excel tools to automatically run StressCheck FE analyses and generate β-factor curves (Figure 14);
3) MSD/MED crack growth analysis program, CanGROW (Figure 14): developed at NRC for MSD evaluation which is capable of growing multiple cracks simultaneously and to calculate the stress intensity factor by compounding a set of β-factors from the β-factor library and/or the FE based β-factor tool [5][9];
4) MSD/MED residual strength analysis tool: using the fracture toughness (Kc) criterion, the net section yield (NSY) strength criterion, and a plastic zone link-up (PZL) criterion. For built-up structures, a FE-based global load reduction factor is also used in the NSY criterion of the cracked component [6][8][9];
5) Monte Carlo based risk analysis method/tool: with enhanced ProDTA and CanGROW, the NRC tools can calculate the MSD/MED crack size distribution with different initial cracking scenarios, and determine the single flight probability of failure (SFPOF) for complex aircraft structures with/without MSD/MED [8][9].

Using the developed methods and tools, NRC provided timely support to the CF RARM (record of airworthiness record management) process on a regular basis and quick turn-around time [9][10][11][12][13] (Figure 15). Recently, these methods/tools were also applied for the CF aircraft service life assessment. NRC expects that these developed methods/tools be applied to help aircraft companies to establish a LOV (limit of validity) and/or operational limit for their existing or future airplanes, with the support of test data or in-service evidence.

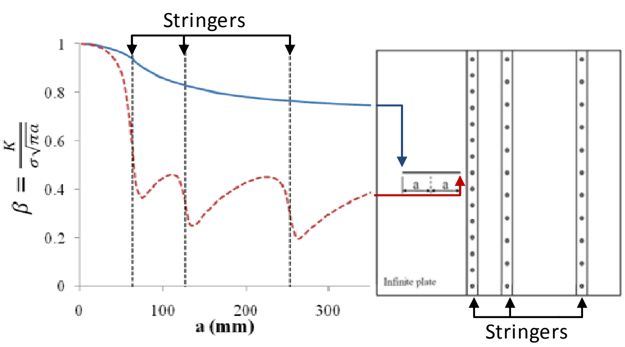
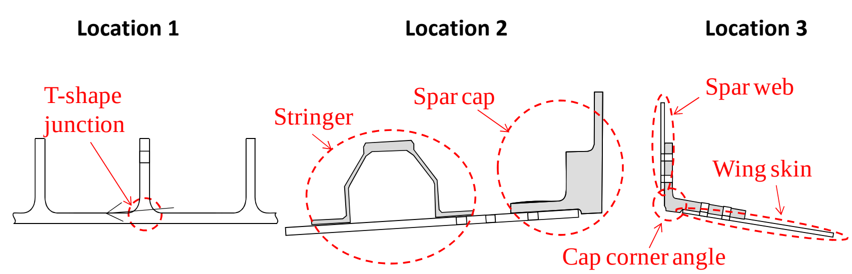
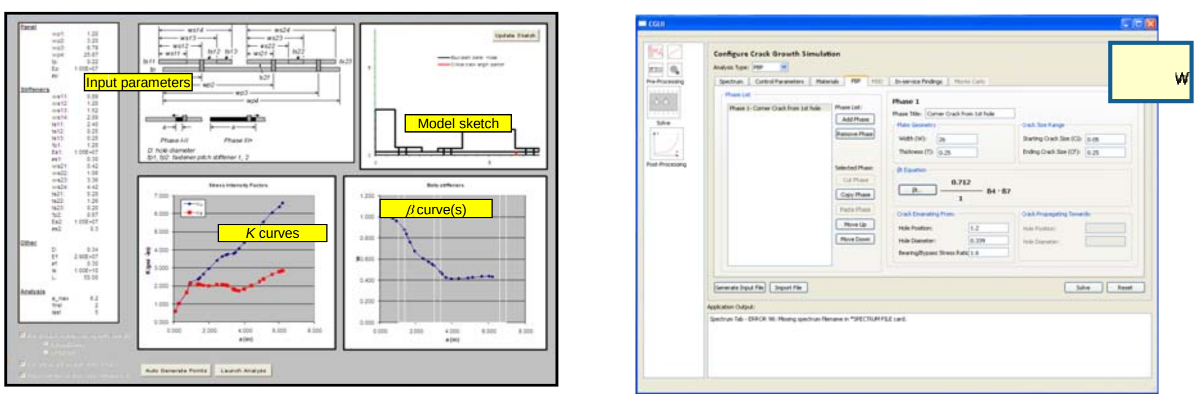


Figure 12. β-factor of a crack growing in an infinite plate with arbitrarily located stringers



a) β: geometry **b) β: adjacent structures** **c) β: adj. struct. + geo**

Figure 13. Three typical complex configurations within NRC generic beta tool



a) Screenshot of the generic FE-based beta tool for Location 2 **b) MSD crack growth code, CanGROW (beta GUI version)**

Figure 14. NRC advanced DTA tools

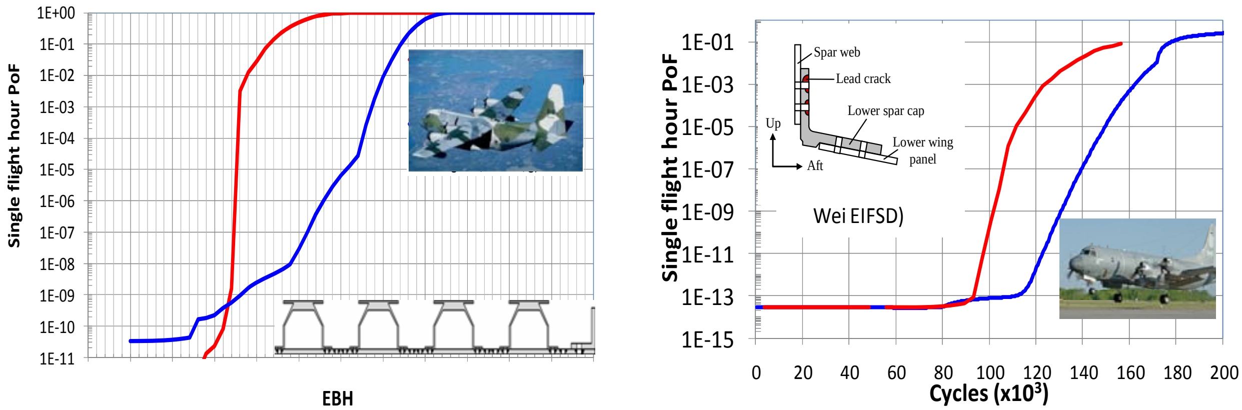


Figure 15. Risk analysis examples for supporting CF fleets

Fracture Mechanics and Crack Propagation Studies

Development of Two Parameter Fatigue Crack Growth Model for fatigue Crack Growth Analysis under Spectrum Loading

Grzegorz Glinka, University of Waterloo

The fatigue process is often considered to be composed of two stages, i.e. fatigue crack nucleation and fatigue crack growth. Two different methods are used in order to analyse fatigue durability associated with each stage of the same phenomenon. This imprecise division is the base for the only contemporary approach to mechanical fatigue life prediction and it has caused a lot of difficulties and errors while used in practice. Therefore a new Fatigue Crack Growth model has been developed based on detailed analysis of the physics of the phenomenon and accurate estimation of the stress and strain history ahead of the notch or crack tip. The novelty of the proposed fatigue crack growth model lies in the fact that the model can be used for assessing fatigue durability of smooth, notched and cracked machine and structural elements without the necessity of considering several vaguely defined fatigue stages. The entire fatigue process is considered as a process of successive crack increments resulting from material damage in a highly stressed region such as the notch or crack tip neighborhood.

The fatigue crack growth was predicted by simulating the stress-strain response in the material volume adjacent to the crack tip and estimating the accumulated fatigue damage in a manner similar to fatigue analysis of stationary notches and smooth machine components. The fatigue crack growth driving force was derived on the basis of the local stresses and strains at the crack tip using the Smith-Watson-Topper (SWT) fatigue damage parameter: D = σmax·Δε/2. It was found that the fatigue crack growth was controlled by a two parameter driving force derived in the form of $\Delta\kappa = K_{\max,tot}^{p}\Delta K_{tot}^{(1-p)}$ contrary to most of the one parameter models involving only the applied stress intensity range ΔK. The two parameter driving force enabled correct prediction of the effect of the mean stress including the influence of the applied compressive stress and tensile overloads, the overload effect and the effect of the internal (residual) stress induced by the reversed cyclic plasticity.

The new element differentiating the proposed model from all contemporary fatigue crack growth models is the use of the resultant maximum stress intensity factor $K_{max,tot}$ and the resultant stress intensity range ΔK_{tot} accounting for residual stresses induced by the reversed plastic deformation in the stress concentration region. An additional novelty is the memory model accounting for the stress history effect on growing (non-stationary) fatigue cracks. In effect the model can be applied for predicting fatigue lives of cracked, notched and smooth machine components subjected to lengthy variable amplitude stress spectra without the necessity of using empirical models based on the closure concept.

The method enables accurate determination of fatigue lives and prognosis of inspection periods for real engineering problems without the necessity of using approximate solutions.

The model requires the knowledge of only the material stress-strain curve and the strain-life curve obtained from smooth material specimen tests or the stress-strain curve and limited constant amplitude fatigue crack growth data.

The model has been successfully validated for a variety of aluminum alloys, steels and titanium alloy under a variety of variable amplitude loading spectra. The model can also be used for the prediction of fatigue crack growth in corrosive environments.

The project is sponsored by the Office of Naval Research, Washington DC, USA.

A Method for Calculation of Stress Intensity Factors and Fatigue Crack Growth Analysis of Arbitrary Planar Cracks Subjected to 2-D Arbitrary Stress Field

Grzegorz Glinka, University of Waterloo

Fatigue cracks in shot peened and case hardened notched machine components are subjected to stress fields induced by the external load and residual stresses resulting from the surface treatment. Both stress fields are characterized by non-uniform distributions and handbook stress intensity factor solutions are in such cases unavailable, especially in the case of planar non-elliptical cracks as that one shown in Figure 16.

The method presented in the paper is based on the generalized weight function technique (Figure 16) enabling the stress intensity factors to be calculated for any Mode I loading applied to arbitrary planar convex and embedded crack. The stress intensity factor can be determined at any point on the crack contour by using one general weight function in the form of expression (1).

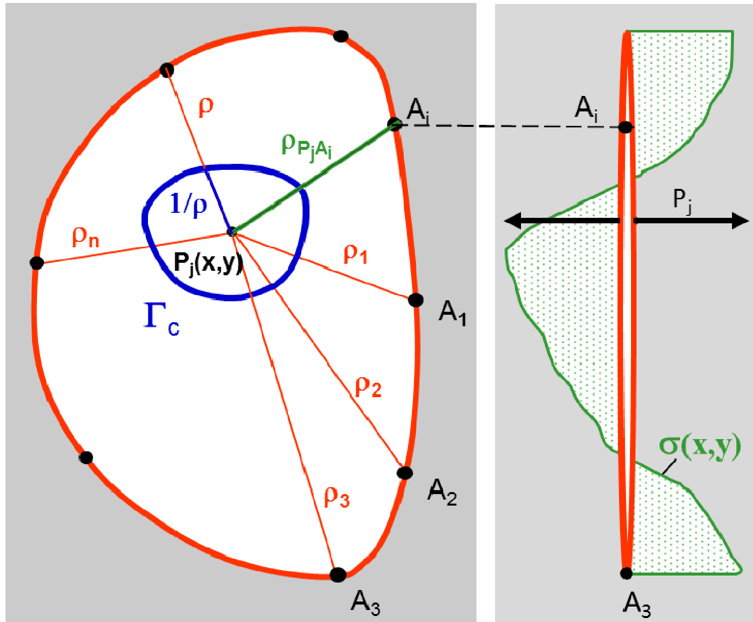


Figure 16. Definition of the weight function parameters and the inverted contour Γ_c

$$m_A(x,y,P) = \frac{P\sqrt{2}}{\pi\rho^2\sqrt{\Gamma_c}} \qquad (1)$$

The weight function, m_A, can be sufficiently well described by two quantities, i.e. the distance, ρ, from the load point, P(x,y), on the crack surface to the point, A, on the crack front where the stress intensity is to be calculated and the length, Γ_c, of the inverted crack contour. The stress intensity factors are calculated by integrating the product of the stress field and the weight function over the entire crack area.

$$K_A = \iint_{Area} \sigma(x,y)\, m_A(x,y)\, dxdy \qquad (2)$$

The general weight function and calculated stress intensity factors have been validated against various numerical and analytical data. The method is particularly suitable for modeling fatigue crack growth in presence of complex 2-D stress fields as that one shown in Figure 17.

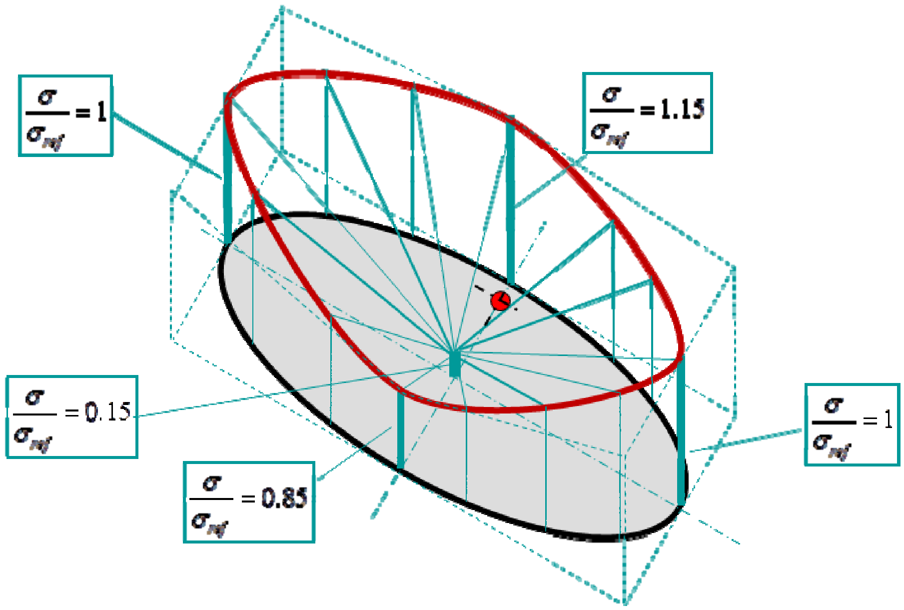


Figure 17. Stress field in a cross section of an engine valve spring containing embedded flaw

Stress intensity factors and subsequent fatigue crack increments are calculated at several control points along the crack front. As a result the fatigue crack shape evolution can be simulated up to the critical stage. The actual crack shape is simulated by a series of linear segments in order to increase the efficiency of the numerical integration of the weight function. An algorithm for the analysis of fatigue crack growth of planar cracks, and validation results supporting the entire methodology have been also developed.

The simple weight function of eq. (1) is accurate and replaces hundreds of simplified solutions collected in handbooks. The new additions to the method is concerned with surface breaking cracks, corner cracks (aviation, nuclear industry) and irregular cracks occurring in welded structures. The method is very efficient and has attracted great interest from both scientists and practising engineers. It has already been included into commercial software packages used and approved as design tools by NASA (in the NASGROW computer software package) and US Air Force (in the AFGROW package).

Short Crack Model Development for Airframe Life Assessment

Min Liao, National Research Council Canada

Under a three-year collaborative DND-NRC project, NRC has been continuing to develop the physics-based short/small crack models for aircraft structural life assessment. Both experiments and analytical modeling are 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. Developed and adopted new test procedures to quantify the residual stress effect on short-long crack growth, along with the quantitative fractography (QF) to support HOLSIP physics-based model development. The materials under investigation are 7050-T7452 and 7075-T73 forging.

The short crack growth tests used both SENT (single edge notched tension) and CT (compact tension) coupons, and four crack growth measuring techniques; a) marker bands; b) silicon-based replica; c) ACPD (alternative current potential drop); and d) ACR (adjusted compliance ratio). In addition, a limited number of ACR tests with a corrosion cell (containing 3.5%NaCl) are ungoing for the 7075-T73 forging material. Quantitative fractography (QF), including marker bands reading, has been performed to determine short crack size and nucleation features including particles/pores and machining marks (Figure 18). The complete test results and analyses are expected to be available before the ICAF2013.

On the modeling side, 1) a parametric StressCheck model on the SENT coupon was built in which the coupon geometry and crack shape, aspect ratio can be adjusted. The model produced accurate stress intensity factor values with only negligible differences from the published results [14]; 2) A probabilistic crack growth tool, which can model material, geometry, and loading variability, was further developed with a GUI in MS-Excel linked with AFGROW (v4.12.15). Additional crack growth rate curves for 7075-T6 (R=0.1) and 4130 Steel (R=0.1, 0.8) were generated from an ASTM Round Robin test program [15]; 3) A CTOD (crack tip opening displacement) model was modified for crack nucleation and short crack modeling considering the microstructural influences [16]; and 4) Fatigue analysis of the CF-18 aft wing fold shear-tie lug was completed using the previously developed IDS-based short-long crack growth model of 7050-T7452 [17] (Figure 19). A three-dimensional (3D) finite element model was developed to simulate the loading applied in the shear-tie lug test and determine the local stress/strain distribution [18]. Short crack data and the subsequent material model, were specifically developed for the lug material (7050-T7452 forging), in which the Holistic Structural Integrity Process (HOLSIP) approach was used to correlate the short crack model with material microstructural features, such as constituent particles and porosities. Based on the failure mechanisms, a simplified single crack model was proposed to analyze the multiple crack growth problem and the life estimation compared well with the test results. This case study showed the benefit and potential of the HOLSIP short crack research for supporting CF aircraft structural life assessment. Also it reveals the complexity of the fatigue analysis involving complex geometry, multiple cracks, and possible residual stress from the forging process, which demand further development of accurate 3D FE modeling with cracks and advanced fracture mechanics models.

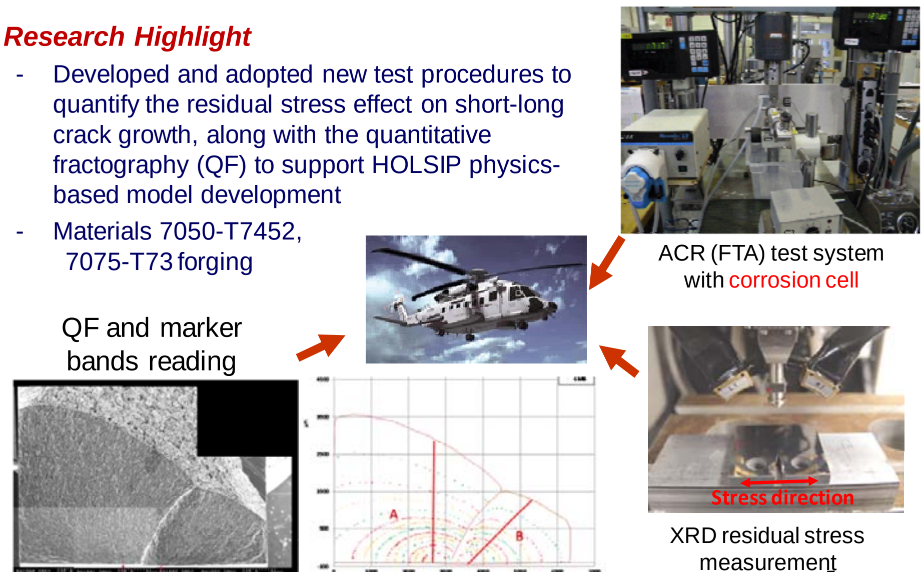


Figure 18. Short-long crack model development

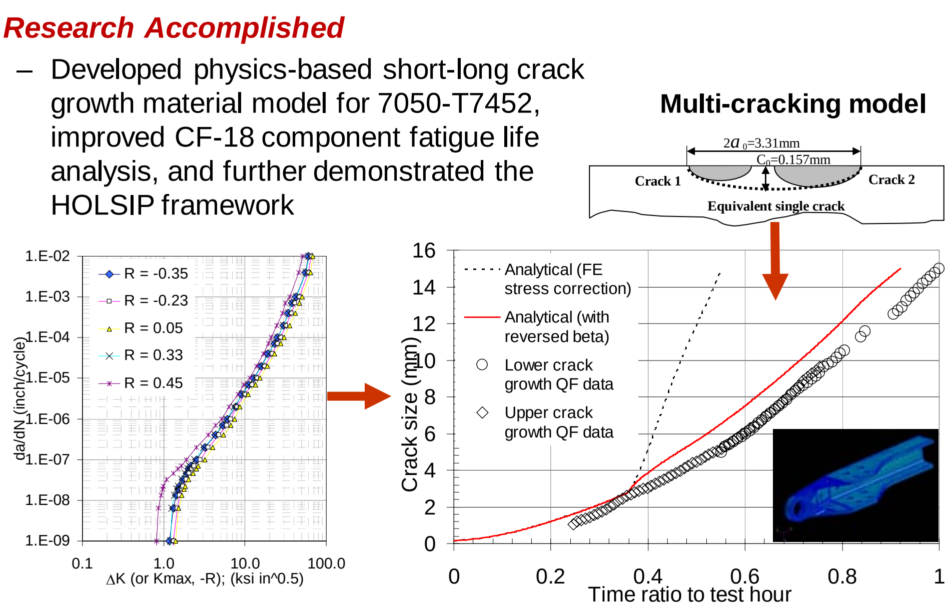


Figure 19. CF-18 component fatigue analysis case study using the HOLSIP-based modeling

Structural Integrity

Environmental Degradation of Helicopter Composite Structures

Cheung (Lucy) Li, National Research Council Canada

Composites materials and structures, in particular sandwich constructions, are widely used as load-bearing structural components on helicopters such as CH149 Cormorant; however, these structures are susceptible to in-service environmental degradation. In addition, rotary-wing aircraft operating in a marine environment may also be affected by salt condensation. Not only may salt-laden water seep through cracks, joints and broken sealant into honeycomb core, salt crystals may also accumulate in the matrix resin of facesheets and facesheet-to-core adhesive. These environmental factors can cause significant degradation and damage to composite materials during service, leading to reduced performance and service life, and in some cases, even catastrophic failure of an aircraft. These environmental factors act separately, but most often they behave synergistically in a complex interplay. Take the effect of freeze-thaw cycling and moisture for example. The moisture ingress may cause microcracking in the matrix and fibre/matrix disbonding. Freeze-thaw cycling may lead to propagation of the cracks and disbond, which in turn accelerates moisture ingress of the composite component. Thus, the synergistic effect can lead to a much greater degree of degradation than the mere sum of the two effects. Since most military aircraft are kept in service well beyond their original design lives, studies of long-term degradation would help address growing concerns about continued airworthiness and structural integrity of these composite structures.

In this study, a preliminary mathematical model was developed to simulate degradation of typical helicopter sandwich structures after long-term exposure to environmental conditions. A building block approach was used, with an initial focus on modelling of material behaviours when subjected to temperature and water, Figures 20 and 21, as well a framework model development with a goal to simulate disbond of a bonded laminate and a pristine sandwich structures under loading. The material behaviour models will then be integrated into the developed model of the sandwich structures to investigate the environmental effects on long-term structural performance. Experiments were conducted to evaluate the environmental effects on the performance coupons, after conditioned to typical and extreme service environments to which the helicopter may be exposed. The performance of the sandwich coupons was evaluated using various techniques that may include, but not limited to, NDI, thermal analysis, digital imaging correlation, fractography and mechanical testing, Figures 22 and 24. The developed model was validated by experimental case studies.

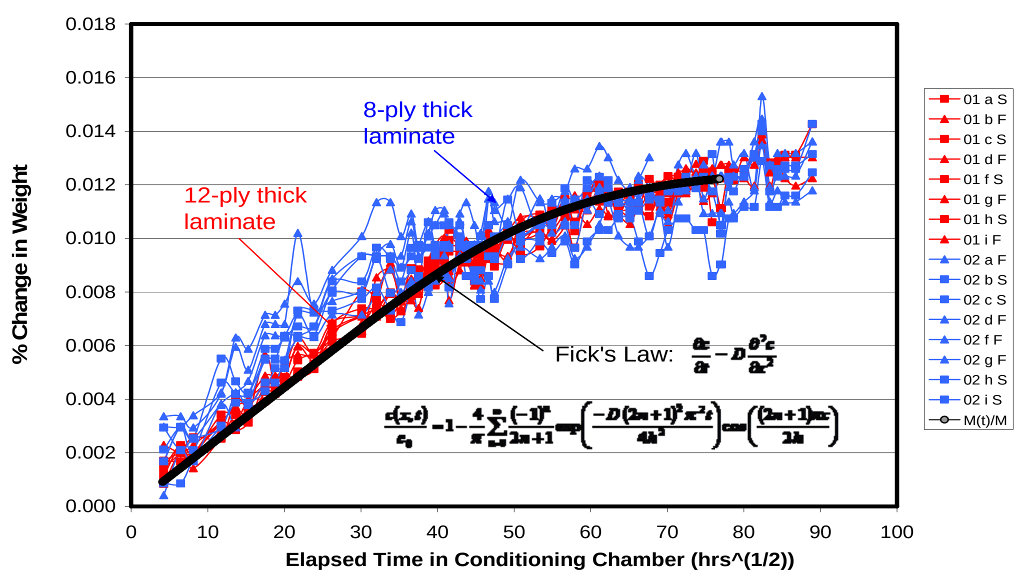


Figure 21. Percentage weight change in monolithic coupons

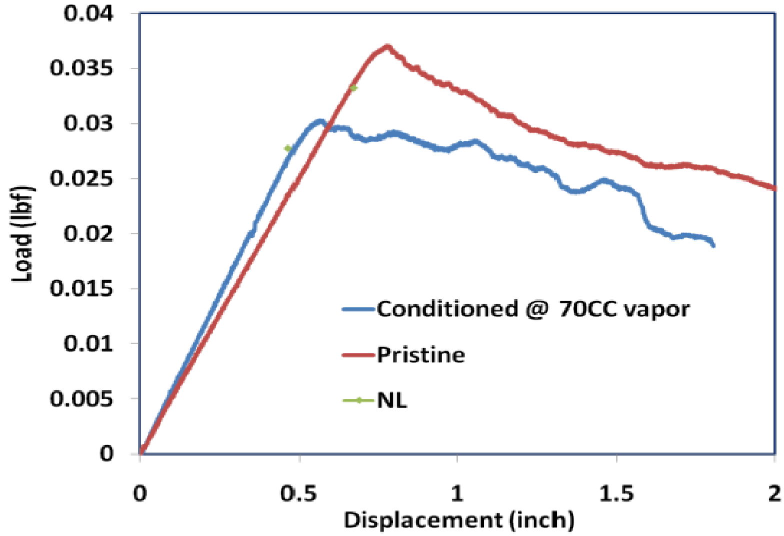


Figure 22. Reduction in fracture toughness and change in failure mode

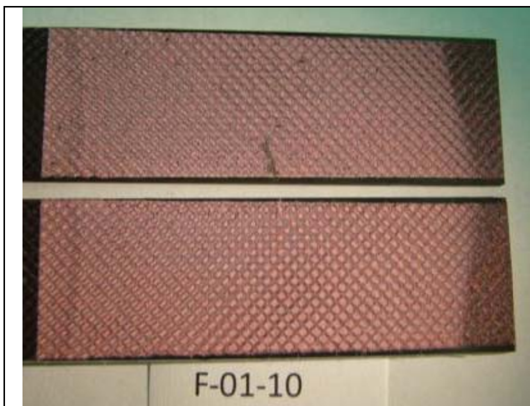
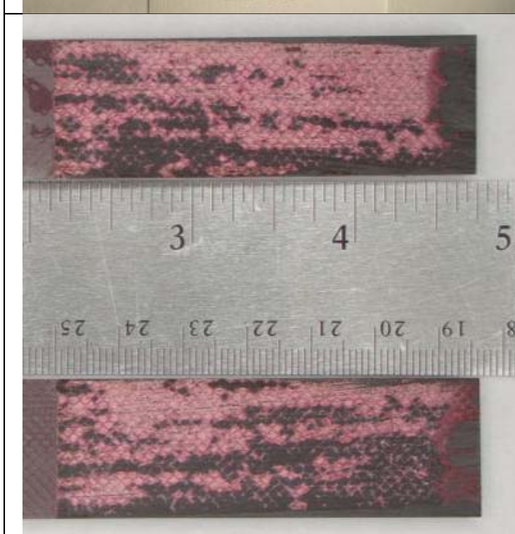
	100% cohesive failure in dry pristine bonded composites laminates
	Mixed cohesive/adhesive failure mode after environmental conditioning

Figure 22. Change in failure modes of conditioned composite coupons.

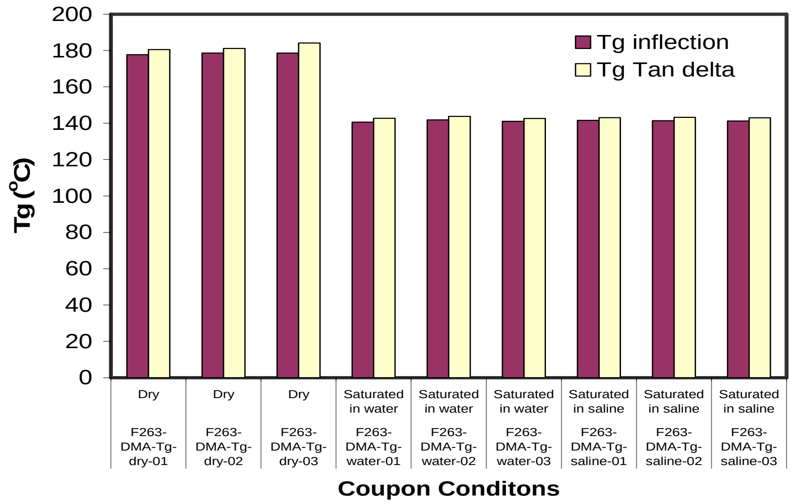


Figure 23. Reduction of Glass transition temperature by 40°C

CC115 Wing Skin Evaluation of Strain on Fatigue Life

Mr. Gamma, Kelowna Flightcraft Ltd.

As part of a recent CC115 repair, the 7079-T6 upper wing skin was plastically deformed. The most deformed areas of the upper skin were replaced with a cut-out repair, but other regions of the wing were left as is. The peak strain in the remaining skin has been conservatively estimated to be 1.75%. To substantiate the repair in fatigue, a testing program was implemented to assess the impact of pre-strain on the fatigue life of 7079-T6. The testing program is ongoing, so no results are yet available for review. The testing involved the generation of S-N and da/dN curves based on three strain levels (0%, 1.75% and 2.63%). A relative assessment of the effect of strain on the fatigue and crack growth response will be possible.

Bell Helicopter Material, Processes and Sub Component Testing

Bell Helicopter Textron Canada Limited

As part of the ongoing development of knowledge on the fatigue behavior of materials and elements subjected to various manufacturing processes, coupon and element fatigue testing was conducted by Bell Helicopter Textron Canada Limited (BHTCL). One such study consisted in determining the effect of exposure to high temperature of shot peened material.

Aging Aircraft Issues

Analysis of Spot Welds in CP140 Nacelle Longeron Repair

Modecai Oore, I.M.P. Group Ltd.

Cracks were found at No.1 Outboard Nacelle in the upper outboard longeron of CP140110. Repair doublers were applied using fasteners while keeping some of the original spot-welds. An approximate analysis was conducted by modelling the spot welds as fasteners with equivalent stiffness. This analysis was overly conservative, complicated and consumed substantial time and effort.

After the original analysis, a study was undertaken to develop an effective methodology for analyzing spot-welds so it could be applied to similar repair configurations in the future. In the study, spot-welds were identified as crack configurations and analyzed for stress intensity factors (K). Comparisons of K_I, K_{II} and K_{III} by various modelling methods were presented for this configuration. It was shown that spot-welds can be analyzed with formulae available for 3D crack geometries or by finite element methods such as MSC NASTRAN or StressCheck. It was found, however, that the crack element in MSC NASTRAN does not give the proper solution for the K_{III} mode of cracking and should not be used as such for spot-weld analyses.

CF-18 Rudder Drying

Capt. Hungler, Royal Military College of Canada

The previous technique for detecting internal damage/defects in rudders required the rudder to be cooled to -20^{o} C, and hence water that had ingressed was frozen. The rudder was then inspected in this state using thermographical technique. The results obtained were questionable.

The USN rudder drying technique involves taking X-rays to determine where water damage exists. Holes are then drilled into the affected cells to allow the water to "drip" out. This technique is not favoured by DND.

DND has developed a drying process in order to improve the results of inspecting of the rudder for internal damage/defects using thermography technique. The steps are as follows:

- A thermal couple is attached to the rudder surface in order to monitor/control the surface temperature.
- A copper plate is place on the surface to spread the applied heat over the areas of interest. The plate is then covered with a heat blanket.
- The rudder assembly is then wrapped in a satin breather cloth which in turn is encased by a vacuum bag.
- The rudder assembly is wrapped in a satin breather cloth and encased in a pre-fabricated vacuum bag, Figure 24.
- Heat is applied to the blanket and the vacuum draws the heated vapour through a condensation chamber, Figure 25, where the moisture extraction can be monitored.

Figure 24. Pre-fabricated vacuum bag installation

Figure 25. Condensation chamber

Joining Techniques

Friction Stir Welding Demonstrator

L. Kok, Bombardier Aerospace

As reported earlier in [19], work continued on the Bombardier FSW demonstrator, Figure 26. The first lifetime of testing, with pressure bending, and torque loading being applied, focused on the damage assessment of an as-built component, with any possible production waivers that may be incorporated to address non-conformance issues. After successfully completing a regional aircraft lifetime, repairs schemes to the basic welded components were made for fatigue crack propagation evaluation. Deliberate damage, cuts and compromises were introduced on the fuselage specimen for a second lifetime of testing. Cracks emanating from such deliberate damage were then monitored. In some cases the crack features continued along the weld in the case of a longitudinal cut, and in other circumferential cases through the lap welded stringer Figure 27. No evidence of rapid un-arrested crack extension was observed, just predictable crack growth rates. In a novel observation a crack emanating from a deliberate 7000 series stringer bulb cut was arrested by a longitudinal lap weld feature and the crack then turned along it. Further investigation on this type of observed behaviour may be warranted Figure 28.

Work in defect detection during processing was also advanced. As reported separately in [20] defects were shown to be observable by control parameter capture during the process and later confirmed by testing Figure 29.

After the second lifetime of testing, further residual strength tests in but the circumferential and longitudinal directions were conducted. Benign extensions were observed much as per current fuselage construction techniques as per Figure 30.

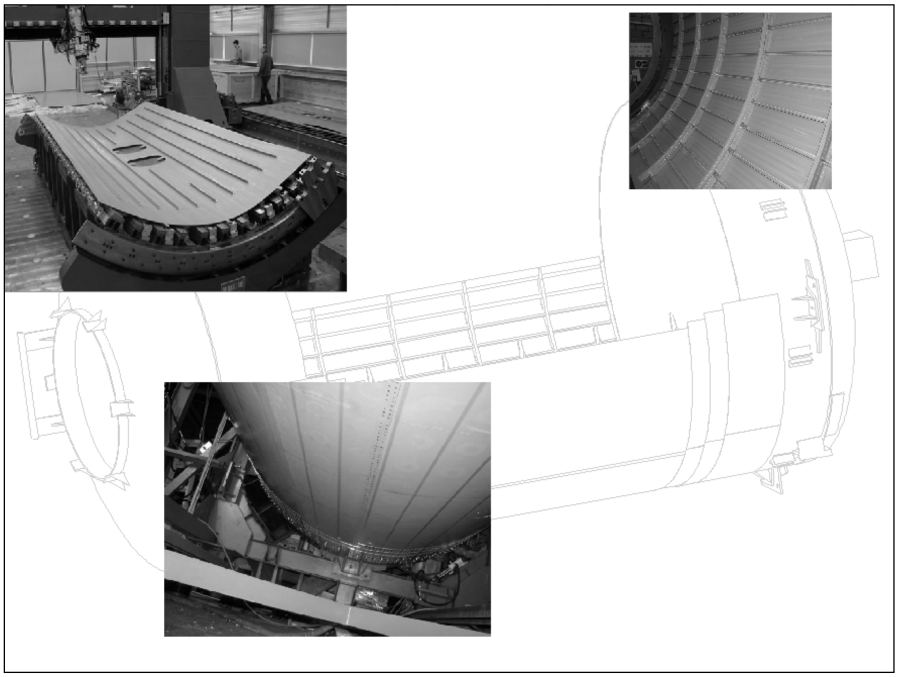


Figure 26. FSW Fuselage Airtank Test

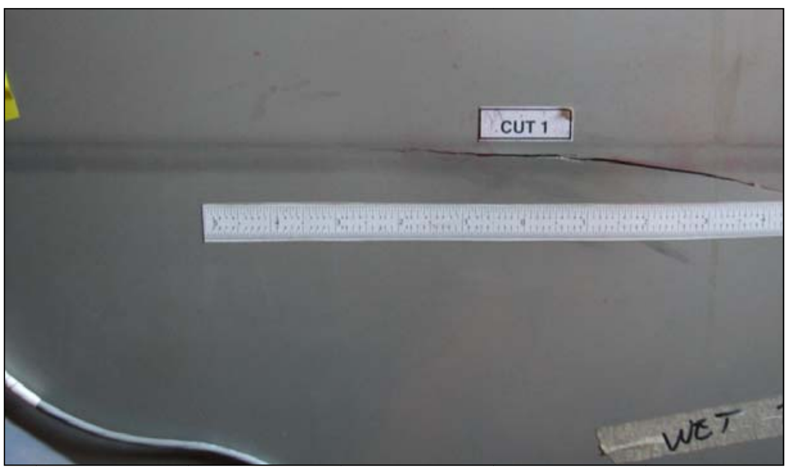


Figure 27. FSW Crack growth from Longitudinal Weld

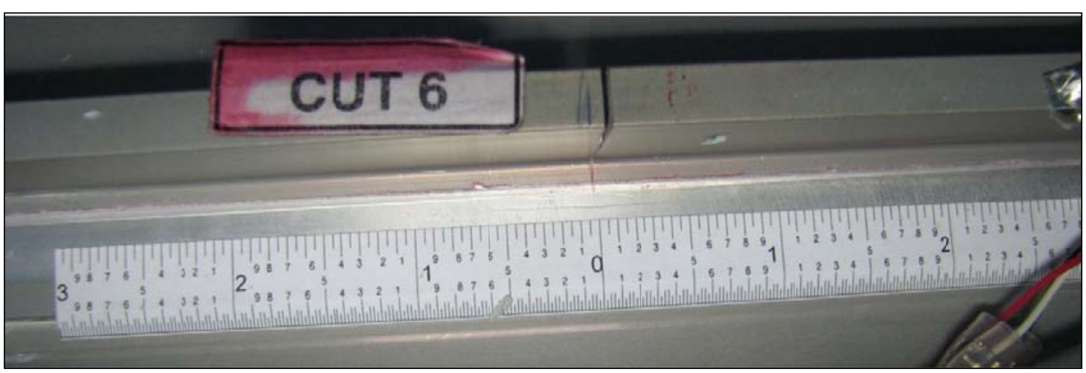


Figure 28. Circumferential Crack into Stringer Lap Weld

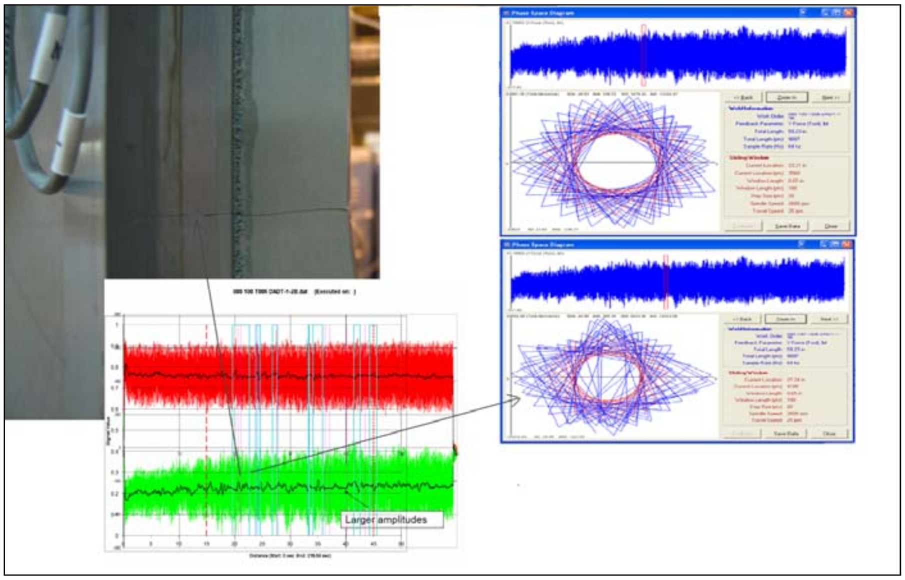


Figure 29. Weld Process Parameter Capture and Weld Defect Control

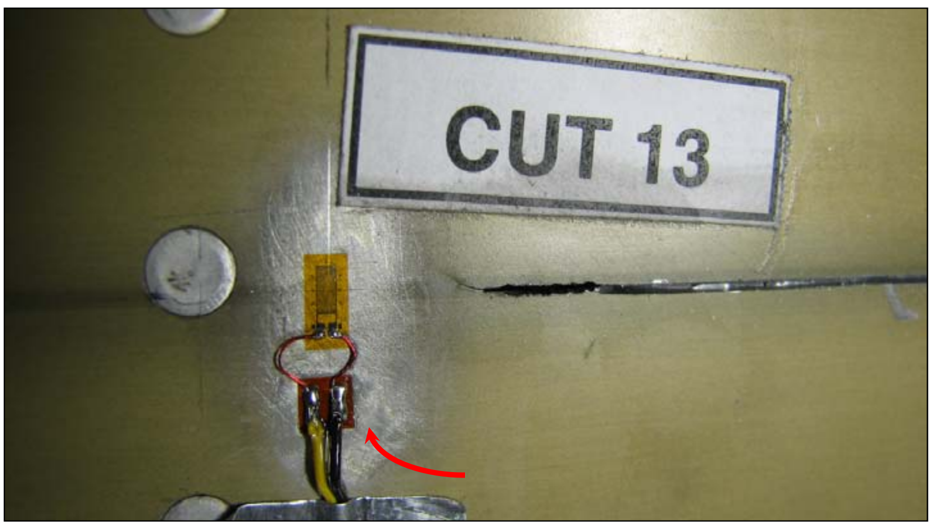


Figure 30. Benign residual strength test crack extension

Bell Helicopter Certification of a Bonded Joint on a Major Airframe Metallic Components

Bell Helicopter Textron Canada Limited

As part of the certification of a new helicopter design, full scale fatigue testing of a bonded joint on a major airframe metallic component is conducted by Bell Helicopter Textron Canada Limited (BHTCL), in collaboration with the full scale test facility of Bell Helicopter Textron Inc. (BHTI), The test specimen includes simulated manufacturing defects. The test is conducted in the low cycle fatigue regime and is still in progress.

Usage and Structural Health Monitoring

Fatigue Usage Characterization

Jean Dubuc, L-3 Communication (Canada) Military Aircraft Services (MAS)

To characterize the usage of structural locations on the CF-18, both a continuous fatigue monitoring (or tracking) program and Master Event Spectra (MES) to represent actual fleet usage were developed. Both of these types of usage characterization aim to improve individual aircraft usage estimates to optimize the structural maintenance plan and substantiate reduction of the Tracking Factor (TrF, which penalizes analyses for lack of usage characterization). Please see previous ICAF reviews for an overview of these topics.

Direct benefits for additional Fatigue Usage Indices (FUI) are to provide individual aircraft cumulated usage (to optimize the SMP and sometimes remove some over-conservatism through use of the TrF) and to help define usage targets for new MES requirements. However, the effort to develop FUIs is considerable and another approach is needed to characterize usage over the vast majority of structure.

For locations that are not tracked (or do not correlate to a tracked location), new MES were developed to estimate individual aircraft usage. This approach required development of several MES to properly characterize different historical usage periods. Moreover, an investigation was carried out to determine the usage target levels required (ex. 90th percentile usage) to warrant reducing the TrF. The development of new MES, involves not only the characterization of the flight envelope usage but also the development of structural fatigue loads. These loads are required for lifing purposes of the aircraft structure including on risk analysis impact.

The requirement for and type of additional usage characterization was determined using the decision process shown in Figure 31.

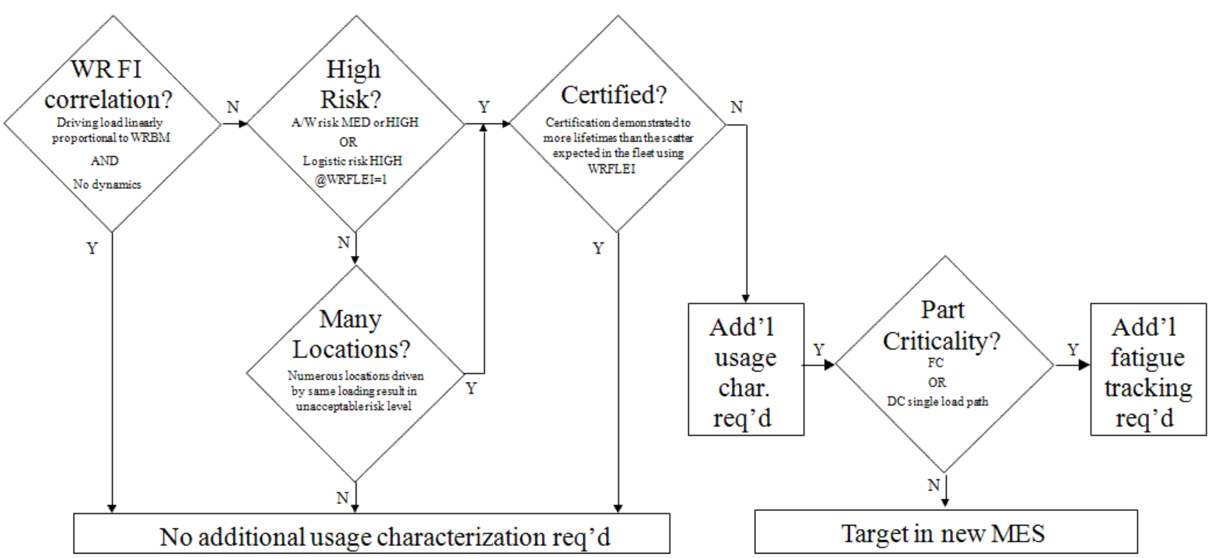


Figure 31. Additional Tracking Indices - Proposed Decision Process

For locations without any fatigue usage characterization, under the current lifing policy, we would be required to life and track these locations using an existing fatigue life expanded index (FLEI) and a TrF. Figure 32 shows an example of a location (Loc B, here Aft Fuselage Splice) that accumulates damage slower than a tracked location (Loc A, here Wing Root). In this particular case, the lifing methodology proves to be overly conservative, resulting in unnecessary maintenance actions for many aircraft.

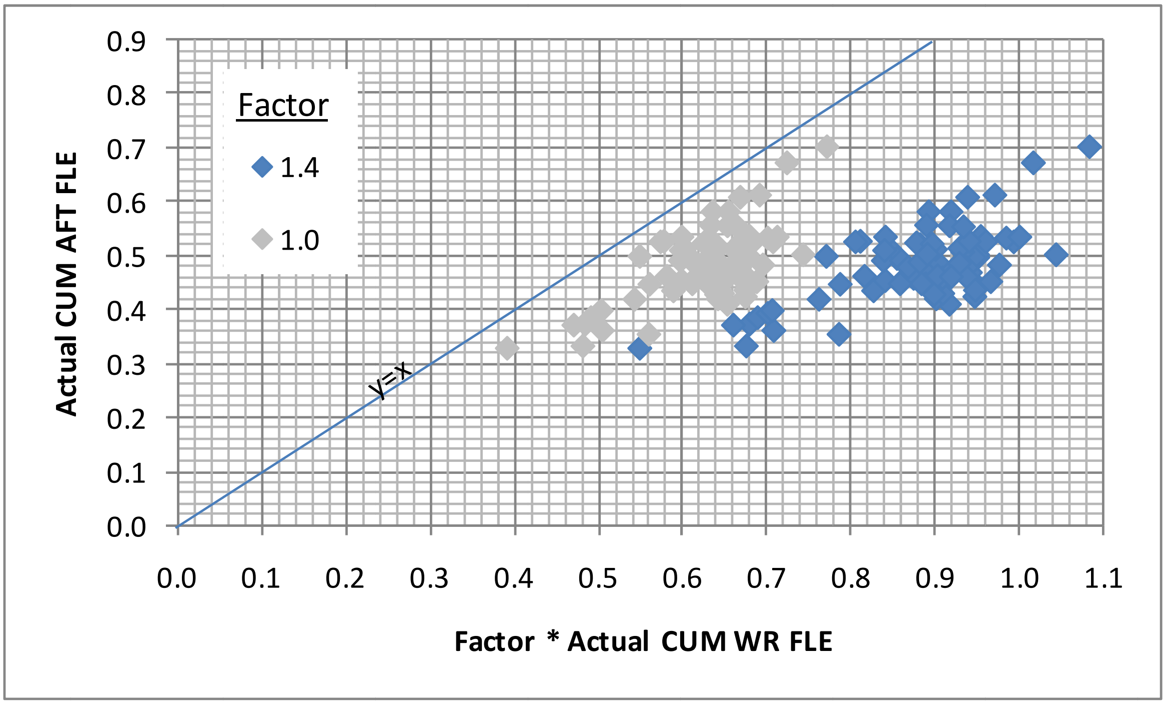
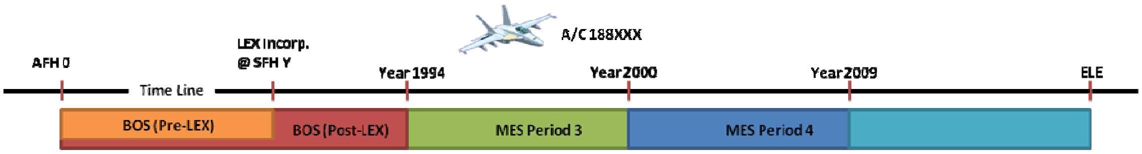


Figure 32. Non-tracked vs. SLMP-tracked Aft Fuselage Splice

Now without going through the effort of developing a FLEI for Loc B, we could develop MES representative of the various important flying periods in the CF-18 history. For the purposes of this example, we assume that 4 periods exist to date:



- the pre-LEX fence usage period;
- LEX fence usage to end 1993;
- LEX fence usage from 1994 to end 2000;
- LEX fence usage from 2001 to today.

The distribution of flight hours in each of these periods for a selection of aircraft is shown in Figure 33.

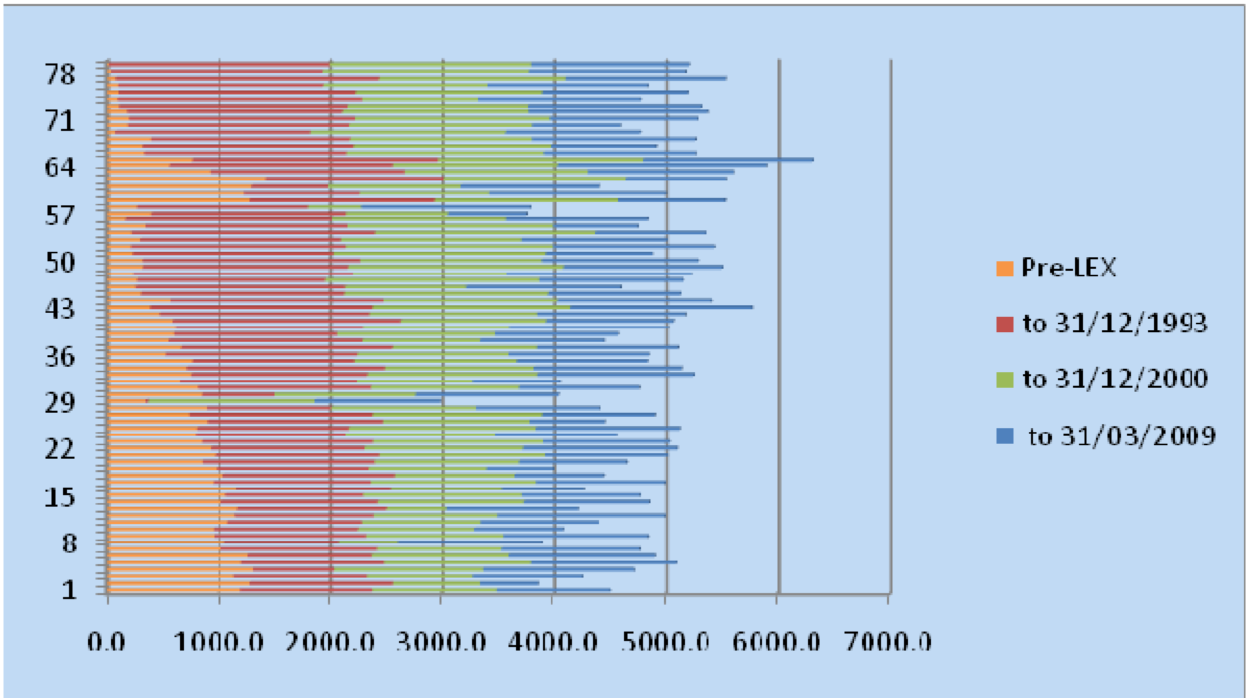


Figure 33. Distribution of CF-18 Fleet AFH over Various Time Periods

Once these periods are established, a MES is built for each that targets its 90th percentile severity for all known fatigue usage indices (and some parameters). Different damage rates are targeted in each time period to best represent fleet usage. We believe that 90th percentile damage rates will result in conservative estimations of overall damage accumulation. These rates are multiplied by the number of FH under each period and summed to give an indication of the fatigue usage accumulation for every aircraft. The results of this calculation are plotted against the actual cumulative Loc B FLE in Figure 34. Note the considerable reduction in over conservatism using this approach versus the current lifing approach for non-tracked locations with TrF=1.4 (recall Figure 32). Individual aircraft usage estimations are compared with the BOS to position aircraft with respect to the design reference. Then, individual aircraft maintenance thresholds and intervals need to be converted in terms of WRFLE and AFH as these are available in the field.

As can be seen in Figure 34, results are conservative for nearly all active aircraft. It is believed that the level of conservatism demonstrated here is acceptable for secondary structure and redundant durability critical parts. For parts with higher criticality, i.e. fracture critical parts and single-load path durability critical parts, referenced individual aircraft tracking is recommended.

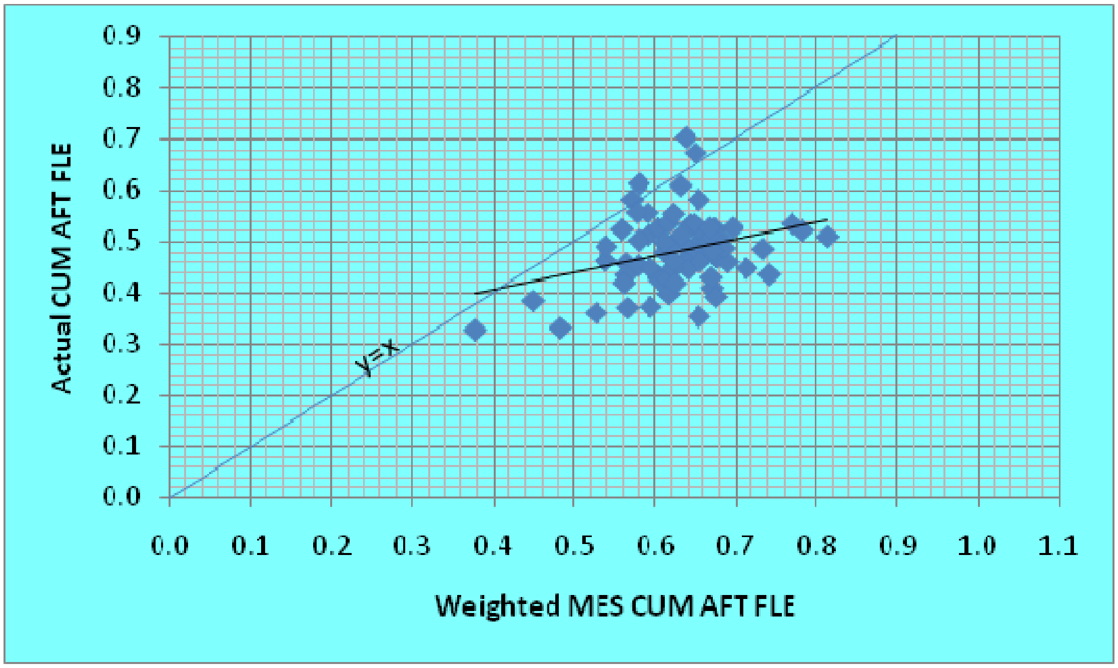


Figure 34. MES Damage Estimations vs. Individual Aircraft Tracking for Loc B

CC115 Buffalo Operation Loads Monitoring Verification of Current Usage–IAR/CF

R.S. Rutledge and C.A. Beltempo, National Research Council Canada

NRC was tasked to support the management of the collection of data from an existing Operational Loads Monitoring (OLM) installation on a Buffalo aircraft, as well as provide training for downloading the data and transferring this data to NRC. NRC converted the output data to a useable format, and modified existing software / write new software to process the data, to enable comparison against the baseline spectrum used in the CC115 Buffalo original certification testing full scale test spectrum. NRC compared previous and current usage finding some differences but the usage was less than the original fatigue testing applied. To date only 9 months of data has been recorded which is insufficient to make conclusions about the current CC115 service spectrum, and review critical Structurally Significant Items (SSI's) with respect to current inspection cycles and existing damage tolerance criteria.

Helicopter Health and Usage Monitoring

C. Cheung, M. Martinez and N. Bellinger, National Research Council Canada

Managing the structural integrity of aircraft platforms hinges on having accurate load spectra data. For rotary-wing aircraft, development of reliable load spectra is still in its infancy. Calculation of loads must include all the dynamic loads caused by the rotating components and these are often not well understood. In addition, the combination of the various loads over the complete flight spectrum is a more complex task than for fixed-wing aircraft. Monitoring of the dynamic loads is also a more difficult task than for manoeuvre loads of a fixed wing aircraft. As a result, there are more approximations, and potential conservatisms, made in developing rotary wing load spectra. NRC has developed a number of technologies and methodologies for fixed wing aircraft, including HOLSIP (the Holistic Structural Integrity Process) as it applies to metallic and composite structures, age degradation effects (including multi-site damage, corrosion and fretting) and patch repair. With knowledge of an accurate loading spectrum, these methodologies may be further applied to rotary-wing aircraft. To more accurately determine the life of components, a life usage monitoring system is being developed that would include the translation of on-board loads monitoring into manoeuvre spectra, which would then be translated into a loads analysis for flight critical components. This would enable life consumption of individual aircraft to be assessed based on actual versus estimated aircraft usage data. NRC is reviewing the current state of affairs of the three helicopter platforms currently being operated by Canadian Forces, namely: CH124 Sea King, CH146 Griffon, and CH149 Cormorant, with the aim to identify potential areas of improvement for the helicopter fleets in their maintenance and management practices for possible cost savings, which may be quite useful for new acquisition fleets, such as the CH148 Cyclone, to establish efficient and effective practices from the outset. To apply the HOLSIP framework to helicopters, usage monitoring procedures/techniques and a condition-based maintenance policy are being developed. The first major component of this strategy is establishing the load spectrum. Usage monitoring data such as flight condition monitoring and flight loads monitoring are input to a transfer function that determines the load spectrum. Options to carry out flight condition monitoring and flight load monitoring are being identified and explored. These options take into consideration the existing aircraft equipment such as the flight data recorder or health and usage monitoring system, the implementation of machine learning methods such as artificial neural networks, and the use of a variety of new sensor and wireless technologies. A collaborative program between NRC and Australia's DSTO (Defence Science and Technology Organization) was set up in 2007 to continue work on loads synthesis for helicopter dynamic components using a variety of machine learning methods. The aim is to determine the loads experienced in the dynamic components from fixed airframe measurements using data obtained from a Black Hawk flight loads survey. This work is being continued with NRC implementing neural network and genetic algorithm techniques to this problem, while DSTO is applying their own genetic algorithms and linear regression methods. More advanced digital signal processing techniques will be applied in the data pre-processing stage and extensive testing of the neural network on various steady state flight conditions is planned. In addition, trials are being run by down sampling the data to a lower frequency rate to correspond to existing flight data recorder sampling rates, and investigate the application of various machine learning techniques to analyze helicopter data. Vibration and other data are also being analyzed to identify and classify particular flight events.

Advanced Loads Monitoring Development and Development of Test Beds for SHM Sensor Verification and Evaluation

M. Martinez, R. Rutledge and M. Yanishevsky, National Research Council Canada

NRC is currently in the second year of a three year program receiving financial support from the Department of National Defence where it is in the process of designing, manufacturing, assembling, instrumenting, and analytically characterizing three structural test bed rigs for use with sensor and technology demonstrations in a structured evaluation. The work includes the development, manufacture and characterization of undamaged and damaged replaceable elements that may or may not include discrete, multiple site, and multiple element damages.

Three test beds are being developed: a simple beam and stiffened skin test bed with limited number of actuators for evaluating the loads monitoring sensor and system capability was developed in the first year, Figure 35; a medium complexity four spar rectangular box beam test bed with multiple actuators for primary sensor performance evaluation is currently being developed in the second year of this program, Figure 36; and a capability to apply simulate realistic flight loads in a controlled environment on a full scale F/A-18 aircraft outer wing box with up to 12 actuators will be developed in the third year, Figure 37. The two latter platforms will have the capacity to have either hidden damage or damage not specifically identified to the sensor / system manufacturer.

The test beds will provide much needed platforms for the demonstration of sensor integrity and integration of technologies on progressively more complicated structural components that have variations in accessibility, while experiencing known realistic loading. As part of these developments, NRC is investigating the development of robust accurate sensors for measuring displacements and calculating structural loads. Successful wireless technologies will also be investigated for transfer and storage of SHM data information. NRC is trialing several technologies and systems, such as miniature gyroscopes, fiber Bragg grating transducers, acoustic emission systems, and eddy current surface mounted sensors for measuring structural displacements and inspecting for damage evolution. As the platforms are completed, they will be made available to external clients and equipment manufacturers to demonstrate the capabilities of their equipment as a precursor to further development and implementation on actual flight aircraft.

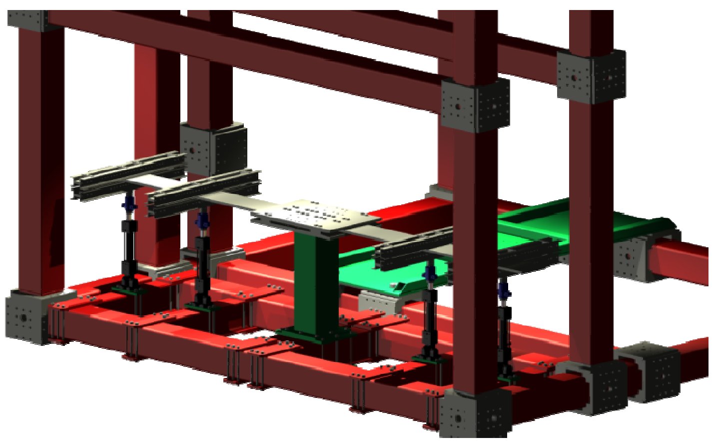


Figure 35. Simple beam bending and torsion platform for evaluating displacement transducer, strain measurement and technologies

Figure 36. A simulated wing box comprised of composite upper and lower skin with aluminum spars and ribs

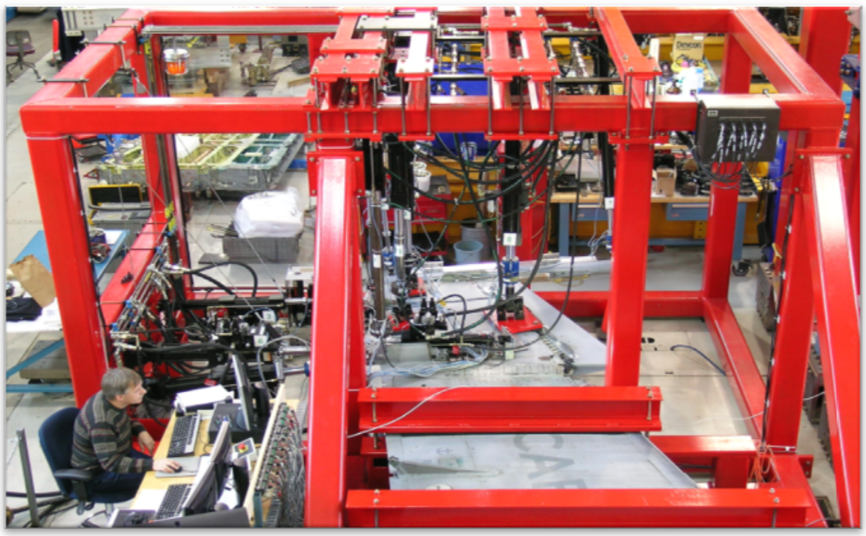


Figure 37. A large scale testing capability for CF188 outer wing spectrum testing

Structural Health Monitoring (SHM) activities at Bombardier Aerospace

J. Pinsonnault, M. R. Mofakhami, Bombardier Aerospace, Core Engineering

The Revised SHM architecture at Bombardier Aerospace is presented in this paper followed by the two recent projects using different sensing technologies.

Structural Health Monitoring is the process of evaluating the structure integrity using onboard systems. Figure 38 presents the architecture of Structural Health Monitoring at Bombardier Aerospace. This architecture has evolved over the past years and tends now to crystallize in the following form.

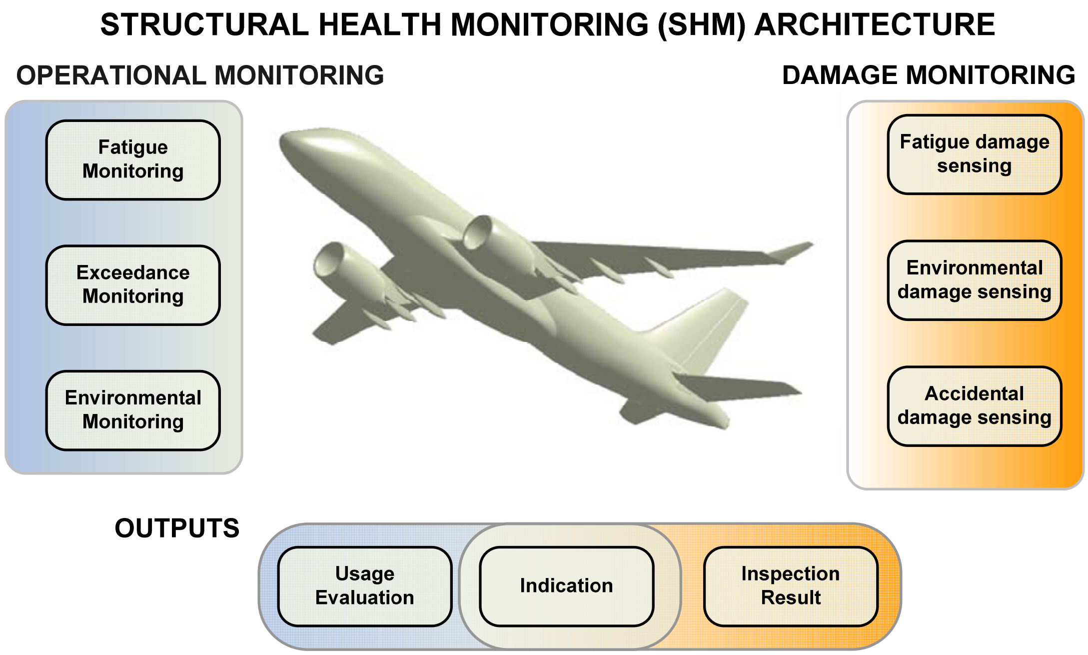


Figure 38. Structural Health Monitoring Architecture at Bombardier Aerospace

Operational Monitoring (OM) regroups all the methods that can contribute to the evaluation of the structure utilization or conditions in-service including fatigue, exceedance and environmental monitoring. OM approaches can be derived from the damage tolerance and fatigue evaluation of the structure. The outputs of OM are indications and/or usage evaluation. The indications are related to abnormal loads or conditions and can be advisory or mandatory. The usage evaluation allows the comparison of design spectra and expected environmental conditions with in-service data in order to increase inspection intervals when margins allow.

Damage monitoring (DM) regroups all the measurement methods for damage detection in the structure that can include damage localization and damage sizing. Damage monitoring covers fatigue, environmental and accidental damages. The outputs of DM are advisory indications and/or inspection results. The indications are related to the presence of potential damages and can be used to improve repair planning. The inspection results are obtained by qualified alternate methods of inspection that cover specific areas on the structure for specific types of damage and sizes.

In-service demonstration of CVM

Two Comparative Vacuum Monitoring (CVM) sensors were installed on a CRJ100 in-service. The sensors were flown between January and July 2009. The objective was to demonstrate the practicality to install and to perform periodic measurements with CVM sensors on an aircraft in-service.

Shown at Figure 39, the two sensors were installed at locations where cracking is expected within the inspection area represented at Figure 40.

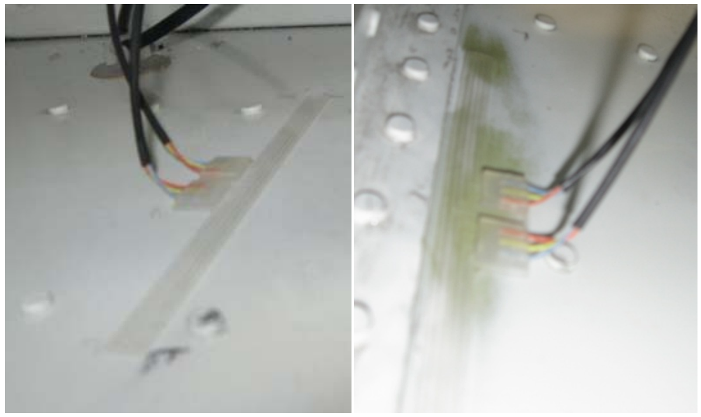


Figure 39. CVM sensors

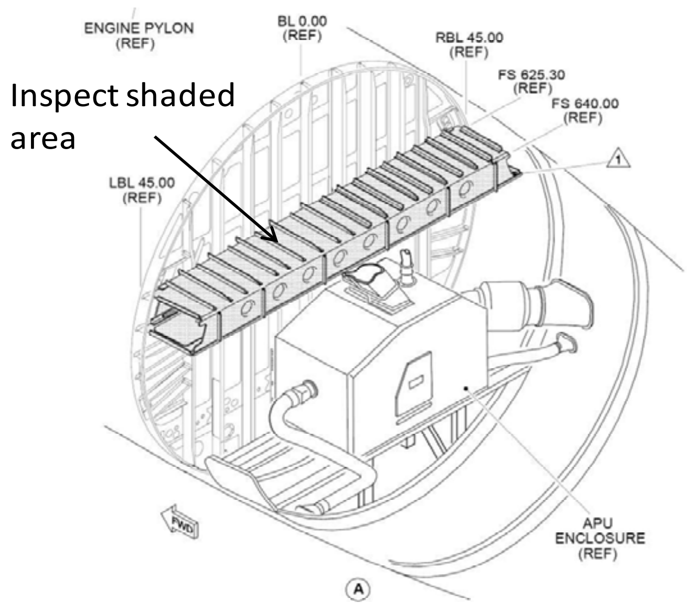


Figure 40. Inspection area

The intended function in this application is to provide an advisory indication of the presence of a possible flaw few days prior to a mandatory inspection. This information can be used to schedule a downtime sufficiently long to allow a possible repair at upcoming mandatory inspection. It was also considered to propose a solution that would meet the full inspection requirements, but the conceptual design showed that the number of required sensors was significantly high and the business case requirements were not met.

Strain distribution measurements using optical fiber sensors

Two optical fibers were installed on a machined wing plank during ground testing to measure strain distribution along a stringer. Shown in Figure 41, one fiber was installed on top of the stringer and one on the other surface of the wing plank under the stringer.

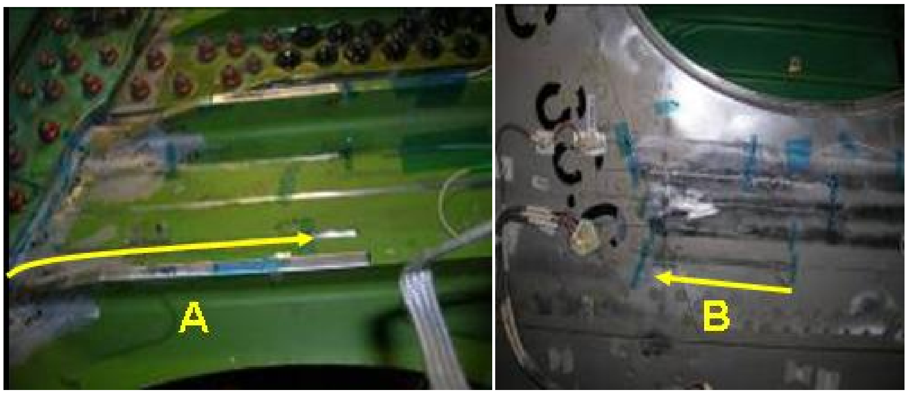


Figure 41. Optical fibers installation on machined a wing plank

Figure 42 presents the strain distribution measured with the two optical fibers.

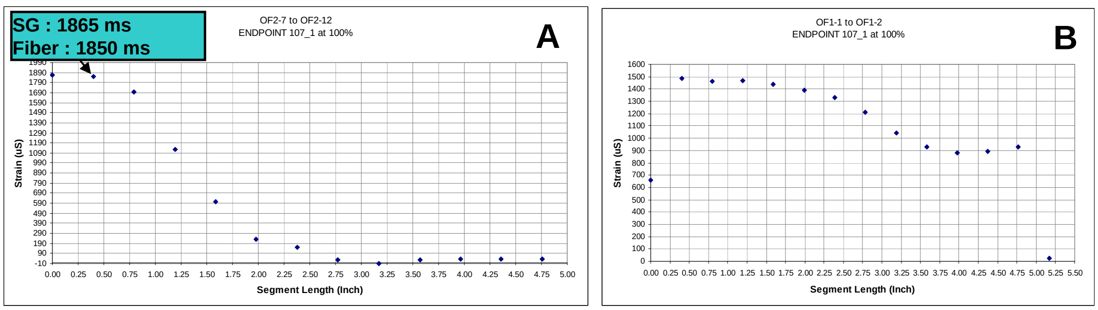


Figure 42. Strain distributions

The first objective of this test was to measure strain distribution for design purposes, but it is expected that this technology can play a key role in damage detection for next generation aircraft.

Nondestructive Inspection and Sensors

New NDT

Jean Dubuc, L-3 Communication (Canada) Military Aircraft Services (MAS)

In 2009, L-3 MAS reported on its effort towards the characterization of probability of detection (POD) on conventional NDT techniques. Since then, L-3 MAS has spent significant effort in order to develop new non-destructive inspection techniques to address specific inspection issues not addressed by conventional methods. The achievements sought for were developing a NDT technique allowing the detection of 0.060" long cracks under fastener heads without fastener removal and developing a technique to detect 0.075" long cracks in a second layer (sub-layer) of material.

Test coupons representing typical CF-18 typical inspection problematic were designed and fabricated by L-3 MAS. These coupons were required to perform a relevant feasibility study with emerging non-destructive technologies. Four coupon assemblies with many holes and different simulated crack configurations were fabricated for the purpose of the feasibility study. The coupons allowed to investigate the effect of plate thickness, fastener hole geometries (countersink or straight), fastener materials (both pin and collar), crack dimensions, crack nucleation sites (emanating from holes or from surfaces), in the case of cracks at holes; cracks in a first layer or second layer of material, in the case of cracks at holes; access on the fastener head or collar side, and sealant presence at faying surfaces.

Three NDT companies performed the feasibility studies in laboratories at their facilities; GE Inspection Technologies (GEIT), Olympus NDT and TecScan Systems. The techniques used are Low Frequency Eddy Current, Ultrasonic Phased Array, Eddy Current Arrays and Pulsed Eddy Current (PEC). Results for cracks under fastener heads without fastener removal and for surface-cracks in a 2^{nd} layer were conclusive. For cracks in fastener holes in a 2^{nd} layer of material, the targeted crack sizes proved to be too challenging and a second feasibility study with longer simulated cracks is currently on-going.

Artificial Seeding of Fatigue Cracks in NDI Reference Coupons

M. Yanishevsky, M.Martinez, C. Mandache, M. Khan, A. Fahr and D. Backman, National Research Council Canada

One of the most challenging aspects of quantifying the performance of Non Destructive Inspection (NDI) techniques for their calibration, detection capability and reliability is the development of representative coupons containing crack-like discontinuities. Electrical discharge machining (EDM) using shim or wire electrodes offers opportunities to develop rudimentary level discontinuities; however, these have several characteristics that differ from actual fatigue cracks. EDM slots are often triangular in an attempt to represent corner cracks or rectangular to represent surface cracks. Unfortunately, these are crude approximations of the quarter-elliptical or semi-elliptical cracks found in service. Deep EDM slots also have a significantly larger width, whereas shallow EDM slots can behave more like a machining mark rather than a sharp discontinuity, thereby affecting the responses of NDI techniques such as eddy current and ultrasonics. Five case studies for which reference NDI coupons were developed with carefully seeded fatigue cracks are presented. The principles used were similar with a few nuances. Typically the specialized geometry was machined undersize which allowed the machining of starter EDM slots. The parts were then subjected to fatigue cycling to generate sharp cracks from the EDM slots. Once the required crack sizes were achieved, the starter EDM slots were machined away removing the evidence of where and how the cracks were introduced. If required, there was post machining of the appropriate geometry and addition of assembly details to create more representative reference coupons and structural design details for assessing the ability of NDI techniques to find simulated in-service cracks. These case studies cover approaches for creating fatigue cracks: at critical radii; inside fastener holes at corners, countersinks and mid-bores; on free surfaces to create surface cracks (ex. for monitoring crack development and growth under bonded repairs); inside simulated gas turbine engine compressor bores; and underneath bushings in simulated helicopter lugs. Examples of the differences that EDM slots and fatigue cracks have on detectability and reliability are provided.

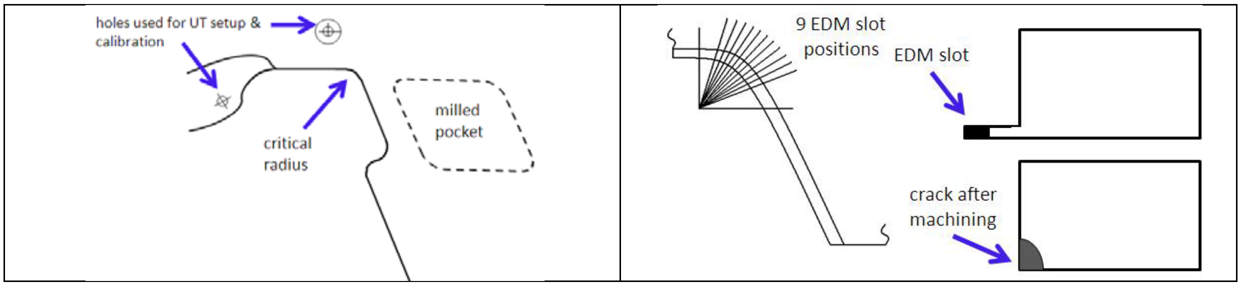


Figure 43. The critical radius in a fighter aircraft structure (left). Close-up of the EDM slot in the fin for creating corner cracks (upper right). The final corner crack after fatigue cycling and machining away of the fin (lower right).

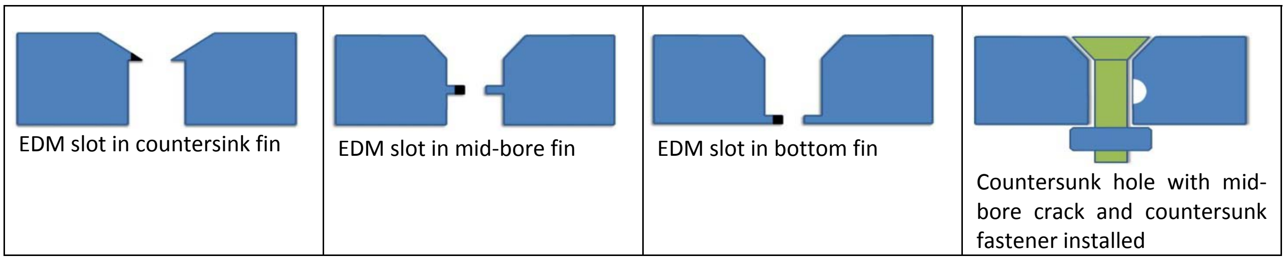


Figure 44. Counterbore machining strategies for locating fatigue cracks in countersunk holes at the: (A) countersink; (B) mid-bore; and (C) bottom surface. A schematic of a final machined hole with mid-bore fatigue crack and countersunk fastener installed (D).

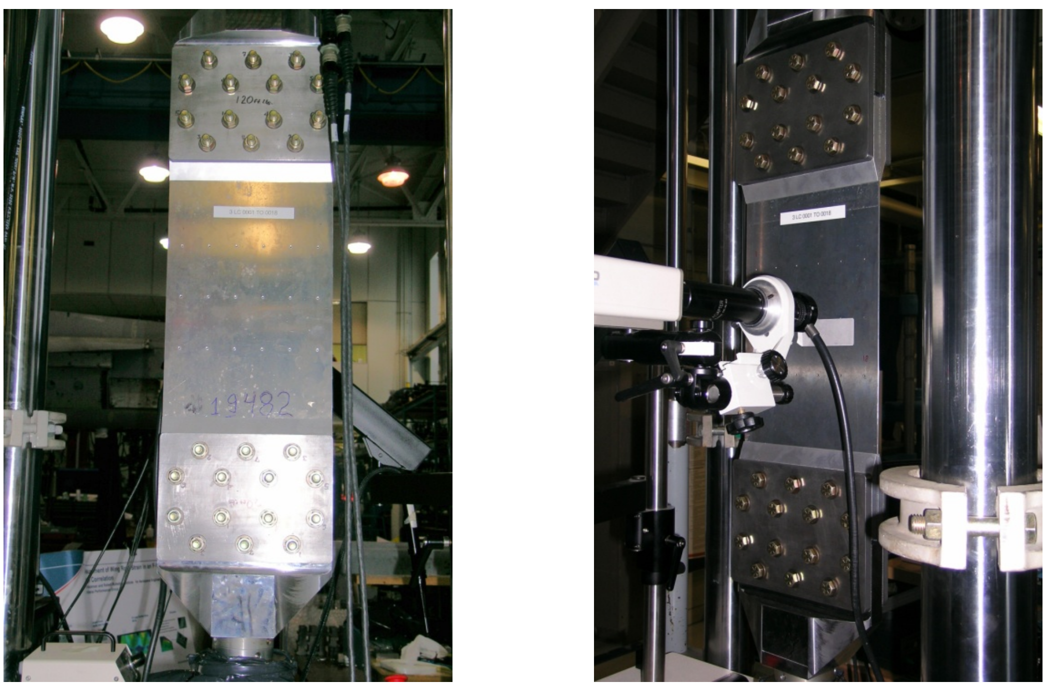
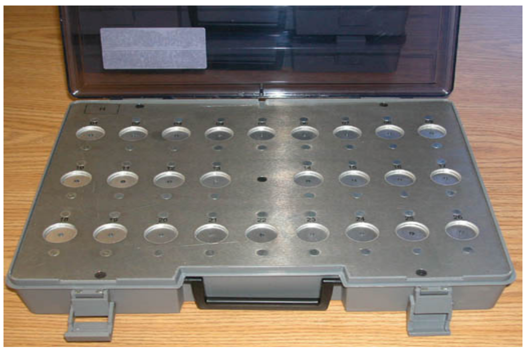


Figure 45. Test panel installed in the servohydraulic load frame (left). A video camera (right) was used to monitor cracks growing from the fins with EDM slots into the holes. Once completed, coupons were later machined from the panel as shown in Figure 22.

(A) single coupon (top view)

(B) inspection set (side view)



(C) inspection sets assembled in box

(D) eddy current inspection

Figure 46. Examples of: (A) single coupon (top view), (B) inspection set (side view), (C) inspection sets assembled in box, and (D) an inspector performing a bolt hole eddy current inspection.

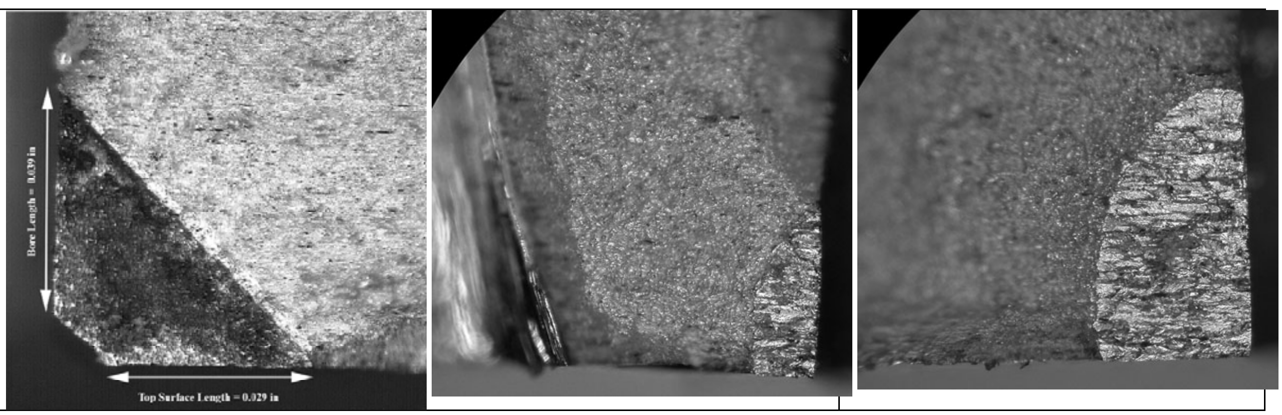


Figure 47. Example of an EDM corner slot 0.029” (0.74 mm) depth / 0.039” (0.99 mm) length. Note remelt material at end of EDM slot (left) Two seeded corner fatigue cracks originating from 0.020” (0.5 mm) starter fins #2006 0.006”(0.15 mm) depth / 0.029” (0.74 mm) length (middle). #2050 0.028” (0.71 mm) depth / 0.049” (1.24 mm) length (right). Aspect (length to depth) ratio is 4.8 at middle; 1.8 at right.

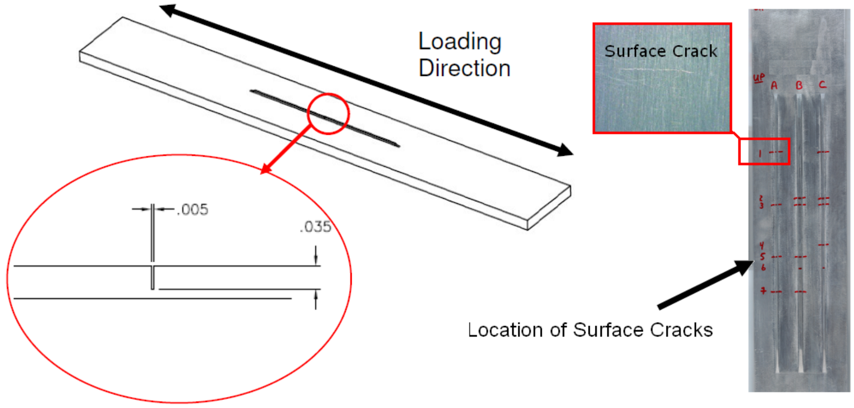


Figure 48. Surface crack specimens. Single rib with single crack (left); multiple ribs with multiple cracks (right).

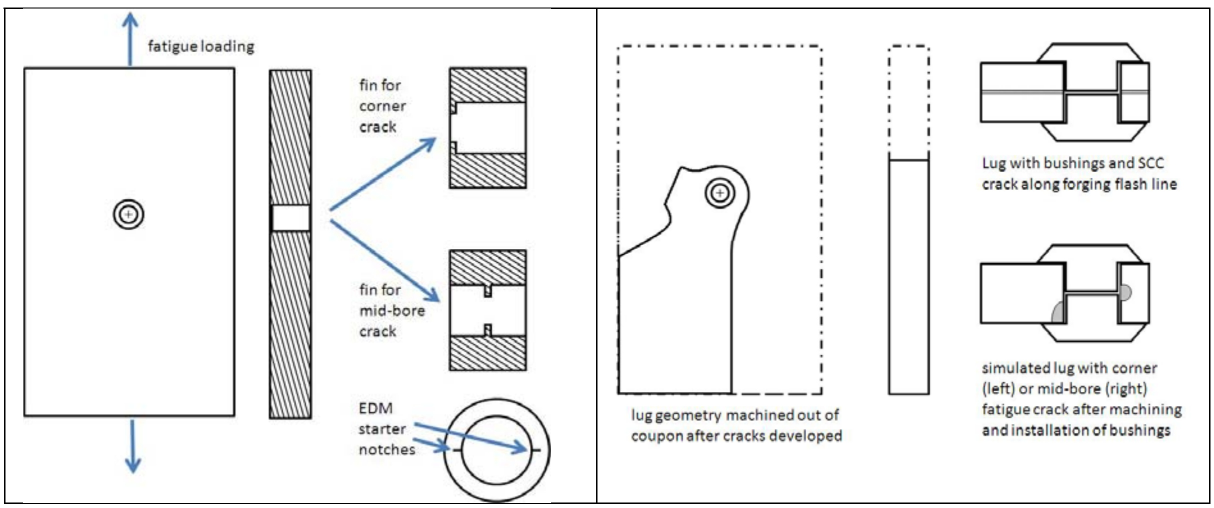


Figure 49. Reference coupon developed to create corner cracks and mid-bore cracks of the pylon hinge hole (left). Schematic of the machined lug geometry out of the coupon (left). The schematic of the original SCC (horizontal) crack that had developed along the forging flash line in the lug is shown at upper right. The schematic for the lug with either corner or mid-bore fatigue crack after machining and installation of bushings is shown at lower right.

References

[1] Bombardier, Y., Liao, M., "*Stress Intensity Factor Solutions for Cracks at a Hole in an Infinite Sheet*", NRC report No. LTR-SMPL-2010-0117, June 2010.

[2] Bombardier, Y., Liao, M., "*Stress Intensity Factor Solution for a Through-the-Thickness Edge Crack Growing Through an Open Hole*", NRC report No. LTR-SMPL-2011-0015, January 2011.

[3] Bombardier, Y., Liao, M., Bisson, S., Beres, W. "*A New Stress Intensity Factor Solution for Through-Thickness and Part-Through Cracks at an Offset Loaded Hole*", NRC report No. LTR-SMPL-2010-0167, September 2010.

[4] Bombardier, Y., Liao, M., Renaud, G. "*Modelling of Continuing Damage for Damage Tolerance Analysis*". Proceedings of the 2011 International Committee on Aeronautical Fatigue Symposium, Montréal, June 1-3 2011.

[5] Bombardier, Y., Liao, M., Renaud, G. "*Stress intensity factors for a crack in a panel with arbitrarily located stringers*", Proceedings of the 2011 Aircraft Airworthiness & Sustainment Conference, San Diego, April 18-21 2011.

[6] Renaud, G., Liao, M., Bombardier, Y. "*Development of an Efficient Methodology and Tool to Determine Stress Intensity Correction Factors for Complex Aircraft Structures"*, Proceedings of the 2011 ICAF, Montréal, June 1-3 2011.

[7] Renaud, G., Liao, M., and Bombardier, Y., *"Development of FE-based Stress Intensity Factor Correction Solutions for Adjacent Structures Effects"*, LTR-SMPL-2009-0240, 12/18/2009.

[8] Liao M., Bombardier Y., Renaud G., Bellinger N.C. and Cheung T., *" Development of advanced risk assessment methodologies for aircraft structures containing MSD/MED"*, The 31st CONFERENCE and the 25th SYMPOSIUM of the International Committee on Aeronautical Fatigue (ICAF), Rotterdam, Netherlands, June 2009.

[9] Liao, M., Bombardier, Y., G. Renaud, *"Advanced Damage Tolerance and Risk Assessment Methodology and Tool for Aircraft Structures Containing MSD/MED"*, The International Council of the Aeronautical Sciences (ICAS) 2010 Congress, Nice France, 2011.

[10] Liao, M., Bombardier, Y., and Renaud, G., *"Quantitative Risk Assessment for the CC-130 Centre Wing Structure (Phase II) - Final Report"*, NRC Report, LTR-SMPL-2009-0098.

[11] Liao, M., Bombardier, Y., and Renaud, G., *"Updated Quantitative Risk Analysis for the CC-130 Center Wing Structures: Part 1 CW-1 Location"*, NRC Report, LTR-SMPL-2010-0222.

[12] Liao, M., Bombardier, Y., and Renaud, G., *"Updated Quantitative Risk Analysis for the CC-130 Center Wing Structures: Part 2 CW-14 Location",* NRC Report, LTR-SMPL-2010-0232.

[13] Liao, M., "*Quantitative Risk Assessment for CP-140 Wing Structures - Updated Results*", LTR-SMPL-2009-0185, 9/25/2009.

[14] Renaud, G., Liao, M., and Bombardier, Y., *"Modeling of Short Crack in 7075-T73 Specimens"*, The 2nd Aircraft Airworthiness and Sustainment Conference, Sandiego USA, April 2011.

[15] Renaud, G., and Liao, M., *"A Probabilistic Crack Growth Modeling Tool Including Material, Geometry, and Loading Variability"*, International Conference on Computational & Experimental Engineering and Sciences (ICCES), Mar, 2010, Las Vegas, USA.

[16] Liao, M, *"Dislocation Theory based Short Crack Model and its Application for Aircraft Aluminum Alloys"*, Engineering Fracture Mechanics; Volume 77, Issue 1, January 2010, Pages 22-36.

[17] Liao, M., and Renaud, G., *"Fatigue Analysis for CF-18 Component: Wing Fold Shear Tie Lug"*, CPR-SMPL-2009-0108, Published in the 10th International Fatigue Congress "FATIGUE 2010", Prague, Czech Republic. June 6-11, 2010.

[18] Renaud, G., and Liao, M., *"Finite Element Analysis of the FT312 Wing-Fold Aft Spar Shear-Tie Lug"*, LTR-SMPL-2010-0052, March 2010.

[19] Jerzy P. Komorowski(Ed), IAR-NRC, Review of Canadian Aeronautical Fatigue Work 2007-2009 Canada, 2009.

[20] Kok, Poston and Moore, *Bombardier Aerospace Demonstrator*, 26th ICAF Symposium, Montreal, 2011.