AEROSPACE – STRUCTURES AND MATERIALS PERFORMANCE (SMP)
NATIONAL RESEARCH COUNCIL CANADA

Review of Canadian Aeronautical Fatigue and Structural Integrity Work 2011-2013

Volume 1 of 1

Report No: LTR-SMPL-2013-0046

Date: March 2013

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National Research Council Canada Conseil national de recherches Canada

Canada

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Classification :	Unclassified	**Distribution :**	Unlimited
For:			
Reference:			
Submitted by:	A. Yousefpour, Acting Director Research and Development, SMMT		
Approved by:	J. Komorowski, Aerospace General Manager		

Pages:	59	**Copy No:**	
Fig.:	56	**Tables :**	2

SUMMARY

This report provides a review of Canadian work associated with aeronautical fatigue and structural integrity during the period 2011 - 2013. All aspects of structural technology are covered including full-scale tests, loads monitoring, fracture mechanics, composite materials and non-destructive inspection.

Organisation Abbreviations Used in Text:

BA – Bombardier Aerospace
RCAF – Royal Canadian Air Force
DRDC – Defence Research and Development Canada (DND)
DND - Department of National Defence
DSTO - Defence Science and Technology Organization
DTAES - Directorate of Technical Airworthiness and Engineering Support (DND)
GE – General Electric
IMT - Innovative Materials Technologies Inc.
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)
SwRI - Southwest Research Institute
USN - United States Navy

Table of Contents

LIST OF ACRONYMS FOR THE TECHNICAL TERMS

ACR	Adjusted Compliance Ratio
AE	Acoustic Emission
ALEX	Aircraft Life Extension Program
ASI	Aircraft Sampling Inspections
ASIP	Aircraft Structural Integrity Program
ASIMP	Aircraft Structural Integrity Management Plan
BoC	Basis of Certification
BOS	Baseline Operational Spectrum
CAE	Computer Aided Engineering
CBM	Condition Based Maintenance
CFH	Component Flight Hours
CFRP	Carbon Fibre Reinforced Polymer
DDTCP	Durability and Damage Tolerance Control Plan
DADTA	Durability and Damage Tolerance Assessment
DT	Damage Tolerance
DTA	Damage Tolerance Analysis
EBH	Equivalent Baseline Hours
EC	Eddy Current
EDM	Electrical Discharge Machining
ELE	Estimated Life Expectancy
EMC	Electro-Magnetic Compatibility
EMI	Electro-Magnetic Interference
FBG	Fibre Bragg Grating
FCS	Flight Control Surfaces
FCGR	Fatigue Crack Growth Rate
FE/FEA/FEM	Finite Element/Finite Element Analysis/Finite Element Model
FISIF	F/A-18 International Structural Integrity Forum
FOS	Fibre Optic Sensors
FSCS	Flight State and Control System
FUI	Fatigue Usage Indices
HCF	High Cycle Fatigue
HOLSIP	Holistic Structural Integrity Process
LPI	Liquid Penetrant Inspection
LIF	Life Improvement Factor
LMS	Loads Monitoring System

MEMS	Micro Electro-Mechanical Systems
MES	Master Event Spectra
MOI	Magneto-optical imaging
MSD/MED	Multi-site Fatigue Damage/ Multiple Element Damage
NDI/NDE	Nondestructive Inspection/Evaluation
OEM	Original Equipment Manufacturer
OLM	Operational Loads Monitoring
POD	Probability of Detection
ROI	Region of Interest
SEM	Scanning Electron Microscope/Microscopy
SENT	Single Edge Notch Tension
SSI	Structurally Significant Items
SF	Severity Factor
SLAP	Service Life Assessment Program
SHM	Structural Health Monitoring
SMP	Structural Maintenance Program
SsCx	Split Sleeve Cold Expansion
SRM	Structural Repair Manual
TRL	Technology Readiness Level
TH	Time History
UAV	Uninhabited Aerial Vehicles
VAPVD	Vacuum Arc Physical Vapour Deposition
WFD	Widespread Fatigue Damage

1.0 INTRODUCTION

Canadian industry, universities and government agencies were solicited for information describing their fatigue technology and structural integrity related activities over the period 2011 to 2013. This review covers work performed or being performed by the following organizations:

Bombardier Aerospace
Department of National Defence (DND)

- Defence Research and Development Canada (DRDC) - Air Vehicles Research Section (AVRS)
- Royal Canadian Air Force (RCAF)
- Director General Air Equipment Technical Management (DGAEPM)
- Royal Military College of Canada

I.M.P. Group Ltd.
Innovative Materials Technologies Inc.
L-3 Communications (Canada) Military Aircraft Services (MAS)
National Research Council of Canada
Carleton University
Marshall Aerospace and Defence Group

Names of contributors (where available) 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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2.0 FULL-SCALE AND COMPONENT TESTING

2.1 CF-18 H-STAB LIFE EXTENSION

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

Historically, the Flight Control Surfaces (FCS) of the CF-18 have not been considered, systematically, as part of the Aircraft Structural Integrity Program (ASIP) effort for the aircraft. Certification was demonstrated by the 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 Original Equipment Manufacturer (OEM) certification limit of 6000 CFH for many FCS.

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

Since ICAF 2011, the global strategy for a life extension beyond OEM 6000 CFH has been revised. This shift in strategy was triggered by exceptionally good sampling inspection results on the RCAF fleet. The complexity and associated cost of certifying composite structures through testing was the second main driver.

This revised strategy is primarily based on the fact that the CF-18 H-STABS are in a superior state than USN stabs in terms of environmental degradation and existing defects/flaws (corrosion, water ingress, blown core, disbonds, etc.). This is mainly due to a more benign environment and different operational usage. The noted flaws/defects are essentially the origins of all failure mechanisms for the built-up honeycomb structure. In the absence of these initiating factors/effects as described above, the HSTAB skin/core assembly is a highly redundant structure that would otherwise not be significantly affected in terms of service life. Thus, the agreed way ahead is to use the substantial amount of work that has been performed under the SLAP and the RCAF FCS program thus far to demonstrate that the RCAF extension from 7000 to 8000 AFHRS entails lower relative risk than the USN demonstrated in-service by going from 6000 to 7000 AFHRS. L-3 MAS is currently setting up this new strategy with the support of NRC.

2.2 HORIZONTAL STABILATOR DAMAGE SIZE EVALUATION AND RESIDUAL STRENGTH TESTING FOR CF-18 AIRCRAFT

C.A. Beltempo, R.S. Rutledge, M. Yanishevsky and M. Genest, NRC-Aerospace.
E. Dionne, N. Mihaylov, Y. Richard, and G. MacLeod, L-3 Com MAS Canada.
Capt. D.W. Chown and P. Londei, DND.

A consortium of NRC, L-3 MAS and DND has carried out CF-18 horizontal stabilator life extension tests at the Ottawa full scale test facility. Significant life extension testing for the CF-18 wing and fuselage was previously performed in Canada at NRC Ottawa, Bombardier Defence Services at Mirabel (now L-3 MAS), and the Defence Science and Technology Organization in Melbourne, Australia; however, these tests did not specifically evaluate the Flight Control Surfaces (FCS). As a result of recent work by L-3 MAS, component histories and fatigue indices for many FCS were found to be approaching or already past the OEM certification limits. In order to avoid very expensive horizontal stabilator procurements, fleet logistic simulations indicated that it was necessary that some fleet horizontal stabilators reach an additional 1/3 lifetime while earlier configuration stabilators reach an additional 1/6 lifetime. In addition, the Structural Repair Manual (SRM) defined relatively small maximum disbond sizes between skin and honeycomb core, beyond which the part is to be repaired / discarded. These limits were validated as a result of additional work performed by the OEM, and were based on skin buckling analysis. However, current RCAF fleet inspections showed that some of these allowable damage sizes had already been exceeded, indicating that the analysis was unnecessarily conservative, particularly considering the complexity of modeling failure.

In order to expand the size of acceptable damages for the horizontal stabilator, the component was statically tested at NRC with induced damages of three increasing sizes at three critical locations to demonstrate the capability of the component to withstand larger damages, as shown in Figure 1. An extensive exercise in damage introduction and inspection technique evaluation was carried out. To establish further confidence following static testing and damage introduction, all five of the once-in-one-lifetime (limit) loads that defined the loading envelope were applied 2000 times to each damage location for a total of 30 000 limit load applications. The intent was to determine if the large induced damages would grow under the extreme loads since these composite structures could exhibit fatigue degradation under high loads, if they are prone to occur. Following this abbreviated limit load cycling, the article was tested to ultimate loads at each critical location. No damage growth at the induced damage sites was detected during any of this testing.

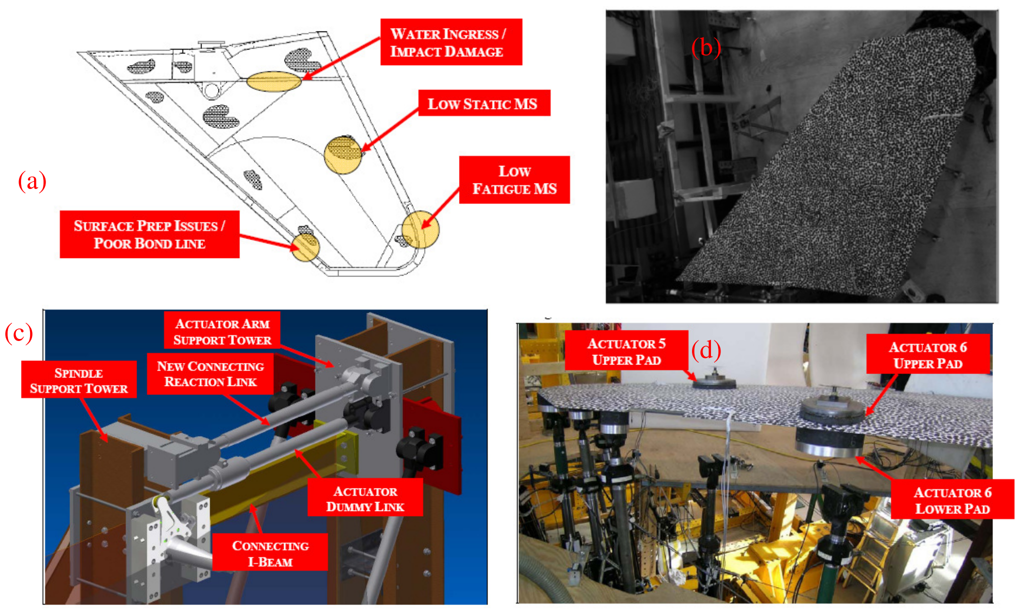


Figure 1. (a) Damage Locations, (b) Image Correlation, (c) Reaction and (d) Test Fixture Setup

Although several nondestructive inspection (NDI) techniques were evaluated, the pulsed thermography, shown in Figure 2, provided the best compromise in terms of speed and contrast for damage measurements. No growth was witnessed at any of the three locations during any of the static or cyclic tests. Following teardown, the horizontal stabilizer spindle was found to have been plastically deformed, which was attributed to the high ultimate load conditions.

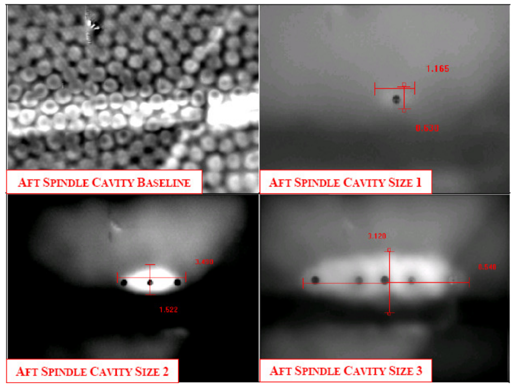


Figure 2. Aft Spindle Cavity Thermography Inspections Following Damage Introduction

Post-test strain and finite element modeling (FEM) comparative analysis led to the conclusion that the two outboard damage locations had been under-loaded based on strain comparisons to digital image correlation results, shown in Figure 3, while the inboard aft spindle cavity location showed strains that were consistent with the target FEM values for all load conditions. The results indicate that the structure is robust and work continues to evaluate flutter and effects of added mass for repairs.

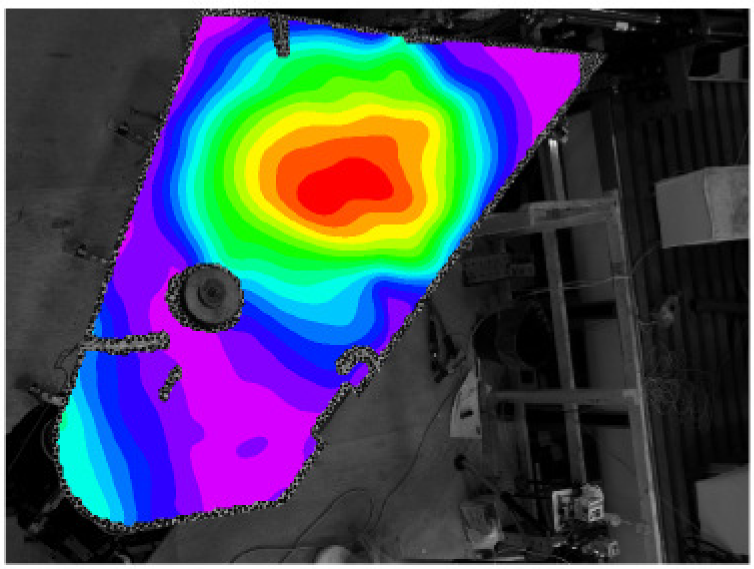


Figure 3. Horizontal Stabilator Upper Surface Digital Image Correlation Strain Measurement

2.3 HIGH CYCLE FATIGUE TEST SUMMARY

J. Rodgers, Innovative Materials Technologies Inc.

To fulfil the compliance requirements as demanded by airworthiness certification authorities (Transport Canada and the FAA), Innovative Materials Technologies Inc. (IMT) worked with NRC to perform coating qualification testing through a collaborative project. This project consisted of three tasks: (1) coating deposition on engine compressor blades and flat specimens using a vacuum arc physical vapor deposition (VAPVD) process, (2) screening of the coatings to identify the best candidate coating for qualification testing, and (3) performing comprehensive coating qualification testing. The coated blades and coupon specimens were then subjected to qualification testing, involving visual and metallographic examination, determination of elemental composition, X-ray diffraction, Vickers hardness testing, scratch testing, salt spray (fog) corrosion testing, erosion resistance determination, natural frequency determination, high cycle fatigue testing, and impact resistance testing.

The high cycle fatigue (HCF) testing was performed on 7 uncoated and 7 coated Stage 1 compressor blades, Figure 4. The results of these tests demonstrate that the difference in the fatigue lives of the coated and uncoated blades are not statistically significant, Figure 5. The nature of the observed fracture in both coated and uncoated blades suggests that the design of the test fixture requires modifications in order to improve the representation of the in-service boundary condition.

Of the 5 failures observed in the uncoated blades, 4 occurred in the root area, below the shoulder and only one occurred in the airfoil section above the shoulder. Two blades ran-out at 10 million cycles. Of the 4 confirmed failures observed in the coated blades, 3 occurred in the root area while one occurred in the airfoil section. No surface cracks were identified on a fifth blade which was deemed to have failed based on a 2% drop in natural frequency. Two coated blades also ran-out at 10 million cycles. The above results point to the presence of essentially two critical stress regions with the current boundary condition: one in the root, close to the blade-fixture contact edge, and the other in the airfoil section. This situation was improved with HCF testing of Stage 2 blades due to modifications that were made to the mounting fixture. The main modification consisted of a deepening of the dovetail groove that lengthened the contact area at the dove tail, in effect bringing the extent of contact close to the blade shoulder.

For Stage 2 blades, HCF testing was performed on ten uncoated and eight coated blades. Statistical testing revealed that the fatigue lives of coated and uncoated Stage 2 blades were indistinguishable. Among the uncoated blades tested, five exhibited cracks in the airfoil, two had cracks in the dovetail region, two showed no damage (although they did meet the 2% natural frequency reduction failure criteria), while one blade ran-out. With the coated blades, two had cracks in the airfoil section, five exhibited cracks both in the airfoil and dovetail, while one ran-out. The test results with the modified fixture demonstrated a considerable improvement in the number of samples that exhibited the desired failure mode.

IMT

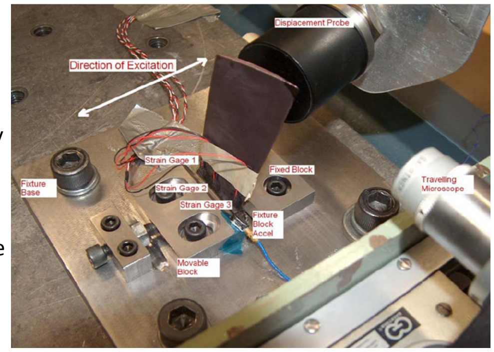
Natural Frequency and High Cycle Fatigue - 1

To analyse the vibration of blades at their first natural frequency till fracture.

NF: to demonstrate that the coating does not change the fundamental natural frequency in a statistically significant manner.

HCF: to demonstrate that there is no statistically significant difference in the fatigue life of coated and uncoated blades

An Unholtz-Dickie / SA30-R16A electrodynamic shaker system (frequency range of 5-2000 Hz and a maximum dynamic force of 10,000 lb) was used for this test.



Compressor blade in test fixture

Figure 4. Natural Frequency and High Cycle Test

IMT

Natural Frequency and High Cycle Fatigue - 2

- Natural frequencies (1st Mode) of coated and uncoated Stage 1 and Stage 2 compressor blades

Stage-1 blades		Stage-2 blades	
Uncoated	Coated	Uncoated	Coated
417.4±3.5	423.8 ±6	531.3 ±9	536.5 ±7.3

Coating does not change the natural frequencies in a statistically significant manner.

- Fatigue life of coated and uncoated blades were indistinguishable, meaning that the coating shows no detrimental effect to the HCF fatigue life

Figure 5. Natural Frequency Test Results

2.4 C-SERIES PRE-PRODUCTION FUSELAGE BARREL TEST

A. Charbonneau, Bombardier Aerospace, Commercial Aircraft

A pre-production fuselage barrel was manufactured and tested as part of BA effort in characterization of Aluminium Lithium alloy and as an up-front validation of design concept, Figure 6. Testing started in 2009 and was completed in 2012.

The objectives of this test were:

- 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 Widespread Fatigue Damage, hence ensure anticipated operating limit.

Figure 6. Pre-production fuselage barrel

To achieve these objectives, a fuselage barrel representative of the production design was manufactured by the fuselage supplier, Figure 7. The test specimen was heavily instrumented. The test plan consisted of various test phases:

- 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 Durability phase (Phase 1) was completed January 2011 with no major finding. Typical structural repairs and allowable damage were incorporated into the specimen during Phase 1. Artificial damage and further instrumentation were incorporated for Damage Tolerance phase (Phase 2) in spring 2011. Testing was completed in spring 2012.

In conclusion, testing simulated three aircraft design lives through 180,000 test cycles with no major finding and all objectives were achieved.

Figure 7. Fuselage barrel test

2.5 BOMBARDIER REGIONAL JET FULL SCALE DURABILITY AND DAMAGE TOLERANCE TESTS

S. 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. An update is provided below.

<u>CRJ700/900/1000 Program – Durability and Damage Tolerance Tests</u>

The three CRJ700 test specimens have completed their fatigue cycling. Table 1 summarizes the status of all complete airframe test specimens.

Table 1. CRJ700/900 Complete Airframe Durability and Damage Tolerance Tests

Test Section	Start Date	Status	Number of Artificial Damage
CRJ700 Forward Fuselage	April 2000	160,000 cycles completed Residual Strength completed Tear Down completed Sent to Storage	47
CRJ700 Centre Fuselage/Wing	March 2000	160,000 cycles completed Residual Strength completed Teardown inspection in progress.	98
CRJ700 Aft Fuselage/Empennage	March 2000	160,000 cycles completed Residual Strength completed Tear Down completed. Sent to Storage	94
CRJ900 Centre Fuselage/Wing	March 2002	160,000 cycles completed Under preparation for Residual Strength	60

The majority of the CRJ700/900 component fatigue tests are completed. Three additional specimens have been added into the test program for the new member of the family: the CRJ1000. Table 2 summarizes the status of all component test specimens.

Table 2. CRJ700/900/1000 Component Durability and Damage Tolerance Tests.

Test Specimen	**Status**
CRJ700 Slat Track 1-2 deployed	160,000 cycles completed Residual Strength completed Tear Down completed Sent to Storage
CRJ700 Slat Track 1-2 retracted	
CRJ700 Slat to Track 1-3 Attachment	
CRJ900 Slat No 3	
CRJ700 Ground Spoiler	
CRJ700 Multi-Function Spoiler	
CRJ700 Inboard Flap	
CRJ700 Inbd Flap Hinge box WS 128	
CRJ700 Outbd Flap Hinge box WS 220	
CRJ700 Outbd Flap Hinge box WS 264	
CRJ700 Winglet	
CRJ700 Aileron	
CRJ700 Rudder	
CRJ700 Elevator	
CRJ700 Fwd Engine mounts	
CRJ700 MLG Trunnion	110,000 cycles completed Residual Strength completed Tear Down completed Sent to Storage
CRJ900 MLG Trunnion	160,000 cycles completed Residual Strength completed Tear Down completed Sent to Storage
CRJ1000 MLG Trunnion	75,000 cycles completed
CRJ1000 Inbd Flap	107,500 cycles completed
CRJ1000 Inbd Flap Hinge Box WS 178	120,000 cycles completed Residual Strength completed

3.0 FATIGUE LIFE PREDICTION AND ENHANCEMENT

3.1 ASSESSMENT OF THE REMAINING LIFE FOR SERVICE EXPOSED COMPONENTS OF TURBINE ENGINES THROUGH ANALYSES AND TESTING

W. Beres, D. Dudzinski, S. Robertson, National Research Council Canada

NRC is performing life updating and damage tolerance assessment of several components from a legacy, turboprop engine. Details of the entire program can be found in [1]–[6]. On the basis of the mission profile analyses as well as thermal and stress analyses of the entire rotor of this turboprop engine, four components were considered to be the most critical, however, both analyses and testing concentrates on the turbine spacer shown in Figure 8.

The aft bore corner which showed the largest von Misses stress was identified as the fracture critical location on this spacer. The stress analyses for the spin rig conditions were performed using the axisymmetric FE model and all calculations were performed in 2D. However, for crack growth predictions, the 3D model of the spacer shown in Figure 9 was used. The crack growth predictions in this location were performed using Zencrack working with Abaqus. The crack growth started from an initial corner crack. Due to the complicated geometry of the spacer the crack growth has to be done in four stages where the results from each stage were used as the initial conditions for the next stage, Figure 10.

Figure 8. Turbine spacer

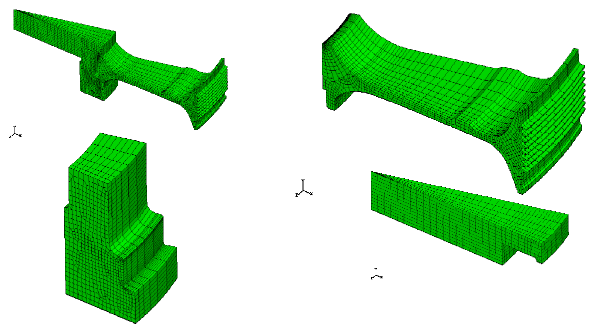


Figure 9. Finite element model of the spacer together with the model of the spin rig arbour

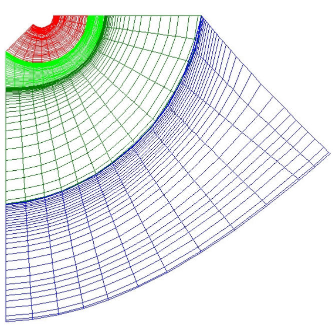


Figure 10. Crack growth at the spacer bore corner from the initial notch to the final size. Four stages of finite element calculations are represented by different colours

Figure 11. Spin rig facility at the NRC

Figure 12. Remnants of the spacer after burst

Figure 13. Microstructure analysis of the fragment of the burst spacer

After the FE analyses of the entire engine rotor and turbine spacer, a set of tests on the spacers was conducted in the spin rig facility shown in Figure 11. The tests were performed in a vacuum at high temperature, in a back-to-back configuration, where two identical spacers were installed on the rotor and tested at the same time [2]. Two tests designated A and B included a total of four spacers. The test runs were periodically interrupted for the NDE. Specific evaluation (inspection) techniques were developed for spacers. They included: (i) visual inspection technique; (ii) liquid penetrant inspection (LPI) technique, and (iii) eddy current (EC) technique. The eddy current technique was conducted using special probes that were designed at NRC.

On the completion of Test A, Spacer 1 showed a significant amount of cracking after a relatively small number of cycles, but Spacer 2, installed on the same rotor, did not show any indication of cracking. Both spacers were subjected to metallurgical analyses and finally destructively examined after the test.

During the second rig test, Test B, Spacer 3 burst. This was a planned event designed to establish the experimental safety margin for the spacers. Figure 12 shows remnants of this spacer partially arranged. Similar to Test A, Spacer 4 in Test B did not shown any visible damage indications after the burst of the other spacer. Remnants of Spacer 3, Figure 13, were investigated using an SEM. Microstructure analyses confirmed that the crack nucleated from spacer aft bore corner. However it was also found that another location, the anti-rotational pin hole side also exhibited significant damage that contributed to the spacer burst pattern. In addition, Test B allowed for the very rudimentary assessment of the crack growth rate. It was found that the crack propagation rate was replicated within a factor of two when compared to the FE predictions. So far, the spin rig based experimental results have not allowed for extending the spacer life, however both analyses and tests revealed significant uncertainties in research methods.

Three uncertain factors have to be taken into account when comparing numerical predictions and the rig test results. The first is the uncertainty of the fatigue crack growth rate for the material in the particular spacer tested. The second is the uncertainty in the record of the total number of cycles since new for these spacers. The test spacers were taken from operation with the estimated time since new. Therefore the actual number of cycles that the spacers experienced before they entered the test was known with significant uncertainty. The third factor contributing to the uncertainty of the results is that the numerical predictions of the crack propagation interval were based on estimating the number of cycles starting from a known initial crack size, a_i, while the rig tests were performed on the spacers that did not show any indication of cracking when three NDE methods were used. Therefore the spin rig results for the burst spacer represent a combined propagation interval comprising of four stages of component damage progression: crack nucleation, small crack growth, large crack growth and unstable fracture. These four factors contribute to the difficulty in interpreting the spin rig results.

3.1.1 References

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[3] Hou, J., Dubke J., Barlow, K., Slater, S., Harris, L., Calcuttawala, S., and Beres, W. ,"3D Crack Growth Analysis and its Correlation with Experiments for Critical Turbine Components under an International Collaborative Program," ASME Turbo Expo 2008: Power for Land, Sea and Air, June 9-13, 2008, Berlin, Germany, GT2008-50548, 2008.

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[5] Beres, W., Dudzinski, D., Tsang, J. , Wu, X.J., Zhang, Z., "Assessment of the remaining life for service exposed components of turbine engines through analyses and testing," SPIE Journal, 2013, (in print).

[6] W. Beres, R.M. Kearsey, X.J. Wu, Q. Yang, D. Dudzinski, J. Tsang "Research on advanced high temperature materials and protective coatings for new and legacy gas turbine engines for military platforms," NATO Symposium on Design, Modelling, Lifing and Validation of Advanced Materials in Extreme Military Environments, Biarritz, France, October 2012, STO-MP-AVT-187, Paper No. 6, pp. 1-28.

3.2 FRACTOGRAPHIC EXAMINATION OF COUPONS REPRESENTING AIRCRAFT STRUCTURAL JOINTS WITH AND WITHOUT HOLE COLD EXPANSION

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

In an effort to establish the potential benefits of hole cold expansion and interference fit fasteners for long term fatigue performance, coupons representing three types of aircraft structural joints were designed, manufactured and assembled. These joints, manufactured using 7075-T6 aluminium alloys for main members and 7075-T73 for doublers or spacers, were representative of: 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. The joints were assembled using Hi-Lok™ interference fit fasteners, with and without FTI SsCx^{TM} hole cold expansion, and were fatigue tested to complete failure using a representative mission mix service load spectrum for a coastal patrol aircraft wing fatigue sensitive location. The failure surfaces were examined with the aid of an optical microscope. The fractographic descriptions for the three types of joints examined, document the crack nucleation sites, mechanisms of crack formation, and the extent of damage at catastrophic failure, including the effects of manufacturing, such as direction of hole cold expansion and direction of interference fit fastener insertion. It was found that interference fit fasteners can damage holes on the insertion side, which can exacerbate fatigue crack nucleation by fretting fatigue mechanisms. The alteration of hole conditions will affect where cracks originate and how they propagate, affecting not only fatigue life, but also requisite inspections to assure safety of flight, see Figure 14 to Figure 16. Caution is advised to investigate material characteristics (grain orientation, ductility, etc) and geometric design details (edge margins) for their hole cold expansion potential before indiscriminate use of this technology to alleviate problematic holes with limited fatigue life. Examination of fracture surfaces for fractographic evidence provides valuable insights into the failure process, which will allow for more accurate and representative life prediction of joint performance.

3.2.1 Reference

Yanishevsky M et al. Fractographic examination of coupons representing aircraft structural joints with and without hole cold expansion. Eng Fail Anal (2013), http://dx.doi.org/10.1016/j.engfailanal.2012.12.008

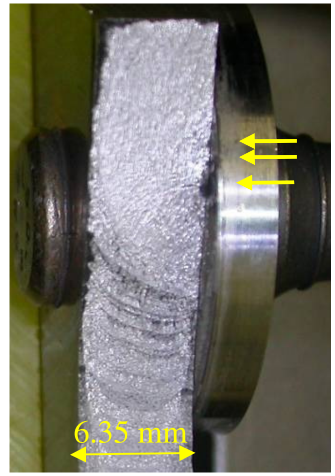
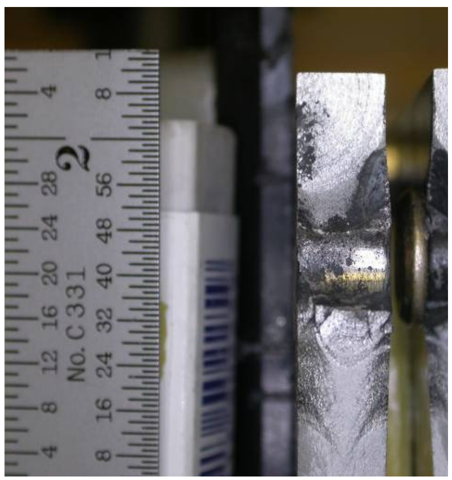
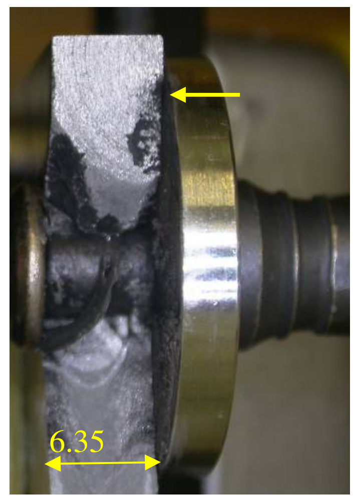
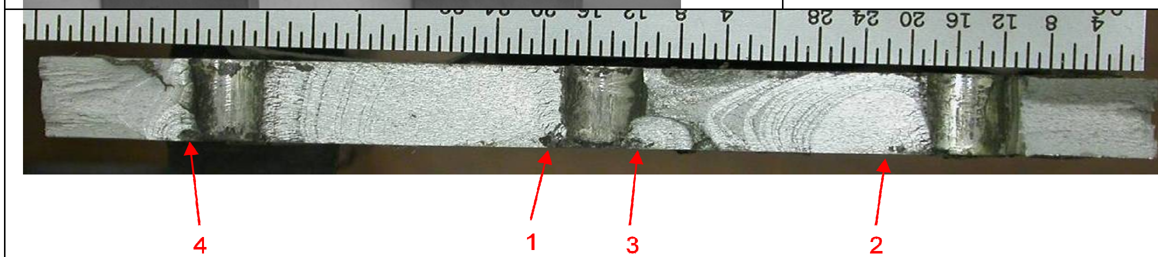
Coupon ID	**Overall photos**	**Failure close-up**	**Comments**
NLT 3D13 C02 e/D = 2.0 SsCx + Hi-Lok			Fretting fatigue cracks nucleated on the plate surface under the disc at contact points with the plate leading to fatigue cracks that joined and grew around the hole.
NLT 3D38 C03 e/D = 2.0 SsCx + Hi-Lok			A fretting fatigue crack nucleated on the plate surface under the disc at the contact point with the plate leading to fatigue cracks that grew into the fastener hole and progressed on the other side of the hole in fatigue.

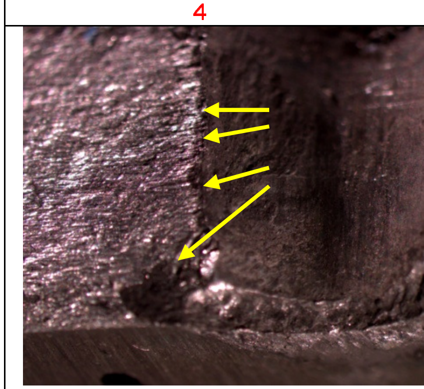
Figure 14. Failure surfaces of NLT coupons with e/D = 2.0, SsCx and Hi-Lok fastener

Coupon 5D43 MLT A05 (e/D = 2.0 Hi-Lok)

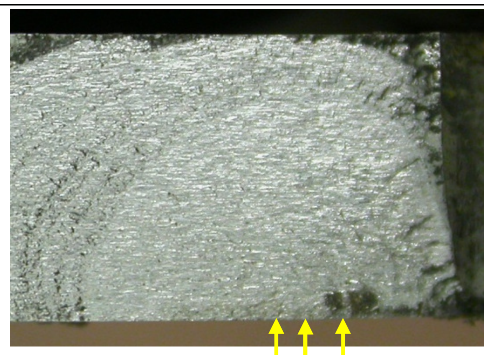
Four cracks formed from multiple sites on the fastener insertion side of the main member due to fretting with the doubler. Crack 1 formed at the surface to the right side of the central hole and grew into the low edge margin hole. Crack 2 formed on the surface away from the low edge margin hole, growing toward the centre. Crack 3 formed at the surface near the central hole opposite to crack 1. Crack 4 formed at the low edge margin hole across from crack 1.



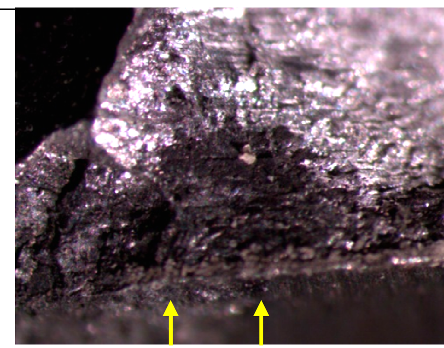
Fastener Insertion Direction



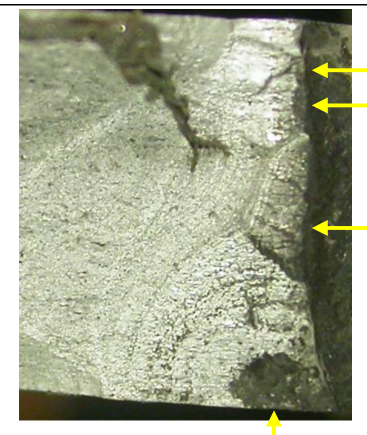
Crack 1 origins



Crack 2 origins

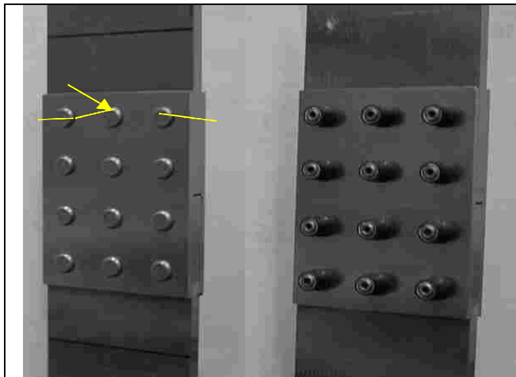


Crack 3 origins



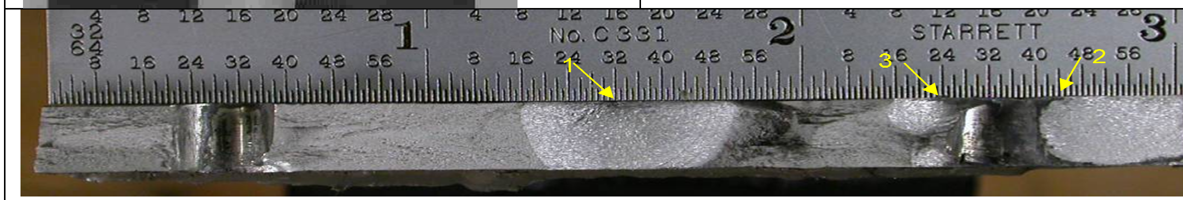
Crack 4 origins

Figure 15. Failure surfaces of MLT test A05 with e/D = 2.0 and Hi-Lok fastener (coupon ID 5D43)



Coupon 4D39 HLT A01 (e/D = 2.0 Hi-Lok)

Three cracks in the vicinity of the upper row of fasteners on the main member formed due to fretting with the doubler on the fastener insertion side. Crack 1 was a surface crack that nucleated above the central fastener hole on the upper most row. Cracks 2 and 3 formed at the surface near a low edge margin hole. Crack 2 grew through to the edge of the coupon; crack 3 grew toward the central hole.



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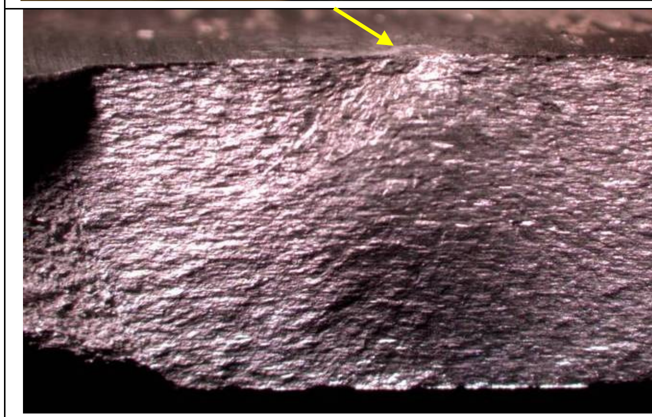
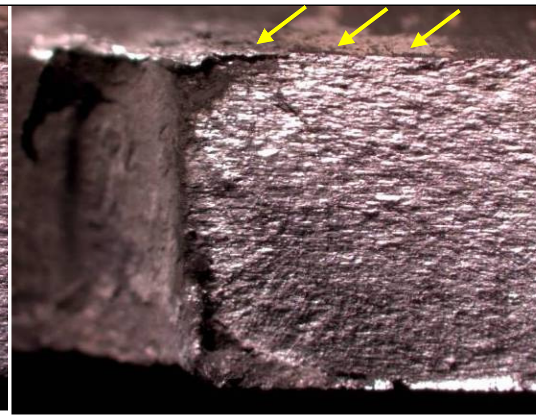
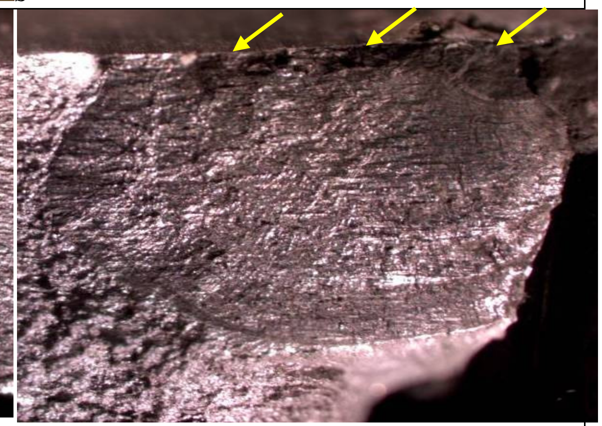
Fastener Insertion Direction



Crack 1 origin



Crack 2 origins



Crack 3 origins

Figure 16. Failure surfaces of HLT test A01 with e/D = 2.0 and Hi-Lok fastener (coupon ID 4D39)

3.3 HOLE REPAIR TECHNIQUES

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 damage 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 past work completed in that area at ICAF 2009 and 2011. These past studies focused on providing life improvement factors for life improvement techniques measured on intact holes.

Recent studies focused on determining the benefit of repair techniques on holes already subjected to in-service fatigue loading. A joint project was completed with other members of the F/A-18 International Structural Integrity Forum (FISIF) in order to determine the benefit of split-sleeve cold working in the presence of small cracks. Test coupons with induced cracks of 0.12 mm and 0.40 mm were cold worked and life improvement provided by cold working was compared to the life improvement factor (LIF) obtained on virgin coupons. It was shown that cold working in the presence of small cracks provides a LIF comparable to the one obtained on virgin coupons. With larger cracks, a certain improvement was measured. However, a large scatter in the post-cold-working life was observed.

Additional work on related topics is currently on-going with the RAAF through the use of the F.I.N.A.L. F/A-18 centre barrel tests completed at DSTO. First and second over sizing of holes were introduced at locations where fatigue cracking is expected. The test will allow evaluating the benefit of these over sizes.

4.0 PROBABILISTIC AND RISK ANALYSIS METHODS

4.1 DEVELOPMENT OF INTEGRATED STRUCTURAL LIFE ASSESSMENT TECHNOLOGIES FOR ROYAL CANADIAN AIR FORCE (RCAF) AIR FLEETS

M. Liao, G. Renaud, and Y. Bombardier, National Research Council Canada

Royal Canadian Air Force (RCAF) aircraft have been certified using different airworthiness standards; some are different from the original equipment manufacturer (OEM) standards. The actual RCAF aircraft usage/loads and in-service damage can be very different from the OEM design conditions including their full scale certification test observations. Some complex damage, such as multi-site fatigue damage (MSD), widespread fatigue damage (WFD) along with Canadian environment related damage modes that can be present in the various materials are not taken into account by the OEM. Meanwhile, the RCAF holds the certification authority and approves the repairs or modifications to all their aircraft in service. Therefore, the determination of the RCAF aircraft service life (operational limit, economic life) considering these new damage scenarios is a practical science and technology (S&T) problem. In addition, the lack of design data/information as well as the lack of OEM timely and cost-effective technical support has resulted in both engineering and economic problems for the RCAF. In the Air Force Science & Technology Implementation Directive (AFSTID, 2010), one of the RCAF Objectives is to reduce the *"ownership and operating" costs by 10% by 2015 and by 20% by 2020*.

Under a 3-year collaborative project (FY11-14) between Defence Research and Development Canada (DRDC) and the National Research Council (NRC), NRC has been carrying out research to improve and integrate NRC airframe life assessment methodologies and tools (see Figure 17), including holistic modeling of fatigue, damage tolerance, and age degradation process on new materials, and complex damage modes. As a result, the RCAF aircraft will receive more reliable and timely S&T support on structural life and risk assessment to help decision-making in airworthiness management and cost-effective maintenance.

The main tasks of this project are highlighted as follows:

- Task I Conduct a brief review/summary of the current RCAF lifing and risk analysis methods/tools. Some of the review/summary are published in [1][2][4].
- Task II Generate fatigue crack growth test data for a new airframe material (7249-T76511). Details related to this are presented in the 2013 Canadian ICAF National Review §5.1.
- Task III Enhance NRC damage tolerance analysis methodologies and tools with expanded beta library and validated material models. Some preliminary results are summarized in the 2013 Canadian ICAF National Review §5.2 and §5.3, as well as in [6][7].
- Task IV Improve Global-Local FE methods for multi-load-path structural system with complex damage mode, such as MSD and WFD. Some preliminary results are summarized the 2013 Canadian ICAF National Review §6.3.
- Task V Improve NRC risk analysis models and tools (ProDTA and CanGROW) for generic fuselage and wing critical locations. Some preliminary results are published in [3][4][5].
- Task VI Demonstrate/validate the developed methods/tools with RCAF aircraft case studies and best practices. Some preliminary results are published in [6][7].

Some of the developed methods and tools have been applied for engineering support to the RCAF aircraft, through separated DTAES contracts/tasks. The final results will be reported in the next ICAF review.

As the principal investigator of this project, NRC is also partnering with DRDC, the Directorate of Technical Airworthiness and Engineering Support (DTAES) of the Department of National Defence (DND), Marshall Aerospace Canada (MAC), IMP Aerospace & Defence, the Royal Military College (RMC), the University of Waterloo and Mississippi State University, while exchanging some information and leveraging some efforts with other the Technical Cooperation Program (TTCP) and NATO Research and Technology Organisation (RTO) member countries.

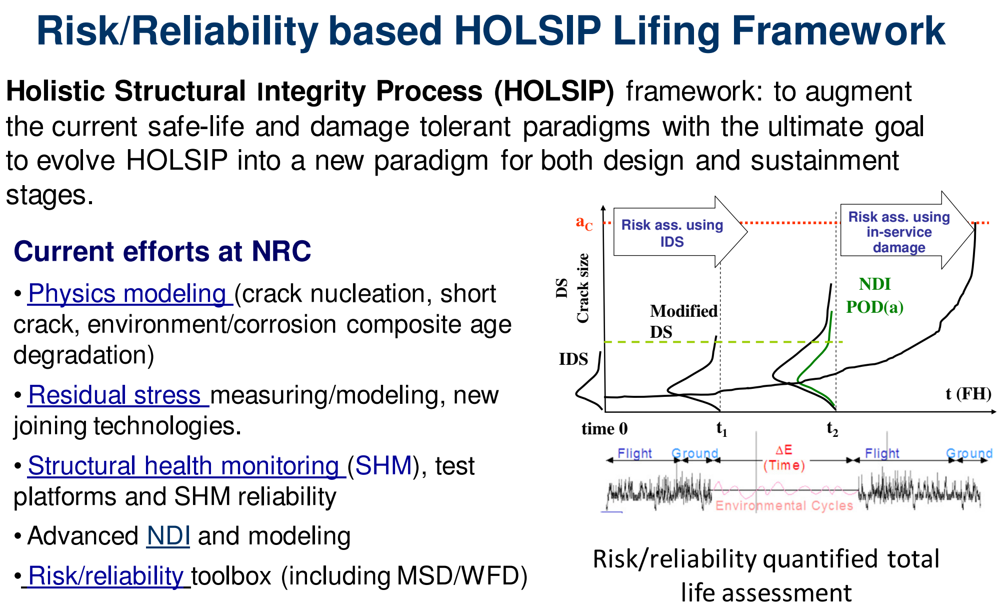


Figure 17. Integrated structural life assessment technologies under development at NRC

4.1.1 References

[1] Liao, M., Bombardier, Y., Renaud, G., and Bellinger, N.C., Advanced damage tolerance and risk analysis methodologies and tools for aircraft structures containing multiple-site and multiple-element fatigue damages, Proceedings of the Institution of Mechanical Engineers, Part G: Journal of Aerospace Engineering, Vol. 226, Issue 11, 2012.

[2] Liao, M., Renaud, G., Bombardier, Y., Bellinger, N.C., Probabilistic Risk Analysis for Aircraft Structural Life Assessment with Limited In-Service Damage Findings, The 28th Congress of the International Council of the Aeronautical Sciences (ICAS 2012), 23-28 September 2012, Brisbane, Australia.

[3] Liao, M., Bombardier, Y., G. Renaud, "Monte Carlo Simulation based Probabilistic Risk Analyses for Aircraft Structures Containing MSD/MED", Proceedings of the 2011 Aircraft Airworthiness & Sustainment Conference (AA&S2011), San Diego, April 18-21 2011.

[4] Liao, M., Comparison of Different Single Flight Probability of Failure Calculations for Aircraft Structural Risk Analysis, to be presented at 2012 Aircraft Airworthiness & Sustainment Conference (AA&S2012), Baltimore USA, April 2012.

[5] Liao, M., A Critical Review of the Single Flight (Hour) Probability of Failure (SFPOF) for Aircraft Structural Risk Analysis, LTR-SMPL-2012-0022, 2012.

[6] Renaud, G., Liao, M., Calibration of AFGROW and CanGROW Models Using the CP-140 SLAP Test Results, LTR-SMPL-2012-0154, 2012.
[7] Liao, M., Renaud, G., Bombardier, Y., Short/Small Crack Model Development for Aircraft Structural Life Assessment, The 13th International Conference on Fracture ICF13, Beijing, China, June 2013.

5.0 FRACTURE MECHANICS AND CRACK PROPAGATION STUDIES

5.1 DEVELOPMENT OF A FATIGUE CRACK GROWTH RATE MATERIAL MODEL FOR A NEW AIRFRAME MATERIAL, 7249-T76511 ALUMINUM EXTRUSION[1]

Y. Bombardier, M. Liao, National Research Council Canada

The 7249-T76511 aluminum alloy is a relatively new alloy and is now being used for new transport aircraft wings and as a 'drop-in' replacement for existing components made of 7075-T6511 aluminum alloy. As such, this alloy was optimized to maintain a high level of strength and fracture toughness, while improving its corrosion resistance over the 7075-T6511 aluminum. Despite the numerous advantages of using the 7249-T76511 alloy over 7075-T6511, no fatigue crack growth rate (FCGR) data could be found in publically available literature, which limits the ability to conduct fatigue life assessment of structures made of 7249-T76511. As it is expected that the 7249-T76511 will be more commonly used for airframe components, there is a need to obtain FCGR data and develop a model for this material to support current and future platforms.

In this work, FCGR tests were conducted for small and long cracks according to ASTM E 647. For long crack data, the FCGR tests were conducted with compact tension (C(T)) specimens for two positive stress ratios: 0.05 and 0.60. The FCGR tests were conducted in laboratory air as well as in a 3.5% sodium chloride (NaCl) solution to quantify the effect of a corrosive environment on the FCGR. The effect of the initial stress intensity factor values for the K-decreasing test phase, the effect of the corrosion cells, and the fixture design were also investigated as part of this test program. A new method, the adjusted compliance ratio (ACR), was also used to determine the effect of remote crack closure on the stress intensity factor. For small crack data, single edge notch tension (SENT) specimens were tested to generate FCGR data from naturally nucleated small fatigue cracks. Residual stresses were also measured using the x-ray diffraction technique and the slitting method to substantiate experimental FCGR results. Figure 18 presents the fatigue crack growth rate data collected using the three C(T) specimens per stress ratio for laboratory environment.

A crack growth material model was developed for the 7249-T76511 alloy using a tabular look-up approach with Walker's equation on a point-by-point basis to interpolate and extrapolate FCGR data at any given R-ratios. To validate the material model, experimental results were obtained by conducting fatigue crack growth tests on C(T) and middle tension (M(T)) specimens loaded using three transport aircraft loading spectra: Mini-TWIST and two transport aircraft wing spectra (tension-tension and tension-compression). Load errors were monitored during the test and their effect was quantified using crack growth simulations. Three load interaction models (Hsu, Willenborg, and Wheeler) were calibrated and evaluated based on their ability to predict the fatigue life for the tested spectra. Preliminary calibration showed the difficulties in selecting a single calibrated load interaction model and its parameters that can provide accurate fatigue life predictions for all tested spectra. Figure 19 shows an example of the calibration efforts conducted using the Hsu and Willenborg closure models based on the results of three specimens tested using the modified Mini-TWIST spectrum.

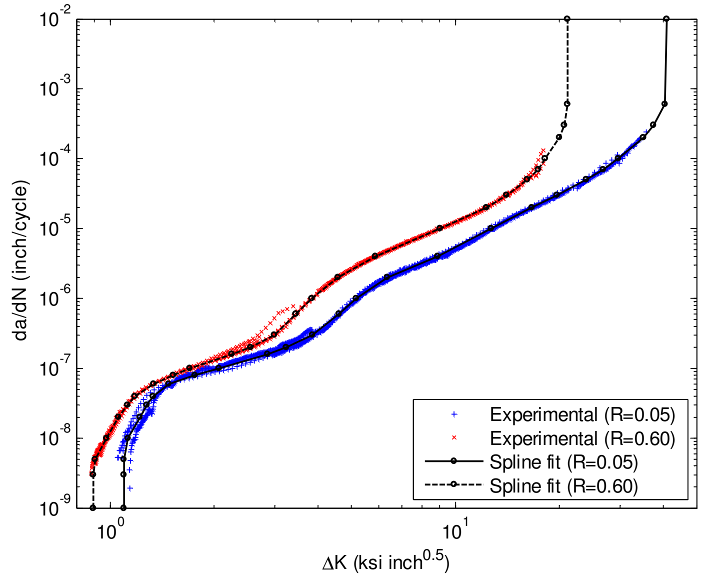
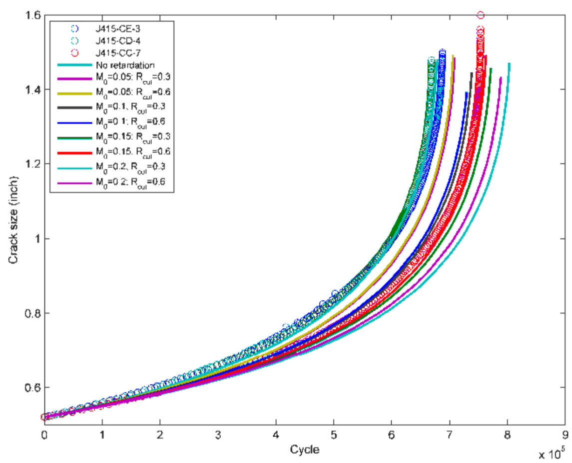
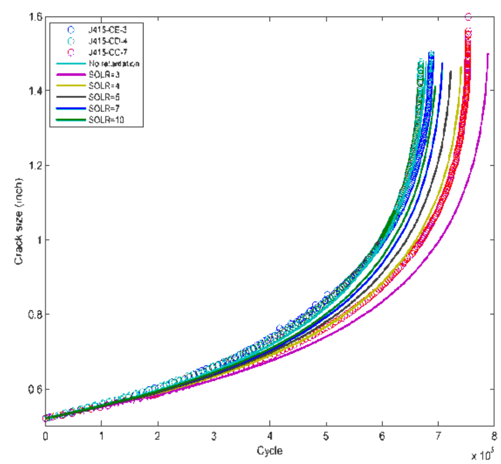


Figure 18. 7249-T76511 fatigue crack growth rate experimental and fitted data for laboratory environment



a) Hsu crack closure model



b) Willenborg crack closure model

Figure 19. Fatigue crack growth predictions

5.1.1 Reference

[1] Bombardier, Y., Liao, M., "Fatigue Crack Growth Rate Testing of 7249-T76511 Aluminium", National Research Council Canada, Report No, LTR-SMPL-2013-0001.

5.2 DEVELOPMENT OF IMPROVED STRESS INTENSITY FACTORS FOR AIRCRAFT STRUCTURAL DURABILITY AND DAMAGE TOLERANCE ANALYSIS

Y. Bombardier, G. Renaud, and M. Liao, National Research Council Canada

The National Research Council Canada (NRC) has been validating and developing stress intensity factor solutions for improving the accuracy of aircraft structural durability and damage tolerance analyses.

The stress intensity factor solution developed by NRC for radial and diametrical cracks at an offset (non-centred) loaded fastener hole[1] (illustrated in Figure 20) has been independently validated by NASA and the Southwest Research Institute (SwRI) as a potential addition to the NASGRO crack growth analysis program. Following good agreement between NRC's solution and NASGRO boundary element solution (BE02), the solution was integrated in NASGRO 7.0 as solution TC23. Compared to the BE02 solution, TC23 is more computationally efficient and more general as it supports in-plane bending and pin-loading in addition to remote tension condition.

Recently, NRC developed a new stress intensity factor solution for cracks located in panels reinforced with arbitrarily located stringers (illustrated in Figure 21)[2]. The new solution was developed by modifying the constitutive equations of a solution for symmetrically and periodically spaced stringers with riveted rigid fasteners. The new solution supports arbitrary stringer locations with respect to the crack location, includes the capability to model compliant fasteners, improves the accuracy of the equivalent stringer compliance by considering Poisson's effect, and allows the fasteners to be arbitrarily located within each stringer. The stress intensity factors calculated using the new closed-form solution were compared to linear static finite element analysis results and very good agreement was obtained for the tested cases.

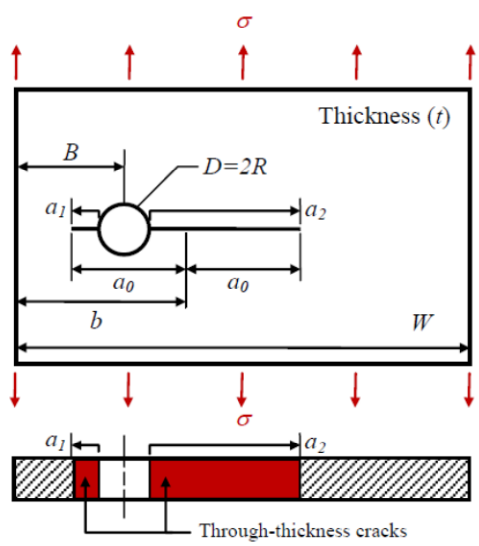


Figure 20. TC23 Solution

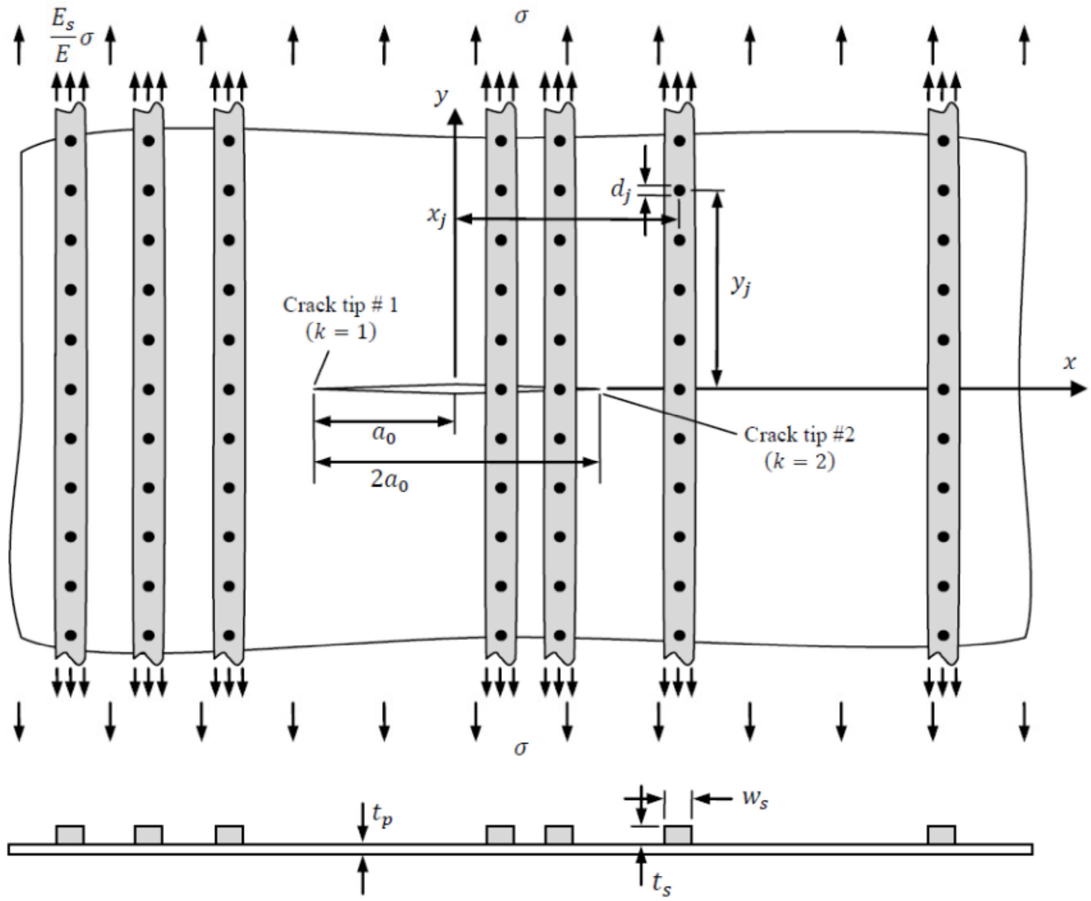


Figure 21. Crack in a panel with arbitrarily located stringers

5.2.1 References

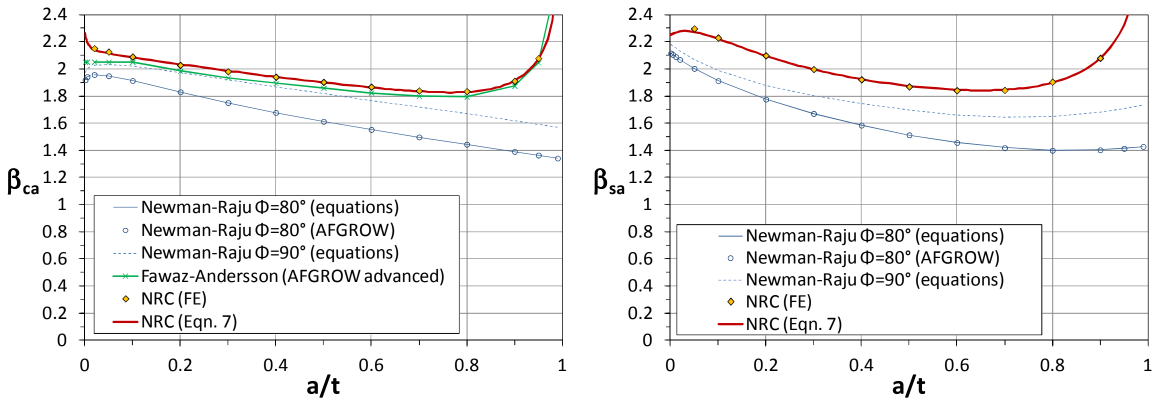
[1] Bombardier, Y., Liao, "Stress Intensity Factor Solution for Cracks at an Offset Loaded Fastener Hole", Journal of Aircraft, vol. 48, No. 3, May-June 2011, p. 910-918.

[2] Bombardier, Y., Liao, M., Renaud, G., "Improved Stress Intensity Factor Solution for Cracks in Panels with Arbitrarily Located Stringers", 53rd AIAA/ ASME/ ASCE/ AHS/ ASC Structures, Structural Dynamics and Materials Conference, 23 - 26 April 2012, Honolulu, Hawaii, AIAA 2012-1699.

5.3 IMPROVED STRESS INTENSITY FACTOR SOLUTIONS FOR CRACKS AT A HOLE AND STUDY OF BULGING FACTOR SOLUTIONS FOR AIRCRAFT DURABILITY AND DAMAGE TOLERANCE ANALYSIS

G. Renaud, Y. Bombardier, and M. Liao, National Research Council Canada

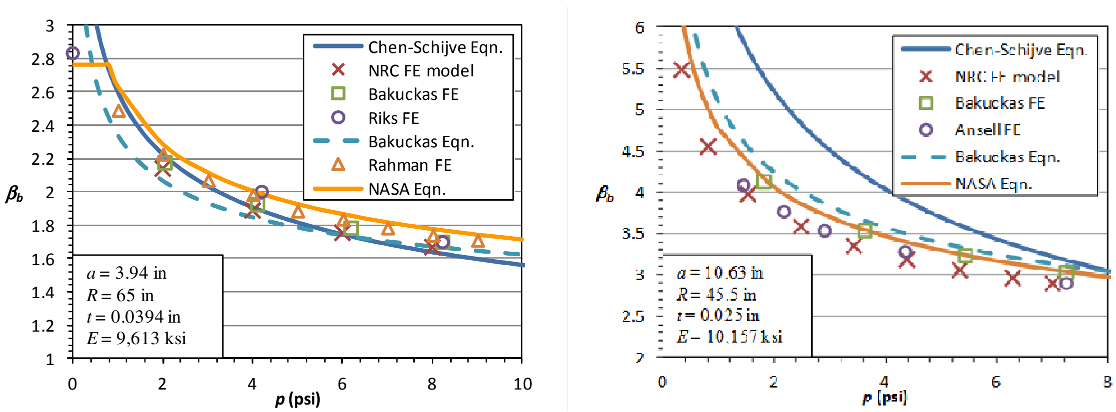
Three-dimensional finite element (FE) analysis of corner and surface cracks at a hole were carried out and improved stress intensity factor solutions for quarter-circular and semi-circular cracks were developed[1]. It was shown that the NRC solutions are more accurate than the Newman-Raju solutions, especially for surface cracks (Figure 22a) and corner cracks in the thickness direction (Figure 22b). The new solutions were also shown to be comparable to the Fawaz-Andersson results tabulated in AFGROW, which do not include the surface crack case and do not cover cracks smaller than 10% of the thickness, which is often necessary for conducting damage tolerance analysis. The developed FE model was also used to calculate the stress intensity factors of naturally nucleated cracks and compared with fractography analyses.



a) Corner crack b) Surface crack

Figure 22. Stress intensity factors (r/t = 1, thickness direction)

Crack growth in a pressurized fuselage panel is complex due to the curved geometry, the presence of biaxial and internal pressure loads, and large bulging (out-of-plane) deformations that develop local membrane and bending stresses. Bulging factor formulations available in the literature were compared with a preliminary parametric StressCheck FE model using published cases and a typical case representative of the CC-130 fuselage[2]. Comparisons showed that the developed model is able to calculate bulging factors comparable to the published unstiffened solutions (Figure 23), while being adaptable to any geometry and expandable to stiffened geometries.



a) Case 1 b) Case 2

Figure 23. β_b vs. p solution comparaison

5.3.1 References

[1] Renaud, G., Liao, M., and Bombardier, Y., “Improved Stress Intensity Factor Solutions for Surface and Corner Cracks at a Hole”, AIAA 202-1700, April 2012.

[2] Renaud, G., and Liao, M., “Bulging Factor Solutions for Cracks in Pressurized Fuselage Panels: A Preliminary Study”, LTR-SMPL-2011-0246, November 2011.

6.0 STRUCTURAL INTEGRITY

6.1 CT114 (TUTOR) AIRCRAFT STRUCTURAL INTEGRITY PROGRAM (ASIP)

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

L-3 MAS conducts a full-fledged ASIP program on the CT114 Tutor fleet on behalf of the Canadian Forces. CT114 ASIP includes the following major tasks:

- Prepare and keep up-to-date the ASIP Master Plan (known as Aircraft Structural Integrity Management Plan – ASIMP in the Technical Airworthiness Manual),
- Prepare and keep up-to-date supporting plans; such as the Structural Maintenance Plan (SMP), Durability and Damage Tolerance Control Plan (DDTCP), and Structurally Significant Items (SSI) database,
- Monitor fleet usage by collecting, processing aircraft usage information, calculating consumed fatigue life, estimating remaining structural life for each aircraft, and providing periodic reports to DND.

The purpose of the CT114 ASIP Master Plan is to define the specific approach used to achieve the required level of structural integrity for the CT114 airframe structure. It provides DND with a comprehensive reference and planning document on the current status of all facets dealing with the aircraft structural integrity. It also keeps track of the structural problems encountered up to date and the way they were dealt.

The CT114 SMP provides a list of SSI based on available failure records from full-scale tests, in-service maintenance data, and available analytical data such as DTA. It also proposes inspection techniques and an inspection interval for every SSI. The DDTCP presents the durability and DTA methodology to be used for the CT114 aircraft, identifies the durability and fracture control requirements for the airframe, introduces the methodology used to classify the structural information gathered up-to-date, and presents the CT114 SSI database structure.

Aircraft usage monitoring is achieved by collecting, evaluating and processing the Operational Loads Monitoring (OLM) system data. Periodically collected aircraft usage data is validated and accumulated fatigue damage is calculated for major aircraft components. In addition, remaining life for every major aircraft component is calculated based on predicted aircraft usage by using the software tool GIFTS. The monitoring program findings and L-3 MAS recommendations are reported to DND on a monthly, semi-annual and annual basis.

6.2 CF-18 AIRCRAFT STRUCTURAL INTEGRITY PROGRAM (ASIP) AND LIFE EXTENSION PROGRAM (ALEX)

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

As part of the System Engineering Support contract (SESC) contract, L-3 MAS conducts a full-fledged ASIP program on the CF-18 fleet on behalf of the RCAF. Most of the recent efforts are dedicated towards interpretation of the IFOSTP, Aircraft Sampling Inspections (ASI), and fleet findings (RCAF and other operators) in order to define and update the Structural Maintenance Program (SMP) of the aircraft, more specifically, the ALEX Program. Considering the RCAF Baseline Operational Spectrum (BOS) and increased scatter factors, ALEX represents a 50% extension in life over the original design. L-3 MAS is finalizing the definition and development of the third 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 RCAF in order to decide on the mitigation strategy or, in some instances, acceptance of the risk. At completion, CP3 is expected to include approximately 50 maintenance packages to be implemented at around 80% of the service life of the aircraft. The CP3 Validation and Verification aircraft was inducted and delivered in 2012. While the first two phases of ALEX concentrated on modifications aimed at meeting the safe-life criteria, CP3 is primarily a safety-by-inspection program comprised mostly 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 RCAF 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.

6.3 ANALYSIS OF THE EFFECT OF MULTI-ELEMENT DAMAGE (MED) ON A CRITICAL LOCATION USING GLOBAL-LOCAL FINITE ELEMENT MODELING[1]

G. Li, G. Renaud, and M. Liao, National Research Council Canada

To study the effect of multiple element damage (MED) on the CC-130 centre wing critical location CFCW-1, finite element (FE) analyses were performed to calculate a load reduction factor (β_{LRF}) that quantifies the load transfer to adjacent wing-box structures upon crack growth. A lower panel crack up to 20 inches long was considered under different MED and loading scenarios. The analyzed cases included local stringer failures and the failure of the front lower cap at the CFCW-14 location (WS 174), under wing-tip load and manoeuvre load conditions. Results showed that: (i) negligible change was induced by the WS 174 front lower cap failure; (ii) little difference was seen in the two loading conditions; (iii) moderate change was found between two panel crack modeling approaches; and (iv) considerable change was caused by local stringer failures near by the CFCW-1 location.

The load reduction factor β_{LRF} at the WS 61 lower panel structure region has been determined. Results (see Figure 24) showed that: (i) negligible change was induced by the WS 174 front lower cap failure; (ii) little difference was seen in the two loading conditions; (iii) moderate change was noted between the two panel crack modeling cases; and (iv) considerable change was created by lower stringer failure at the CFCW-1 location.

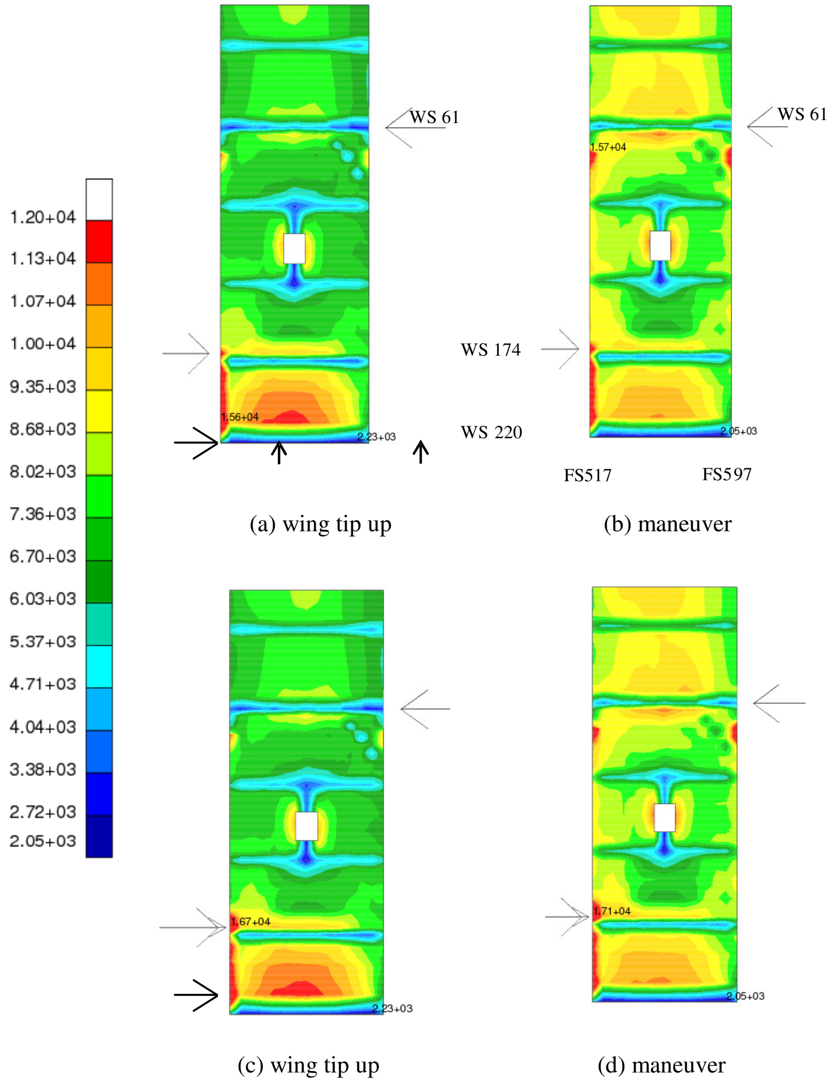


Figure 24. Full-field maximum principal stress contours (psi) on the lower wing skin without crack for (a) and (b) when the WS 174 front lower cap is intact; (c) and (d) when the WS 174 front lower cap is broken

6.3.1 References

[1] Li, G., Renaud, G., and Liao, M., Analysis of load profile at the CC-130 CW-1 critical location using global finite element modeling, LTR-SMPL-2012-0100.

6.4 BONDED REPAIR DEVELOPMENT FOR METAL SANDWICH PANELS

Directorate of Technical Airworthiness and Engineering Support (DND)

DND/DTAES is developing bonded repairs using a Sol-Gel surface prep process, which is to be carried out by units or contractors in the field. These repairs are aimed at restoring the structural integrity of helicopter metal sandwich panels which has been reduced by the different types of damage that are present. These include dents, punctures, cracks, and disbonds, Figure 25. Several repairs have already been carried out on damaged panels. The bonding process is being evaluated "as we go" due to the urgency of the repairs. Coupons (SLS and wedge) are produced with each repair and sent to the Quality Engineering Test Establishment of the Department of National Defence for testing to evaluate the quality and strength of the bond. Coupon tests are interesting because they represent a cross-section of field-level bonding (different technicians, different equipment, different locations, different ambient conditions, etc). Results so far have been very encouraging; with the worst-case SLS results on the order of 3-5 times the strength required to fail the parent material. Development of a training and qualification program is desired for future applications on the aircraft.

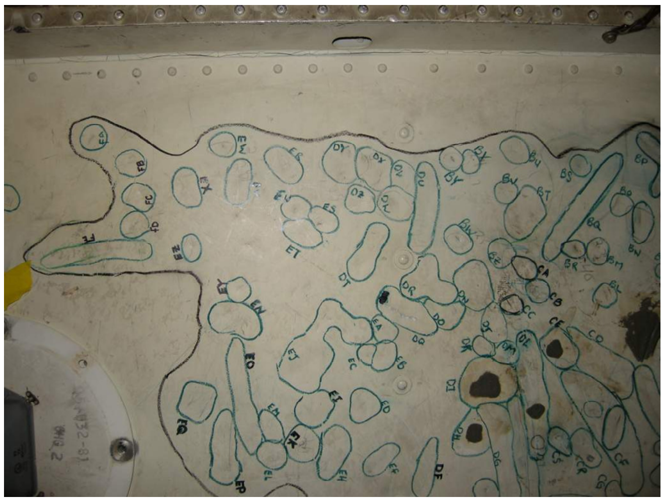


Figure 25. Image showing damage in metal sandwich structure

Currently Bell Helicopter Textron Canada (BHTC) is performing FE analysis on certain panels of the aircraft in an attempt to increase the damage limits. Current focus is on small damage (incidental, caused by FOD in off-airport landings, etc) rather than "abuse damage" (larger damage). In addition BHTC, NRC and DTAES have been evaluating different materials that can be used to protect some of the sandwich panels. Multiple protection schemes for different locations on the helicopter are currently being implemented, or are in development or certification.

7.0 AGING AIRCRAFT ISSUES

7.1 UPDATE OF INSPECTION PROGRAM FOR PRIMARY STRUCTURE ON THE RCAF CC130 HERCULES FLEET

Marshall Aerospace and Defence Group

The Royal Canadian Air Force (RCAF) operates a fleet of CC130 H-model Hercules aircraft, Figure 26, primarily in the Search and Rescue (SAR) and Air-to-Air Refuelling roles. Until recently this fleet included older E-model aircraft and also conducted tactical operations and strategic air lift across Canada as well as abroad. The latter roles have been transferred from the retiring E-models to the recently acquired J-model fleet.

Figure 26. CC130 H-model Hercules aircraft

In 2009, in preparation for this transition, the RCAF identified a need to review and update the inspection requirements for primary structure, as part of the on-going Aircraft Structural Integrity Program (ASIP), which is modeled on that defined in MIL-STD-1530C. These inspection requirements (i.e. areas, NDI methods and intervals) are based on damage tolerance analysis (DTA) of principal structural elements provided by the OEM, Lockheed Martin. These DTAs determine the critical crack length and safety limit as a function of the mission profiles, the mission mix and the environmental criteria. Being a transport aircraft, the loads in the CC130 primary structure are primarily affected by the gust environment.

The RCAF has used an onboard Loads Monitoring System (LMS) since 1997. These data were used to update the mission profiles and mix and to prepare a matrix of recorded flight data (primarily vertical acceleration exceedance data as a function of airspeed and altitude), Figure 27. Using these inputs, the OEM updated the RCAF Durability and Damage Tolerance Assessment (DADTA). The resulting report identified new inspection intervals and also incorporated refinements to inspection areas and methods as a result of in-service experience collected from other operators. The inspection intervals are reported in so-called Equivalent Baseline Hours (EBH), which is a standardized measure of usage that accounts for the severity of flying. EBH are determined by multiplying the duration of a flight by a Severity Factor (SF) specific to the mission profile. The LMS data is used to determine the most representative mission profile (from among 64 variants) and thus the SF and EBH for each flight. Cumulative EBH are tracked by aircraft and used to determine when inspections of primary structure are due.

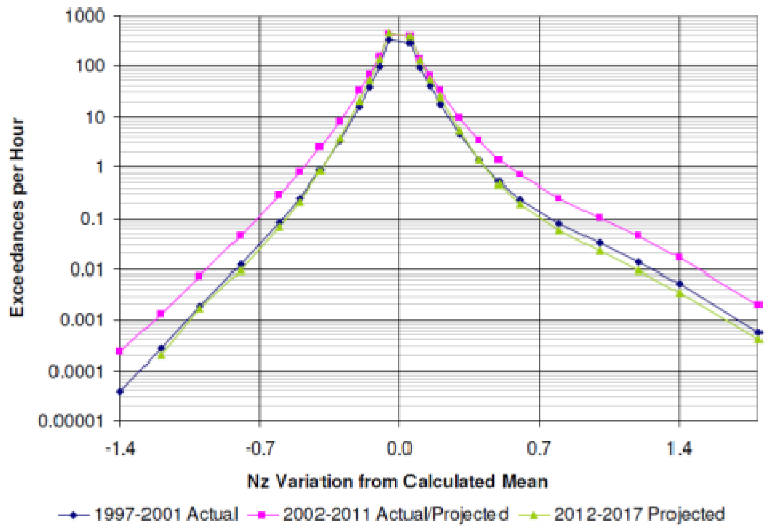


Figure 27. Historical and projected total exceedances

Implementation of the DADTA Update has required incorporating the new intervals into the inspection program, but was complicated by a parallel change in the definition of the baseline itself. The latter was previously specific to the RCAF; however an OEM standard baseline will now be used. This will simplify implementation of OEM Service Bulletins and comparison with other military operators. To change the baseline it has been necessary to review all historical usage data and convert it from the old baseline to the new. Implementation of these changes will be completed early in 2013. It should be noted that initial review of the DADTA Update determined that the inspection intervals for several locations had decreased, Figure 28. This raised the possibility that inspections were due sooner than anticipated, or even overdue on some older aircraft. A rough and necessarily conservative estimate of usage in the new baseline was used to identify special inspections to be performed as an interim measure, pending full implementation of the updated requirements in the maintenance schedule. These inspections were completed with no faults found.

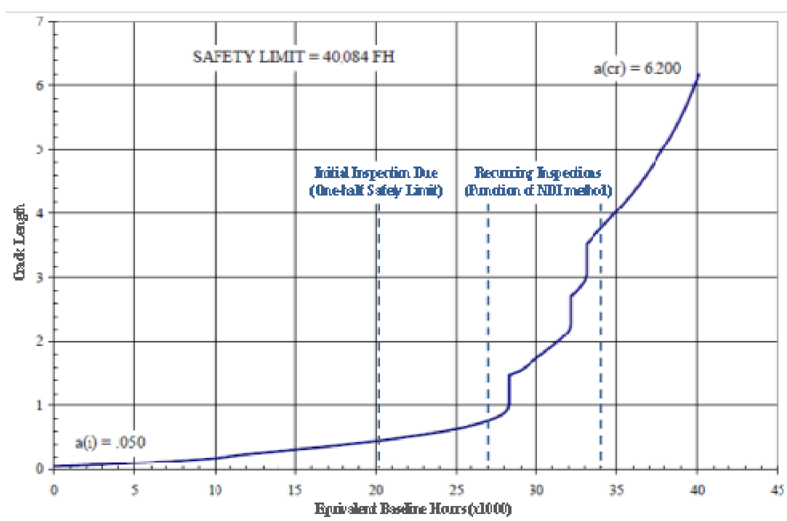


Figure 28. Initial DADTA update

Implementation of the revised inspection requirements from the DADTA Update ensures that the structural integrity of the CC130H fleet is maintained in its new role. The CC130H fleet is currently scheduled to operate until at least 2020.

7.2 ANALYSIS OF SPOT WELDS IN A NACELLE LONGERON REPAIR

M. Oore, IMP AEROSPACE

Cracks were found at No.1 Outboard Nacelle in the upper outboard longeron of a four-engine long-range patrol aircraft. Repair doublers were applied using fasteners while keeping some of the original spot-welds. Approximate analysis was conducted by modelling the spot welds as fasteners with equivalent stiffness. This analysis, however, was overly conservative, complicated and consumed substantial effort and time.

The focus of this study was to develop an effective analysis methodology, Figure 29, for spot-welds so it can be applied in future similar repair configurations. Spot-welds, Figure 30, were identified in the present study as cracked configurations and analyzed for stress intensity factors (K).

Comparisons of KI, KII and KIII by various modelling methods were presented for this configuration, see Figure 31. It is shown that spot-welds can be analyzed with some formulae available for 3D crack geometries or by finite element methods such as MSC NASTRAN and StressCheck. However, it was found that the crack element in MSC NASTRAN does not give the proper solution for KIII mode of cracking and should not be used for the spot-welds KIII analysis or other configurations that require a solution for KIII mode of cracking.

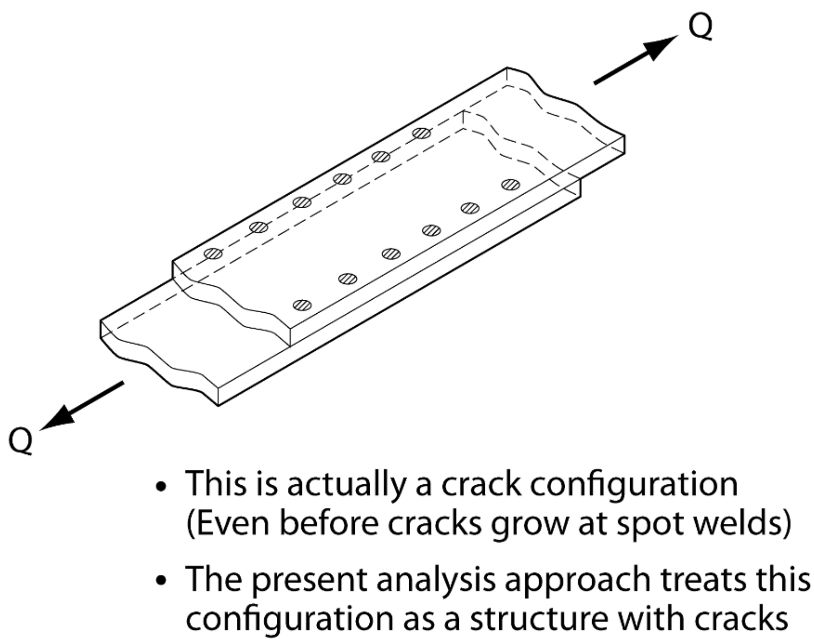


Figure 29. Simplified model of longeron section with spot welds (Two plates connected with spot weld and loaded by shear)

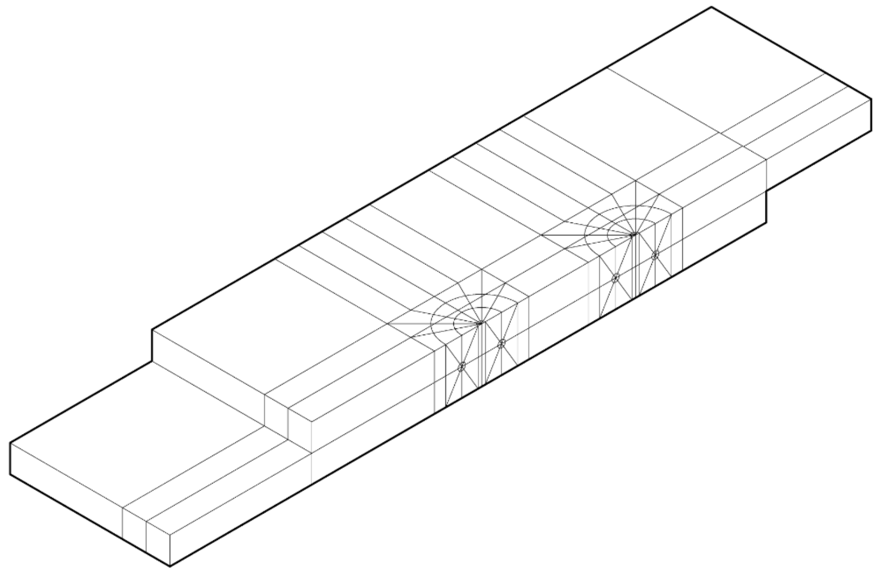


Figure 30. StressCheck 3D model of two plates connected with two spot welds

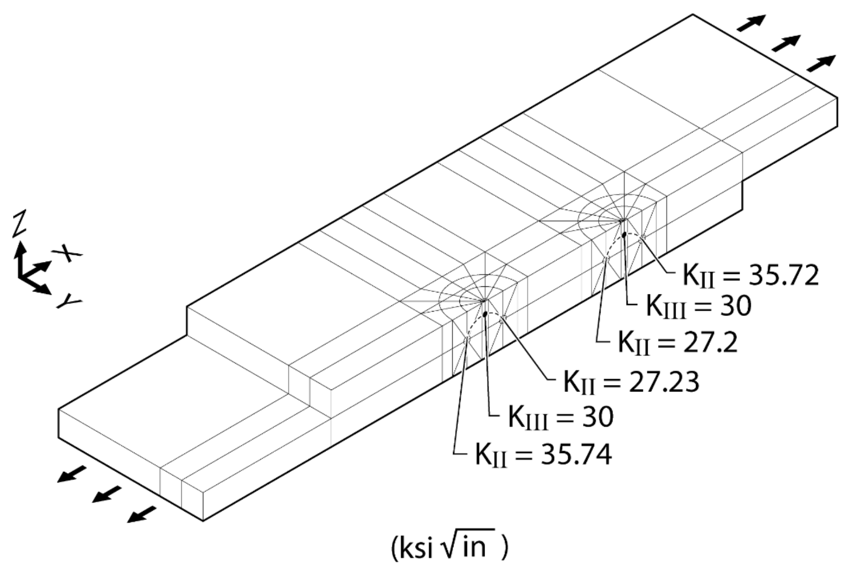


Figure 31. StressCheck results

7.3 IMPROVING BOND-LINE INTEGRITY IN COMPOSITE BONDED REPAIR

C. Marsden, R. Underhill, Royal Military College of Canada

There is an increasing use of fibre-reinforced composites as skins for aircraft structure and composite honeycomb panels used in aircraft. These materials will become damaged and require repair. In many cases it will be necessary or highly desirable to perform these repairs in situ on the aircraft in an out of autoclave (OOA) process. However, the adhesives for aerospace applications are designed for autoclave processing. When these materials are used for OOA processing, bond-line issues such as high voids content, can jeopardize the performance and long term durability of the repair. In addition, because there is often access to only one face, strong thermal gradients can form, especially in thick composite structure. This also can lead to bond line issues.

There have been various studies from other nations, in particular the US, Australia and Germany on bond-line quality improvement. Some techniques such as adhesive handling, vacuum optimization, and resin flow control have proven effective in reducing bond line voids. However, there has been no systematic study of how such defects affect the performance of the bond especially the long term performance. Bond-line requirements, such as a limit on the number and size of voids, are currently based on empirical tests, not on a true understanding of how such bond defects affect bond performance. The Royal Military College (RMC) is carrying out a project to redress this problem by developing both a clear understanding of how process variables affect the properties of the bond-line but also how the size, number and distribution of bond-line defects affect the performance of the repair.

The project will concentrate on thicker composites such as the glass and carbon fibre composites that are used in the floor panels of the Comorant helicopter or the Kevlar/carbon material used in the window frames of the same helicopter. These materials are readily available and the lessons learned should be readily transferable to other fibre-epoxy systems such as the carbon fibre/epoxy systems found in the CF-18 and potentially to composites found in the new F-35s.

Work will concentrate on applications to scarf joints. This kind of joint is particular desirable because it does not alter the moments about the repair. Some work on other simpler joint geometries will occur during the early stages of the project, especially where the results of test are unlikely to be geometry specific.

The first part of the project will concentrate on understanding how the processing parameters affect void formation and initial bond strength. This serves a dual purpose. In the first place, a clear understanding of the factors affecting void formation will allow RMC to develop techniques that minimize void production. In the second place, this understanding will allow them to be able to create specific void distributions. Since an important goal of this project is to understand how void size and spatial distributions affect bond strength and long term durability, it is essential that techniques be developed that allow the reliable production of specifically defective specimens. Without this capability, it will be impossible to develop clear models of how voids affect bond strength and durability because it will not be possible to sample a sufficiently large data space.

The second part of the project will concentrate on understanding how bond-line voiding affects both the initial bond line strength and the long term durability of the bond when exposed to moisture. Quantitative models will be developed to describe these effects.

This project has only just begun and thus the results will be presented during the next ICAF review.

8.0 UNMANNED AIR VEHICLES

8.1 A PROCESS FOR THE DESIGN AND MANUFACTURE OF ROBUST PROPELLERS FOR SMALL UNINHABITED AERIAL VEHICLES

B. Rutkay, MASc Candidate and J. Laliberte, Carleton University, Department of Mechanical and Aerospace Engineering

Small uninhabited aerial vehicles (UAVs) are currently being developed for civilian missions that require long endurance and range. One major factor influencing the range and endurance of an aircraft is its power plant and for propeller-driven aircraft there is a direct relationship between the efficiency of the propeller and the aircraft's performance. To allow Brican Flight Systems, a Canadian UAV manufacturer, to improve the performance of their UAV through increased propeller efficiency, a process to design and manufacture small batches of propellers was created. Key to this was a computer program that could rapidly and accurately predict propeller performance, check the manufacturability of the design, and estimate the strength of the blade, thus enabling a designer to quickly create an efficient and flight-worthy propeller. To validate the accuracy of the results provided by the computer program and to assess the quality of candidate materials and manufacturing methods, a test program based on showing compliance with applicable airworthiness standards (e.g. fatigue and damage tolerance) for propellers was undertaken. This test program included material and manufacturing trials and used a low-speed wind tunnel to collect simulated flight test data.

8.2 INSERT DESIGN AND MANUFACTURING FOR FOAM CORE COMPOSITE SANDWICH STRUCTURES

A. Lares, MASc graduate, J. Laliberté, P.V. Straznicky, Carleton University, Department of Mechanical and Aerospace Engineering

Sandwich structures have been used in the aerospace industry for many years. The high strength to weight ratios that are possible with sandwich constructions makes them desirable for airframe applications. While sandwich structures are effective at handling distributed loads such as aerodynamic forces, they are prone to damage from concentrated loads at joints or due to impact. This is due to the relatively thin face-sheets and soft core materials typically found in sandwich structures.

Carleton University's Uninhabited Aerial Vehicle (UAV) Project Team has designed and manufactured a UAV (GeoSurv II Prototype) which features an all composite sandwich structure fuselage structure. The purpose of the aircraft is to conduct geomagnetic surveys. The GeoSurv II Prototype serves as the test bed to advance UAV technologies. Those areas of research include: low cost composite materials manufacturing, geomagnetic data acquisition, obstacle detection, autonomous operations and magnetic signature control.

In this work a methodology for designing and manufacturing inserts for foam-core sandwich structures was developed. The results of this research work allow a designer wishing to design a foam-core sandwich airframe structure, a means of quickly manufacturing optimized inserts for the safe introduction of discrete loads into the airframe.

The previous GeoSurv II Prototype insert designs (v.1 & v.2) were tested to establish a benchmark with which to compare future insert designs. Several designs and materials were considered for the new v.3 inserts. A plug and sleeve design was selected, due to its ability to effectively transfer the required loads to the sandwich structure. The insert material was chosen to be epoxy, reinforced with chopped carbon fibre. This material was chosen for its combination of strength, low mass and also compatibility with the face-sheet material. The v.3 insert assembly is 60% lighter than the previous insert designs.

A casting process for manufacturing the v.3 inserts was developed. The developed casting process, when producing more than 13 inserts, becomes more economical than machining.

An exploratory study was conducted looking at the effects of dynamic and fatigue loading on the v.3 insert performance. The results of this study highlighted areas for improving dynamic testing of foam-core sandwich structure inserts.

Correlations were developed relating design variables such as face-sheet thickness and insert diameter to a failure load for different load cases. This was done through simulations using Computer Aided Engineering (CAE) software, and experimental testing. The resulting correlations were integrated into a computer program which outputs the required insert dimensions given a set of design parameters, and load values.

9.0 USAGE AND STRUCTURAL HEALTH MONITORING

9.1 FATIGUE DAMAGE ESTIMATION OF HELICOPTER ROTATING COMPONENTS FOR HEALTH AND USAGE MONITORING

C. Cheung, B. Rocha, J.J. Valdes, A. Stefani, M. Li, National Research Council Canada

Operational requirements are significantly expanding the role of military helicopter fleets in many countries. This expansion has resulted in helicopters flying missions that are beyond the original design usage spectrum. Therefore, the current life usage estimation for the fatigue critical components may no longer have the required low probability of failure, or conversely, operational lives could be expanded if actual mission flight load spectra are considerably less severe. Due to this, there is a need to monitor individual aircraft usage to compare with the original design usage spectrum to more accurately determine the life of critical components, within a HOListic Structural Integrity Process (HOLSIP) approach. One of the key elements of tracking individual aircraft usage and calculating component retirement times is accurate determination of the component loads. These loads can be measured through installed sensors; however, the installation and operation of a sensor suite on an individual aircraft basis is challenging and expensive, particularly when rotating components are considered. Challenges include imposed requirements of considerable low weight and small dimensions for the sensor system and difficulties in the transmission of power and data between the rotating and stationary frames (rotating components and fuselage, or base station). Traditional measurement systems for dynamic components include slip rings or telemetry systems; however, these measurement methods have not proven to be reliable, since they introduce considerable noise levels and suffer from loss of sensor signals. These systems are difficult to maintain and, particularly when telemetry systems are considered, the transmission of data is not possible at all times during the operation of the helicopter, due to conflicting requirements and potential interference with other onboard systems vital for the operation of the aircraft. Therefore, an accurate and robust process to estimate these loads indirectly is a practical alternative, which can also be used to supplement existing methods or predict future operational data. Load estimation methods can utilize data obtained from existing, centralized, aircraft sensors, such as standard flight state and control system (FSCS) parameters, thereby minimizing the requirement for additional sensors and avoiding the high costs associated with additional instrumentation installation, maintenance and monitoring.

In the sequence of the collaborative program between NRC and Australia's DSTO (Defence Science and Technology Organization), set up originally in 2007 to undertake loads synthesis for helicopter dynamic components obtained from stationary airframe sensors through the application of a variety of computational intelligence techniques, the work performed by NRC during the last two years was focused on using S-70-A-9 Australian Black Hawk flight loads survey data for estimating several main rotor loads for several flight conditions using only data from the FSCS parameters. This work investigated the use of various computational intelligence and statistical techniques, including genetic algorithms and neural networks, to explore the data and build models (Figure 32). As a result, it has been shown that reasonably accurate and correlated predictions for the examined main rotor loads could be obtained when compared with data obtained from strain gauges installed on these components during the flight loads survey.

The obtained estimates focused on generating a time signal prediction for the main rotor loads. From an operator's point of view, the load versus time signal alone does not provide enough practical information. The useful metric or performance measure in helicopter life cycle management is the component retirement time or fatigue damage accumulation which can be calculated from the load time signal. This work therefore evolved to attain a complete approach to load monitoring that includes estimating the load versus time signal and calculating the subsequent fatigue damage, particularly for rotating components, while avoiding the use of additional sensors, which is specifically challenging when dynamic components are considered.

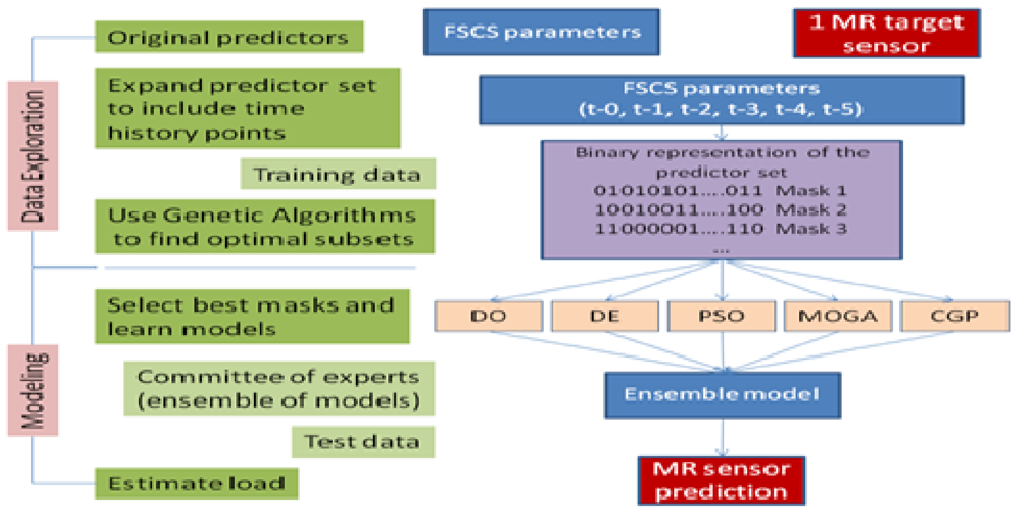


Figure 32. Methodology for load versus time signal prediction

The traditional algorithm for estimating fatigue damage is depicted in Figure 33 and Figure 34. Given the time history of the equivalent stress for a component, relevant stress amplitudes are determined and for each amplitude the number of stress cycles is counted. Rainflow counting is one commonly used method of cycle counting, with this phase being followed by the application of a damage theory for the calculation of a damage fraction due to a load cycle and subsequent damage accumulation due to a load versus time signal train. Material data is required to relate the stress amplitudes and the number of cycles to fatigue failure (S-N curve) and the Palmgren-Miner Rule can be applied for calculation of damage accumulation, with the subsequent estimation of component life.

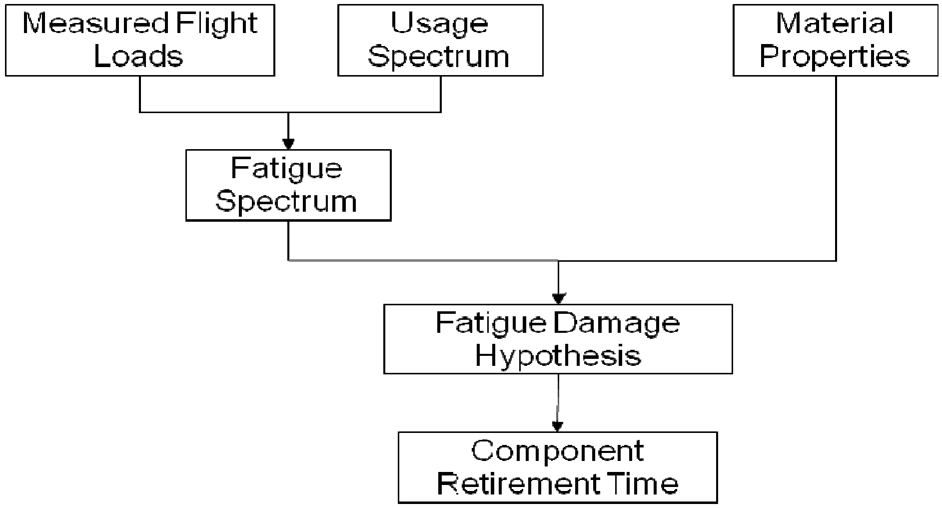


Figure 33. Safe life methodology

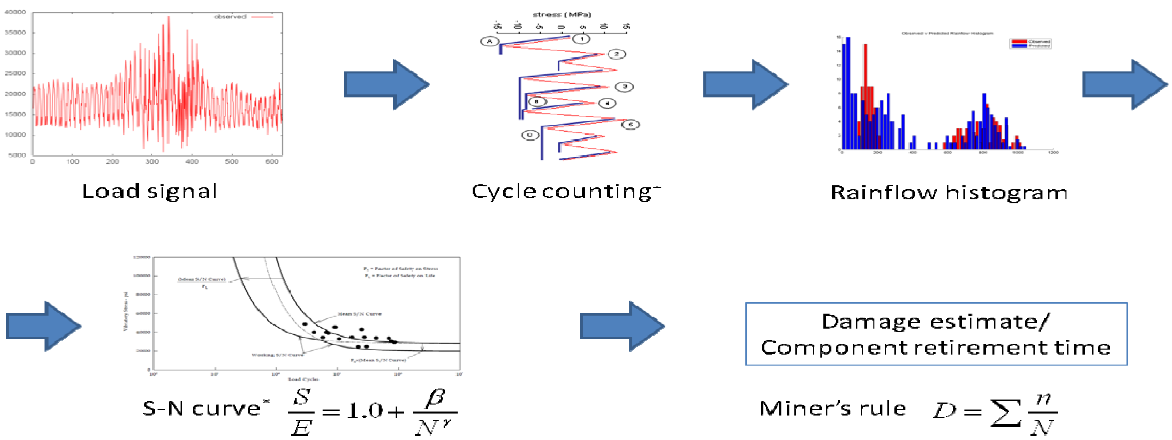


Figure 34. Fatigue damage calculation process

Initially, the standard method of fatigue damage calculation was followed. The starting point was the load versus time signal for a particular flight manoeuvre and flight record. A Rainflow histogram was constructed from the predicted load versus time signal, dividing the time signal into a specified number of bins all having the same bin width. This bin width is related to the amplitude range of the load versus time signal and the number of bins considered. The details of this binning process (width, number) are, however, proprietary to the original equipment manufacturer (OEM). Consequently, if the number of bins considered by the OEM is not known (usually not disclosed by the OEM), an error associated with a different estimate of the number of bins and their consequent width can result. Load amplitude counting could fall in different bins, with different magnitude ranges, and therefore with different estimates of accumulated fatigue damage and remaining life.

While this process for calculating fatigue damage is well known and theoretically straight forward, to apply the process to in-service load versus time data is not so straightforward without the essential and precise knowledge of the bins' width and number of bins to use. If the fatigue damage estimate is the desired end result, it is possible to reformulate the modelling process to minimize the Rainflow histogram error (difference between the predicted and observed histograms) by tuning the model. However, a number of issues were encountered with this approach including training set selection, determining the binning details without OEM information, as explained previously, and accounting for overpredicted signal amplitudes.

Due to the challenges encountered with using Rainflow histograms, an alternative cycle counting method was explored to move this work forward. In this approach, no histograms were constructed and the signal amplitudes were mapped directly to the S-N curve models to calculate fatigue damage, with minimal added computational time with respect to the computationally generated load versus time signal prediction for the main rotor loads. Two S-N curve models were used to determine the number of cycles to failure for a given stress amplitude. Using this approach, the damage fraction was estimated based on the load versus time signal predictions generated for several Black Hawk main rotor parameters. These estimates compared favourably with the fatigue damage accumulation calculated from the observed load versus time signals.

The final objective is for the developed methodology to be applied to the current and future helicopter platforms operated by the Canadian Forces. At present this work is being developed targeting the CH146 Griffon fleet.

9.2 FULL TIME-HISTORY FATIGUE USAGE APPROACH TASK

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

L-3 MAS has supported the Royal Canadian Armed Forces (RCAF) F/A-18 (CF-18) fleet management effort for over 20 years now. It remains to establish the final Structural Maintenance Plan (SMP) to maintain airworthiness and maximize aircraft availability to fleet retirement. More than a decade ago, the RCAF adopted a unique Basis of Certification (BoC) for the CF-18 aircraft. Part of this BoC is a Baseline Operational Spectrum (BOS) that reflects actual CF-18 usage. It is based on representative flights from the most severe squadron in the fleet for the period 1989-1990.

This BOS was used to develop the fatigue certification test spectrum and is used as the design spectrum for all fatigue analyses on the CF-18. As such, it needs to be representative of average fleet usage from day one to fleet retirement. However, we know that the fleet has not flown exactly like the BOS over the past 20 years. Therefore, the fatigue usage of critical areas on individual aircraft is monitored to track their progressions with respect to the design baseline.

To characterize the usage of structural locations on the CF-18 a continuous fatigue monitoring (or tracking) program was developed to represent actual fleet usage. This type of usage characterization is aimed at improving the SMP as follows:

- Maximize the life of the CF-18
- Optimize maintenance thresholds / intervals
- Prioritize special inspections.

Fatigue Usage Indices (FUI) not only provide direct estimates of individual aircraft cumulated usage (to optimize the SMP and sometimes remove some over-conservatism through reduction of the TF), but they also help define usage targets for new MES requirements. However, simplifications of the fatigue characterization model are required to render this approach practical on an operational daily basis. In some instances, these simplifications can have an impact on the estimated FUI that need to be evaluated against a more robust approach.

L-3 MAS has proposed to develop an approach that involves the generation of the full Time-History (TH) usage of each aircraft using more robust load formulation methodology. This approach is called the Time-History (TH) method. Although it offers improvements over the continuous fatigue monitoring (FUI) approach it is more cumbersome and requires much more effort to process and validate, which is the reason why it is not currently considered a viable option on an operational daily basis.

This Time-History (TH) approach was used to develop spectra of select F/A-18 fleet aircraft at Vertical-Tail and Vertical-Tail Stub locations common with existing tracking FUIs. Results from this study allowed for the ranking of the selected aircraft amongst each other and with the rest of the fleet. As well, this study permitted to evaluate the accuracy of the continuous fatigue monitoring FUI and to determine if these need to be adjusted to match the Time-History results which are considered more representative of the actual usage.

9.3 GUIDE WAVE RESEARCH

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

L-3 MAS is carrying out research related to the development of Structural Health Monitoring solutions. The use of guided waves is investigated in order to detect defects in complex aircraft structures. The use of such techniques was shown to be successful for detecting disbonds in composite structures via PZT-generated Lamb waves.

9.4 ADVANCED STRUCTURAL HEALTH MONITORING CAPABILITY DEVELOPMENT AND DEMONSTRATION INCLUDING TEST BEDS FOR SENSOR VERIFICATION AND EVALUATION

B. Rocha, A. Beltempo, R. Rutledge and M. Yanishevsky
National Research Council Canada

The National Research Council Canada (NRC) completed a three year program, receiving financial support from DND/DRDC, where it designed, manufactured, assembled, instrumented, and characterized, both analytically, numerically and experimentally, three structural test beds. The objective was to have a set of well characterized commissioned platforms, representative of the different complexity levels that can be found in current and future aerospace structures, to enable sensor and technology assessment, development and demonstrations, in structured evaluations. The work also included the development, manufacture and characterization of undamaged and damaged replaceable elements that may or may not include discrete, multiple site, and multiple element damages. Altogether, this enables the evaluation of the applicability of sensor systems to the complexity of aerospace structures, not only in terms of sensor systems' dimensions, weight and most of all performance, but also opening ways to perform evaluations considering the aircraft environment, in terms of temperature, humidity, pressure and vibration. Beyond an important assessment of sensor and associated systems, this framework enables the evaluation and development of candidate sensor systems for a gradual implementation into full scale ground testing, as part of the design and development, and operation of aircraft and components, with the final objective of using sensors and associated systems in flight, both during testing and aircraft operation; prior to more complex introduction into aircraft flight testing.

Several test beds were developed:

- The first test bed, Figure 35, consists of either a simple aluminium beam with constant, solid, rectangular cross-section or a stiffened aircraft representative aluminium skin with riveted z stringers. The test bed, through the use of up to four hydraulic actuators, can be used to evaluate load monitoring sensor and system capabilities in bending and/or torsion loading configurations, in quasi-static and low frequency conditions, simulating realistic loads and either constant amplitude or realistic operational spectra [1].
- The second medium complexity structural platform is representative of a hybrid wing box with internal aluminium structure and composite material removable skins, Figure 36, with the capability to introduce / replace damaged structural components, such as aluminium internal spars, or aluminium skins. This test bed is dedicated to the evaluation of both load and damage monitoring / damage detection systems [2].
- The third platform is able to apply simulated realistic flight loads in a controlled environment on a full scale F/A-18 aircraft outer wing box with up to 10 actuators, Figure 37. This platform includes the capability to incorporate damaged and undamaged components, with hidden damage already existing in the structural platform. These damages can be grown by applying loading spectra. This platform enables both the assessment of load monitoring and damage detection / damage monitoring technologies in a complex and realistic aerospace structure, in which existing damage locations and characterization is either not disclosed, or is not specifically identified to the sensor / system under evaluation.

The test beds provide much needed platforms for the demonstration of sensor integrity and integration of technologies on progressively more complicated structural components that have varying degrees of accessibility, while experiencing known realistic loading. As part of these developments, NRC investigated the development of robust accurate sensors for measuring deformation and calculating structural loads; and for damage detection and monitoring. Wireless technologies were also investigated for transmission and storage of sensor data for Structural Health Monitoring (SHM) systems, while other aspects, such as power requirements and transmission were also initially accounted. NRC, in collaboration with various SHM equipment manufacturers, developers and users, examined a number of technologies and systems, such as miniature gyroscopes and accelerometers, consisting of Micro Electro-Mechanical Systems (MEMS), particularly for load monitoring; Fibre Optic Sensors (FOS), such as Fibre Bragg Grating (FBG) transducers and distributed sensing, for both load monitoring and damage detection and monitoring; Acoustic Emission (AE) systems for damage detection and localization; and eddy current surface mounted sensors for measuring structural displacements and inspecting for damage and monitoring damage growth.

Future capabilities for including environmental effects relevant to aircraft service are being developed around these platforms to enable sensors and systems to be advanced through demonstrated trials further up the Technology Readiness Level (TRL) scale.

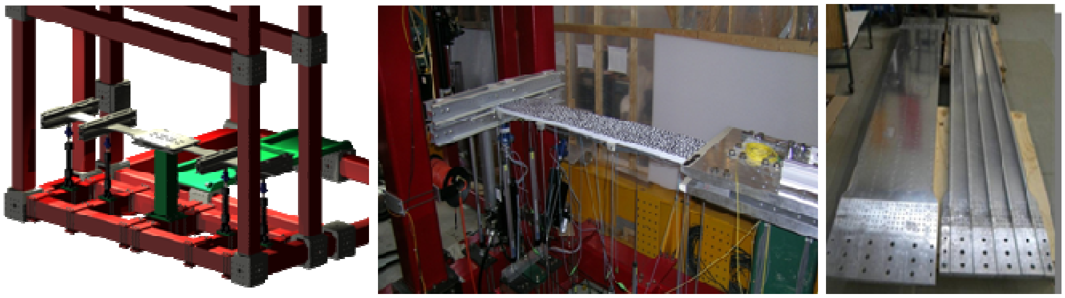


Figure 35. Simple beam and stiffened aircraft skin bending and torsion platform for evaluating load monitoring technologies

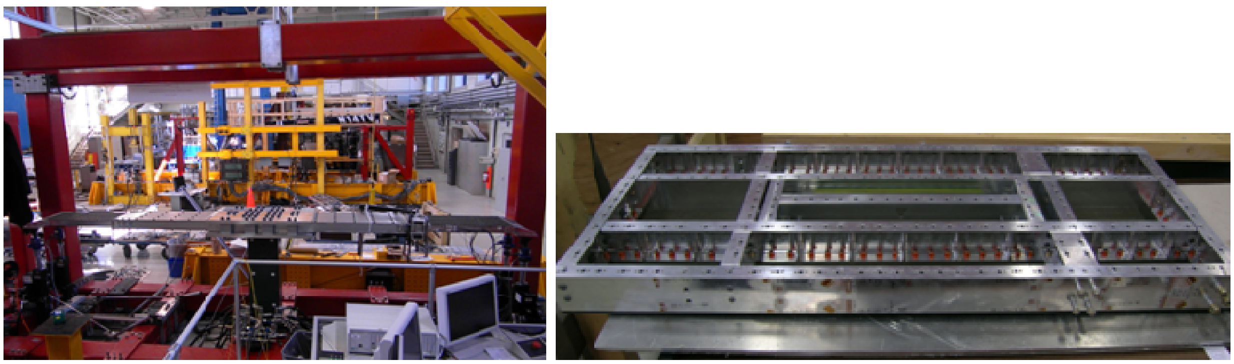


Figure 36. A simulated wing box comprised of composite upper and lower skin with aluminium spars and ribs

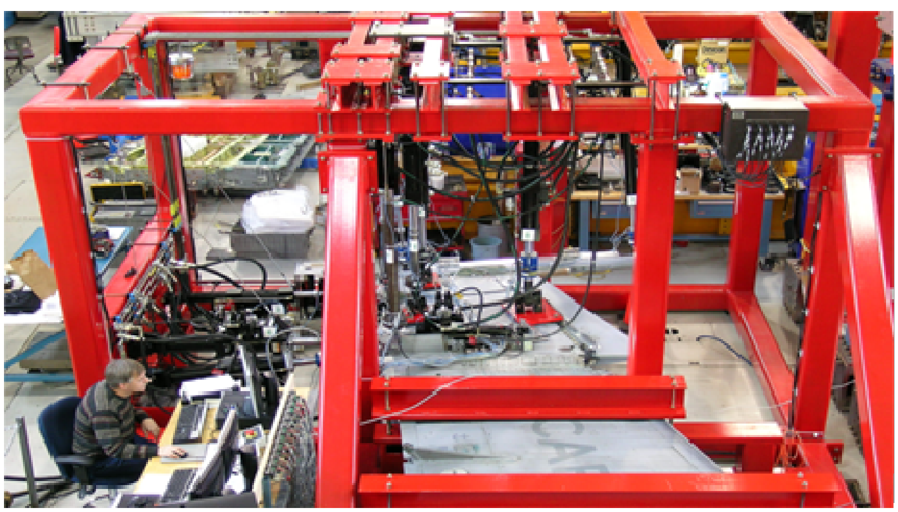


Figure 37. A large scale testing capability for F/A-18 outer wing spectrum testing

9.4.1 Acoustic Emission System Assessment with Cranfield University

An assessment of an Acoustic Emission system for crack detection was performed in collaboration with Cranfield University using the developed second SHM structural test bed; the medium complexity platform representative of a hybrid structure wing box, with internal aluminium structure and replaceable composite skins. The structure was fully instrumented with strain gauges and the test rig incorporated the use of four loading actuators. The internal aluminium structure was prepared to include an additional spar, consisting of an aluminium C-channel with embedded damage. This damage consisted of an artificially seeded fatigue crack which was grown from a sawcut introduced at the edge of an undersized hole in an extended length C-channel, Figure 38, in accordance with approaches developed and documented in [3]. After growing the crack to the appropriate size, the C-channel was cut to final length, a simulated fuel transfer hole cutout was introduced into the web and all the necessary fastener holes drilled to final size. In the case of the initial undersized pre-cracked hole, the final drilling process eliminated all evidence of the crack origin leaving a small fatigue crack at the final drilled fastener hole edge for subsequent testing of the AE system.

To perform a realistic evaluation of the AE system capabilities, the characteristics of the crack in the C-channel, such as its orientation, length, and its location were not disclosed. During the assessment, spectra with realistic loads were applied, inducing strains in the test structure typical of the ones observed during aircraft operation. This evaluation ended with interesting, positive and promising results, with the evaluated use of the AE system enabling the identification of the structural area where the original introduced fatigue crack existed and was growing. The system was also able to identify potential additional damage growing in the test structure; the latter indication was confirmed using a non-destructive inspection (NDI) bolt hole eddy current technique[4].

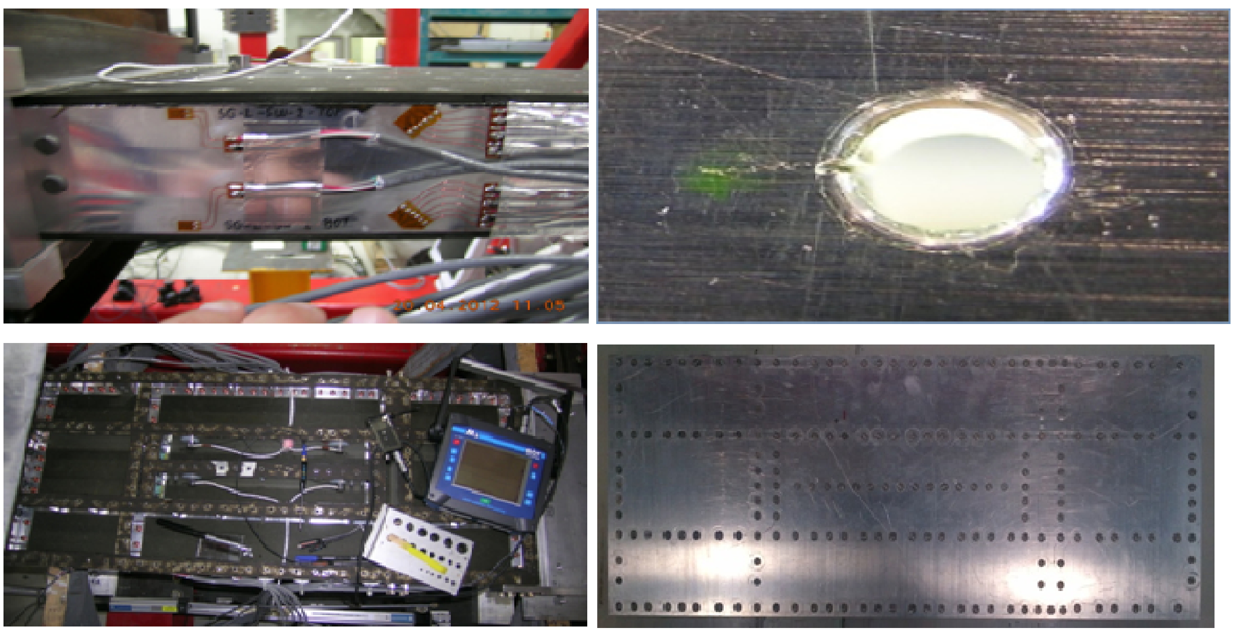


Figure 38. Acoustic Emission system evaluation on representative hybrid material wing box test structure

9.4.2 Development and Assessment of a Micro Electro-Mechanical Systems based Sensing Technique for Load Monitoring

A sensor system based on Micro Electro-Mechanical Systems (MEMS) tri-axial accelerometers and gyroscopes is being developed at NRC for structural load monitoring. By measuring the accelerations and angular velocities in the three spatial directions, the sensor nodes are able to output their spatial angular orientation, which can then be utilized to derive structural deformations by using several sensor nodes in the load monitoring system. Through the obtained deformations, structural strains are calculated and by using analytical and numerical methods, applied loads and loading configurations can be estimated. Several methods were conjointly developed and incorporated in the sensor system, including filters, and optimization and processing techniques, such as quaternion math and a Kalman filter, by which the different system measurements (obtained by the gyroscopes and accelerometers) can be correlated enabling their measurement errors and noise to be decreased, leading to more accurate results[5].

The developed sensor nodes are miniature in size and lightweight, with the possibility of being either bonded, or mechanically attached to the structure being monitored. The required power is relatively small, making it possible for the sensor units to be powered either by using aircraft power or by a small local battery for considerable periods of time. The latter powering option avoids the use of extensive electrical wiring networks, reducing overall system weight with reduced risk of Electro-Magnetic Interference (EMI) and Electro-Magnetic Compatibility (EMC) issues with other avionics. In addition, the small power requirement, being satisfied by a local battery power supply, enables the potential for energy harvesting local to each sensor node, from vibration, solar, RF or thermal sources among others. This feature is currently being explored by NRC.

The sensor nodes have incorporated local data storage and computational processing capabilities, enabling sampling rates in the order of hundreds of Hertz. In addition to wired data transmission, for instance using a USB interface, a wireless data transmission capability was developed and incorporated in each sensor node. The developed wireless system enables the transmission of measured and calculated data in real time; or by transmitting stored data at opportune moments during sensor system and aircraft operations. With this latter option, conflicting use of the sensor system wireless capability with aircraft operations and other aircraft related systems is avoided.

This MEMS based load monitoring system, Figure 39, was installed in the first SHM structural test beds and tested through the application of bending, torsion and coupled bending and torsion loading configurations. These load conditions were applied quasi-statically and at low frequencies, typical of manoeuvring loads. Constant amplitude and/or frequency, and varying amplitude and/or frequency loading spectra were applied, with the results from the developed load monitoring system under assessment being compared with the measurements obtained from the platforms' resident displacement, strain gauge and load application baseline instrumentation. These assessments were concluded with the developed sensor system results were similar to the measurements of the baseline instrumentation with a relative difference of 4% on average and with no phase shifts. These promising results have led to the installation of the system to assess its load monitoring performance on the complex F/A-18 outer wing SHM test bed at NRC.

Figure 39. MEMS based load monitoring system under assessment

9.4.3 Load Monitoring and Damage Detection using Fibre Optic Sensor Systems

Fibre optic sensing systems present several advantages for SHM of aerospace structures. Fibre optic sensors (FOS) are compact, light weight and they can be readily embedded into structures or bonded to their surfaces with minimal physical interference. Since these sensors use optical/light power, they present excellent Electro-Magnetic Compatibility (EMC) and Electro Magnetic Interference (EMI) characteristics, not affecting, or being affected by neighbouring electrical systems or avionics. NRC has been assessing, developing and evaluating different FOS solutions for both load monitoring and damage detection and monitoring. These solutions include point sensing with the use of Fibre Bragg Grating (FBG) sensors and distributed sensing with technologies based in Rayleigh backscatter.

The NRC developed SHM structural test beds were used in the evaluation of the FOS based load monitoring systems, for both FBG[6] and distributed sensing based systems[7]. Different loading conditions were applied, ranging from pure bending and pure torsion, to coupled bending and torsion, applied quasi-statically and at low frequency, using sinusoidal and representative operational spectra. Examples of the applications on the different platforms are presented in Figure 40 and Figure 41. In Figure 42, the strains measured by the distributed sensing system are presented together with the strains measured by baseline strategically positioned strain gauge sensors. The relative differences between the measurements are presented as a percentage difference. It can be observed that the maximum differences in strain were approximately 14 $\mu\varepsilon$, corresponding to less than 4% with respect to the strain gauge values. The maximum relative differences of approximately 20% correspond to small strain differences of around 4$\mu\varepsilon$. Similarly, in Figure 43, the measurements for the FBG based system are presented.

A damage detection hybrid system based on FBGs and piezoceramic transducers is being developed and the proof of concept has been experimentally tested at NRC, as shown in Figure 44. The initial trials were performed in a quasi-isotropic Carbon Fibre Reinforced Polymer (CFRP) panel with a drilled hole to simulate damage.

Figure 40. Optical fibre sensor for distributed sensing installed in the first SHM structural test bed

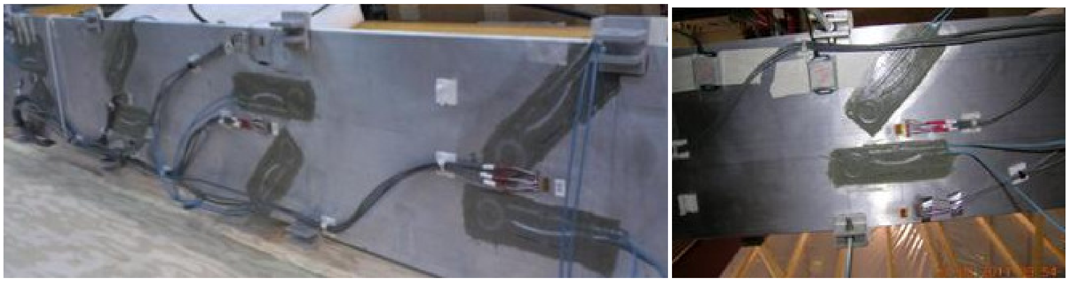


Figure 41. FBGs installed in the first SHM structural test bed, for point strain sensing

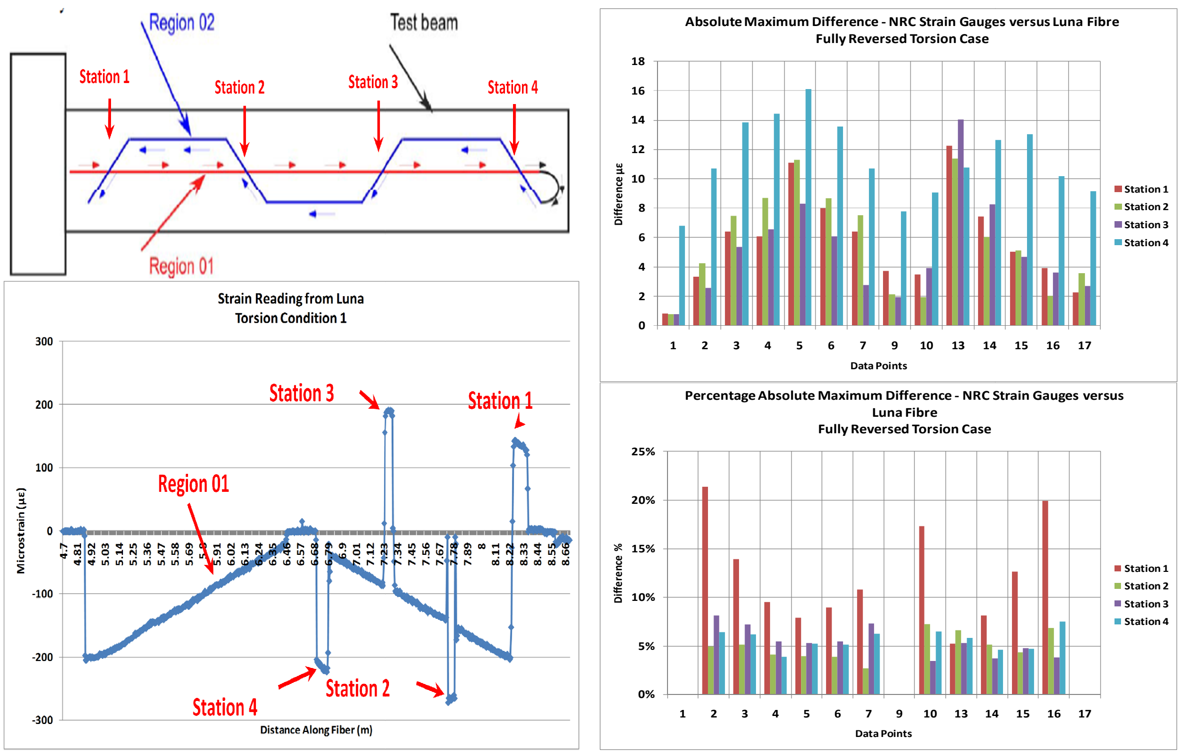


Figure 42. Strains measured by the FOS distributed sensing system

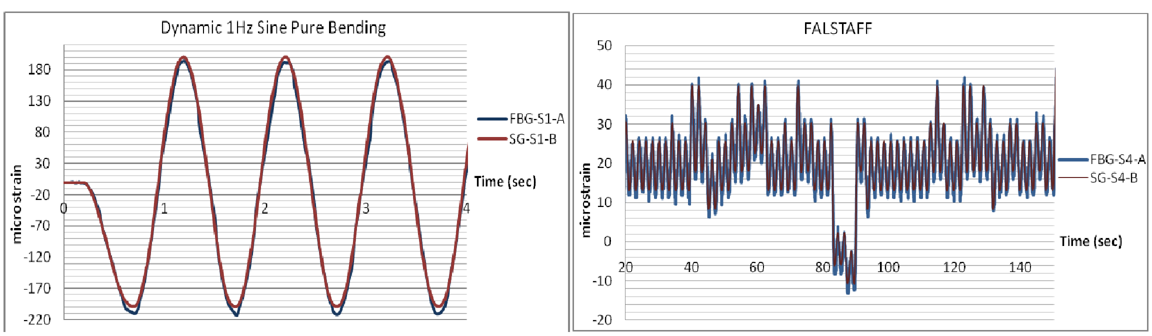


Figure 43. Strains measured by the FOS FBGs

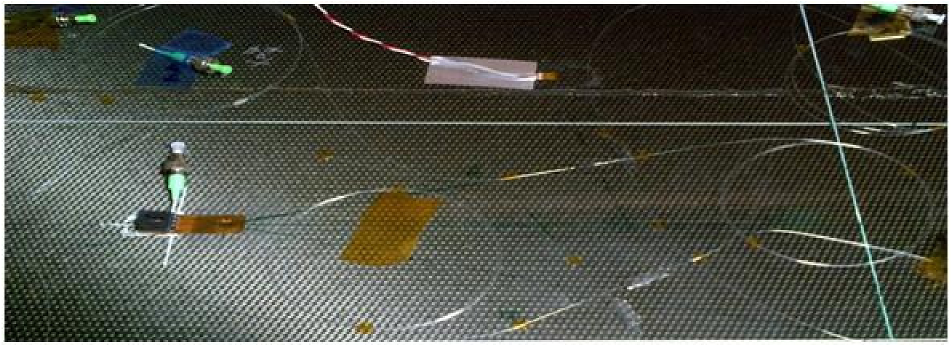


Figure 44. Proof of concept setup with CFRP skin for testing a hybrid FBG/piezoceramic damage detection system

9.4.4 References

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9.5 STRUCTURAL HEALTH MONITORING (SHM) ACTIVITIES AT BOMBARDIER AEROSPACE (BA)

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

9.5.1 RESEARCH PROJECTS

Bombardier Aerospace is the industrial lead for a research project entitled “Characterization of Guided Waves Propagation in Aircraft Structures”, partially funded by the Consortium for

Research and Innovation in Aerospace (CRIAQ) and launched in September 2012. The project is knowledge-gap oriented and aims at providing the industry with a knowledge base of typical cases for propagation of guided waves and their interaction with defects in aircraft structures. The waves will be generated by an actuator and measured using a vibrometer. The test articles will be selected in a building block approach and will be representative of real aerospace structures.

In addition another research project is being carried out to investigate the SHM benefits to future Aircraft entitled "Design Optimization of an aerospace structure for Structural Health Monitoring". This project will investigate how the design philosophy of an aircraft structure can be evolved from a "design for inspection" to a "design for SHM" in order to address possible impacts on the structural performance, weight and maintenance planning compared with conventional design philosophy.

9.5.2 INTEGRATED SHM TECHNOLOGIES

Technology Readiness Roadmaps were developed for the most focused approaches in the SHM field including Fatigue Monitoring and damage monitoring i.e. Acousto-Ultrasonics, Fiber Optics distributed strain sensors, Eddy Current Foil and Comparative Vacuum Monitoring. Among them Acousto-Ultrasonics technology has been selected to demonstrate some of the captured gaps in the technology roadmap.

A project is also being carried out that included flight testing of the Acousto-Ultrasonics technology in an operational environment. In this project sensors will be installed on flight test vehicles in order to demonstrate the system prototype for the specific application. Sensor performance will be monitored for the course of two years flight testing in an operational environment, e.g. non-pressurized zone, non-controlled temperature, severe humidity environment and normal operational vibration.

9.6 STRUCTURAL HEALTH MONITORING FOR GENERAL AVIATION AIRCRAFT

S. Pant, PhD Candidate, J. Laliberte, Carleton University, Department of Mechanical and Aerospace Engineering and M. Martinez, TUDelft

The overall life cycle cost of an aircraft could be reduced with the use of a Structural Health Monitoring (SHM) system as part of a Condition Based Maintenance (CBM) approach. In a CBM approach the aircraft is taken out of service only when potential damage is detected by the SHM system. In a typical SHM system, the structure is monitored using on-board sensors continuously or at discrete intervals. The data gathered by the sensors can be processed on-board or sent to the ground control station for evaluation. The evaluation results are used to identify damage occurrences and inform the operator and maintenance personnel about the location and severity of the damage.

There are several means of detecting and analyzing damages using SHM systems. One of such techniques is by using acoustic-ultrasonic Lamb waves, which are generated and acquired by piezoelectric actuators/sensors installed on-board the structure. Lamb waves are ultrasonic guided waves that are highly sensitive to interference on the propagation path and can travel long distances. Damage can be detected using the difference between the group velocity of a Lamb wave on damaged and un-damaged baseline specimens.

The propagation characteristic of Lamb waves is given in the form of dispersion curves, illustrating the plate-mode phase and group velocity as a function of the excitation frequency. In order to generate such dispersion curves material elastic constants are required. A Matlab-based algorithm has been developed using the 3D linear elastic model in order to generate the dispersion curve of both metallic and n-layered composite laminate. Also as a part of the SHM system development; in-situ methods for characterizing the elastic properties of structure using ultrasonic waves is currently being developed and tested for metallic specimen. The method is being tested and verified experimentally using piezo transducers and receivers and numerically in ABAQUS. Once proven the method will be expanded to composite structure.

10.0 NONDESTRUCTIVE INSPECTION AND SENSORS

10.1 MAGNETO-OPTICAL IMAGING FOR EASY USE IN AIRCRAFT STRUCTURE INSPECTION

M.Genest and C. Mandache, National Research Council Canada

Magneto-optical imaging (MOI) is a relatively new electromagnetic non-destructive inspection (NDI) technique that can be applied in most cases where eddy current and magnetic particle inspections are viable. The majority of reported applications are referring to MOI's use in aerospace NDI for detection of cracks, Figure 45, and corrosion at rivet sites.

Although not necessary more capable than eddy current inspections, MOI provides definite advantages over conventional surface scan eddy current, such as: fast inspections – an area of a few cm^2 being inspected in a matter of seconds, and intuitive data interpretation – the inspection result is in image form instead of signals. These advantages could reduce the aircraft down time and increase the NDI reliability. In its current state, the wide acceptance of the MOI technique is hindered due to noise, lack of recordable results, and impossibility of data post-processing.

NRC has developed a set of software/hardware add-ons on the commercial MOI 308/37 system (by Quest Integrated). The hardware includes optical encoders, custom imaging head fixture, video capture card and a laptop computer. The software includes Matlab-built graphical user interfaces that allow video recording of the inspection, image stitching and different processing algorithms, such as background subtraction, de-noising, and image enhancement.

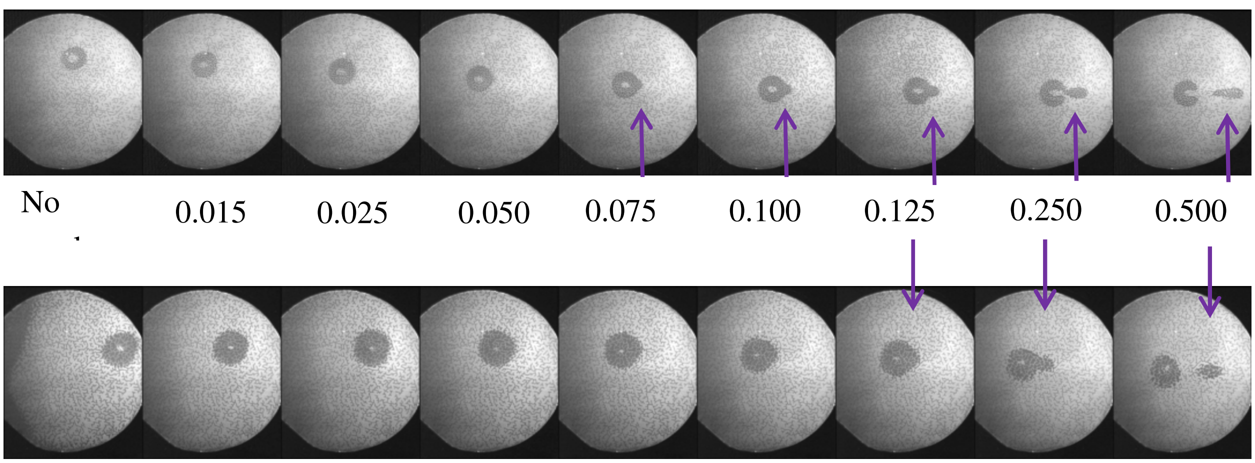

Figure 45. Example of a MOI results for the inspection of a row of 9 rivets with cracks in first layer (up) and second layer (bottom)

10.2 ESTIMATING PROBABILITY OF DETECTION FROM IN-SERVICE NON-DESTRUCTIVE INSPECTION DATA

M. Khan and M. Liao, National Research Council Canada

NRC has been evaluating the capability of a new probability of detection (POD) estimation approach proposed by Berens to estimate an effective POD from in-service inspection data (Figure 46). Non-destructive inspection (NDI) plays a critical role in the Damage Tolerance (DT) life cycle management of aircraft structures. The DT approach relies on periodic NDI of fracture critical components. The reliability of these techniques is quantified in terms of the probability of detection (POD) from which the crack size at a 90% POD with a 95% confidence level can be determined.

The standard POD estimation approach (following MIL-HDBK-1823 guidelines) requires a statistically valid number of simulated specimens with representative materials, flaws, geometry and inspection procedures. The inspections are usually carried out in a laboratory environment, but all variables should be controlled so as to simulate in-service conditions as closely as possible. Both hit and miss crack data from inspection results are required to construct the standard POD and its confidence bounds. This approach is both time consuming and expensive. However, most representative cracked airframe structures cannot be properly simulated by laboratory generated coupons with electric discharge machining (EDM) notches. In-service crack data often does not provide any information about sizes of non-detected (missed) flaws unless a full teardown is performed. Therefore the standard POD approach is difficult to apply for in-service inspection data. Estimating POD using in-service inspection data is considered to be an alternative, cost-effective approach when data is available. This work was carried out to evaluate the capability of a POD estimating approach based on Berens' effective POD model using in-service crack detection data without the detailed information of missed cracks.

Several case studies were carried out to assess the capability of this approach, using both laboratory generated fatigue crack and in-service crack data sets (Figure 47). The studies demonstrated the promising results from this approach to estimate POD using only the detection data, without the need for the sizes of missed cracks. It also identified a few strengths and limitations of this approach which may be useful to future applications of this approach and interpretation of result.

10.2.1 References

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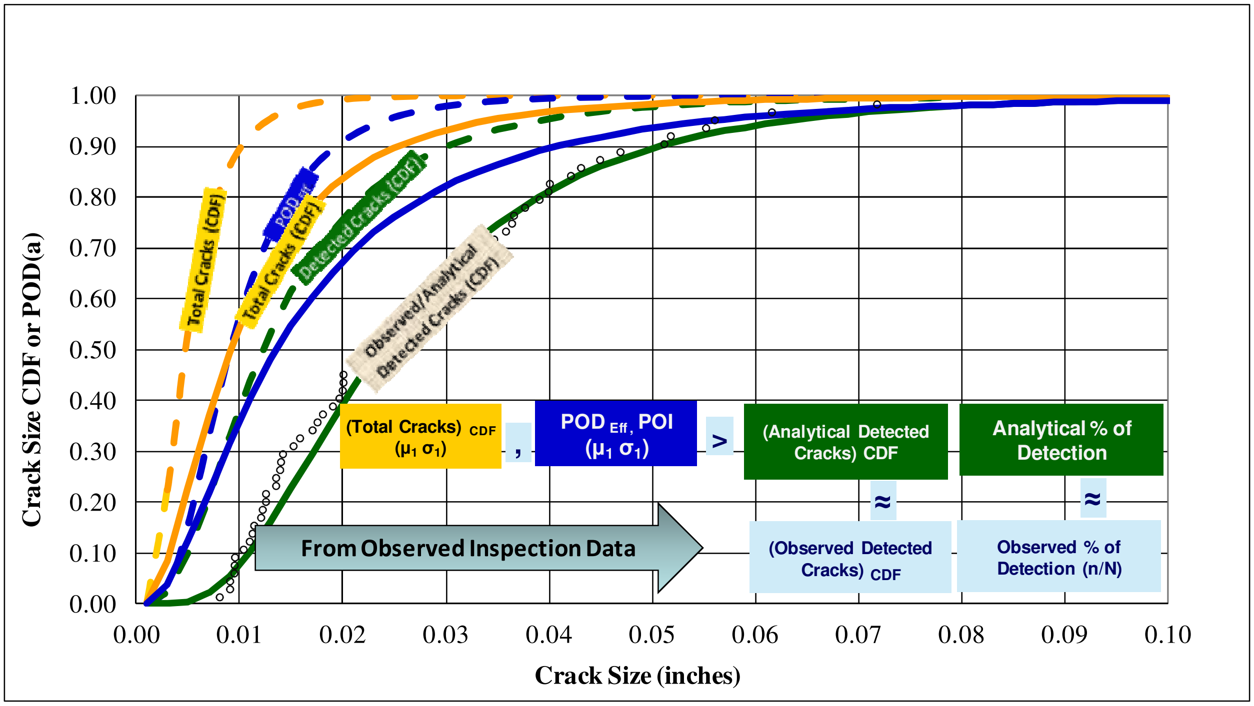


Figure 46. Illustration of iterative process to find best fit characteristic of effective POD

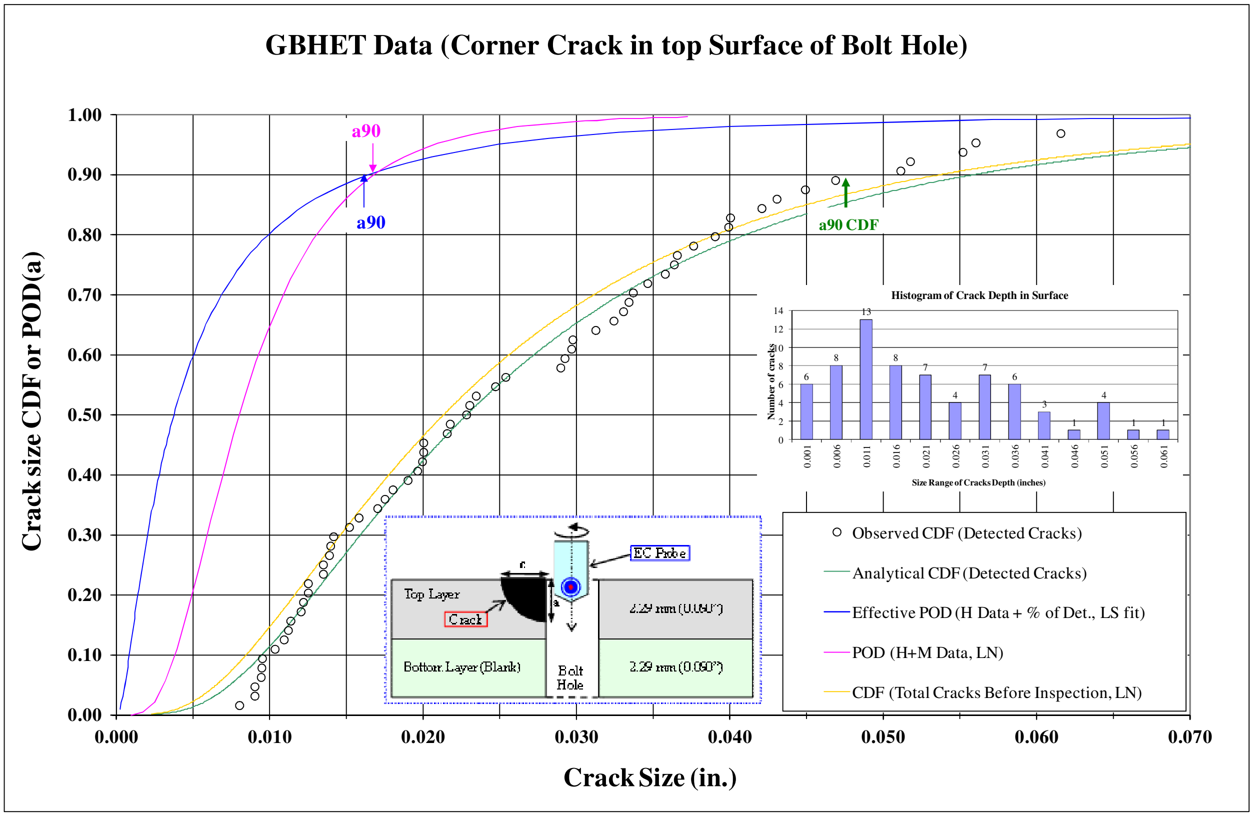


Figure 47. Example curves of standard POD, effective POD, and CDFs of actual and detected cracks

10.3 DETECTION OF FRETTING CRACKS IN AIRCRAFT STRUCTURES USING NRC ULTRASONIC SENSING TECHNOLOGIES

Kuo-Ting Wu[a], Zhigang Sun[a], Cheng-Kuei Jen[a], Alain Blouin[a], Nezih Mrad[b] and Hugo Bélanger[c]

[a]National Research Council Canada
[b]Department of National Defence, Air Vehicles Research Section
[c]L-3 Communications MAS (Canada)

Fretting fatigue originates from cycling stress exerted on a surface. In this study, we were interested in detection of fretting cracks around rivet holes. Among existing in-situ non-destructive crack detection means, Eddy Current (EC) technique and liquid penetrant inspection technique have gained much popularity due to their ease of use and capability of detecting surface cracks. However, the application of the EC technique requires removal of the fastener, which is certainly not desirable particularly for riveted areas where the rivet removal process itself could be more damaging. Liquid penetrant technique requires direct access to the inspected area and its applicability is hindered if cracks are hidden under the fastener or covered by another component (for instance, skin). The objective of this project was to find ways to detect these cracks without the need of removing any components.

Since ultrasound has the potential to detect flaws remotely without the need for removing fasteners or other structure components for gaining access, it should be the first choice among all options if it can produce satisfactory results. A multi-point ultrasonic detection approach was developed consisting of generating and detecting shear horizontal surface acoustic waves and inspecting the region of interest (ROI) from an outside accessible area by manually moving the sensor on the structure surface towards or away from the crack. Figure 48 to Figure 51 show, respectively, a test piece (aircraft stabilizer former), cracks on the test piece, schematic of intended implementation method of the detection approach, and detection results. In the motion scan picture (Figure 51), crack 2 indicated in Figure 49 was clearly identified.

The amplitude of ultrasound propagating through a heterogeneous medium can be attenuated by the presence of dissimilar parts due to acoustic scattering loss at the boundaries between the dissimilar materials. When micro cracks form in a matrix material, the region containing these cracks becomes heterogeneous. When an ultrasonic wave propagates through this region, it will be attenuated more than it would be for a same distance in a pristine region. This means that by monitoring the strength of diagnostic ultrasonic waves through a material, micro fretting cracks may be detected at an earlier stage in the fretting fatigue life before they evolve into “macro-cracks”, which are only detectable much later with conventional NDE techniques. To verify this assumption, a 20-MHz longitudinal wave transducer was placed on the test piece near the edge of a hole to generate a diagnostic wave that went across the plate thickness and then captured the echo signal from the opposite side of the plate (Figure 52). One hole with cracks around and another hole without cracks were selected. A total of 24 positions starting from position 1 (0°) with an increment of 15º from one position to the next were evaluated for each selected hole. In the case of a hole with cracks, position 1 was where the centre crack was located. In the other case, a similar position was taken as position 1. In the case without fretting cracks, the echo amplitudes measured at the 24 positions around the hole were quite uniform (Figure 53). In the presence of natural cracks, the signals obtained at position 24 and positions 1 up to 6 were significantly lower than those obtained at other positions (Figure 53). The weakening of signals at these locations is believed to be caused by natural cracks in the area. Based on this observation, an SHM approach for crack detection or crack formation and growth monitoring was proposed (the test piece (V-tail stub) is used as example);

- An ultrasonic transducer network is fabricated on an annular thin metallic substrate using NRC paint-on sensor fabrication technology (Figure 54). This sensor network is then dry-coupled to the surrounding of a rivet (or bolt) hole on the outboard flange of the V-tail stub where crack formation and growth are to be monitored. Each sensing element can be used individually to check the integrity of the material underneath or jointly with some of the other sensing elements for sensor diagnosis and calibration. The sensor network can be made as thin as 120 µm. Moreover, according to a recent lab study, the paint-on sensor could sustain up to 500 bars of compression pressure when placed on a flat surface. Figure 55 shows a trial version of the sensor network (“smart washer”). Figure 56 displays echo signals obtained by applying this sensor network to a 6.35-mm thick aluminum plate.

The sensor network, being permanently mounted on the structure, can be hooked up to a main ultrasonic measurement unit only when a routine checkup has taken place after the aircraft is grounded, eliminating the risk of electromagnetic interference with on-board flight control and the need of on-board ultrasonic equipment. In addition to the possibility of providing early warning of micro-crack formation before the crack progresses to a macro level, the technology could also be used to early detection of a loose fastener.

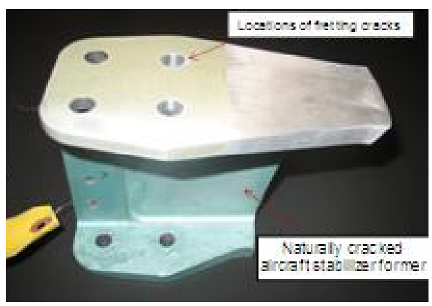
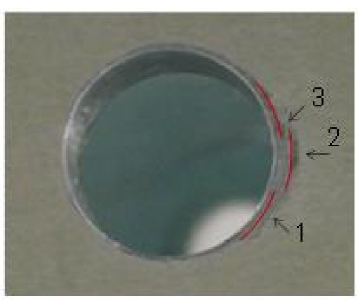


Figure 48. Naturally cracked aircraft stabilizer former (V-tail stub)



Crack ID	Length (Inch)	Minimum depth (Inch)	Distance to hole edge (Inch)
1	0.212	0.030	0.035
2	0.216	0.030	0.060
3	0.215	0.030	0.034

Figure 49. Fretting cracks on the stabilizer former sample

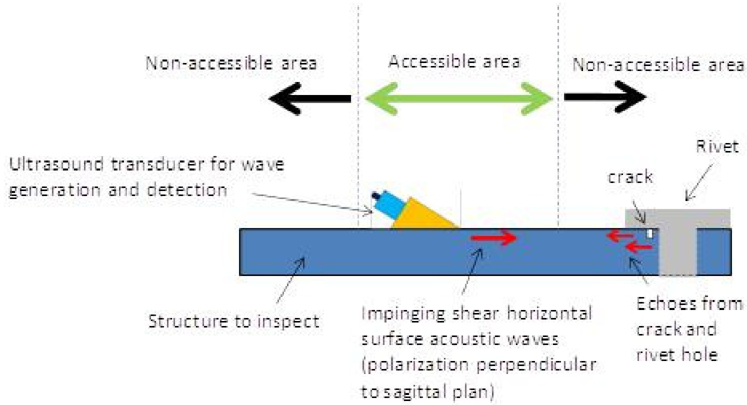
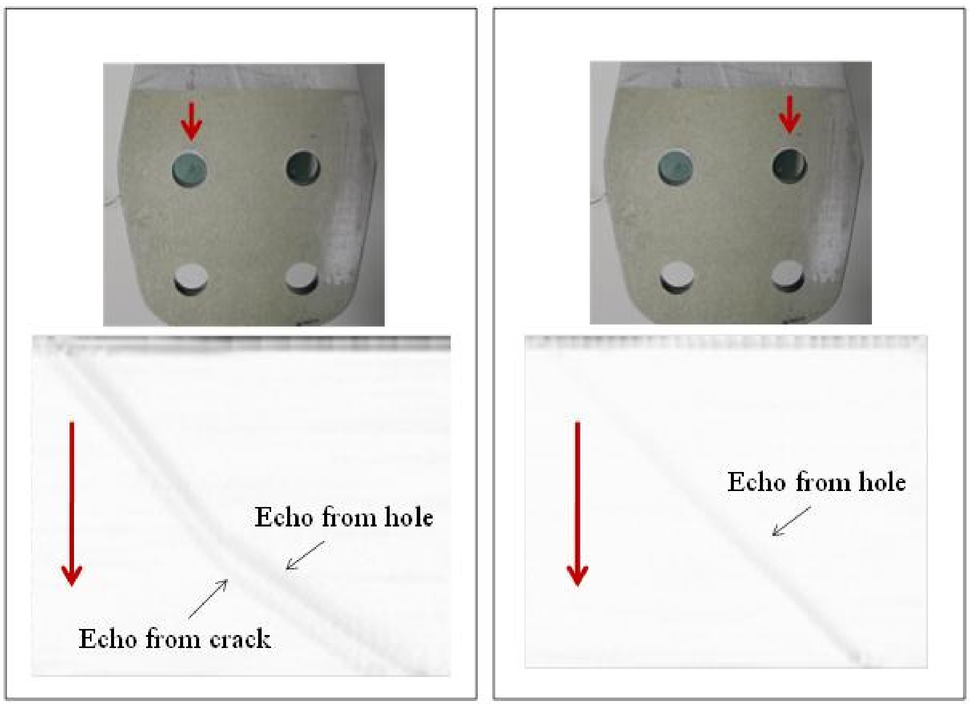


Figure 50. Intended implementation method of the sensing approach. Multiple inspections are to be conducted by moving the ultrasound transducer in the accessible area towards or away from ROI



(a) (b)

Figure 51. Motion scan pictures obtained by moving the ultrasound transducer towards a ROI. Scan results (a) on the crack side and (b) on the side without crack. The red arrows indicate sensor and echo moving directions

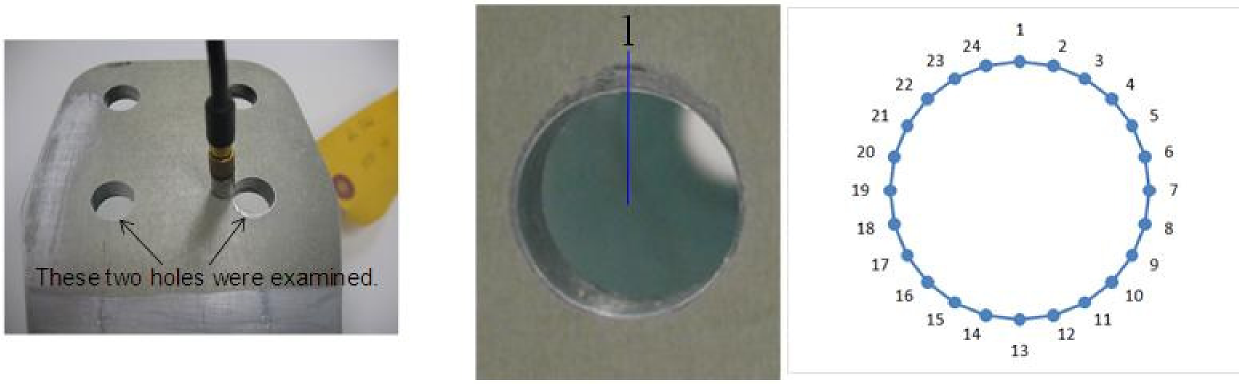


Figure 52. Experimental setup. Left: Transducer and its location with respect to the edge of a hole under examination; Middle: Definition of position 1 (0°); Right: 24 positions examined

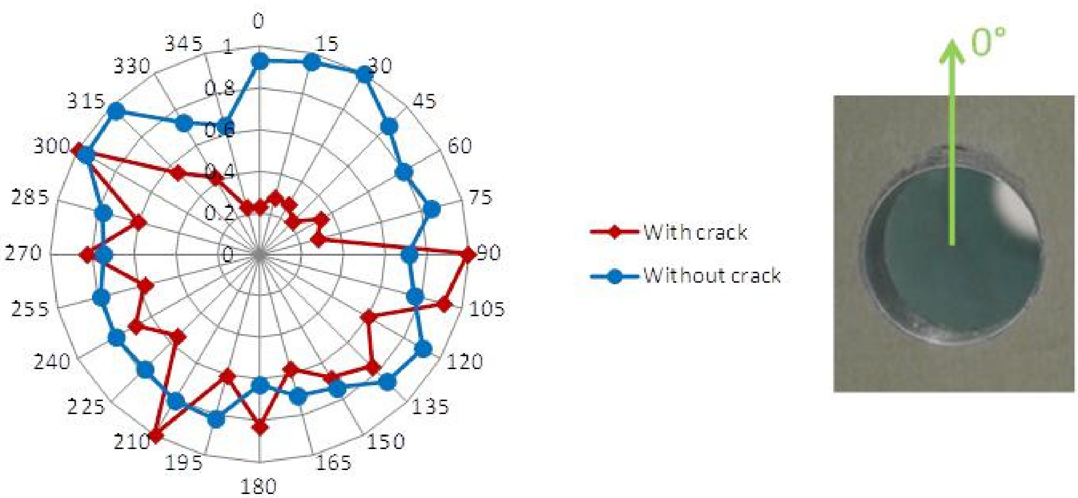


Figure 53. Normalized echo amplitude measured at 24 positions around holes with and without cracks. Solid points represent measured data

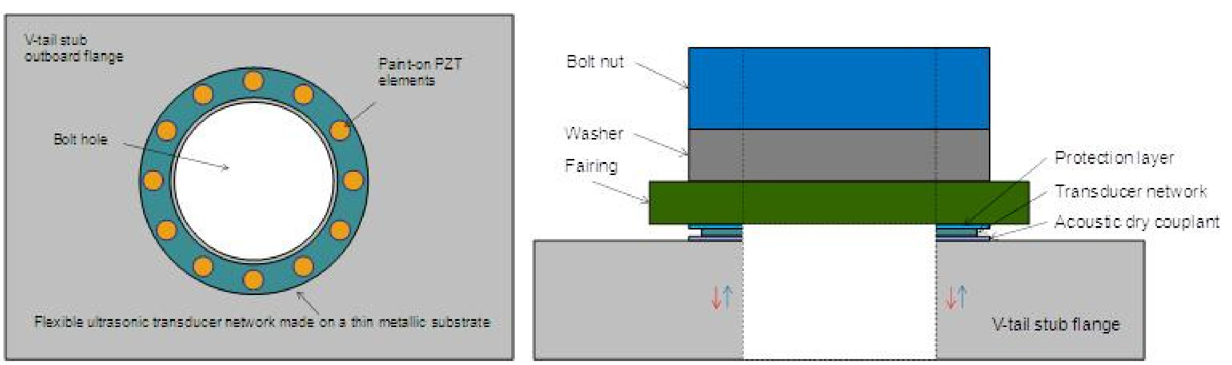


Figure 54. A proposed sensor network layout and sensing configuration for SHM of fretting cracks

Figure 55. A trial version of the proposed sensor network ("Smart washer") for SHM of fretting cracks

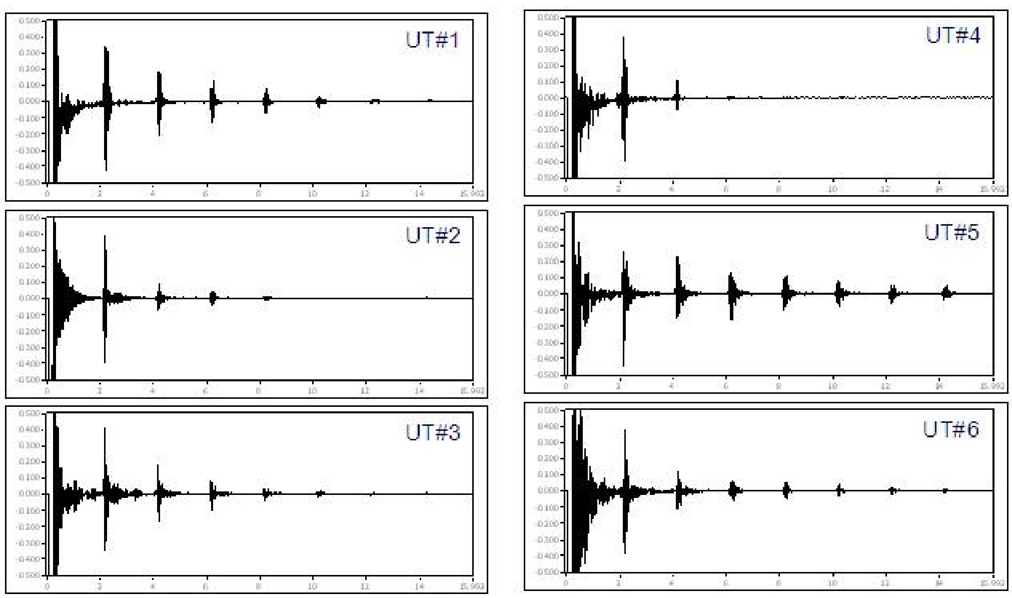


Figure 56. Echo signals generated by the sensor network displayed in Figure 55